Description of document:	10 (ten) reports produced for Heavy Lift and Propulsion Technology (HLPT) Tradeoffs Contracts awarded by the National Aeronautics and Space Administration (NASA) Marshall Space Flight Center (MSFC), 2019
Requested date:	12-November-2018
Release date:	25-October-2019
Release date:	15-November-2019
Posted date:	02-December-2019
Source of document:	FOIA Request NASA Headquarters 300 E Street, SW Room 5Q16 Washington, DC 20546 Fax: (202) 358-4332 Email: hq-foia@nasa.gov

National Aeronautics and Space Administration

Headquarters
Washington, DC 20546-0001

October 25, 2019

Office of Communications

FOIA: 19-MSFC-F-00098

Thank you for your Freedom of Information Act (FOIA) request dated November 12, 2018, and received November 13, 2018, at the George C. Marshall Space Center FOIA Office. Your request was assigned FOIA Case Number 19-MSFC-F-00098 and was for:

> A copy of each report produced under each of these Heavy Lift and Propulsion Technology (HLPT) tradeoffs contracts awarded by NASA (MSFC)
>
> NNM11AA09C awarded to Boeing
> NNM11AA08C awarded to ATK Launch Systems
> NNM11AA10C awarded to Lockheed Martin Corporation
> NNM11AA06C awarded to Analytical Mechanics Associates, Inc.
> NNM11AA05C awarded to Aerojet-General Corporation.
> NNM11AA16C awarded to United Launch Alliance, LLC
> NNM11AA07C awarded to Andrews Space Inc.
> NNM11AA11C awarded to Northrop Grumman Systems Corp.
> NNM11AA14C awarded to Pratt & Whitney Rocketdyne/ United Technologies
> NNM11AA13C awarded to Orbital Sciences Corp.

Please be advised that the search for responsive records has concluded and a total of 1,644 pages have been located. We have reviewed the responsive records under the FOIA to determine whether they may be accessed under the FOIA's provisions. Based on that review, this office is providing the following:

Boeing report (369 pages):

__69_ page(s) are being released in full (RIF);[1]
__29_ page(s) are being released in part (RIP);
_271_ page(s) are withheld in full (WIF).

[1] All page counts above are provided in approximate numbers.

ATK Launch Systems Report (195 pages):

20 page(s) are being released in full (RIF); [2]
6 page(s) are being released in part (RIP);
169 page(s) are withheld in full (WIF).

Lockheed Martin Corporation Report (142 pages):

58 page(s) are being released in full (RIF);
2 page(s) are being released in part (RIP);
82 page(s) are withheld in full (WIF).

Analytical Mechanics Associates Report (197 pages):

0 page(s) are being released in full (RIF);
0 page(s) are being released in part (RIP);
197 page(s) are withheld in full (WIF).

Aerojet-General Corporation Report (149 pages):

33 page(s) are being released in full (RIF);
0 page(s) are being released in part (RIP);
68 page(s) are withheld in full (WIF).

United Launch Alliance Report (92 pages):

15 page(s) are being released in full (RIF);
3 page(s) are being released in part (RIP);
74 page(s) are withheld in full (WIF).

Andrews Space Report (166 pages):

29 page(s) are being released in full (RIF);
5 page(s) are being released in part (RIP);
131 page(s) are withheld in full (WIF).

Pratt & Whitney Report (71 pages):

28 page(s) are being released in full (RIF);
2 page(s) are being released in part (RIP);
41 page(s) are withheld in full (WIF).

[2] All page counts above are provided in approximate numbers.

Orbital Sciences Report (184 pages):

__129__ page(s) are being released in full (RIF); [3]
__23__ page(s) are being released in part (RIP);
__32__ page(s) are withheld in full (WIF).

Please be advised that release of report NNM11AA11C, awarded to Northrop Grumman Systems Corporation, will be released separately within the next 10 business days.

NASA redacted from the enclosed documents information that fell within the following FOIA Exemptions explained below.

Exemption 3, 5 U.S.C. § 552(b)(3)

Exemption 3 concerns matters that are "specifically exempted from disclosure by statute … provided that such statute (A) requires that matters be withheld from the public in such a manner as to leave no discretion on the issue, or (B) establishes particular criteria for withholding or refers to particular types of matters to be withheld." *See* 5 U.S.C. § 552 (b)(3). Pursuant to the Export Administration Act of 1979, in conjunction with the Export Control Act of 2018 (P.L.115-232, Subtitle B, Part I), NASA withholds export controlled information, including items on the Commerce Control List (15 C.F.R. § 774).

Exemption 4, 5 U.S.C. § 552(b)(4)

Exemption 4 protects trade secrets and commercial or financial information obtained from a person that is privileged or confidential. *See* 5 U.S.C. § 552(b)(4). Courts have held that this subsection protects (a) confidential commercial information, the disclosure of which is likely to cause substantial harm to the competitive position of the person who submitted the information and (b) information that was voluntarily submitted to the government if it is the kind of information that the provider would not customarily make available to the public. Thus NASA invokes Exemption 4 to protect contractor proprietary information.

Exemption 6, 5 U.S.C. § 552(b)(6)

Exemption 6 allows withholding of "personnel and medical files and *similar files* the disclosure of which would constitute a clearly unwarranted invasion of personal privacy." *See* 5 U.S.C. § 552(b)(6). NASA is invoking Exemption 6 to protect personal signatures.

You have the right to treat this delay as a denial of your request. Under 14 CFR § 1206.700, you may appeal this denial within 90 calendar days of the date of this letter by writing to:

Administrator
NASA Headquarters
Executive Secretariat

---

[3] All page counts above are provided in approximate numbers.

[1] All page counts above are provided in approximate numbers.

MS 9R17
300 E Street, SW
Washington, DC 20546
ATTN: FOIA Appeals

The appeal should be marked "Appeal under the Freedom of Information Act" both on the envelope and the face of the letter. A copy of your initial request must be enclosed along with a copy of the adverse determination and any other correspondence with the FOIA office. In order to expedite the appellate process and ensure full consideration of your appeal, your appeal should contain a brief statement of the reasons you believe this initial decision to be in error.
For further assistance and to discuss any aspect of your request you may contact NASA's Chief FOIA Public Liaison at:

Stephanie Fox
Chief FOIA Public Liaison
Freedom of Information Act Office
NASA Headquarters
300 E Street, S.W., 5P32
Washington D.C. 20546
Phone: 202-358-1553
Email: Stephanie.K.Fox@nasa.gov

Additionally, you may contact the Office of Government Information Services (OGIS) at the National Archives and Records Administration to inquire about the FOIA mediation services it offers. The contact information for OGIS is as follows: Office of Government Information Services, National Archives and Records Administration, 8601 Adelphi Road-OGIS, College Park, Maryland 20740-6001, e-mail at ogis@nara.gov; telephone at 202-741-5770; toll free at 1-877-684-6448; or facsimile at 202-741-5769.

Important: Please note that contacting any agency official including the undersigned, NASA's Chief FOIA Public Liaison, and/or OGIS is not an alternative to filing an administrative appeal and does not stop the 90 day appeal clock. If you have further questions, please feel free to contact me at martha.e.terry@nasa.gov or 202-358-2339.

In accordance with § 1206.804 (c), after consultation with the NASA Marshall Space Flight Center General Counsel Office, I am the official responsible for the denial of your request. If I can be of further assistance, please feel free to contact me at martha.e.terry@nasa.gov or Stephanie Fox at the contact information provided above.

Sincerely,

Martha Terry
NASA FOIA Officer
Headquarters, Office of Communications

National Aeronautics and Space Administration

Headquarters
Washington, DC 20546-0001

November 15, 2019

Office of Communications

FOIA: 19-MSFC-F-00098

Thank you for your Freedom of Information Act (FOIA) request dated November 12, 2018, and received November 13, 2018, at the George C. Marshall Space Center FOIA Office. Your request was assigned FOIA Case Number 19-MSFC-F-00098 and was for:

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> NNM11AA07C awarded to Andrews Space Inc.
> NNM11AA11C awarded to Northrop Grumman Systems Corp.
> NNM11AA14C awarded to Pratt & Whitney Rocketdyne/ United Technologies
> NNM11AA13C awarded to Orbital Sciences Corp.

On October 25, 2019, responsive documents for all companies listed in your request were released to you. We informed you at that time that the report under contract NNM11AA11C (awarded to Northrop Grumman) would be released separately. The program office located a total of 79 pages in response to your request for the Northrop Grumman report. Processing of these pages is now complete. We have reviewed the responsive records under the FOIA to determine whether they may be accessed under the FOIA's provisions. Based on that review, this office is providing the following:

__64__ page(s) are being released in full (RIF);[1]
__14__ page(s) are being released in part (RIP);

[1] All page counts above are provided in approximate numbers.

___1_ page is withheld in full (WIF).

NASA redacted from the enclosed documents information that fell within the following FOIA Exemptions explained below.

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In accordance with § 1206.804 (c), after consultation with the NASA Marshall Space Flight Center General Counsel Office, I am the official responsible for the denial of your request. If I can be of further assistance, please feel free to contact me at martha.e.terry@nasa.gov or Stephanie Fox at the contact information provided above.

Sincerely,

Martha Terry
NASA FOIA Officer
Headquarters, Office of Communications

Broad Agency Announcement NNM10ZDA001K 1 June 2011

Heavy Lift and Propulsion Technology Systems Analysis and Trade Study

FINAL REPORT

ATK

Heavy Lift and Propulsion Technology Systems Analysis and Trade Study

FINAL REPORT

1 June 2011

Submitted to:

NASA/Marshall Space Flight Center
Office of Procurement
Marshall Space Flight Center, AL 35812

Submitted by:

ATK

Alliant Techsystems

620 Discovery Drive, Bldg. 2, Suite 200
Huntsville, AL 35806

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ACRONYMS

AR&D............Automated Rendezvous and Docking
BAA..............Broad Agency Announcement
CRL...............Capability Readiness Level
DDT&E..........Design, Development, Test, and Evaluation
DRA..............Design Reference Architecture
DRD..............Design Requirements Document
ESAS.............Exploration Systems Architecture Study
FOM..............Figure(s) of Merit
GEM..............Graphite Epoxy Motor
HLLV............Heavy Lift Launch Vehicle
HLPT.............Heavy Lift and Propulsion Technology
HVFU............High Voltage Firing Unit
IOC................Initial Operational Capability
IMLEO...........Initial Mass in Low Earth Orbit
I_{sp}....................Specific Impulse
ISRU.............In-Situ Resource Utilization
LEO...............Low-Earth Orbit
LH_2................Liquid Hydrogen
LOM..............Loss of Mission
LOX..............Liquid Oxygen
MRL..............Manufacturing Readiness Level
MMH.............Monomethylhydrazine
NEO..............Near Earth Object
NTO..............Nitrogen Tetroxide
NTR...............Nuclear Thermal Rocket
ORSC............Oxygen Rich Staged Combustion
PRL...............Process Readiness Level
RBM..............Re-Boost Module
RP..................Refined Petroleum (RP-1)
RSRM...........Reusable Solid Rocket Motors
SDLV............Space Shuttle-Derived Launch Vehicle
SOW..............Statement of Work
SSME............Space Shuttle Main Engine
TIM...............Technical Interchange Meeting
TRL...............Technology Readiness Level
VASIMR........Variable Specific Impulse Magnetoplasma Rocket

Executive Summary

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SYSTEM ANALYSIS

1 TECHNICAL APPROACH

Alliant Techsystems Inc. (ATK) conducted system analyses and trade studies to objectively trade appropriate key attributes that support human space flight exploration. ATK followed a systems analysis approach portrayed in Figure 4 that uses probabilistic analysis to effectively and objectively allow system architecture trades of key decision attributes/figures of merit/measures of effectiveness, ground rules and assumptions, and weighting factors. The cyclic nature of this approach also allows for the process to be repeated as knowledge and insights increase.

The architecture trade space started with an initial assessment of the NASA architectures summarized in the Heavy Lift Launch Vehicle (HLLV) study provided with the broad agency announcement (BAA). Upon completion of the first technical interchange meeting (TIM-1) ATK narrowed the architecture trade space to the leading candidate architectures and refined the figures of merit (FOM) based on what was learned through the initial iteration of the trade study and development of the probabilistic analysis tool. Additionally, an evaluation of potential in-space mission architectures was performed to identify capability gaps that must be bridged to achieve manned exploration beyond Earth orbit.

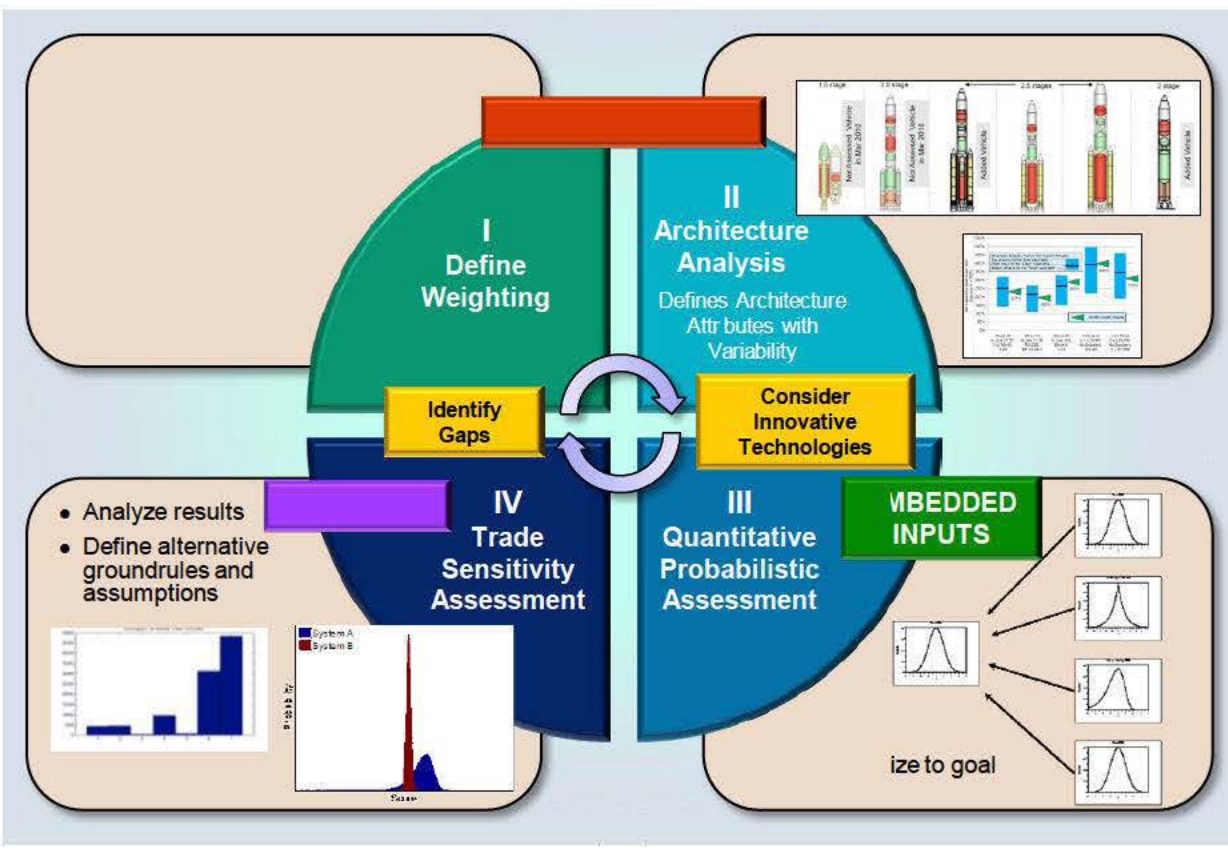

Figure 4. System Optimization Tool and Trade Study Approach

2 PROBABILISTIC ASSESSMENT TOOL

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3 MISSION DEFINITION

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4 PROBABILISTIC ASSESSMENT OF HEAVY LIFT LAUNCH VEHICLE

4.1 Heavy Lift Launch Vehicle Characteristic Architectures

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The results of the IOC FOM as prepared for TIM-1 are listed in Table 11. For TIM-2, no update was performed on the analysis. The data for the three down selected architectures remained unchanged.

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4.2.4 Cost

Three FOM were established to assess affordability for this study. These were the Funding Profile FOM, Life Cycle Cost FOM, and the DDT&E FOM

4.2.4.1 Funding Profile

4.2.4.1.1 Funding Profile FOM rationale

The Funding Profile FOM was initially established with a weighting of 15% based on the desire to meet an affordable funding profile. The concern here is to not have an architecture that requires more funds in a given annual budget cycle than can be expected.

4.2.4.1.2 Analysis Description

The FOM was designed to capture the impact aggressive schedules for the IOC FOM might have on the year to year funding availability. IOC was calculated without regards to the availability of funds in any given year; therefore a given architecture could theoretically have an early IOC but require substantially more funds than are available during its compressed development timeline.

The available funding profile was calculated by taking the Human Exploration Framework Team (HEFT) projected budget bogeys and their Design Reference Mission 4 Heavy Lift Launch Vehicle yearly allocation, adjusted with a knockdown factor based on the projected available budget (ie if HLLV had $500 allocated in 2015, while all of HEFT had $1,000 and there was only $800 available, the overage was allocated equally such that the HLLV line was adjusted down with a 20% knockdown factor to $400). The assumed available funding profile is shown in Table 12 below (values in millions). Note these values are in line with or substantially less than the amounts authorized in the National Aeronautics and Space Administration Authorization Act of 2010 for the first 3 years covered by the act.

Table 12. Available Funding Profile (assumed)

Year	2012	2014	2015	2016	2017	2018	2019	2020
Funding	$ 1,825	$1,896	$1,753	$1,708	$1,863	$1,774	$1,941	$1,874

Next a crude funding required table was built for each architecture based on evenly distributing the DDT&E cost over the time from ATP (assumed to be the start of FY12) to the IOC date for each architecture. This was intended to be a crude approach for the TIM-1 analysis, with the idea of coming back during TIM-2 to build more realistic time-phased cost models. However, the results from this crude approach (shown in Table 13 below) revealed that only the architecture with the shortest development schedule (shuttle derived w/ no upper stage) managed to trip the funding profile, however this overage occurs after the IOC date and is therefore not relevant. Based on these results, it was determined that our cost models were far enough below NASA's estimates that the funding profile was not likely to be tripped even with more accurate time-phased cost models. In order to make the funding profile FOM a useful discriminator between architectures, we would have needed to come up with a new, lower assumed funding profile. Instead, we chose to eliminate this FOM and reallocate its overall weight as discussed in the FOM weighting Section 4.3.

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4.2.4.2 Life-Cycle Cost

4.2.4.2.1 Life Cycle Cost FOM rationale.

In the HLPT BAA request, NASA identified low DDT&E and Life Cycle Costs as parameters of primary importance. The Life Cycle FOM was established with weighting of 15% based on the desire to meet a total life cycle cost.

4.2.4.2.2 Life Cycle Cost Analysis Description

All costs were determined through research of current and historical programs, open source data and engineering estimates. Two different business practice models were assumed: a government and commercial model. The primary difference between the two business models are in the area of government oversight which also contribute to the staffing levels required by the contractor. Since the tool used in this study is based on a probabilistic assessment, cost variation inputs were based on the fidelity of the costs source data, the vehicle complexity and the known demonstrated performance of the various vehicle elements. All vehicles were given a -10% variation on the low side since under running a complex technical development program has been shown to be optimistic. The variation on the high side (Overrun) differed with each vehicle concept. Appendix B.2.2 contains the detailed costs assumed for each of the vehicle elements for both DDT&E and recurring costs and the source of the data.

The life cycle for the launch vehicle portion of this study was assumed to be the DDT&E and flights required to place 800 metric tons of payload in a 30 X130 nm LEO. The data was gathered in two areas; 1st for the DDT&E phase of the vehicle design and 2nd the projected recurring costs times the number of flight required to place the required amount of payload into orbit.

4.2.4.3 Design, Development, Test and Evaluation Cost

4.2.4.3.1 DDT&E FOM rationale.

The Life Cycle FOM was established with weighting of 15% based on the desire to meet a DDT&E cost target. This FOM is one of three FOM established to assess affordability for this study.

4.2.4.3.2 DDT&E Analysis Description

All costs were determined through research of current and historical programs, open source data and engineering estimates. Two different business practice models were assumed: a government and commercial model. The primary difference between the two business models are in the area of government oversight which also contribute to the staffing levels required by the contractor. Since the tool used in this study is based on a probabilistic assessment, cost variation inputs were based on the fidelity of the costs source data, the vehicle complexity and the known demonstrated performance of the various vehicle elements. All vehicles were given a -10% variation on the low side since under running a complex technical development program has been shown to be optimistic. The variation on the high side (Overrun) differed with each vehicle concept. Appendix B.2.2 contains the detailed costs assumed for each of the vehicle elements for both DDT&E and recurring costs and the source of the data.

The DDT&E cost target is for the development of the vehicle to includes full development and resulting in the first flight of the resulting vehicle. The target level for this FOM was $11.5B for this effort.

4.2.4.4 Life Cycle cost and DDT&E FOM Results

The initial results of both the Life Cycle Cost FOM and the DDT&E FOM are shown below in Table 14 and Table 15

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Table 16 and Table 17 summarize the results of the study for the architectures down selected for TIM-2. The cost assessment results indicate that the three remaining architectures all meet the DDT&E requirement of being equal to or less than $11.5B. On a life cycle cost the SDLV scored the best.

Table 16. Final Commercial Cost Model Results (TIM-2)

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Table 17. Final Government Cost Model Results (TIM-2)

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5 READINESS LEVEL ASSESSMENT SUMMARY

5.1 In-space architectures

The entirety of the in-space propulsion related architectures are currently at very low readiness levels. All of these technologies have been discussed at length in academic papers and theoretical research. Some have been previously tested, including nuclear thermal Rockets, but do not have a current use or industry base to draw from. Especially in the case of nuclear thermal rockets, there may even be significant restrictions on the development and testing of these technologies.

Other technology gaps, related to propulsion, are the need for reliable automated rendezvous and docking, advanced propulsion in-space cryo-cooling and zero gravity fluid transfer. NTR and LOX/LH_2 propulsion require large amounts of LH_2 to be stored in the hostile environment of space. LH_2 must be kept pressurized and at a temperature of -252.87°C (-423.17°F). Hydrogen boil off is a common occurrence due to heat leaks and it is estimated that a hydrogen loss of 0.5 to 1.0% per month would not be uncommon. The technology for cooling of large amounts of propellant for a period of over two years needs to be established and qualified. Some additional information can be found in Appendix A.

Heavy lift launch capability is an enabling technology for providing opportunities to develop and test other technologies in a real space environment. Technology development missions also serve to exercise and mature the heavy lift launch system in preparation for manned missions beyond Earth orbit.

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7 CONCLUSIONS & RECOMMENDATIONS

This study shows that the probabilistic analysis tool can be instrumental in guiding decision processes, and can provide NASA with additional insight into the uncertainties and risk associated with varying trade space options, ground rules and assumptions.

A Space Shuttle derived architecture utilizing heritage hardware and infrastructure is consistently identified as the most cost effective solution with the least schedule uncertainty. Opportunities exist for significant reduction in heritage Space Shuttle program costs through reduced material costs, process optimization, design-for-cost modifications, controlled government-contractor interactions and utilization of right-sized facilities. Additionally, Space Shuttle derived architectures provide the optimum use of solid and liquid propulsion to minimize launch delays, maximize workforce retention, and provides a clear path for evolvability from 70mT to 130+mT payload capabilities.

When evaluating reliability, the SDLV were competitive with the other architectures. The small gains in reliability that may be achievable through non-heritage technology come with considerable cost investment and schedule risk. The maturity growth of new systems significantly impacts the probability of success for large, multiple launch, missions such as manned exploration of Mars. This risk can be bought down through additional, and smaller, pre-cursor missions. These precursor missions could include demonstrations of on-orbit assemblies or destinations such as the International Space Station, Moon or near Earth objects.

Space Shuttle derived architecture can be operational by 2016 and stay within the projected budget constraints. ATK encourages NASA to verify this study, and others like it, by challenging the assumptions and techniques used. Finally, ATK recommends that NASA proceed with development of a Space Shuttle derived heavy lift launch vehicle leveraging existing contracts to the maximum extent possible; targeting operational capability in 2016.

DRDS/DELIVERABLES

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APPENDIX A DETAILED IN SPACE-ANALYSIS

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APPENDIX B DETAILED FIGURES OF MERIT ANALYSES & DATA

B.1 MASS TO LEO

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APPENDIX C SUPPORTING TECHNOLOGIES

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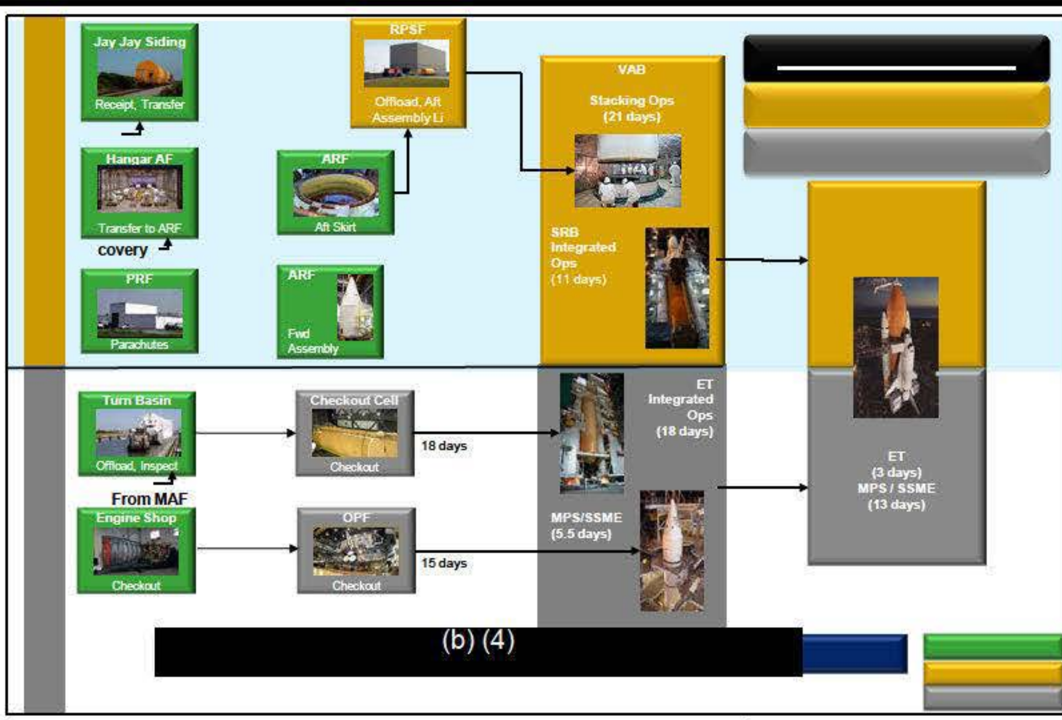

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NASA/Final Report – DRD 1372MA001—3 June 2011

AEROJET

Heavy Lift and Propulsion Technology

Systems Analysis and Trade Study

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Executive Summary

NASA has a mandate for human exploration beyond Earth's orbit. In the last four decades, NASA's two biggest obstacles to achieving its mandate have been obtaining adequate funding and capturing the public imagination with its human exploration missions. Aerojet's approach to this study was to focus, not on expanding the limits of current technology, but rather on developing a sustainable and affordable architecture that would capture the public's interest and that would be within the financial reach of NASA's expected budgets. To accomplish this, Aerojet developed several core tenets of affordable and sustainable beyond LEO space transportation architectures.

Tenet #1 – The architecture must support early (this decade) missions of public interest and must continue regularly scheduled missions that engage the public. Aerojet chose a series of exploration missions beginning with a crewed long duration lunar orbit mission in 2018 and then increasing in challenge and public interest up to a Mars surface landing in 2033. In addition, each exploration campaign involves significant milestone launches and mission events occurring at a regular tempo from the initial launch up to and including the human exploration phase. The Mars surface campaign begins in 2025 with the first cargo launches occurring while the human Phobos exploration campaign is still underway. The Mars surface campaign lasts until 2036 when the crew returns from their 500+ day stay exploring the Martian surface.

Destination	Human Arrival Year
Long Duration Lunar Orbit	2018
NEO (2008 EV5) Surface	2024
Phobos Surface	2025
Mars Surface	2033

Tenet #2 – The in-space architecture must be flexible and each element must support multiple missions. NASA cannot afford to develop unique elements for each mission. The elements must be linked to the selected missions and demonstrated with an incremental approach to permit the step-wise development of multipurpose architectural elements. Our architecture maximizes the use of common elements such as the 70 mT launch vehicle, the common cryogenic in-space propulsion stage, the 150kW SEP Module, and the Space habitat to keep the major engine, stage, and habitat production lines active with continuing production to support a series of missions using the same common elements. This also avoids having years with no production, and the associated sustaining engineering costs of keeping the production lines available.

Tenet #3 – The driving cost of missions is the cost of placing payloads in Low Earth Orbit (LEO). The architecture must reduce the mass required in LEO through the use of near term technology. Prepositioning cargo at the destination enables most of the mission elements to be transported to the destination using mass-efficient space electric propulsion (SEP). This greatly reduces the mass required in LEO. As a slower method of payload delivery, SEP also has the benefit of spreading mission milestones and spending over a longer time base, rather than requiring a large salvo of launches all in one year. Propellant logistics at the destination can also be used to reduce the mass required in LEO. The propellant for crew return from Mars to Earth can be prepositioned using mass-efficient SEP tugs, and verified ready before the crew departs Earth. The use of in-situ manufactured propellants for Mars ascent is also important for saving the mass of Mars ascent propellant delivered from Earth. Finally, for truly robust crew transportation infrastructure beyond 2033, capable of providing crewed missions to Mars at every launch opportunity, the fuel efficiency provided by nuclear thermal rockets is indispensible.

Tenet #4 – Because the driving cost of missions is the cost of placing payloads in LEO, the launch architecture must also minimize infrastructure costs by making maximum use of commonality with other launch systems and with the in-space systems. Aerojet also found that with SEP cargo delivery to the destinations, a 70 mT launch vehicle is capable of meeting all the mission requirements up to the Mars surface campaign. The 70 mT vehicle size also allows hardware commonality with other users such as launch vehicle engines, in space engines, and stage hardware shared by Air Force and commercial users. This distributes the fixed costs of production among users outside the NASA budget and lowers recurring costs for all users. The use of multiple, modular launch vehicles also increases the production rate of each launch vehicle element enabling lower costs and competition during production.

Another key feature of the commonality in the architecture is the separation of the earth departure stage from the launch vehicle stage. This separation allows a smaller upper stage engine on the launch vehicles and also allows the tailoring of the earth departure stage to the payload size in Earth orbit.

Commonality is increased through the separation of crew and cargo. This separation allows a smaller launch vehicle for the crew and allows the larger cargo vehicle to avoid the extra expense of man-rating.

Using these tenets, Aerojet developed an architecture that meets our goal of fitting within NASA's Exploration budget while providing regular exciting missions. **Figure 1** provides a summary of the mass-produced hardware elements, and how they are shared with other users. The recommended engines are summarized in **Figure 2**, and explained in detail in Section 6. Other architecture elements are common to multiple missions, but their production rates are not very high. Steady, medium-rate production of the engines and vehicles for the exploration architecture and other users will ensure hardware reliability and workforce stability.

Figure 1. **Exploration Architecture Hardware Commonality**

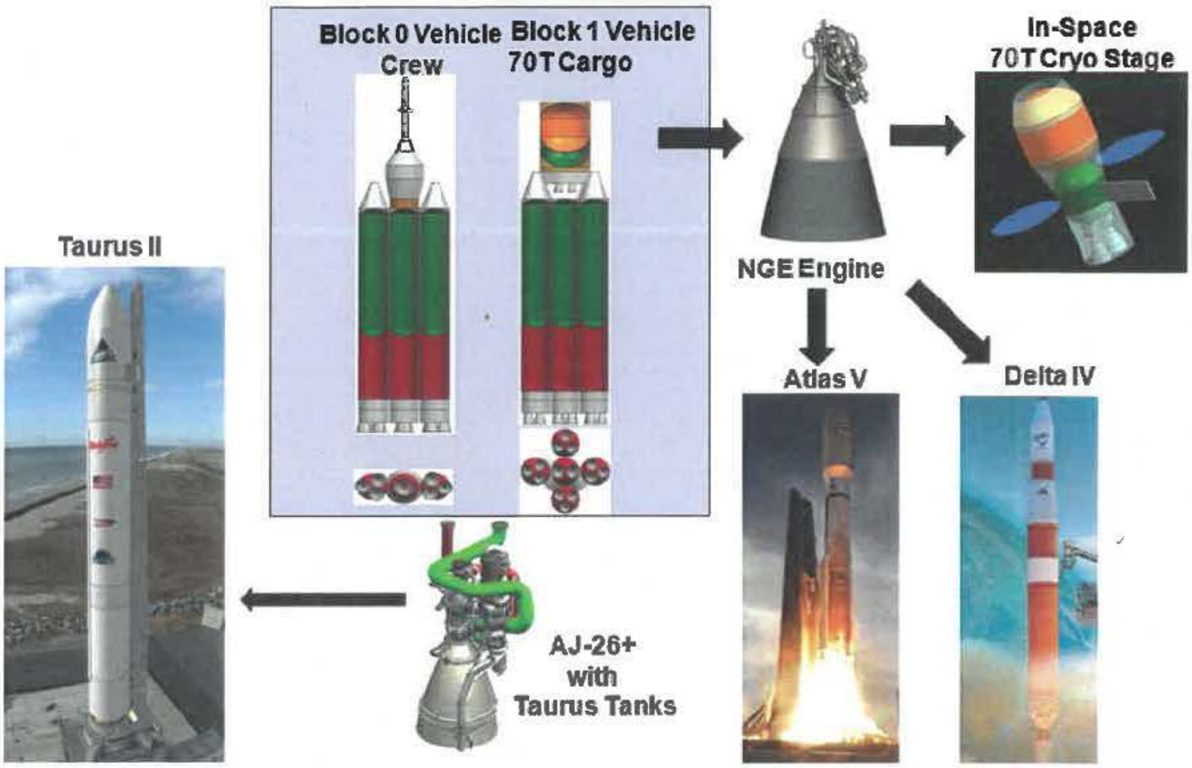


Figure 2. **Exploration Architecture Engines**

Aerojet Engines Discussed in HLPT Report						
						
Engine Name	AJ26-500	NGE	Mars Ascent Engine	SEP 150 kW Module	NTR	AJ-1M
Propellants	LOX/RP	LOX/LH2	LOX/ Methane	Kr or Xe	LH2; (U, Zr)C in Graphite Composite	LOX/RP
Cycle	Ox-Rich Staged Combustion	Augmented Expander	Augmented Expander	Hall Thruster	Expander	Ox-Rich Staged Combustion
Thrust	500,000 lbf (SL)	35,000 lbf (Vac)	35,000 lbf (Vac)	1.35 lbf (Vac)	20,000 lbf (Vac)	1,000,000 lbf (SL)
Specific Impulse (Vac)	332 sec	467 sec	375 sec	3,000 sec	913 sec	331 sec
Mixture Ratio	2.7	5.88	3.5	0	0	2.7
Chamber Pressure	2,280 psi	1,800 psi	1,800 psi	N/A	1,000 psi (2,700 K)	2,280 psi
Area Ratio	27	288	288	N/A	300	27
Dry Mass	4,495 lbm	655 lbm	835 lbm	300 lbm	5,700 lbm	9,800 lbm
Throttling Range	50% to 108%	70% to 100%	Non-throttling	10% to 150%	Non-throttling	50% to 100%
Exit Diameter	69 in	51 in	60 in	40 in	58 in	69 in (150 in overall dia.)
Length	170 in	100 in	105 in	24 in	208 in	174 in

The architecture is sustainable over the long haul. Launches are scheduled in almost every calendar year, and at least one human exploration campaign is always in progress. Figure 3 shows a summary schedule for the overall architecture. A fleet of space electric propulsion (SEP) tugs will be built to ferry vehicles and supplies to Mars and back. Each tug is capable of three round trips to Mars in its 15 year design life. Crew transportation to Mars will be done initially with cryogenic propulsion for the Phobos mission, and later with nuclear thermal rockets that will be capable of mounting a crew mission to Mars with as few as three launch vehicles, and will be positioned to continue crewed Mars exploration on two-year centers after the first Mars surface mission in 2033.

This architecture also meets NASA budget profile as shown in Figure 4. Average spending from 2012 through the Mars surface campaign in 2033 is \$2.7b in constant 2010 dollars, inclusive of all launch vehicle and in-space hardware development. Peak spending is \$5.1b in 2022, which can be leveled by early procurement of launch vehicle and in-space hardware.

Figure 3. **Exploration Architecture Schedule**

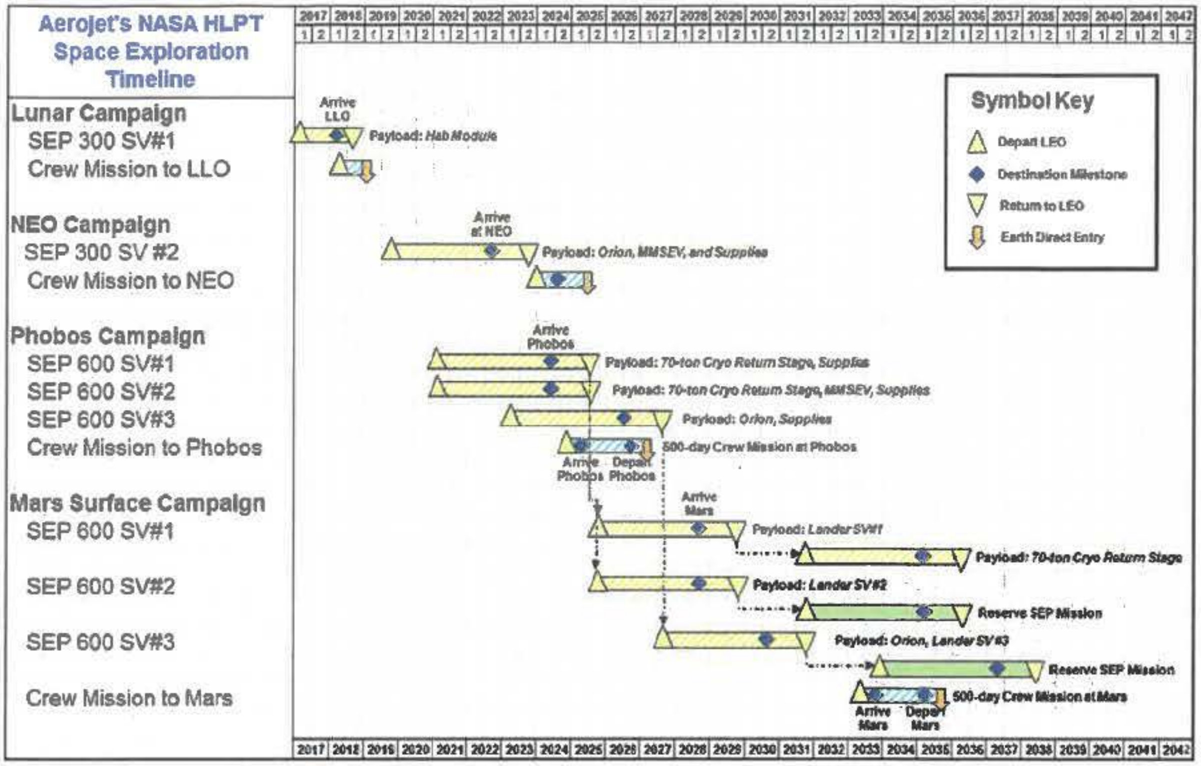
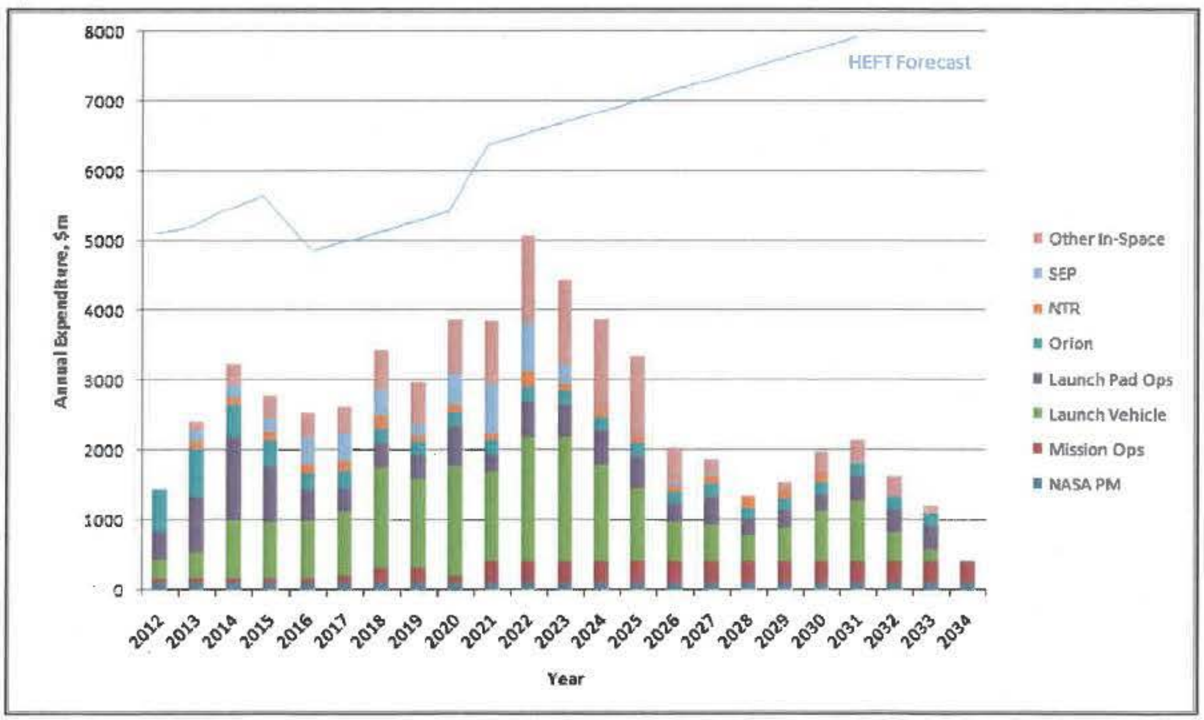


Figure 4. **Budget Profile**



In this report, we have described a plan and architecture that meets the ultimate destination goal of putting humans on the surface of Mars. We have shown that this can be done with a 70 mT launch vehicle. We have shown that we can do it within the time and budgetary constraints, including even more severe assumptions than what the NASA HEFT team assumed.

How is it that we were able to do this? We did it the way that Werner von Braun and Ernst Stuhlinger said it should be done. Von Braun was famously quoted as saying "I would not be at all surprised if we did not fly to Mars electrically." And he dispatched Stuhlinger to go off and explore the potential of electric propulsion – which resulted in his classic book on ion propulsion. The Von Braun team was also heavily involved in the planning and motivation for the NERVA program to develop the nuclear thermal rocket. They knew that these technologies were critical to any exploration program that proposed to take humans further into the solar system.

In doing this we paid attention to three fundamental subjects: Physics, Economics, and History.

The physics of transporting large amounts of material to destinations far beyond low earth orbit drove us to consider highly efficient forms of in-space transportation such as SEP and NTR. We took maximum advantage of these in our architecture. This is necessary to limit the amount of propellant required to accomplish these transfers, which becomes the predominant driver of the Initial Mass in LEO (IMLEO) and therefore the cost of executing the missions.

The economics of producing hardware in sufficient quantities to achieve an affordable unit cost is what drove our approach of commonality (including looking beyond just NASA to DoD and commercial users). It is also what made us select a launch tempo and production rate that kept the workforce productively employed and not reliant on large periods of sustaining engineering.

And finally history showed us that the right path is an evolving one. Just as Apollo did not start with Saturn V launches to the moon, but rather with a Mercury / Redstone launch into a suborbital trajectory, followed by subsequently more capable Mercury / Atlas flights and then by a whole series of Gemini / Titan flights – each one proving critical elements and steps along the path to the moon. This was our motivation for our evolving path to Mars. Each one of the missions in our manifest proves another element or critical aspect before we attempt the next more ambitious mission.

We at Aerojet are grateful for the opportunity to share our thoughts with NASA, and we look forward to continued collaboration with NASA as we implement this exciting vision of human exploration beyond low earth orbit.

Section 1.0—Introduction

The overall objective of this study is to identify an affordable and sustainable architecture, including all launch and in-space elements, for human exploration beyond Low Earth Orbit (LEO), including high Earth orbits, cis-lunar space, Near-Earth Objects, the Martian moons, and ultimately the surface of Mars. A key to the study is an understanding of how the in-space elements directly impact the launch requirements: use of innovative but practical in-space power and propulsion systems dramatically reduce the launch vehicle requirements, resulting in an affordable and sustainable human exploration campaign. This report summarizes the assessment of the Concept of Operations (CONOPS) considered by Aerojet, including architecture element data, in-space transportation options, trade evaluations, and results which demonstrate that the human deep-space exploration can be realistically accomplished within the current NASA exploration budget by properly investing in the architecture elements.

All exploration architecture assessments must start by selecting the target destinations and establishing the delivered mass requirements to complete the planned missions. Once the required delivered mass is established, the transportation architectures are traded to identify the most affordable and sustainable delivery approach which accomplishes the mission objective. The transportation trades are performed by selecting propulsion options, establishing mass models for the transportation vehicles, and using appropriate mission analysis and trajectory codes to establish vehicle requirements for each option. It is critical to remember that the ΔV for a given mission depends on the selected transportation architecture, departure dates, and other constraints (gravity losses, orbital alignment, trip times, etc). The result of this effort is a table of launch mass and date requirements and trip times between the Earth departure orbit and the destination. These results represent the delivery requirements for the launch vehicle trade study: how much do you need to launch and when do you need to launch it? The results of the overall architecture trades are then evaluated against the affordability and sustainability requirements described above. It is critical to note that all costs must be included in this evaluation: technology development, vehicle design, development and build, all launch and ground operations costs (including all required demonstration missions), and the costs of the exploration missions themselves. The fact that many required capabilities and vehicles do not yet exist drives a phased approach to human deep-space exploration: it is critical that the early missions demonstrate the capabilities required by later missions.

1.1 The von Braun Paradigm

In 1994 the phrase "von Braun paradigm" was coined to describe the early approaches laid out by Dr. Werner von Braun for human space exploration that has been largely followed by NASA until present day. This approach is as follows:

- Build a Space Shuttle to enable routine space access
- Build a Space Station to learn how to work in space for long durations and to stage future missions
- Build a lunar outpost to learn how to work off-world and to stage large missions beyond Earth orbit
- Explore Mars

This approach, while not intended to be strict or exactly linear, was leap-frogged by the Apollo program. In the years after Apollo, NASA went on to develop the Space Shuttle and International Space Station (ISS) both of which were disconnected from either Moon or Mars exploration, but which have served to establish capabilities for in-space assembly of large structures and long-term human space habitation. Among the issues plaguing the Space Shuttle and ISS programs are the lack of gains in propulsion beyond pure chemical approaches, the combination of crew and cargo missions, and the enormous ground infrastructure required to support vehicles that are not used by any organizations other than NASA. Only recently, with the advent of the European ATV and Japanese HTV, has NASA started launching crew and cargo separately to take advantage of the lower cost of unmanned launchers. While many studies have considered mission to the Moon and Mars, this work primarily relied on the most recent ESAS[1] and DRA 5.0[2] reports from NASA. The work presented herein represents a reevaluation of the von Braun paradigm using new propulsion technologies and separate crew and cargo approaches to establish an affordable and sustainable human exploration campaign.

1.2 Apollo Reference

The Apollo mission to the moon provides an excellent reference for capabilities and a maturation plan as shown in Table 1-1. This section documents some of the basic aspects of the Apollo space architecture for later comparison.

Table 1-1. **Apollo Reference Metrics**

Architecture Element	IMLEO, mT	Function
Apollo CM	5.8	Crew habitat for up to 6d
Apollo SM	24.5	Lunar capture and Earth Return (AJ10-137 NTO/Aerozine 50 @ 271s Isp)
Apollo LM	14.7	Lunar descent/ascent/surface habitat for 3 day stay
S-IVB	120	Earth departure (J-2 LOX/LH2 @ 421s Isp)

It is important to note that Apollo did not start with a Moon landing – they developed and demonstrated increased capabilities in a series of increasingly difficult missions, starting with. This did not stop after Apollo 11, but rather continued with the introduction of improved engines and other capabilities providing ever greater capability, culminating in a 3-day, multi-excursion stay on the lunar surface with Apollo 17.

1.3 Study Design Approach

Time constraints dictated that our study evaluate the launch and in-space architectures in parallel, following the core tenets of our study described above: using the in-space architecture to drive toward smaller launch vehicles, which in turn enables launch vehicle element commonality across NASA, DoD, and commercial markets to dramatically reduce the overall cost of exploration. For each architecture considered, we carefully matched the boundary conditions for the launch vehicle and in-space elements. The design and analysis approach for the space architecture and exploration campaign are as follows:

1. Design the Architecture
    a. Define the candidate destinations and delivered mass requirements
    b. Define the propulsive mission phases (from launch to Earth return)
    c. Estimate the ΔV requirements for each mission phase and destination
    d. Define the architecture trade space

2. Analyze the Architecture
    a. Optimize propulsion options for each mission phase
    b. Perform sensitivity analysis for each mission
    c. Estimate the maximum condition of each architecture element
3. Design the Exploration Campaign
    a. Define the campaign objectives
    b. Design each mission
        i. Define the mission objectives
        ii. Define the concept of operations
        iii. Estimate the manifest
        iv. Estimate the mission timeline
        v. Estimate the mission cost
    c. Estimate the campaign timeline
    d. Estimate time-phased exploration campaign cost

The following sections review the detailed approach and results for each of these sections, beginning with the launch vehicle.

Section 2.0—Launch Vehicle Architecture Design

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Section 5.0—Exploration Campaign Design

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Section 6.0—Capability Gap Assessment

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Section 7.0—Costing Analysis

7.1 Costing Approach

One of the major objectives of this study was to perform a life cycle cost analysis on the selected exploration architecture. The study divided the architecture costs into categories, including propulsion, launch vehicles, in-space elements, ground operations, mission operations, and NASA program management. Each category was investigated in detail, based on the assumed exploration destinations, mission plans, and required launch manifests as defined elsewhere in this report. Constant 2010 dollars were used throughout the study. The study cost categories are addressed separately in the following sections.

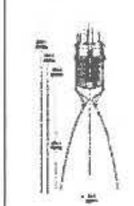
7.2 Launch Vehicle Costing

The two main drivers for the launch vehicle costs were the booster and upper stage engines, and the launch vehicles themselves. Ground operations were a third significant cost element on the launch vehicle side. The derivation of cost estimates for these categories is described in the following sections.

7.2.1 Engine Cost Model

Aerojet provided an engine development and recurring cost model to NASA and the other study participants at the beginning of the present study. We used our own model for estimating the engine costs of the selected architecture, including booster engines, upper stage engines, and the in-space engines for applications including cryo stages and Mars descent and ascent vehicles. Our costs for the nuclear thermal rocket program were developed using a different model, which was created based on work from a concurrent proposal to NASA. The resulting engine life cycle costs are shown in Figure 7-1.

Figure 7-1. **Main Engine Life Cycle Cost Summary**

Main Engine Life Cycle Cost Summary						
Costs Included From Start of Engine Development Through 2033 or End of Program						
Engine	AJ26+	NGE	Mars Ascent Engine	SEP 150kW modules (HCT, PPU, XFC)	NTR	AJ-1M (Optional)*
Life Cycle Cost	$5,580m	$1,749m	$510m	$540M	$1,765m	$3,341m

In the Aerojet engine cost model, a production rate of 10 units per year was assumed sufficient to meet the minimum recurring cost for a given engine. Based on the required launch manifest, this production rate was met for both the AJ26-500 booster engine and the NGE upper stage and in-space engine. The life cycle costs in Figure 7-1 do not reflect additional production of these engines required to meet requirements of other users. For example, the AJ26-500 engine also serves the Taurus II program and potentially the EELV program. The NGE also serves the EELV program with projected demand of 9 engines per year. In fact, production levels significantly above 10 units per year would result in additional decreases in the engine unit cost, but this decrease was not reflected in the engine cost model.

The non-recurring cost for the NGE was assumed to be paid by the US Air Force for their EELV upper stage engine requirement. This is a longstanding program requirement that must be filled for the national interest, whether an exploration program uses the engine or not. We assumed a thrust level for the NGE that is compatible with the Air Force customer requirements, even though a somewhat higher thrust level would be better for the exploration architecture from a purely technical standpoint.

The launch manifest included several years with no launches. This did not result in interruptions to the engine production lines for AJ26-500 and NGE. The production lines were slightly front loaded with higher production rates in the early years to meet the overlapping NEO and Phobos campaigns, then reduced to lower levels that were still above 10 units per year through 2033. As a result, we found that there were no sustaining engineering costs associated with maintaining these engine production lines in years with no production – thus illustrating one advantage of a modular, hardware rich launch vehicle architecture.

The AJ-1M one million pound thrust engine was costed on the basis of its use in the Block II 100 mT and Block III 130 mT class launch vehicles. The number of engines required was calculated based on a revised flight manifest that used fewer launches to put essentially the same set of payloads into the reference LEO. The average production rate was just below the 10 unit benchmark for minimum unit cost. However, in this scenario, development of the AJ26-500 engine was still required to meet the requirement for a 2017 demonstration flight. The AJ26-500 was also used for crew launches in the Block 0 launch vehicle through the Mars surface campaign in 2033. This approach had the advantage that the AJ-1M was never required to be a man-rated engine. The drawback was that the AJ26-500 became a low rate production engine, with average production of about three units per year, and resulting unit cost increased.

The other engines in Figure 7-1 also had lower production rates. The SEP 150 kW module was built for the SEP tugs, which were reuseable with a 15 year service life. Therefore, we assumed this line would be shut down after the launch of the last reuseable tug. Small sustaining costs were included to keep the line open during years of no production between the initial 300 kW tug production, and the later 600 kW tug production. The Mars lander engine was included in the production totals for the NGE, since the lander engine was an NGE modified for 2:1 throttling. The Mars ascent engines assumed very low production, since the architecture only included one NTR vehicle and one Mars ascent vehicle. We deferred sustaining engineering costs for these engines, since the definition of exploration activity beyond the first Mars surface landing was outside the scope of the study. In a robust exploration program enabled by the approaches described in this study, it would be anticipated that additional NTR and Mars ascent engines would be built on 2-year or 4-year centers, resulting in production and sustaining engineering costs for both lines in later years.

7.2.2 Launch Vehicle Cost Model

For launch vehicle cost modeling, Aerojet worked with Ares Corporation to take advantage of their experience with the Launch Vehicle Cost Model (LVCM) tool. LVCM was used by the Air Force to estimate the costs for the Evolved Expendable Launch Vehicle (EELV), and later updated by the National Intelligence Center (around 1998). Recently, LVCM was updated and used, along with other cost models, by NASA to estimate the cost of the Constellation Program. LVCM uses Cost Estimating Relationships (CERs) based on actual costs from the Atlas IIAS, Delta II, Titan IV, Minuteman, Peacekeeper, Space Shuttle, and IUS programs. The CERs in the LVCM use parameters that are known to drive costs, e.g., thrust for engines, bit-rate for telemetry, and loads for structural elements. Recurring

CERs in the model estimate a first unit cost that includes direct labor, material, other direct costs, overhead, general and administrative (G&A), fee, and commission. The non-recurring CERs use the first unit cost and the number of development units. Three development were assumed for the present study. A learning curve of 85% was also added for similar elements within a single launch vehicle such that multiple tanks and boosters within a Vehicle Block would benefit from the "rate-effect." Aerojet used an internal model for engine costs, as described above, rather than the LVCM results.

For conservatism, the LVCM first unit cost for launch vehicle elements was applied for all vehicles produced throughout the exploration architecture. No credit was taken for economy of scale at higher production rates of multiple launch vehicles per year. The learning curve of 85% was only applied within a single launch vehicle; for example, the Block I launch vehicle with five nearly identical liquid booster cores benefitted from this. On the other hand, the LVCM derives avionics costs from data rate as the key requirement driver. Since we did not specify data rate for the LV at the time of the study, the model defaulted to the lowest value and produced fairly low avionics costs.

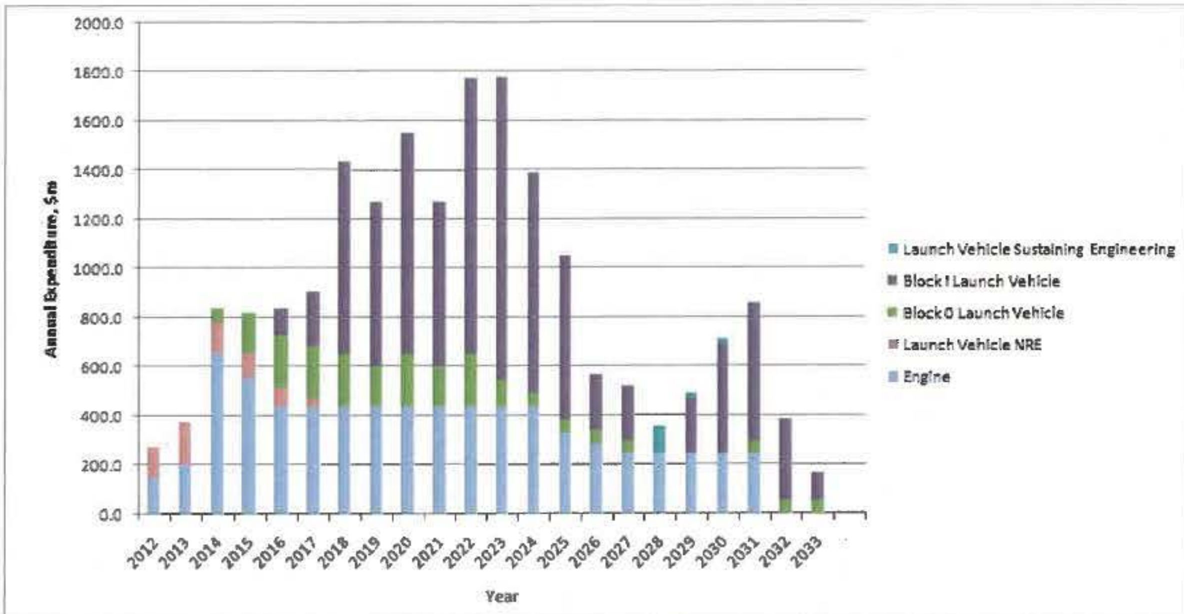
Other groundrules and assumptions for launch vehicle costing included a development timeframe of 5-6 years, and a production cycle length of three years for stages and two years for engines. Sustaining engineering labor costs of 40% of production cost were applied in years with no launch vehicle production. However, this only added a total of $155m to the program for the years 2028-2030 leading up to the Mars surface campaign. LVCM included the cost of spares at the stage level. One unassigned or "white-tail" Block I launch vehicle was also added to the program cost to allow for prompt recovery and relaunch in case of a failure.

The results of the launch vehicle cost analysis are shown in Table 7-1. Launch vehicle spending over the duration of the exploration architecture is shown in Figure 7-2.

Table 7-1. **Launch Vehicle Cost Model Results**

LV Element	NRE	Recurring Cost of Block I Cargo LV (70 mT)	Recurring Cost of Block 0 Crew LV (25 mT)
LRB Module	912.3	329.9	196.2
LV Stage 1	102.5	17.4	17.6
Upper Stage	129.3	90.2	
LV Other	240.2	88.6	19.4
Total	1384.3	526.1	233.2

Figure 7-2. **Projected Launch Vehicle Spending 2012-2033**



Cost Element	Engine	Launch Vehicle NRE	Block 0 Launch Vehicle	Block I Launch Vehicle	Launch Vehicle Sustaining Engineering	Total Per Year
2012	150.0	125.0				275.0
2013	200.0	175.0				375.0
2014	658.6	125.0	54.0	0.0		837.6
2015	558.6	100.0	161.9	0.0		820.5
2016	435.9	75.0	215.9	112.2		838.9
2017	435.9	34.0	215.9	224.4		910.1
2018	435.9		215.9	785.4		1437.1
2019	435.9		161.9	673.2		1271.0
2020	435.9		215.9	897.6		1549.3
2021	435.9		161.9	673.2		1271.0
2022	435.9		215.9	1122.0		1773.7
2023	435.9		107.9	1234.2		1778.0
2024	435.9		54.0	897.6		1387.4
2025	326.9		54.0	673.2		1054.1
2026	286.0		54.0	224.4		564.4
2027	245.2		54.0	224.4		523.5
2028	245.2		0.0	0.0	111.3	356.5
2029	245.2		0.0	224.4	21.6	491.2
2030	245.2		0.0	448.8	21.6	715.6
2031	245.2		54.0	561.0		860.1
2032			54.0	336.6		390.6
2033			54.0	112.2		166.2
Cost Element Total, $M	7329.0	634.0	2104.7	9424.8	154.5	19647.0

7.2.3 Launch Operations Cost Model

Aerojet's approach for the ground operations cost estimate was to perform analysis by similarity to an operationally successful commercial program using similar types of hardware. Sea Launch performed 30 launches of commercial satellites between 1999 and 2009. Although price competition with subsidized competitors led to their eventual bankruptcy, from an operational standpoint the Sea Launch program was a success. Owing to the bankruptcy, some financial data about the venture reached the public domain. Sea Launch sold 30 launches at market rates averaging approximately $65m per launch, for total revenue of $1.95b. Their total accumulated debt at bankruptcy was $2.02b, for total program expenses of revenue plus debt of $3.97b. In July 2010, Sea Launch's Kjell Karlsen stated publicly that the reorganized company can operate profitably on two launches per year, and that their operating costs are $2m per month. Given the current market pricing of commercial launches at $100m, this implies that a total revenue of $200m will cover $24m of operating costs plus two complete launch vehicles at $86m apiece. Projecting these vehicle and operating costs backwards over the 30 launches in 10 years, we obtain a total cost of $2.84b for vehicles and operations. The balance of total program cost, $3.97b minus $2.84b or $930m, represents the approximate non-recurring cost to establish the program. The estimated Sea Launch costs were then scaled for vehicle size and complexity as shown in Table 7-2 to estimate the costs for an analogous approach for the Block 0/Block I launch vehicles.

Table 7-2. **Ground Operations Cost Estimate Breakdown**

Cost Element ($m)	Assumed Sea Launch Cost	Sea Launch Scope	LV Size Factor	Complexity Factor	Estimated Block 0/Block I LV Cost
Development/NRE Costs					
Launch Control Center	233	Command Ship		2.0	466
Launch Platform Acquisition	93	Converted Drilling Platform			N/A
Launch Vehicle Design Adaptation	93	Upper Stage Mods, Fairing, Adapter	2.0	2.0	372
Ground Facilities Development	233	Port, Processing Facility	6.0		1398
Launch Platform Modification and Installation	278	Automation, Fueling Equipment, Propellant Supply	6.0		1668
Total NonRecurring	930				3904
Recurring Costs					
Annual Operations	24	Two launches per year	5.0	2.0	240

The cost exercise we performed assumed that launch vehicle operations development, planning, and execution would be conducted on a commercial basis by one of the major US prime contractors, much as Boeing handled the development of the Sea Launch facilities. The level of insight and oversight exercised by NASA was assumed to be less than was done for the shuttle and other past programs. The resulting annual operations cost of $240m is approximately 50% of the expected operations cost to maintain a shuttle derived architecture. This is a reasonable result in terms of the number of facilities supported. The shuttle derived architecture uses two full service pads, the VAB, several mobile service platforms, and the command center. The architecture of this study uses only one pad, converted to a clean pad with no payload access capability, a minimum size launch vehicle integration facility, the command center, and retires the remaining shuttle program assets.

7.2.4 *Alternative Launch Vehicle Architecture With 130 mT Payload Capability*

Aerojet studied an alternative launch vehicle architecture using the Block II 100 mT and Block III 130 mT launch vehicle designs described earlier in this report. These vehicles used an 8.3 meter diameter core stage, with the 6.8 m diameter hydrogen upper stage identical to the version used on the Block I launch vehicle. The Block III differed from the Block II only by the addition of two liquid rocket booster stages based on the common booster module of the Block 0/Block I launch vehicles.

The Block II/Block III launch vehicle architecture required the development of a one million pound class booster engine, the AJ-1M, described in detail above. This was required to reduce the total number of engines per vehicle back to quantities comparable to the Block I launch vehicle for equivalent engine reliability. Additionally, development of the Block II/Block III did not obviate the need to develop the Block 0 launch vehicle with its AJ26-500 engine. Both the separation of crew from cargo philosophy and the need for a 2017 initial launch capability required the development of the Block 0 launch vehicle, even though the Block I was never developed or used in the alternative architecture. This resulted in a requirement for two sets of launch pad and ground processing infrastructure, since the Block 0 and Block II/III vehicles were not geometrically similar and could not use the same launch pad and ground processing infrastructure.

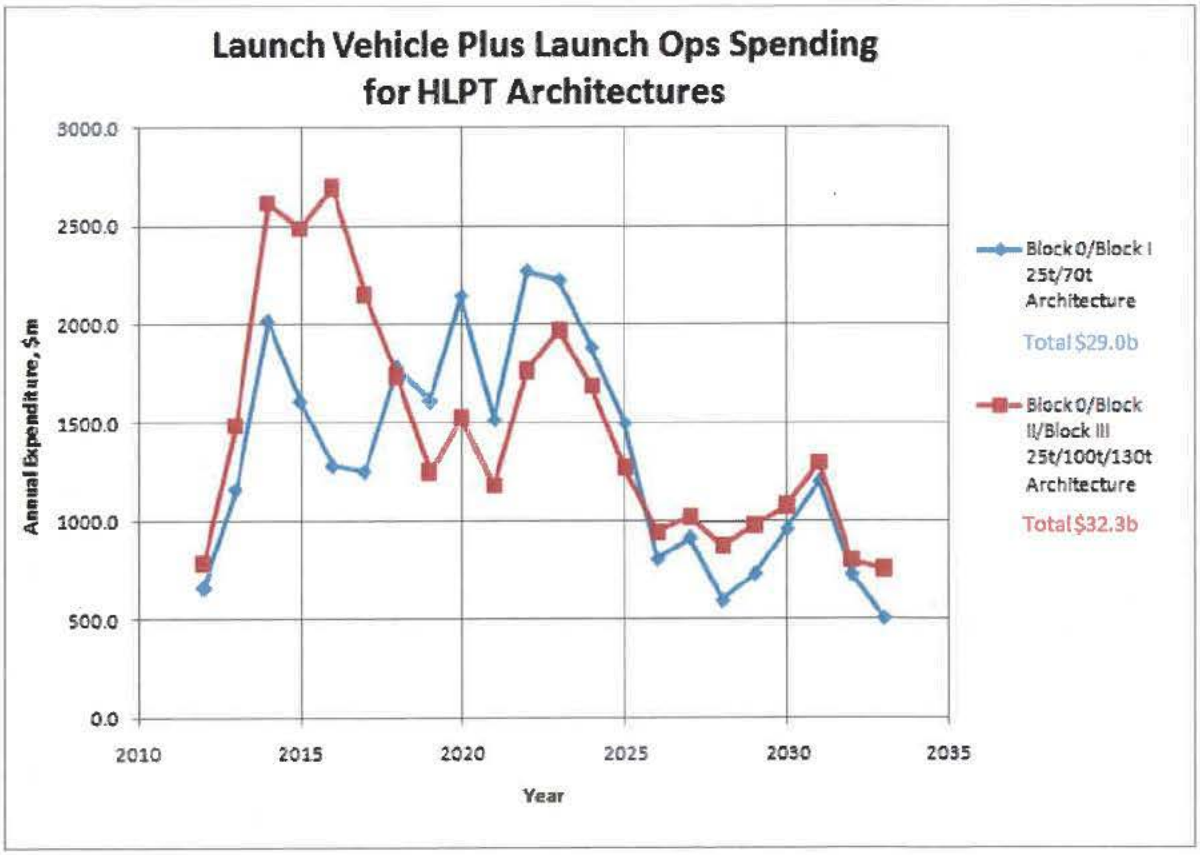
A cost comparison was performed by allocating the required manifest of in-space elements to Block II/Block III launch vehicles instead of the 70 mT Block I vehicle. Mission elements were combined to take maximum advantage of the 100 mT and 130 mT vehicles. The 100 mT vehicle was used when possible to eliminate the cost of the strap-on liquid boosters for those launches. Table 7-3 shows the comparison of the two alternative launch manifests. The total number of launches required was reduced from 41 to 26 by the use of the larger launch vehicles.

Table 7-3. **Comparative Launch Manifests for 70 mT and 130 mT LVs**

Year	Baseline Launch Campaign			Alternative Launch Campaign		
2011	Block 0 LV	Block I LV		Block 0 LV	Block II LV	Block III LV
2012						
2013						
2014						
2015						
2016	1			1		
2017	2			2		
2018	1	1		1	1	
2019	1	1			1	
2020	2	5			1	2
2021						
2022	2	3			1	1
2023	1	3		1	1	1
2024	1	4		1	1	2
2025		4				2
2026						
2027	1	2				1
2028						
2029						
2030						
2031		2			2	
2032		2				1
2033	1	1		1	1	
Subtotals	13	28		7	9	10
Grand Total		41				26

The launch vehicle and ground operations combined costs were compared for the two alternative architectures. Launch vehicle and engine costs were built up for the 100 mT and 130 mT launch vehicles using the same approach as for the 70 mt launch vehicle architecture. Sustaining engineering costs of engine and launch vehicle production lines were more significant than for the 70 mT architecture, and amounted to \$1.2b over the 18 year period of analysis. This was due to generally lower quantities of flight hardware and the corresponding lower production rates. Operations costs assumed development and operation of two launch vehicle processing facilities and two pads. The cost of the second facility and pad was assumed to be 50% of the cost for the first, due to commonality of infrastructure and support personnel. The results of the 130 mT architecture study are shown in Figure 7-3.

Figure 7-3. **Comparative Costs of 70 mT and 130 mT LV Architectures**



From the figure, the 130 mT architecture, in red, has a significant spending peak in the early years due to additional engine, launch vehicle, and ground facility development. The 70 mT architecture, in blue, has a higher peak in the middle years, due to the higher numbers of launches required to meet the NEO and Phobos campaign requirements. Finally, the sustaining costs of the additional engine lines and ground facilities lead to higher costs for the 130 mT architecture in the later years, due to the lower launch rate. The total spending for the two architectures over the 18 year period was approximately the same. Our conclusion was that the 130 mT architecture did not show promise of significant cost savings over the 70 mT architecture, despite the lower number of launches.

7.3 In-Space Element Costing

The in-space portion of the exploration architecture contained a number of various vehicle types. We defined the schedule and cost of these architecture elements using NAFCOM-based estimates or scaling from comparable programs, including consultation with other contractors. The hardware demand list used for costing the in-space part of the architecture is shown in Table 7-4.

Table 7-4. Hardware Demand List for In-Space Architecture

Destination	Year	Launch Vehicles			In-Space Elements														
		Block 0	Block I	Orion	SEP 300	SEP 600	Xenon Tank	Equipment Pallets	Space Habitat	Cryo Stage 70 t	Cryo Stage 50 t	Mars Sample Return Mission	Multi-Mission Space Exp. Veh.	Mars Descent Stage/Lander	Mars Ascent Stage	Mars Surface Habitat	Rovers	ISRU Plant	Mars Surface Power System
ISS	2016	1		1															
Lunar Orbit	2017	2			1		1	1	1										
	2018	1	1	1						1									
NEO	2019	1	1	1	1		1	1			1		1						
NEO/Phobos	2020	2	5			2	2	2		2		1	1						
	2021																		
	2022	2	3	1		1	1	1		2									
	2023	1	3	1					1	2									
Phobos	2024	1	4	1					1	3	1								
Mars	2025		4				2						1	2	1	1	2	1	1
	2026																		
	2027	1	2	1			1						1	1			1		
	2028																		
	2029																		
	2030																		
	2031		2				1			1									
	2032		2							2									
	2033	1	1	1					1		1								

The development timeframe of most elements was 5-7 years, although 10 years was assumed for the NTR due to its unique attributes. The production cycle for in space elements was three years, and recurring costs were divided evenly across the three years preceding flight. When gaps in production occurred for elements in ongoing serial production, sustaining engineering costs equivalent to one unit per year were added to maintain the line. It was assumed that contractor personnel beyond this level would be shifted to other customers' programs due to commonality. Finally, spare hardware was allocated to allow for recovery from some anomalies. One unassigned 70 mT space cryogenic stage was costed, and each SEP tug included one spare 150 kW engine module in the flight configuration to accommodate failure of other modules.

7.4 Other Exploration Architecture Cost Elements

Other exploration architecture costs included the Orion program, NASA program management expense, and mission operations. For the Orion program, we estimated the non-recurring cost to complete the design as $2b, with a recurring cost per unit of $200m based on two units per year. For most of the architecture, Orion production was one unit or zero per year, so sustaining costs were imposed at 40% of production cost during inactive periods. Total Orion expense worked out to $1.6b for the campaign, with sustaining costs equal to $2b spread over the 18 year period considered.

NASA program management costs were estimated at $100m per year on average. This is a significant change from the shuttle derived baseline value of $419m per year, and reflects implementation of a commercial approach to procurement of mass-produced launch vehicle and in-space hardware. For major hardware items in serial production, the NASA level of participation was assumed to be equivalent to a commercial customer, with attendance at major reviews, but reduced oversight into production details and no parallel analysis.

Mission operations costs were also assumed to be reduced compared to past practice. A mission ops budget was assumed to begin at $50m per year in 2012 to support planning, then rise to $300m at steady state during the exploration missions. This compared to $1.1b per year for the shuttle derived baseline. The assumed reduction reflected the reduced operations tempo compared to the STS era, smaller mission operations staffing, and greater use of autonomy by the exploration crews, partly to save costs and partly of necessity due to the great distances involved.

7.4.1 Cost Sharing

In a few areas, it appeared reasonable to share exploration architecture costs with other budget accounts or different end users. The most important example of this was the NGE engine development. The US Air Force has a firm requirement for a production hydrogen/oxygen upper stage engine in the 25 klbf to 35 klbf thrust class. The incumbent RL-10 engine is not in current production, and a new engine production line will have to be established for future hardware deliveries in the next few years, whether the incumbent RL-10 engine line is re-established, or a new engine such as the Aerojet NGE is built. NASA could greatly benefit from splitting the cost for this engine, as well as reaping the performance dividend of a new engine designed from the beginning with modern technology. For our cost analysis, we assumed use of the Air Force's NGE with all non-recurring cost paid by the Air Force given their firm requirement for this engine.

Another area where cost sharing was assumed was in the technology development and validation programs for some of the in-space elements. In particular, the Mars sample return mission in 2020 was assumed to be partly paid for by the NASA technology or space science budgets, rather than the human exploration budget. The Mars sample return mission will demonstrate some important technologies applicable to human exploration, such as subscale aeroshell and lander design, and a pilot plant for Mars ISRU.

For development of the nuclear thermal rocket engine, the nuclear fuel is one of the most expensive program elements, but we assumed the fuels cost would come from the NASA exploration architecture, and these costs are reflected in our cost projection. However, some of the engine technology cost and test facilities cost was assumed to be shared with the DoE and NASA technology budget.

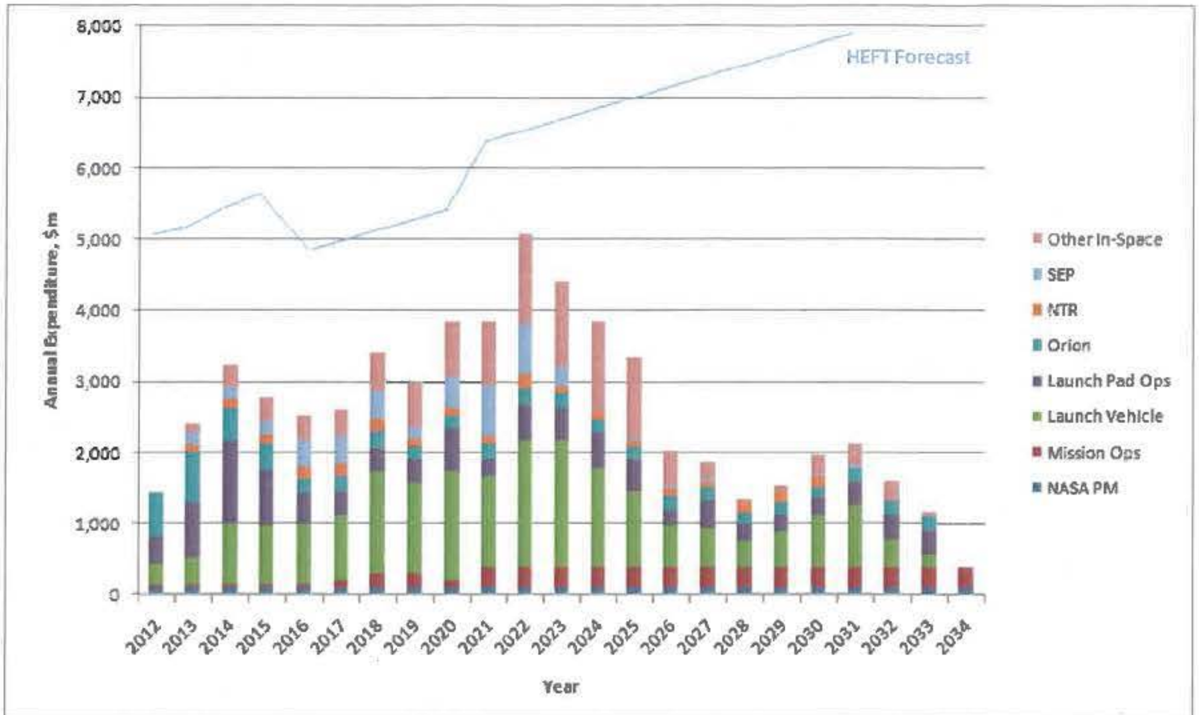
7.5 Costing Analysis Results

Based on the cost analysis described in the previous sections, an overall exploration architecture spending profile or "sand chart" was generated. This allowed for a comparison of projected spending against the previous budget guidance supplied by the Human Exploration Framework Team, or HEFT. Figure 7-4 shows the projected total program spending.

The sand chart had three distinct peaks of spending. The first occurred in 2014 and reflected the significant non-recurring spending required for engine, launch vehicle, and ground facilities development. The second peak occurred in 2022 with a number of important launches required to support the NEO and Phobos exploration campaigns, which had some overlap. All of the years from 2020 to 2025 had a higher total spending figure than the 2014 peak. Finally, a third peak occurred in 2031 during the Mars surface campaign. This peak, and spending in general from 2026-2033, was much lower than the earlier spending because by that point all of the non-recurring expense was retired for launch vehicle and in-space elements. Additionally, the 2033 Mars launch window was much more favorable than the 2024 Phobos window, and required significantly fewer launches to meet. It should be noted that the highest peak in the sand chart, around 2024, could be leveled without additional expense by bringing production of some launch vehicle and in-space hardware forward ahead of the need date.

The sand chart in Figure 7-4 assumed that only the Block 0 (25 mT) and Block I (70 mT) launch vehicles were developed. All of the exploration missions were flown using a maximum payload bit size of 70 mT, although the nominal performance of the Block I launch vehicle was actually 77 mT. Substantially all the exploration equipment, with the exception of the crew space habitat itself, was prepositioned at each exploration destination using space electric propulsion tugs. This allowed the mass of the crewed deep space transportation system to be absolutely minimized. Crew transportation to the NEO, Phobos, and Mars was performed using long term storable cryogenic stages with NGE engines. However, given the need for a more robust crew transportation system permitting more frequent and less costly Mars transit opportunities, the cost of developing the NTR engine and transportation stage was also included in the sand chart, including one NTR flight stage. This stage would be available to support the Mars surface campaign beginning in 2033. For the sand chart, the cost of both the cryogenic stages and the NTR stage were included. The launch vehicles needed for the heavier cryogenic stage approach were used in the cost projection.

Figure 7-4. **Overall Cost Projection for Exploration Architecture**



Cost Element	NASA PM	Mission Ops	SEP	Other In-Space	Launch Vehicle	Launch Pad Ops	Orion	NTR	Total per Year
2012	100	50	10	0	275	393	600	0	1,428
2013	100	50	176	105	375	787	700	100	2,393
2014	100	50	176	310	838	1,180	467	120	3,240
2015	100	50	185	330	821	787	367	140	2,779
2016	100	50	369	345	839	443	213	160	2,519
2017	100	100	391	390	910	344	213	170	2,618
2018	100	200	358	574	1,437	344	213	200	3,426
2019	100	200	150	624	1,271	344	187	100	2,976
2020	100	100	431	794	1,549	594	187	100	3,855
2021	100	300	709	912	1,271	244	213	100	3,849
2022	100	300	709	1,256	1,774	494	240	200	5,073
2023	100	300	273	1,210	1,778	444	213	100	4,418
2024	100	300	5	1,279	1,387	494	187	100	3,852
2025	100	300	32	1,126	1,054	444	187	100	3,343
2026	100	300	32	496	564	244	187	100	2,023
2027	100	300	27	240	524	394	187	100	1,871
2028	100	300	0	40	357	244	160	150	1,351
2029	100	300	5	72	491	244	160	150	1,522
2030	100	300	32	259	716	244	160	150	1,961
2031	100	300	27	320	860	344	187	0	2,138
2032	100	300	0	286	391	344	187	0	1,607
2033	100	300	0	93	166	344	187	0	1,190
2034	100	300	0	0	0			0	400
Cost Element Total, $M	2,300	5,050	4,097	11,061	19,647	9,738	5,600	2,340	59,833

Section 8.0—Advanced Propulsion Technology

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Section 9.0—Conclusion

9.1 Conclusion

Aerojet has completed an assessment of an integrated architecture for human deep space exploration, from launch to Earth return. We have established that we can meet the objects of an affordable and sustainable deep space exploration campaign by focusing on the tenets described at the beginning of this report:

- Support early (this decade) missions of public interest and continue regularly scheduled missions that engage the public
- Use an in-space architecture that minimizes the required launch mass, is flexible and for which each element supports multiple missions
- Use a launch architecture that minimizes the cost of placing payloads in Low Earth Orbit (LEO), and
- Selecting a launch architecture that minimizes infrastructure costs by making maximum use of commonality with other launch systems and with the in-space systems.

While our initial trade space was quite broad, the absolute requirement for affordability quickly narrowed the field to those systems with large impact on launch cost:

- Use of a hydrocarbon-based launch vehicle and maximize vehicle commonality across NASA/DoD/Commercial markets to distribute fixed costs and ensure continuous production
- Separate cargo and crew, and utilize the most efficient propulsion system available for each – high thrust LOX/H2 or NTR for crew, Solar Electric Propulsion for cargo

Following these principles we have demonstrated that an exciting and viable human space exploration campaign, with missions to the Moon, NEOs, Phobos, and the surface of Mars, can be readily accomplished within NASA's Exploration budget.

Our studies revealed the following critical technology developments:

- The next generation hydrocarbon ORSC launch vehicle engine to provide for increased launch vehicle commonality
- Long-term (>1yr) storage of liquid hydrogen in space or a means for local production at the destinations
- A high power Solar Electric Propulsion Tug for efficient cargo transport
- A high performance LOX/H2 engine for crew transport
- Nuclear thermal rockets for sustainable crew exploration of the surface of Mars.

With these technologies, all of which are well within our reach with credible development plans, humanity can embark on a true exploration of the solar system.

Section 10—References

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REPORT DOCUMENTATION PAGE	Form Approved OMB No. 0704-0188

1. REPORT DATE *(DD-MM-YYYY)*: 22-06-2011

2. REPORT TYPE: Final Scientific and Technical

3. DATES COVERED *(From - To)*: Nov 2010 - Jun 2011

4. TITLE AND SUBTITLE

Heavy Lift and Propulsion Technology Systems Analysis and Trade Study

5a. CONTRACT NUMBER: NNM11AA05C

5b. GRANT NUMBER

5c. PROGRAM ELEMENT NUMBER

6. AUTHOR(S)

Jennings, Pete D.; and Carpenter, Christian

5d. PROJECT NUMBER

5e. TASK NUMBER

5f. WORK UNIT NUMBER

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Aerojet-General Corporation
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Rancho Cordova, CA 95742-6000

8. PERFORMING ORGANIZATION REPORT NUMBER

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

NASA/Marshall Space Flight Center
Marshall Space Flight Center, AL 35812

10. SPONSORING/MONITOR'S ACRONYM(S): MSFC

11. SPONSORING/MONITORING REPORT NUMBER: 1372MA-001

12. DISTRIBUTION/AVAILABILITY STATEMENT

Distribution authorized to U.S. Government agencies and private individuals or enterprises eligible to obtain export-controlled technical data in accordance with OPNAV Instruction 5510.161.

13. SUPPLEMENTARY NOTES

14. ABSTRACT

Aerojet conducted a launch vehicle and in-space architecture study for future human space exploration. The study evaluated numerous space launch and in-space propulsion systems to define the overall architecture propulsion requirements for a representative set of exploration destinations during the period 2016-2033. Mission concepts were defined to complete the exploration missions within projected funding restrictions. Key figures of merit included overall affordability and required initial mass in low Earth orbit. The launch vehicles assessed had a payload capacity of 70 - 130 metric tons. The study identified options to reduce sustaining costs by maximizing commonality of flight hardware within the exploration architecture and with other users. Space electric propulsion was used extensively to preposition major cargo elements at the destinations prior to crew arrival. Development approaches were described for key propulsion technologies, including ox-rich staged combustion engines, as well as in-space chemical, space electric, and nuclear thermal propulsion elements.

15. SUBJECT TERMS

Heavy Lift Launch Vehicle, Ox-rich Staged Combustion, Space Electric Propulsion, Nuclear Thermal Propulsion, Hardware Commonality, Crewed Mars Mission

16. SECURITY CLASSIFICATION OF: a. REPORT	b. ABSTRACT	c. THIS PAGE	17. LIMITATION OF ABSTRACT	18. NUMBER OF PAGES	19a. NAME OF RESPONSIBLE PERSON / 19b. TELEPHONE NUMBER (Include area code)
U	U	U	UU		Miriam Klier / (916) 355-3074

Heavy Lift and Propulsion Technology Final Report

DRD No. 1374MA-003

DOCUMENT NUMBER:	RELEASE/REVISION DATE:
AS-4107-RPT-00003	**31 May, 2011**
Original Release Date **31 May, 2011**	**Contract Number (if applicable)** NNM11AA07C
Content Owner: Andrews Space, Inc.	
Windchill Location: /Windchill/NASA HLPT/Document Management/Released	
Access Restrictions: Restricted to Andrews Space, Inc. employees and selected subcontractors as required.	
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Document Revision History

Rev	Approval Date	Changes	CCN No.	Sections / Pages	Approved CM
A	05/31/2011	Initial Release	n/a		dp

Acronyms and Abbreviations

AHP – Analytical hierarchy process
CBC – Common booster core
CCDev – Commercial crew development
CEV – Crew exploration vehicle
CM – Command module
CONOPS – Concept of operations
CPS – Cryogenic propulsion stage
CUS – Common Upper-Stage (ULA concept)
DDT&E – Design, development, testing, & evaluation
DoD – Department of defense
DRM – Design reference mission
DTC – Design to cost
ECLSS – Environmental control and life support system
EDL – Entry, descent, and landing
EDS – Earth departure stage
EELV – Evolved Expendable Launch Vehicle
ELF – Electrodless Lorentz Force
ET – Shuttle external tank
FOM – Figure of merit
GCR – Galactic cosmic radiation
GEO – Geostationary Earth orbit
GN&C – Guidance, navigation, & control
GPS – Global positioning system
GRC – NASA Glenn research center
GSE – Ground support equipment
GTO – Geostationary transfer orbit
HEFT – Human exploration framework team
HEO – Highly elliptical orbit or High Earth Orbit
HLV – heavy lift vehicle
HLLV – Heavy lift launch vehicle
HLS – Heavy launch system
HMO – High Mars orbit
IMLEO – Initial mass in low Earth orbit
IMU – Inertial measurement unit
INL – Idaho National Labs
IOC – Initial operational capability
ISS – International space station
I_{sp} – Specific impulse
ISRU – In-situ resource utilization
IVHM – Integrated vehicle health management
JSC – NASA Johnson space center
KDA – Key decision attribute
KSC – NASA Kennedy space center
L2 – 2nd Lagrange point in the Sun-Earth-Moon system
LAS – Launch abort system
LCC – Life cycle cost
LEO – Low Earth orbit
LH2 – Liquid Hydrogen
LOC – Loss of crew
LOM – Loss of mission
LOX – Liquid Oxygen
MET – Microwave electrothermal thrusters
MLP – Mobile launch platform
MMSEV – Multi mission space exploration vehicle
MSFC – NASA Marshall space flight center
MTBF – Mean time between failures
NAFCOM – NASA-Air Force Cost Model
NEO – Near Earth object
NEP – Nuclear electric propulsion (tug)
NSTS – National Space Transportation System (Space Shuttle)
NTR – Nuclear thermal rocket (tug)
O&M – Oversight and management
OPF – Orbiter processing facility
ORSC – Oxygen-rich staged combustion
PAM – Propulsion / Avionics Module
PGM – Platinum group metals
P_{LOC} – Probability of Loss of Crew
P_{LOM} – Probability of Loss of Mission
P/R – Partially reusable
RBS – Reusable booster system
RP – Liquid Kerosene
RSRM – Reusable solid rocket motor
SRB – Solid rocket booster
SRM – Solid rocket motor
SDV – Shuttle derived vehicle
SEP – Solar electric propulsion (tug)
SLS – Space launch system
SM – Service module
SOC – Space operation center
SOI – Sphere of influence
SOTA – State of the art
SSME – Space shuttle main engine
TEI – Trans-Earth injection
TPS – Thermal protection system
TRL – Technology readiness level
ULA – United Launch Alliance
USAF – United States Air Force
VAB – Vehicle assembly building
VASIMR – Variable specific impulse magnetoplasma rocket
Vinf – Hyperbolic excess velocity

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1. Executive Summary

1.1 Introduction

Andrews Space, Inc. is pleased to present this report at the completion of its study of Heavy Lift and Propulsion Technologies. This study has been far-reaching, encompassing human exploration elements, in-space propulsion elements, as well as earth-to-orbit launch elements as part of a plan to provide heavy-lift capability to support human exploration for the next several decades.

Andrews has developed and compiled assessments of a wide range of both in-space and launch elements, leading to recommendations for architecture development paths as well as technologies that will most benefit those paths. This report is a compilation of the processes, analyses, and assessments of those architectures and technologies.

1.2 Study Process

At the beginning of the study, top-level objectives were identified to provide direction for subsequent architecture and technology evaluation and selection. It was Andrews' goal to provide an unbiased look at heavy-lift propulsion technologies, both for in-space and Earth to orbit capability, to provide an affordable and sustainable path for human exploration. Those top-level objectives include:

- Determine capabilities required to support innovative human space exploration
- Ensure capability to multiple destinations: Moon, Mars & environment, near-Earth asteroids, and Lagrange points
- Determine technology, research, and development required to meet system goals
- Determine heavy-lift launch vehicle and in-space propulsion elements for a complete architecture
- Explore multiple alternative architectures, including expendable and reusable elements
- Assess affordability, operability, reliability, and commonality of the considered architectures
- Determine space launch propulsion technologies that will enable a more robust exploration program, support commercial ventures as well as related national security needs.

1.3 Architecture Assessment

Andrews' approach during the HLPT study was to research potential human exploration missions, define required mission elements required for those missions, derive the necessary transportation requirements, and then define and assess a variety of in-space and earth-to-orbit transportation architectures that would meet those requirements.

1.3.1 Mission Models

A variety of Design Reference Missions (DRMs) were developed with the goal of eventually supporting a flagship human mission to Mars, the Moon, or other exploration destination (Figure 1-1). Precursor missions provide a near-term testbed for technologies and processes eventually needed for the flagship mission. Four potential flagship missions were studied, encompassing widely different mission goals: A science mission to a near-earth asteroid investigating solar system origins, a shorter term science outpost at a Lagrange point, a lunar mining mission that involves the retrieval of rare elements and minerals that are currently running in short supply on Earth, and a mission to Phobos, with tele-operated Mars rovers and science equipment emplaced on Mars. The goal of understanding these mission models was to derive requirements for the heavy-lift and in-space transportation systems that will be needed to carry out the missions.

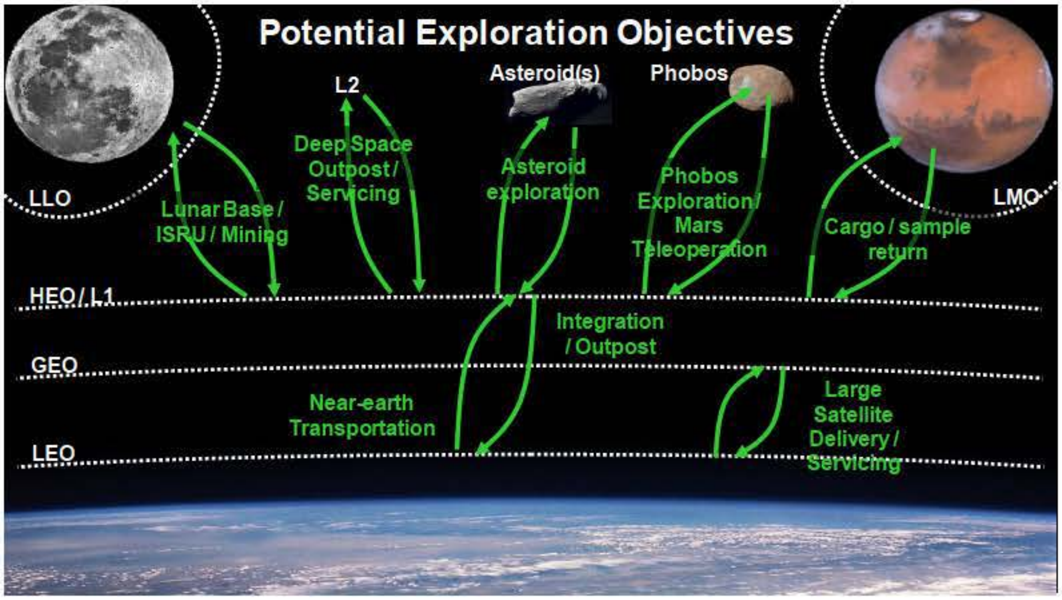


Figure 1-1 Possible Human Exploration Destinations

1.3.2 Mission Elements

The selection of mission elements early in the transportation architecture development cycle is important since mission element dimensions and masses drive requirements to in-space propulsion systems, Earth to orbit launch vehicles, and eventually ground systems and operations. Andrews derived mission element designs from previous studies for both lunar and Mars missions, and developed a database of potential mission elements, with mass, dimensions, and functionality characteristics that would drive transportation needs.

1.3.3 In-space Propulsion Elements

The development of an efficient, reusable in-space propulsion technology is needed for any affordable long term human mission architecture. Four in-space transportation options were studied to determine suitability for use in the proposed DRMs (Figure 1-2).

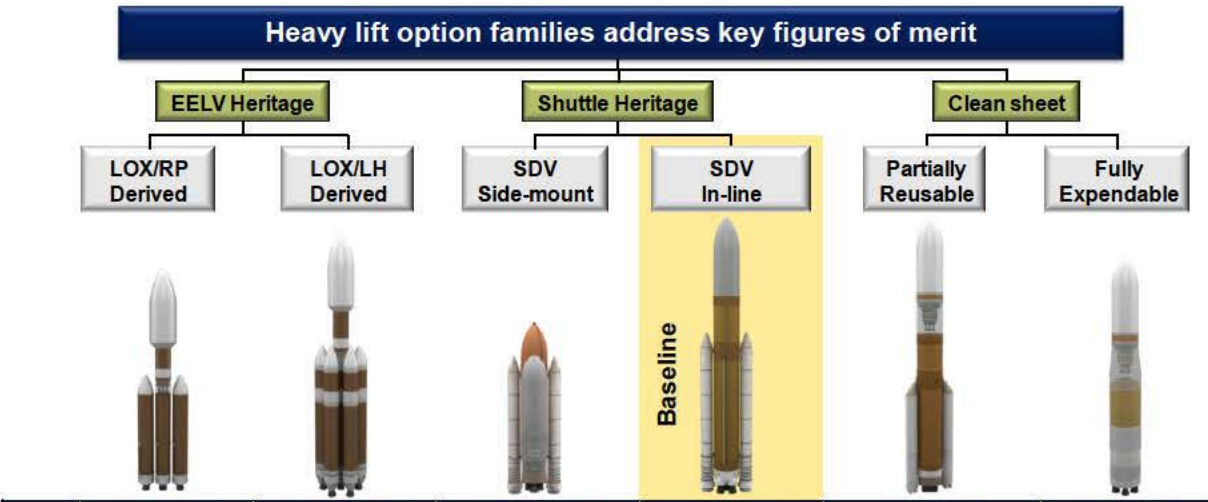
Propulsion Configuration	Description	Baseline Performance	Key Technologies	Benefits
Solar Electric (Baseline)		1.2 Mwe tug 25 mT dry mass	• FAST compact array • Channel HET, ELF, or VASMIR hrusters	• Compact w/ High radiation resistance • Long-life array and thrusters
Nuclear Electric		1.8 Mwe tug 8.0 kg/kWe α 5-8 year life	• Fast Spectrum Reactor • Redundant Brayton cycle conversion	• Deep space capability • Electrical power generation
Nuclear Thermal Rocket		1.04 GW_t, 5 mission life, 600 to 972 sec I_{SP}	• Bi-modal Tungsten-CERMET reactor	• Deep space capability • Dual cycle for electrical power generation
Chemical (LOX/LH2) EDS		0.88 propellant mass fraction, 454 sec I_{SP}	• 30-70 klb_f expander cycle LOX/LH2 engines	• Simplifies infrastructure

Figure 1-2 In-Space Propulsion System Options

A conventional LO2/LH2 chemical stage was compared in terms of performance, cost, reliability, and mission time metrics to stages using solar electric propulsion, nuclear electric propulsion, and nuclear thermal propulsion. Andrews also looked at the current state of technology readiness for each of these propulsion types in order to understand the development needs.

1.3.4 Earth-to-orbit Launch Elements

The primary focus of the HLPT study was to determine an optimum heavy-lift launch system with its associated ground operations, and to understand its technical and performance requirements, while meeting serious budget constraints. Variants of six launch vehicle architectures were considered by Andrews over the course of this study. Each vehicle family was developed to address one or more Figures of Merit (FOMs), with the ultimate goal to understand how a heavy-lift launch system might be developed to meet human exploration needs and still be affordable under the expected budget constraints.



	Option 1	Option 2	Option 3	Option 4	Option 5	Option 6
IMLEO	30-130 mT	35-100+ mT	65 – 85 mT	25 - 130 mT	25 - 130 mT	35 - 130 mT
Primary FOM	Extensibility	DDTE Costs	DDTE, Infrastructure Costs, Early IOC	Reliability, Risk	Performance, Extensibility	Reliability, Volumetric Effic.
Nominal Design	• 8.4-m shroud • 2 to 6 CBC LRBs • RS-84 class engine (2x ea) • 6xRL-10C EUS	• 8.4-m shroud • 2 to 6 CBC LRBs • RS-68A LOX/LH engine (1x ea) • 6x RL-10C EUS	• 8-m shroud • 4-seg RSRMs • 3x RD-25D (1x PAM) • 5x RL-10C EUS	• 8.4-m shroud • 5-segment SRMs • 5x RS-25E • No upper stage	• 8.4 m shroud • 3x RS-25D (1PAM) • Zenit-1 boosters (Reusable RBS for DoD) • 6x RL-10C EUS	• 8.4-m shroud • 6x RS-84 LOX/RP engine • 6x RL-10C EUS

Figure 1-3 Candidate Launch Vehicle Options

The characteristics of each launch architecture are:

- **Option 1** – EELV-derived LOX/RP modular booster with LO2/LH2 upper stage
- **Option 2** – EELV-derived LOX/LH2 modular booster with LO2/LH2 upper stage
- **Option 3** – Shuttle-derived LO2/LH2 booster with side-mounted payload carrier and LO2/LH2 upper stage
- **Option 4** – Shuttle-derived LO2/LH2 booster with in-line payload fairing
- **Option 5** – 8.4-m LO2/LH2 partially-reusable core vehicle with LO2/RP strap-on boosters and LO2/LH2 upper stage
- **Option 6** – Fully expendable 10-m LO2/RP first stage with LO2/LH2 upper stage

1.3.5 Ground Systems and Operations

The infrastructure needs and associated costs of ground systems and operations were assessed for each of the launch architectures and compared to current Space Shuttle operations. Comparisons were made based on vehicle dimensions, mass, complexity, and facility usage. Fixed costs were assessed to understand where improvements could be made, and to understand what changes would need to be made to reach operations and recurring cost goals.

1.3.6 Heavy Lift and Propulsion Technologies

The ultimate goal of the study was to identify and quantify the technologies and processes needed to move forward with a heavy lift capability that will meet the needs of a robust human exploration program. Andrews has assessed the mission needs, in-space transportation needs, and the launch architecture needs to arrive at a set of technologies that can be implemented within the budget constraints.

1.4 Study Recommendations

1.4.1 Mission Model Development

A near term investment in serious systems engineering studies to reach consensus on what the flexible path to human space exploration should be undertaken in the near term. These systems engineering studies should be widely subscribed to and include international partners. The systems engineering studies should be used to determine: if future lunar resources are adequate to justify going back to the Moon, why humans are needed for Near-Earth Objects (NEO) exploration, and how a future Mars mission can be made affordable.

1.4.2 In-Space Propulsion Technology Application

In terms of technology development, the most important technologies relate to the in-space propulsion system. Early development of modular solar electric power (SEP) systems and variable Isp thrusters are necessary to meet future exploration needs and keep launch costs to a minimum. Later development of nuclear power systems will enhance the capability to explore further into the solar system. Modular SEP designs will allow incremental development of array and thruster technologies with potential commercial applications. In concert with this, any planned Mars exploration will require aero-capture and entry, descent, and landing (EDL) technologies to drastically reduce the amount of transportation propellants required.

1.4.3 Launch Capability

A primary finding of the Andrews HLPT study was that by developing a high-performance in-space propulsion system, the need for super-heavy-lift (100 mT to 150 mT class) launch systems goes away. In fact, all expected launch requirements for human exploration missions can be met by an 80-mT class launch system. This not only helps the program meet development cost goals, but it also maintains a more robust flight rate, improving the efficiency of the ground operations. Expected benefits of payload margin and assembly reliability of the larger capacity launch vehicle are outweighed by the cost, risk, availability, and extensibility benefits of the smaller launch vehicle.

1.4.4 Modularity and Extensibility

Another recommendation is to develop a modular launch vehicle capable of launching a wide range of payload classes, ranging from 25 mT to 80 mT capability. This allows for cost reductions over the course of the system lifetime through increased flight rate of vehicle elements, and it improves the sustainability of the launch system by meeting stakeholder needs other than human exploration. Andrews' preferred launch concept is a modular LO2/RP booster with a LO2/LH2 upper stage. Multiple booster and upper stage engines allow engine-out capability off of the launch pad, dramatically improving the overall mission reliability.

Andrews recommends the development of a LO2/RP staged-combustion engine in the 500 – 600 klb thrust class, derived from existing engine technologies (AJ26). Using a derivative RL10 upper stage engine draws on the heritage of the RL10 and provides maximum extensibility across all national upper stage assets.

1.4.5 Operations

Thirdly, operations costs must be drastically reduced from current Shuttle levels by implementing automated launch processes and vehicle health monitoring systems to reduce the number of personnel and facilities needed to support ground and launch operations.

1.4.6 Launch Vehicle Technology Application

For launch systems, Andrews found that most technologies exist or are relatively mature to implement cost-effective launch architectures. Combining existing "best-practice" automated ground processes and production capability, vehicle health monitoring and system management processes, as well as automated mission planning processes will allow the Heavy-lift system to be cost effective and capable of meeting human exploration needs.

2. Andrews HLPT Study Process

The NASA Heavy Lift and Propulsion Technology Study, while focused on transportation for human exploration, became a far-reaching study due to the fact that future human exploration goals and objectives are not well-defined. Andrews Space undertook this study with an eye to providing an unbiased look at how those goals might drive the transportation system design and development.

2.1 Study Philosophy

Because future human exploration paths are not well-defined, Andrews embarked on the HLPT study with a few basic principles in mind. These principles are derived in part from our understanding of NASA's desires and goals, as well as our past experience in system architecture development.

- **Requirements**: Mission objectives drive transportation requirements; therefore a study of transportation needs must start at the end, not at the beginning. Understanding the mission requirements and the logical building blocks needed for exploration missions is critical to understanding how those mission elements will be transported from earth to their destinations.
- **Cost:** HLPT is a design to cost study in that costs must be kept under the budget limits. Early in the study it became clear that we couldn't keep trying the same approach and expect different (cheaper) answers.
- **Commercialization**: Some developments may be best shared between NASA and other commercial/government interests. If commercial interests are involved, they must be able to operate the system at a profit. This requires adequate flight rates and revenue potential for those interests to even consider involvement. If commercial interests are met, NASA may be able to benefit by significantly reduced operations and/or hardware costs.
- **Technology application**: Selected high risk, high payoff technologies (i.e. game changing) can make a big difference in the outcome of the program. If new technologies are to be applied, they must "buy" themselves onto the program. The benefits must outweigh the risk and cost.
- **Sustainability:** A key stakeholder is the public. Keep the public engaged with significant program milestones and clear progress toward a worthwhile goal. This also relates to political support.
- **Flexibility:** Maintain a flexible path that includes options and feedback loops. Expect setbacks and include backups in the plan.
- **International Participation:** Plan for potential international participation, but with well defined interfaces

2.2 Architecture Development Process

Using standard systems engineering principles, it was advantageous to construct the overall architecture by working from a set of Design Reference Missions (DRMs) to determine which mission elements drive the sizing of the transportation elements. In the case of human exploration there are the additional in-space Transportation elements, which can vary the Integrated Mass In Low Earth Orbit (IMLEO) mass by about 50% even after the mission elements are defined. Therefore Andrews' approach was to research various candidate human exploration missions, define required mission elements (e.g. Mars Transit Habitat) and then define the initial in-LEO configuration for each of the candidate In-space transportation concepts. This process is shown schematically in Figure 2-1. Once we understood what needed to come together in LEO it was possible to design the HLV options and associated ground infrastructure to meet the range of sizes and masses needed to be launched to enable on-orbit assembly of an Exploration mission package.

It was apparent early in the study that launching an entire mission package in one launch was not possible due to volumetric and mass requirements (masses in excess of 200 metric tons are required). It also became apparent once costing of heavy lift architectures was initiated that modifications to ground infrastructure could be a major contributor to nonrecurring costs, and that the HLV program would need to share development costs with other launch systems or utilize existing hardware elements (e.g. existing stages, engines, tooling, etc.) as much as possible to make any HLV architectures affordable. This is reflected in the design-to-cost feedback loop in Figure 2-1.

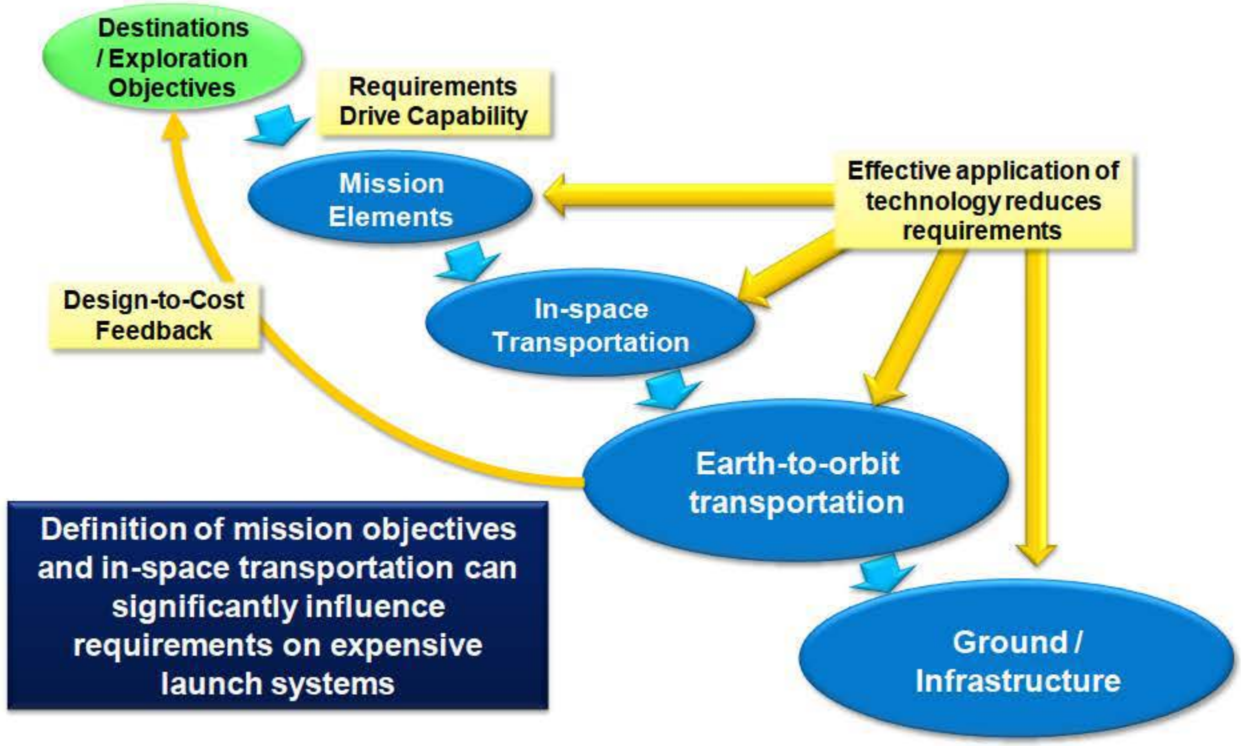


Figure 2-1 Exploration Requirements Flow Diagram

With the basic study philosophy in place, Andrews determined the architectures that would support a set of human flagship missions, whether they be to the moon, to Near-Earth Asteroids, or to Mars. In order to prepare for the ultimate flagship missions, as well as for an exploration path beyond those missions, a series of incremental steps must be taken. Those steps must include development of the mission elements and transportation systems themselves, early robotic precursor missions to understand environments and resources, then early human precursor missions to test system functionality while still in a relatively safe environment (Figure 2-2). With these elements in mind, Andrews developed full exploration architectures that included these steps, and that figured into our assessments.

2.3 Architecture Assessment Process

Andrews Space typically uses a complete systems engineering process to conduct conceptual trade studies and assessments (Figure 2-3). This process includes requirements development, mission model development, trade option development, then assessment of the options against a set of FOMs. We then use an Analytical Hierarchy Process (AHP) that allows us to compare trade options and make decisions based on a set of defined Figures of Merit (FOMs). With the AHP we are also able to perform sensitivity analyses and understand the influence of technology applications.

For the HLPT study, this process was used at several levels to assess in-space transportation options as well as heavy-lift launch vehicle options. We used this process to assess in-space transportation options, as well as Heavy-lift Launch Vehicle options.

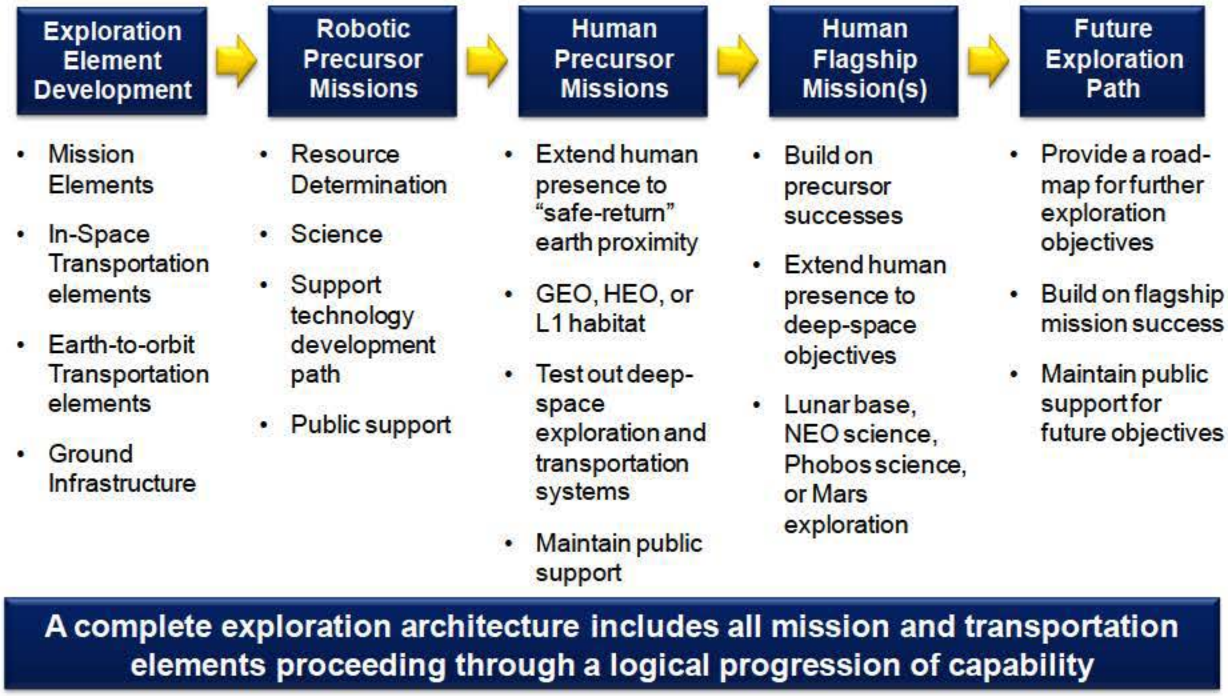


Figure 2-2 Exploration Architecture Development Path

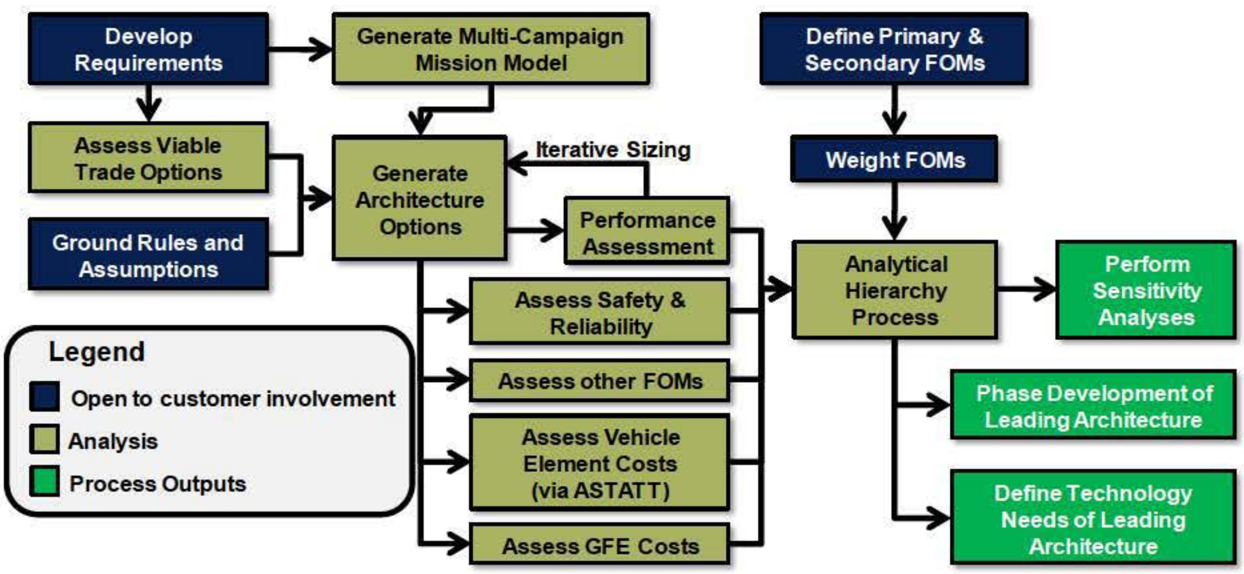


Figure 2-3. Andrews Systems Engineering Process.

2.3.1 Figure of Merit (FOM) Definition

NASA Objective 1: Provide a recommended list of key decision attributes and rationale associated with each.

The first step in the HLPT AHP process was to define and weight Figures of Merit. Primary figures of merit (FOMs) were developed to encompass a large variety of system attributes. Primary FOMs are typically generic and can be applied to almost any trade-set. These FOMs, along with a rationale for their applicability can be seen in Table 2-1. The primary FOMs were then broken down further into secondary FOMs, each identified with metrics that were used to assess the trade options. Secondary FOMs are typically more specific to a particular trade-set, and may vary depending on the options being studied. In this case, the FOMs were driven by perceived NASA objectives, as well as the expected system attributes.

Table 2-1. Figure of Merit Definition.

Primary FOM	Rationale	Secondary FOM	Proposed Metric
1.0 Cost	Cost is the primary driver in the development and operation of an extensive transportation system. The system must fit within the available NASA budget constraints.	1.1 DDTE Cost	DDTE cost, excluding infrastructure
		1.2 Infrastructure Cost	Facilities and GSE cost
		1.3 Recurring Cost	Expendable Hardware + operations cost
2.0 Operability	This FOM is a measure of how well the system can be operated, both in terms of routine flight operations, as well as availability to meet the desired flight rate.	2.1 Hands-on crew time	Total crew hours to prepare launch
		2.2 Availability	Min. time between launches
3.0 Performance	This is a measure of how well the operational system meets the system objectives, in terms of mass, volume, and throughput capability.	3.1 Capability	Delivered payload / System total weight (ref. missions)
		3.2 Volumetric efficiency	Available payload diameter and shroud height
4.0 Safety and Reliability	Reliability relates to the operational system. High reliability is a must for human exploration transportation systems, especially for relatively low flight-rate systems.	4.1 Loss of mission	Percent mission success (PLOM)
		4.2 Loss of Crew	Catastrophic failure rate (PLOC)
5.0 Mission Capture	This FOM applies to the development of the system, in how quickly and how effectively the system captures the desired mission model; as well as how the system can be applied over a range of payloads.	5.1 Time to human flight	Months to first human flight
		5.2 Extensibility	Range of addressable payload capability (minimum to maximum)
6.0 Risk	An important attribute of any system development is risk, in terms of technical and programmatic risk. This FOM applies mainly to the system development.	6.1 Technical Risk	Risk score of performance or cost drivers
		6.2 Programmatic Risk	Combined risk score of non-technical cost risks
		6.3 Sustainability	Average time between exploration milestones

2.3.2 Figure of Merit Weighting

NASA Objective 2: Provide a recommendation for the weighting of the recommended key decision attributes.

Determination of the weightings of each of the FOMs was performed by pair-wise comparison using Expert Choice software. These weightings were used for the initial assessment of each architecture option. For the HLPT study, it was determined that there are two potential paths that might lead to different weightings of the FOMs. One case would relate to the launch of crew on board the heavy lift system, as opposed to being launched on a separate crew-launch system. The other case would relate to a system that is strictly used for cargo, and not crew.

The relative weightings for the case of crew launching on the newly designed HLV can be seen in Figure 2-4. In this scenario, crew safety and system reliability is paramount, with cost being a close second place, and risk coming in third. When broken down to secondary FOMs, again loss of crew is most important, with sustainability (risk) and DDTE cost being highly valued. Alternately, weightings for cargo-only usage of the HLV can be seen in Figure 2-5. Weightings for each of the secondary FOMs are also shown. In this case, cost is most highly weighted, with design, development, testing, and evaluation (DDTE) cost as the most significant portion; followed by Risk (sustainability) and mission capture (Time to flight and Extensibility). A later variation of the cost weighting placed higher emphasis on recurring cost as opposed to DDTE cost, but the primary FOM weighting remained the same. Going into the HLPT study, Andrews assumed as a baseline case that crew would not be launched aboard the HLV, but rather on a separate launch vehicle.

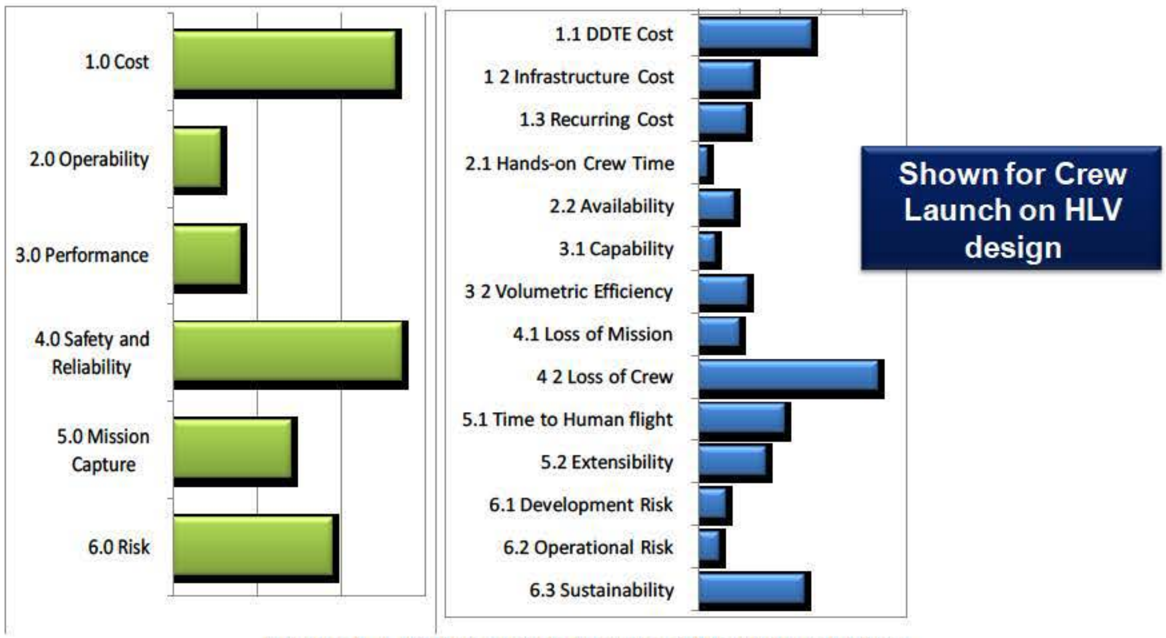


Figure 2-4 Crew Launch Figure of Merit Weightings

Figure 2-5 Cargo Launch Only Figure of Merit Weightings

3. Mission Model Development

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4. Mission Element Definition

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4.2 Mission Element Descriptions

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5. Transportation Architecture Definition

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Table 5-6.

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6. Transportation Architecture Assessment

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8. Summary and Conclusions

Observations made over the course of the HLPT program as a result of analysis are summarized in the following sections.

8.1 Mission Model Observations

The following observations were made in regard to overall mission model considerations:

- Mission objectives drive the in in-space transportation requirements, which in turn drive Earth to orbit heavy lift requirements
- Current mission models for human exploration result in launch rates of less than one launch per year on average, and require 500-800mT to be delivered to LEO (more for chemical in-space propulsion).
- Flexible path human exploration objectives can be met with a variety of in-space propulsion types. SEP provides the best mixture of cost, reliability, and performance (50% mass and 25% volume reduction when compared to chemical option)

8.2 Architecture Assessment Observations

The following observations were made in regard to results of the architecture assessments:

- The winning HLV option appears to be Option 1 – a modular LO2/RP first stage with LO/LH2 upper stage. This option is preferred only when equipped with multiple engines per stage (e.g. AJ26 derivatives) to improve launch reliability with engine-out capability. This option also has good recurring cost, extensibility, and sustainability characteristics due to the relatively small common stages and "dial-up" payload capability.
- Launch vehicle fixed and operating costs must be dramatically reduced (<10% of Shuttle levels) in order to allow the HLV and exploration program to meet program goals. This would apply to any HLV option. This is more of a political issue than it is a technical design issue.
- HLV modularity allows for commonality with other stakeholder needs and can save up to 11% of the total exploration launch life-cycle cost.
- HLV development must maximize the use of existing assets and design knowledge in order to meet schedule and cost goals (EELV and Shuttle derived). The preferred Option 1 vehicle can be built using EELV production capacity (5-m stage diameter), and may use Delta-IV logistics infrastructure.
- An incremental HLV upgrade process can meet exploration objectives while maintaining development cost limits. An engine block change for Option 1 could allow early flights of Block I vehicles using existing RD-180 engine technology, then transition to more reliable Block II vehicles with multiple smaller engines for increased reliability.

8.3 Propulsion Observations

The following observations were made in regard to propulsion system requirements:

- Near term needs (80mT or less) can be met by existing main engine technology. New booster engines are only needed to mitigate supply risks (RD-180) or to increase performance (RS-25E) for some vehicle Options.
- New upper stage engines are not needed for payloads <80mT. RL10 derivatives (RL10C) provide good commonality with other national launch needs and will reduce cost. Multiple upper stage

engines also provide capability for engine-out in the event of engine losses, increasing upper stage reliability overall.

- In-space propulsion development should focus on solar-electric in the near-term, then nuclear for later term needs. Low-thrust, high performance propulsion technologies provide a clear reduction in earth-to-orbit mass requirements, and allow the in-space elements to be reusable, further reducing life cycle costs.

8.4 Technology Development Needs

The following technologies are considered by Andrews to be enabling of future Exploration missions:

- Large lightweight solar power and supply systems
- Variable I_{sp} electric propulsion systems
- Integrated vehicle health monitoring (IVHM) for launch vehicles and in-space systems
- Non-toxic RCS propellants for launch vehicles and in-space systems
- Zero-g propellant management for in-space low thrust systems
- Automated rendezvous and docking for assembly of exploration elements
- Automated launch pad fluid and monitoring systems

8.5 Recommendations

Andrews makes the following recommendations for heavy-lift development strategies.

8.5.1 Mission Model Development

A near term investment in serious systems engineering studies to reach consensus on what the flexible path to human space exploration should be undertaken in the near term. These systems engineering studies should be widely subscribed to and include international partners. The systems engineering studies should be used to determine: if future lunar resources are adequate to justify going back to the Moon, why humans are needed for NEO exploration, and how a future Mars mission can be made affordable.

8.5.2 In-Space Transportation Technologies

The exploration infrastructure will take considerably longer to develop than a new launch system, so the launch system decision should be postponed until consensus is reached on what needs to be launched. The in-space transportation segment is an integral part of the flexible path so it needs to be addressed during the systems engineering studies. In the short term, it makes sense to start some planned technology developments to support the most likely future transportation scenarios. Andrews would recommend technology development studies for Electrodeless Lorentz Force (ELF) Thrusters, a Solar Electric Tug (SEP is identified as a good option to support science missions and as a bridge to future missions), and generic high-performance Nuclear Electric Power-plants.

8.5.3 Launch Capability

A primary finding of the Andrews HLPT study was that by developing a high-performance in-space propulsion system, the need for super-heavy-lift (100 mT to 150 mT class) launch systems goes away. In fact, all expected launch requirements for human exploration missions can be met by an 80-mT class launch system. This not only helps the program meet development cost goals, but it also maintains a more robust flight rate, improving the efficiency of the ground operations. Expected benefits of payload margin and assembly reliability of the larger capacity launch vehicle are outweighed by the cost, risk, availability, and extensibility benefits of the smaller launch vehicle.

8.5.4 Modularity and Extensibility

In terms of an Earth to orbit launch system that will enable a wide variety of mission types, it is recommended to begin development of an incremental or modular vehicle family similar to that of our study Option 1 vehicle. This family should be extensible over a wide range of payload classes and maximize the use of existing best practices and facilities (experience, design, production) in order to constrain costs.

Andrews also recommends development of a oxygen-rich, staged combustion hydrocarbon rocket engine in the 500 to 600 klb thrust-class, suitable for both exploration and USAF reusable booster programs. This recommendation is based on the finding that an HLV with engine-out capability has significantly better reliability and safety than the HLV options with million pound engines.

8.5.5 Operations

Serious effort must be placed into launch cost reduction strategies. The current high operating costs of the Shuttle architecture is driven by jobs (15,000 at KSC in 2007). It is recommended that some form of austere operations (<500 crew) be implemented combined with automated launch processing (e.g. SeaLaunch) in order to sharply reduce costs. These technologies exist – they need to be implemented in any new launch system.

8.5.6 Launch Vehicle Technology Application

For launch systems, Andrews found that most technologies exist or are relatively mature to implement cost-effective launch architectures. Combining existing "best-practice" automated ground processes and production capability, vehicle health monitoring and system management processes, as well as automated mission planning processes will allow the Heavy-lift system to be cost effective and capable of meeting human exploration needs.

Appendix A

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Appendix B

HUMAN EXPLORATION OF THE MOON AND MARS SYMPOSIUM (A5)

Long Term Scenarios for Human Lunar Presence (2)

SPACE COLONIZATION, A STUDY OF SUPPLY AND DEMAND

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Abstract

The last fifty years has nurtured the dream of living and working in space. Unfortunately, that dream appears to be moving further and further into the future, as financial resources become increasingly scarce and space program budgets shrink. This paper steps back and looks at the fundamental economics of people working (and playing) in space, and shows scenarios where colonization could, and should, succeed. The key to success for any economic scenario (plan) is correctly predicting supply and demand versus various pricing points. We based our supply and demand analyses on dozens of previous publications and surveys as well as extensive personal experience. Economic scenarios evaluated include commercial development of lunar resources from lunar LOX, through platinum group metals, energy metals (uranium and thorium), and He-3. Various tourism-based scenarios have been examined; from space hotels in LEO, through lunar tourism, and space settlements for telecommuters. Eventually, if one or more of these scenarios are successful, enough people will be living in space to justify pure colonization, where people migrate to space to provide goods and services to other people living in space. There are numerous near-term technologies that are important to driving down costs and improving the safety and reliability of transportation system elements as well as some surface elements. The cost and impact of these technologies are shown. Also the cost and impact of some more speculative technologies like space elevators are included.

Problem Statement

Space Colonization requires Fully Reusable Earth-to-Orbit Systems (FRETOS) and a market large enough to purchase frequent flights (i.e. daily launches). FRETOS have been studied since the late 1970s[1-4], but a large profitable space market has never developed. Space tourism is often mentioned as the anchor tenant for the enabling space market, but the required destination space resorts are still on paper[5-6]. Space Solar Power Satellites could provide another large potential market, but ground solar thermal combined with natural gas seems to have stolen the market opportunity [7-9].We would propose that mining increasingly scare rare metals on the moon and Near Earth Objects (NEOs) may provide the anchor tenancy required to enable commercial development and operation of FRETOS, and thereby open up space to the masses. We begin by making the economic case for using space resources on earth.

Limits to Growth

Numerous studies[10-13] have shown that there are limits to the population that the planet can sustain at a decent Gross Domestic Product (GDP) per capita. The underlying issue is whether improvements in manufacturing, mining, farming, and pollution prevention technologies over time can overcome diminishing nonrenewable resources and eroding or degrading farmlands. We explored this issue using an updated version of the World3 model available from the Laboratory for Interactive Learning (603) 852-2186. We programmed the world model to create the current 2010 situation with a 4.0 %/year technology improvement rate, and the simulation showed a population collapse in 2035 from starvation (too much capital required to obtain increasingly scarce resources and prevent crippling levels of pollution). Even introducing worldwide family planning immediately and focusing capital resources on replacing fossil fuels with renewable energy sources, only delays the population collapse until 2040. Significant changes to the scenario are required.

Antimony	15-20 years
Gallium	5 years
Hafnium	10 years
Indium	5-10 years
Platinum	15 years
Silver	15-20 years
Tantalum	20-30 years
Uranium	30-40 years
Zinc	20-30 years

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The significant change we propose to explore is to move the mining and smelting of key nonrenewable resources to the moon. This accomplishes two things. First, we get access to very high grade ores which reduces both the capital cost and energy cost of obtaining critical resources, and second, we're polluting the moon instead of the earth which reduces the capital cost of pollution prevention on earth. The materials we propose to import are elements critical to modern technologies and that will run out far sooner than 2040 because their usage is increasing with expanded technology as shown in figure 1 on the left. None of these elements will be used up completely, but as the richer ore bodies are depleted, the prices will soar. For example, the price of rhenium, used for fuel-efficient aircraft engines, has jumped to a record $11,250/kg, almost 12 times its price in 2006. It is now only half the price of gold, which is a major boon to the main countries that mine rhenium ore: Chile and Kazakhstan. Reserves of indium, used for solar cells and LCDs, along with those of hafnium, an essential component of computer chips and in nuclear control rods, may literally run out within 10 years[14].

2006 GNP	$37,574	2000 $
	kg Lifetime	kg per year
Antimony	7.13	9.16E-02
Tin	15	1.93E-01
Tantalum	0.18	2.31E-03
Gallium	0.005	6.43E-05
Gold	0.04	5.14E-04
Platinum	0.045	5.78E-04
Hafnium	N/A	N/A
Germanium	0.01	1.29E-04
Uranium	5.95	7.65E-02
Rhodium	0.004	5.14E-05
Silver	1.58	0.02
Nickel	58.4	0.75
Lead	410	5.27
Indium	0.032	4.11E-04
Zinc	349	4.5
Chromium	131	1.7
Copper	630	8.1
Phosphorous	8322	107.0
Aluminum	1576	20.3

Small amounts of these PGMs and rare earth metals are required in today's smart infrastructure (see figure 2 on the right) and the amount returned from the moon would eliminate the upcoming shortages and enable the high productivity the world is going to need in the future.

Our hypothesis is that if we add lunar resources, we can afford to maintain the switch to plug-in hybrids throughout the world, thereby increasing productivity and reducing persistent pollution. Even though the flow of lunar resources is relatively small, these critical metals have a large impact on productivity, resulting in a potential soft landing for the world population.

This shows the magnitude of the problem, but can space resources provide a solution? **About two-thirds of all known meteorites contain iron-nickel (FeNi) metal. "Iron-nickel" means that the metal is mostly iron but it contains 5-30% nickel as well as a few tenths of one percent cobalt, plus high concentrations (by terrestrial standards) of** strategic metals such as the platinum group, gold, gallium, germanium, **iridium, and others.** Interestingly enough the lower the Fe-Ni metal content in the meteorite, the *more* enriched the Fe-Ni metal is in these rare and precious metals and elements. These elements readily dissolve into the metal that exists, and the less metal that exists, the less diluted they are. Many asteroids are richer in most of these precious metals than the richest Earth ores which we mine. Further, these metals all occur in one ore when it comes to asteroids, not in separate ores. Many of the richest ore bodies on earth are meteorite impact craters from geological times. In particular, the impact crater at Sudbury, Ontario, is rich in iron, nickel, cobalt, copper and platinum group metals. Are similar metals present in impact craters on the Moon? Geologists at Sudbury say that the valuable metals at Sudbury did not come from the impactor, but welled up from deep within the Earth. If this is so, why don't more volcanic upwellings contain rich ores of nickel, cobalt, copper and PGMs? Until we go to the Moon and study more impact craters to determine whether or not they are rich in these metals we cannot be certain. Particles of Fe-Ni metal make about 0.5% of the regolith and can be magnetically separated, but copper is present only in traces in the regolith.

The amount of minable rare earth elements on the moon is unknown, but all asteroid impacts on the moon since its formation are still on or near the surface since the moon has no plate tectonics. Therefore there is a high probability the rare earth metals are present, but they may not be present in minable concentrations. For the purpose of this study we modeled mining the PGMs and assumed a comparable effort could obtain similar quantities of rare earth metals (a stretch goal).

Lunar Mining

Many unknowns exist regarding mining on the Moon, beginning with what's there to exploit. As discussed, one promising source is the asteroidal nickel-iron that is found (as far as we know) all over the lunar surface. Metallic asteroid impact sites exist on Earth; some are exploited as ores. However, over geologic time Earth impact sites will be submerged by plate tectonics and nickel-iron, being a siderophile, will find its way to the Earth's core. The Moon does not exhibit plate tectonics. Metallic asteroid impacts, depending on impact velocity, will scatter metals globally over the Moon, or more locally. Some material will be launched to escape velocity by the impact and will not return. Sample data, as noted, indicate the regolith averages about 0.5% nickel-iron. Not very many sites have been sampled and those that have been are all near the equator. Therefore, our knowledge of "rich" ores, if there are such, is nil.

The regolith itself is about 10% iron, but our interest in asteroidal metal is that it also contains nickel, cobalt, and platinum-group metals. The composition varies considerably, and it is reported that in meteorite samples, the metallics richest in platinum-group metals are found in meteorites with less total metal, as if the platinum-group metals are absorbed by whatever nickel-iron is available. Cobalt content is also reported as quite variable. Therefore, it is vital to perform prospecting as the first step to strategic metals acquisition from the Moon, to find good sites with adequate amounts of nickel-iron of favorable composition.

We constructed a simple cost model to explore the characteristics of mining and processing lunar materials for shipment to Earth. The model did not include descriptions of processes, but characterized the end-to-end process as consisting of six steps, each with six generic characteristics, and seven cost elements. The processes are mining, hauling, preprocessing (such as beneficiation), processing (to final product(s)), packaging, and shipping. The output of each step is the input to the next. The output of the final step is delivery of the product to Earth.

Descriptions of the six steps are given in Table 1, the six characteristics in Table 2, and the seven cost elements in Table 3. The output of each step is the input to the next. The model is solved in reverse. It is explicit; no iteration is required. This model is easily implemented on a spread sheet.

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	Table 3: Cost Parameters
Specific Cost	Cost of equipment built on Earth for shipment to the Moon, in dollars per kg. Does not include transportation cost to the Moon.
Lunar Specific Cost	Cost of equipment built on the Moon, for now in dollars per kg but we really need a measure that's lunar operations oriented.
Resupply Cost	Cost per kg of resupply; transportation cost computed separately
Support Cost	Cost per kg of product for support operations on Earth; this is mainly a labor cost
Transport Cost to the Moon	Net cost of space transportation to the Moon; per kg of usable delivered product, applied to equipment built on Earth and to resupply.
Annual Writeoff Rate	A percentage applied to all nonrecurring costs such as cost of equipment and for its transportation to the Moon as applicable.
Product Value	Feasible selling price for the product delivered to Earth.

As an illustration, the model was applied to a case of platinum-group metals extracted from lunar asteroidal material. Results are given in the two bar charts below, figures 4 and 5.

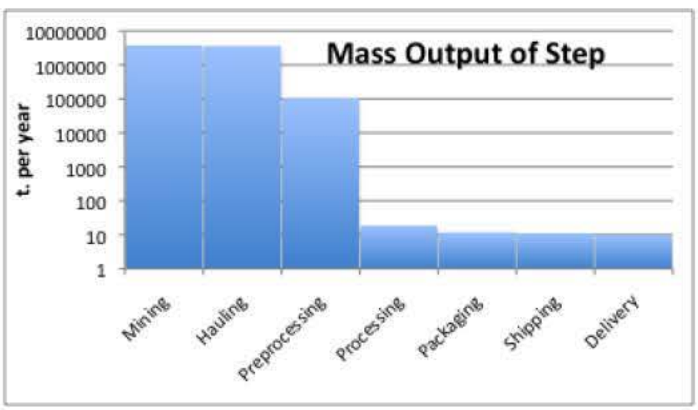


Figure 4. Mass Output through Process Steps

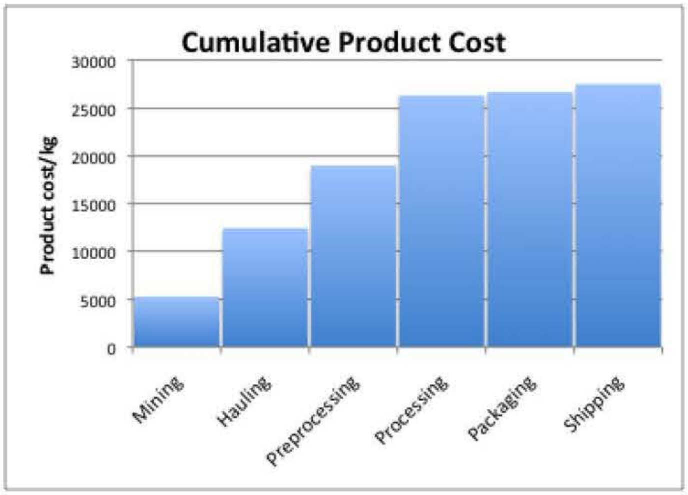


Figure 5. Cumulative Cost of Product for $30,000/kg Lunar Delivery

The two large stepdowns in mass throughput are because (1) the fraction of regolith that is asteroidal nickel-iron is small; we assumed 3% which is a relatively rich ore; the average is less than 1%; (2) the fraction of nickel-iron that is platinum-group metals is very small; we assumed 200 ppm which again is a rich ore.

The cumulative product cost shows how much cost has been added by each step. For this product, which is present in the regolith in small amounts, the cost added by the early steps is large even though these are relatively simple steps; the reason is that a lot of material must be handled. Also, it is essential that these steps involve very little equipment made on Earth, and very little resupply or support from Earth.

Costs of equipment that are feasible are a few thousand per kg, whether made on Earth or the Moon. This example used transport cost to the Moon of $30,000 per kg. The model is also sensitive to that figure. Cutting it in half reduced the final product cost by about 25%.

This is an example of production of a scarce but very valuable material. Platinum itself currently sells for close to $40,000 per kg, and the price is escalating because the supply is small and there are many industrial uses. For products that are more plentiful but less valuable, such as cobalt which currently sells for tens of dollars per kg, the final steps of packaging and shipping, which do not appear very important here, become very important.

A very important lesson learned is that most resource production equipment must be made on the Moon. Shipping it is unaffordable. For example, we assumed 90% of the mining and hauling equipment was made on the Moon. Even though this equipment is very productive (our estimates were that a miner or hauler could process on the order of 100,000 times its own mass in a year) so much of it is needed that indications are about 90% must be made on the Moon. That probably means electronics, electric motors, gears and bearings made on Earth and everything else on the Moon.

A likely early product, much easier to exploit, is propellant made on the Moon. Producing oxygen and hydrogen from water will use processes we understand. Since water and its constituents are a single compound, once the water is extracted and purified equipment we understand can produce the propellants. Propellants made and used on the Moon have the value of the cost of shipping them there from Earth, comparable to the value of platinum on Earth. Propellants made on the Moon and used elsewhere in cislunar space are of less value because the cost of shipping from the Moon must be added in, and the cost of delivery from Earth is less.

Our present ignorance is great.

(1) We do not know the quality of ores or their location. This is even true for early products such as propellants made from water. We know water is there, but we don't know the form (ice may be a reasonably safe assumption), how deep, what fraction of the regolith where it resides, what other volatiles are present, and how accessible.

(2) We have not tested any processes in situ. Some oxygen-making processes such as hydrogen reduction and carbothermal reduction have been tested on Earth, but if significant amounts of water are present, we can probably just use electrolysis, which is one of the steps in these other processes, and it will presumably work just as it does on Earth.

(3) We do not know the properties of asteroidal nickel-iron as made into metal products. We need to know the variability of properties as dependent on ore composition and production means, and whether those properties will suffice for making parts and equipment, or whether (and to what degree) we will need to produce engineered alloys to get the properties we need.

Delivery to Earth

The enriched rare metals will be sent to earth using high technology slings. Modern high strength fibers can sustain the loads to whirl a 180 kg nickel-iron balloon at 3050 m/sec using a 200 m radius sling. The balloon is weighted with rare metals foam for rigidity and to maintain a predetermined center of gravity for earth entry. The balloons contain a thick central disk the diameter of the balloon that contains cast pressure bottles and distributes the rotation loads. On each side of the disk, foil hemispheres of nickel are welded on. The hemispheres are made by spraying molten nickel into a hemispherical mold using plasma/powder metallurgy. The forward section of the disk is filled with foamed rare metals for structural support and placement of the center of gravity. The cast pressure bottles are welded together, and pressurized to high pressure with nitrogen gas propellant. The entire balloon is pressurized to approximately one atmosphere with nitrogen to maintain stiffness for Earth entry. The front half of the sphere is covered with sintered regolith ablator and back half of the central disk has redundant recoverable control packages embedded in it. Each control package has a deployable antenna, solar cells plus battery, small GN&C, and a steerable nozzle for cold gas rocket RCS and OMS. The mass of each control package is 3 kg and the propellant gas is 24 kg. The control packages can vent gas from the balloon during aerodynamic heating and refill the balloon from the bottles as the balloon cools off after entry.

A 200 m radius sling shown in figure 6 will generate 4750 gees at the payload just prior to release. A 180 kg counterweight of lunar-produced iron is carried on the counter arm and is captured in an impact tunnel (the counterweight is destroyed each launch. The counterweight arm is fully functional and serves as a backup in case of problems.

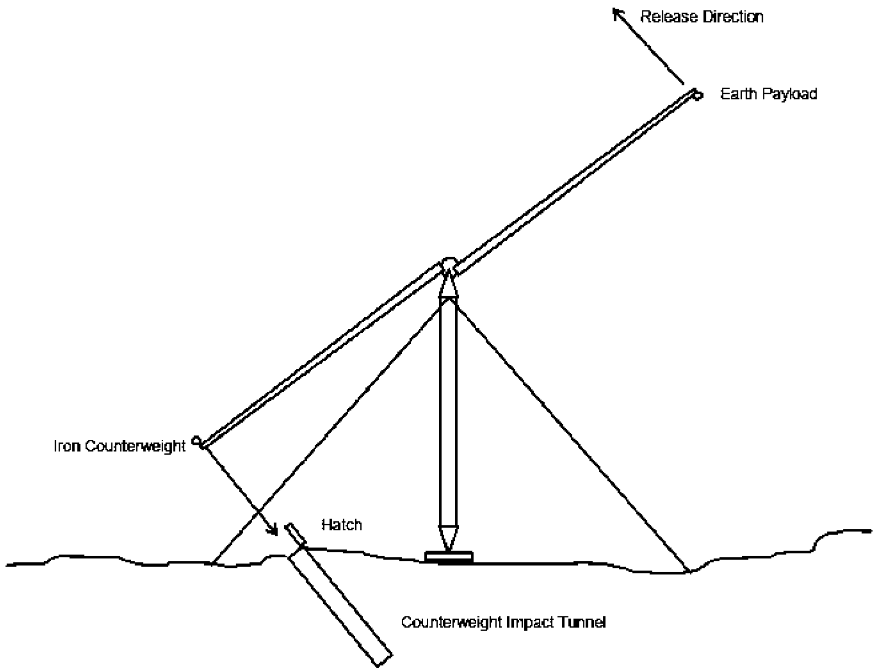


Figure 6. Lunar Sling for Launching Metal Products to Earth (Shown with 200 m radius and launching from 45 deg latitude)

Spinning up the sling requires 3600 kW-hrs of power of which about 2400 kW-hrs is recoverable with 80% efficient combination motor generators and flywheels. Assuming quad-redundant 200 kWe motors, it will take 4.5 hours to load, spin up and release, and three hours to slow down and stop while recovering the energy. During the 4.5 hours spin up, a 300 kWe array could make up the energy shortfall. At a rate of one launch every 8 hours during daylight, a single sling delivers seventy-five tons of useful metal every year. Assuming 10 tons is PGM and rare earth metals and 65 tons is mostly nickel with some cobalt (~ 500 kg), the current value per year per mine is about $400M for the PGM, $1.4M for the nickel, and $45,000 for the cobalt. These numbers will mostly likely be factors higher in twenty years. The hub with motors, arms, flywheels, and power control units masses about ten tons and would be delivered in one piece. The tower, the base, and the guy wires would be lunar manufacture.

The concept of operations (CONOPS) is to launch the balloon spacecraft into an earth intercept trajectory and then fine tune the trajectory as it flies by the L1 transfer station, so that the balloon enters into restricted regions of the Pacific and Indian Oceans. With a ballistic coefficient of 20 kg/m^2 (4 m diameter) the balloons should survive earth entry and be salvageable after a splashdown at 22 m/sec (the foil hemisphere entry body halves will probably tear away on impact, but the built-in flotation foam will keep the core disk floating).

Delivery to Moon

Earth-to-Orbit - The goal is to achieve truly low cost access to space though a combination of a FRETOS coupled with a tether upper stage (TUS) and optimized in innovative ways to get the best use of available technologies. The proposed FRETOS concept uses near-term, low risk, propulsion elements in a simple, robust operationally efficient launcher, with streamlined operations designed to achieve a 24 hour turnaround. Once fully matured, a fleet of 5 FRETOS launchers could support a flight rate of 1,000 launches per year using four tethers. At that flight rate, the target price would be $250 per kg for 15 metric tons of useful payload delivered to either a 1,400 km circular orbit or to a 26,750 km highly-elliptical-transfer orbit (HETO). This price point is considered the "holy grail" of space operations that would enable several viable commercial space markets such as supplying orbital propellant depots, lunar and NEO in-situ resource utilization, building massive space based solar power systems, as well as assembling large space exploration systems for human and robotic missions beyond LEO.

Our FRETOS concept works well because the tether upper stage allows payloads to achieve orbit with an ideal delta V of just 7.0 km/sec, which is 30% less than conventional Earth to Orbit launchers. This lower delta V relaxes the required launcher propellant mass fraction from an extremely challenging 0.945 to very doable 0.865. The proposed FRETOS space segment involves 4 main elements… a Skyhook capture device located at 300 km orbital altitude, a LEO Station located at 1,000 km, a Powered Winch Module located at 1,700 km, and a Counter-Balance located at 2,400 km orbital altitude. The total mass of the fully operational FRETOS space segment is estimated to be 190 metric tons, including 2,100 km of tether lines, ultra-high speed winches, motor/generators, power generation arrays, energy storage flywheels, counter balances, and various station "housekeeping" components. A schematic for the LEO Station portion of the space tether concept is shown in Figure 7 and the overall TUS in figure 8.

Figure 7. TUS Schematic (Not to Scale)

The proposed TUS is designed for quick deployment (ten Heavy EELV launches), so that there is a fast return on the large capital investment. The TUS components are launched separately and docked together in a 400 km orbit. The assembled system is then slowly boosted to 1,400 km for deployment using solar electric propulsion. Once at 1,400 km, the entire tether system is deployed and is ready for operation.

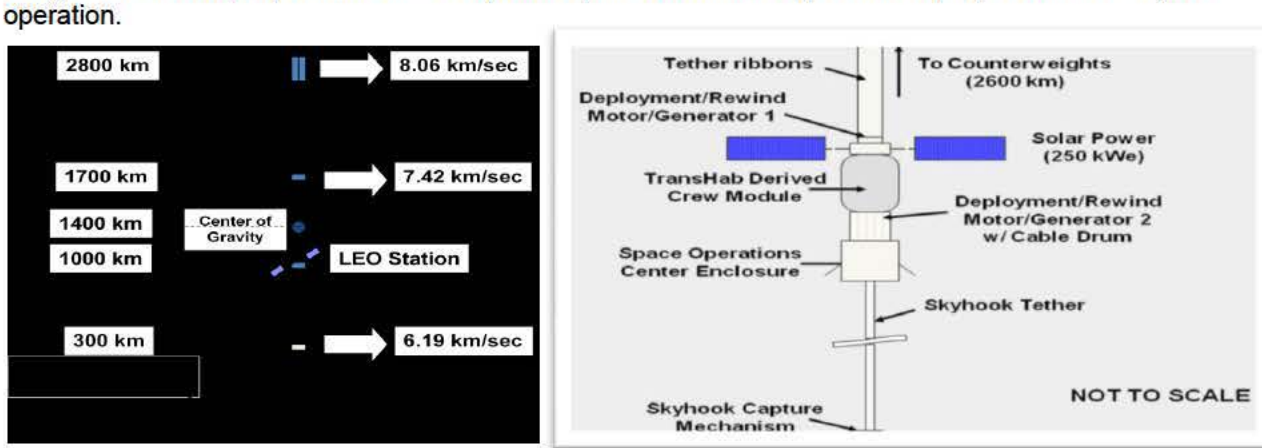


Figure 8. TUS Schematic showing Acceleration Loads

As currently envisioned, the Skyhook B space elements are connected by 3 sets of non-rotating tether lines totaling 2,100 m long. The tether lines are made of high strength, Spectra (Dyneema™ SK75), a commercial product used primarily for high strength marine applications because of its low density (it floats on water) and its high UV tolerance. Each tether line is designed with a safety factor of 2, and uses redundant cabling to reduce the potential for space debris damaging or severing it. In addition to this safety factor and redundancy, the tether is based at the LEO Station roughly 1,000 km high (it moves up and down during operations), which is above 99% of the LEO space debris threat.

In operation, the Skyhook Tether process starts with all tether lines in the positions described in Figure 8. The FRETOS launcher delivers fifteen tons of useful payload to the Skyhook and then reenters for turnaround and reuse. The capture process uses the very robust "probe and gate" approach used successfully for decades on a wide variety of dynamic capture applications such as air-to-air refueling.

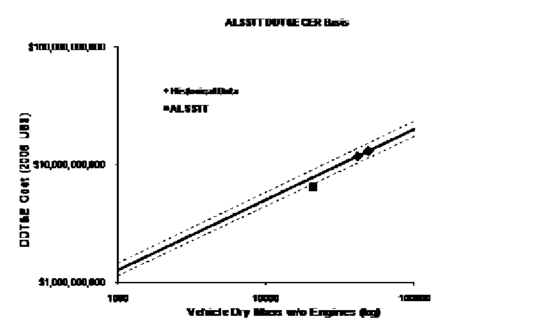
(b) (4)
[redacted]

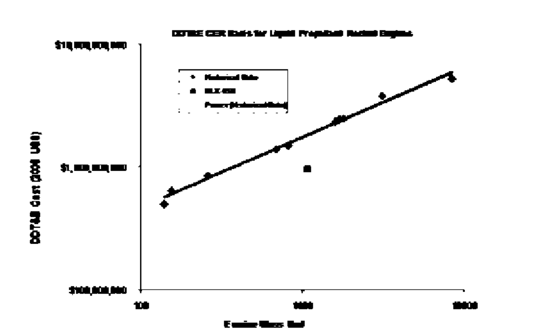
[redacted] [5]. The staging condition for this first stage is nominally Mach 0.56 at 12.6 km altitude which is low enough to use well proven, low cost aircraft designs and standard aircraft recovery with the second stage still attached. This would allow a simple abort capability if for some reason the second stage rocket fails to start or the stage fails to separate as planned.

[redacted] [16] at 80% power, and provides 6.870 km/sec (22,537 feet per second) of ideal delta velocity (much less than any of the previous major FRETOS concepts), with engine-out capability from ignition. After payload capture, the second stage renters and glides to a landing at the launch and recovery base which is located roughly 3,200 nm west of the launch site. The 5 meter diameter payload fairing, (massing 1.8 tons or 4,000 lbm) can be either jettisoned to be destroyed during reentry about 1,500 nm from the launch point, or retained as a shipping container and carried to the final destination.

The FRETOS launcher concept of operations (CONOPS), starting from launch go ahead, is to install the deliverable payload into the Payload Fairing, mate the payload fairing onto the front of the 2nd stage, mate the 2nd stage under the carrier aircraft wing, fuel all systems, and then takeoff and climb to the launch initiation point. Once at the launch initiation point, the carrier aircraft lights its rockets and does a zoom climb maneuver to the staging point where the 2nd stage lights its engine, separates, and boosts the Payload package to the tether rendezvous point. The shroud fairings nominally separate at an altitude of about 80 km during this boost phase. Once at the tether rendezvous point, the 2nd stage aligns the payload with the tether capture "gate" and separates as capture takes place. The 2nd stage then reenters the atmosphere, slows down at very high angle of attack, and glides on to the recovery base about 3,200 nmi down range. The carrier launch aircraft meanwhile flies back to the launch base, and begins refurbishment of its hybrid rocket motors. The flowtime at this point is less than one hour. At the recovery base, the second stage is mated to a transport version of carrier aircraft, which takes off and flies back to the launch base. At the launch base the 2nd stage is de-mated, the inflatable nose is deflated and stowed, and payload processing started. Meanwhile the transport Carrier is refueled and flies back to the recovery base to prepare for the next flight. Our current assessment indicates the entire recovery, turn around, launch and tether lift cycle could be completed within 24 hours, enabling 1,000 missions per year with a fleet of 5 FRETOS launchers and four tethers.

The TUS is an integral part of this low cost access to space architecture. It allows the FRETOS launcher to be built and operated efficiently using existing propulsion and materials technologies. Capturing payloads with the TUS saves roughly 1.7 km/sec (5,566 fps) or about 25% of ideal delta velocity to reach orbit. It also reduces the required propulsion





Figures 9 and 10. Airframe and rocket engine DDT&E Costs are reduced by improved practices and large design margins

system mass fraction from an extremely difficult 0.945 to very doable 0.864. The combined effect reduces the carry mass of the 2nd stage from 300 tons to 200 tons, with a proportional decrease in the development risk and cost.

FRETOS Costs – The single most expensive transportation element to develop is the Air-Launched Single-Stage-to-Tether Vehicle. It has new long-lived Rocket Engines and uses LOX-Hydrogen propellants. The estimated Design, Development, Test and Evaluation (DDT&E) cost estimates for the airframe and engine in 2006$ versus historical programs is shown in figures 9 and 10. The ALSSTT falls below the historical line because much of the design and development process has been automated and fewer manhours are required.

Earth Orbit to Lunar Surface

(b) (4) [redacted]

(b) (4) [redacted][15]. The ReSET characteristics are shown in figure 11 to the right. The trip time from tether release orbit to L1 is about 145 days. The goal is to eventually fuel the ReSET using lunar-generated water.

ReSET Delivery to L1	
ReSET ΔV (SHRO to L1 +10%), m/sec	1,800
Transfer Time, days	145
M in Skyhook Release Orbit Tug, kg	180,000
Peak Thruster Power, kW	2,500
MET Isp, seconds	800
Thruster/PPU Efficiency	0.90
MET Thrust/Weight	1
Array Specific Power, kW/kg	0.130
Flywheel Specific Power, kW-hr/kg	1
Array Busbar Power, kWe	300
Mass Ratio (M0/M1)	1.26
Thrust, N	637.32
Initial Acceleration, m/sec2	0.003541
Powerplant Mass, kg	3,872.7
Array Mass, kg	2,308
Flywheel Mass, kg	1,500
Thruster Mass, kg	65.0
Ascent Prop plus Tank Mass, kg	48,431
ReSET Dry Mass, kg	7,460
P/L + Propellant Launch Mass	178,549
Useful Payload	130,118

Figure 11. ReSET Characteristics

L1 Station – The transfer station at L1: 1) collects payloads coming up from earth and repackages them unto lunar landers going to individual mining sites, tourist destinations, or research centers, 2) collects, stores, and processes propellants for lunar landers, ReSETs, and probably deep space exploration missions, and 3) directs traffic transiting through L1 space, especially the metal balloons leaving the lunar surface and targeted towards areas of the Pacific and Indian Oceans. The L1 station would use ISS type elements launched to the tether and outfitted on-orbit. Most of the processing would be tele-operated from earth, but we still mass allocated 50 tons for L1 Station to support occasional crew.

Lunar Lander – Our lunar lander is a heavy mass version of the Altair Lander designed by NASA. It lands with 60 to 65 tons of payload instead of 12 tons, because landing is by far the riskiest part of the entire mission, and we wanted to minimize program risk. A mass of 180 tons leaving L1 station also allows us to design the lander with engine-out mission completion using proven RL-10 LOX-LH_2 engines. Also, with 65 tons of payload, one lander could provide enough equipment to outfit a mining site to initial operations.

Supply and Demand

We built a ROM economic model of the entire infrastructure needed to acquire and return lunar resources and operated it over a twenty year period. Past twenty years we assumed the resource flow continued to grow at the same rate (constant number of mines added each year). The twenty-year Life cycle Cost (LCC) is shown below (2000$).

(b) (3), (b) (4)

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Conclusions

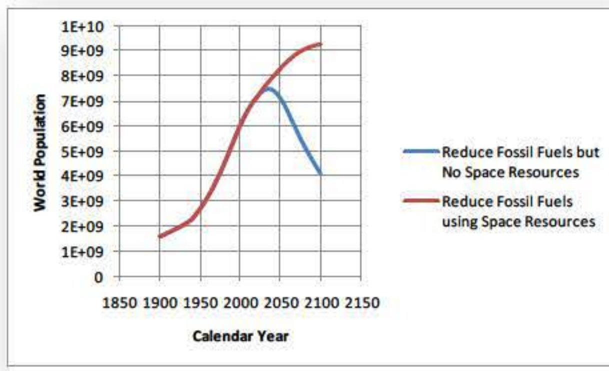

Figure 13 World3 Population Simulation with and without Lunar Resources

Using the World3 model, our best efforts to stabilize the world's population led to a collapse about 2040 caused by a lack of capital to maintain the high crop yields with degrading arable land. We purposely limited growth/year in technology effectiveness to 4% because the critical metals to maintain technology development and expand the replacement of fossil fuels will be extremely scarce and therefore too expensive for general use in twenty years. This scenario is the blue line in figure 13.

If we obtain sufficient lunar resources, there will be abundant CMs to keep the world's rolling stock and smart infrastructure expanding. Therefore, we allowed the growth/year in technology effectiveness to increase to 9% over a twenty year period and avoided much of the cost of obtaining increasingly scarce nonrenewable resources. Together, these two changes make all the difference in generating enough capital to keep expanding crop yields, reducing pollution, and increasing standard of living (GDP/capita). The red line in figure 13 represents this scenario. Even though the population reaches nine billion in this scenario the GDP/capita has reached the level where the population is starting to fall under standard demographics.

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June 3, 2011

Heavy Lift & Propulsion Technology Final Study Report

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0014	Initial	
		For Document Release Use

CONTENT OWNER:

Space Exploration – Exploration Launch Systems

All revisions to this document must be approved by the content owner before release.

National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama
Contract #: NNM11AA09C DRD #1376MA-003

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Document Information

Original Release Date	Contract or CDRL Number (if required)

Authorization for Release

AUTHOR:	*/signed/* [redacted]	AV-2J-AD0R Org. Number	Date
APPROVAL:	*/signed/* [redacted]	AV-2J-AD0R Org. Number	Date
DOCUMENT RELEASE:		Org. Number	Date

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ACRONYMS

BAA	Broad Agency Announcement
BEO	Beyond Earth Orbit
CP	Chemical Propulsion
CPS	Chemical Propulsion Stage
CS	Core Stage
DAC	Design Analysis Cycle
DDT&E	Design, Development, Test and Evaluation
DRM	Design Reference Missions
EELV	Evolved Expendable Launch Vehicle
ELV	Expendable Launch Vehicle
ET	External Tank
FOM	Figures Of Merit
HEFT	Human Exploration Framework Team
HLPT	Heavy Lift and Propulsion Technology
HLV	Heavy Lift Vehicle
Isp	Specific Impulse
IUAC	Instrument Unit Avionics Contract
LCC	Life Cycle Cost
LEO	Low Earth Orbit
LRB	Liquid Rocket Booster
LV	Launch Vehicle
MPCV	Multi-Purpose Crew Vehicle
MSFC	Marshal Space Flight Center
mT	Metric Tonne
NDV	NASA Design Vehicle
NEO	Near Earth Orbit
NEP	Nuclear Electric Propulsion
NRE	Non-Recurring Engineering
NTR	Nuclear Thermal Reactor
SEP	Solar Electric Propulsion
SLS	Space Launch System

SOW	Statement of Work
SRB	Solid Rocket Booster
SSME	Space Shuttle Main Engine
US	Upper Stage
USG	United States Government

EXECUTIVE SUMMARY

The Boeing Company has completed the Statement of Work (SOW) associated with the Heavy Lift and Propulsion Technology (HLPT) Broad Agency Announcement (BAA) Contract #NNM11AA09C. The results of this study are summarized in the main body of this report, with the details provided in multiple separate referenced technical reports which are attached to this report.

During the HLPT BAA study period of performance the issues surrounding our nation's Heavy Lift Launch Vehicle have become increasingly urgent. Boeing has endeavored to inform the government's approach to provide a Launch Vehicle (LV) with capabilities which meet the nation's exploration objectives. Key findings include:

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1.0 SUMMARY OF STUDY TASKS

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APPENDIX A - DETAILED SOW COMPLIANCE

		References Attached #	TIM2	Final Report
4.0.	Study Tasks *HLPT Statement Of Work)	1	✓	✓
4.0.a	The Contractor shall identify and analyze multiple alternative architectures (expendable, reusable, or some combination) on which a Heavy Lift System addressing the NASA HLPT study technical objectives can be based. The "identify" requirements below pertain to topics to be included in the final reports and TIM briefing packages rather than to the manner of the performance of systems analysis or trade study. .	1	✓	✓
4.1.	Key Decision Attributes and Weighting Assessment	2	✓	✓
4.1.a.	The Contractor shall provide a recommended list of key decision attributes and rationale associated with each.	2	✓	✓
4.1.b.	The Contractor shall identify how alternative Heavy Lift System solutions address key decision attributes (figures of merit, measures of effectiveness, etc.).	2	✓	✓
4.1.c.	The Contractor shall provide a recommendation for the weighting of the recommended key decision attributes.	2	✓	✓
4.1.d.	The Contractor shall identify how changes to the weighting of key decision attributes affect the identified alternative architectures.	2	✓	✓

Figure 1-20. Detailed SOW Compliance – Task 4.0 to 4.1.d.

		Reference #	TIM2	Final Report
4.2	Alternative Ground Rules and Assumptions Assessment	5	✓	✓
4.2.a.	The Contractor shall define sets of nominal and alternate ground rules and assumptions for deriving design reference missions (DRMs), architectures, and Heavy Lift System solutions.	5	✓	✓
4.2.b.	The Contractor shall identify how alternative ground rules and assumptions impact the identified alternative Heavy Lift System solutions.	5	✓	✓
4.3	Alternative Architecture TRADES	1,3,5,6,7,8	✓	✓
4.3.a.	The Contractor shall define a set of primary and secondary DRMs derived from the study nominal and alternate ground rules and assumptions.	5	✓	✓

		Reference #	TIM2	Final Report
4.3.b.	The Contractor shall define multiple alternative architectures potentially capable of performing some to all of the primary and secondary DRMs.	1	✓	✓
4.3.b.1	At least one architecture shall include one or more in-space propellant depots.	6	✓	✓
4.3.b.2	At least one architecture shall include a gateway node (e.g., ISS) to serve as a place of rendezvous and/or assembly. For example, between launch vehicle and in-space vehicle elements, or between in-space vehicle element and propellant depot.	6	✓	✓
4.3.c.	The Contractor shall derive and assess Heavy Lift System solutions for the multiple alternative architectures.	3	✓	✓
4.3.c.1	The Contractor shall allocate performance and other functions among the launch vehicle element, in-space vehicle element, and any other architectural elements (e.g., propellant depot) for each alternative architecture.	7	✓	✓
4.3.c.2	The Contractor shall assess incremental launch capability for one or more Heavy Lift System solutions.	8	✓	✓

Figure 1-21. Detailed SOW Compliance – Task 4.2 to 4.3.c.2.

		Reference #	TIM2	Final Report
4.4	Innovative or Non-Traditional Processes Assessment	9	✓	✓
4.4a.	The Contractor shall identify how innovative or non-traditional processes or technologies can be applied to the Heavy Lift Systems to improve its affordability and sustainability.	9	✓	✓
4.5	Commonality Assessment	10	✓	✓
4.5.a.	The Contractor shall identify how aspects of a Heavy Lift System (including stages, subsystems, and major components) could have commonality with other user applications.	10	✓	✓

Figure 1-22. Detailed SOW Compliance – Task 4.4 to 4.5.a.

		Reference #	TIM2	Final Report
4.6	Heavy Lift System Capability Gap Assessment	11	✓	✓
4.6.a.	The Contractor shall identify capability gaps associated with the Heavy Lift System, and for each capability gap identify specific areas where technology development may be needed.	11	✓	✓
4.6.a.1	For those capability gaps identified as requiring technology development, the Contractor shall quantitatively evaluate each capability gap using established metrics: NASA Technology Readiness Level (TRL), Capability Readiness Level (CRL), Manufacturing Readiness Level (MRL), and/or Process Readiness Level (PRL).	11	✓	✓
4.6.a.2	The Contractor shall identify manufacturing and launch process capability gaps for each Heavy Lift System studied.	11	✓	✓
4.6.b.	The Contractor shall identify capability gaps associated with the first-stage main engine functional performance and programmatic characteristics required to support each Heavy Lift System studied.	11	✓	✓
4.6.b.1	The minimum set of functional performance characteristics identified shall include engine thrust, specific impulse (Isp), mixture ratio, mass, throttle range, and physical envelope estimates.	11	✓	✓
4.6.b.2	This assessment shall include, but is not limited to, LOX/RP main engine systems.	11	✓	✓
4.6.b.3	The minimum set of programmatic characteristics identified shall include overall life cycle cost, development schedule, and production rate estimates.	11	✓	✓
4.6.b.4	The Contractor shall identify any impacts to overall life cycle costs of the Heavy Lift System based on the first-stage main engine studied.	11	✓	✓
4.6.c.	The Contractor shall identify capability gaps associated with the upper-stage main engine functional performance and programmatic characteristics required to support each Heavy Lift System studied.	11	✓	✓
4.6.c.1	The minimum set of functional performance characteristics identified shall include engine propellants, thrust, specific impulse (Isp), mixture ratio, mass, throttle range, and physical envelope estimates.	11	✓	✓
4.6.c.2	The minimum set of programmatic characteristics identified shall include overall life cycle cost, development schedule, and production rate estimates.	11	✓	✓

		Reference #	TIM2	Final Report
4.6.c.3	The Contractor shall identify any impacts to overall life cycle costs of the Heavy Lift System based on the upper-stage main engine studied.	11	✓	✓

Figure 1-23. Detailed SOW Compliance – Task 4.6 to 4.6.c.3.

		Reference #	TIM2	Final Report
4.6.d.	The Contractor shall identify capability gaps associated with non-engine technical aspects of the heavy lift launch vehicle element of the Heavy Lift System: tanks, propellant and pressurization systems, integrated system health management, auxiliary propulsion systems, avionics and control systems, and/or structures.	11	✓	✓
4.6.d.1	The Contractor shall identify test and integrated demonstrations to mitigate risk associated with the identified capability gaps.	11	✓	✓
4.6.e.	The Contractor shall identify capability gaps associated with the in-space propulsion elements functional performance and programmatic characteristics required to support each Heavy Lift System studied.	11	✓	✓
4.6.e.1	This assessment shall include, but is not limited to, LOX/H2 and LOX/CH4 propulsion systems.	11	✓	✓
4.6.e.2	The minimum set of functional performance characteristics identified shall include propellant definition, thrust, specific impulse (Isp), mixture ratio, mass, throttle range (if any), and physical envelope estimates.	11	✓	✓
4.6.e.3	The minimum set of programmatic characteristics identified shall include overall life cycle cost, development schedule, and production rate estimates.	11	✓	✓
4.6.e.4	Identify any impacts to overall life cycle costs of the Heavy Lift System based on the engines studied.	11	✓	✓

Figure 1-24. Detailed SOW Compliance – Task 4.6.d to 4.6.e.4.

		Reference #	TIM2	Final Report
4.6.f.	The Contractor shall identify capability gaps associated with non-engine technical aspects of the in-space propulsion element of the Heavy Lift System: tanks, propellant and pressurization systems, cryogenic fluid management, integrated system health management, auxiliary propulsion systems, avionics and control systems, structures, and/or autonomous rendezvous and docking.	11	✓	✓
4.6.f.1	The Contractor shall identify test and integrated demonstrations to mitigate risk associated with the identified capability gaps.	11	✓	✓
4.7	Test and Demonstration Assessment	12	✓	✓
4.7.a.	The Contractor shall identify how incremental development testing, including ground and flight testing, of Heavy Lift System elements can enhance the heavy lift system development.	12	✓	✓
4.7.b.	The Contractor shall identify those in-space space propulsion elements (if any) for demonstration via space flight experiments.	6, 12	✓	✓

Figure 1-25. Detailed SOW Compliance – Task 4.6.f to 4.7.b.

REFERENCES ATTACHED

HLPT No. ITD-HLP-0001	*Heavy Lift Vehicle Alternatives*
HLPT No. ITD-HLP-0002	*Key Decision Attributes and Weighting Assessments*
HLPT No. ITD-HLP-0003	*Heavy Lift Solutions and Architecture Assessments*
HLPT No. ITD-HLP-0005	*Design Reference Missions*
HLPT No. ITD-HLP-0006	*Enabling Architecture Elements Gateways and In-Space Propulsion*
HLPT No. ITD-HLP-0007	*Key Architecture Elements Functional Analysis*
HLPT No. ITD-HLP-0008	*Heavy Lift Incremental Development for Launch Capability*
HLPT No. ITD-HLP-0009	*Innovative and Non-Traditional Approaches*
HLPT No. ITD-HLP-0010	*Commonality Assessment*
HLPT No. ITD-HLP-0011	*Heavy Lift Vehicle Capability Gap Assessment*
HLPT No. ITD-HLP-0012	*Test and In-Space Demonstrations*

HLPT-1

Heavy Lift Vehicle Alternatives

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0001	June 2011	

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National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama Contract #NNM11AA09C: DRD #1376MA-003

Document Information

Original Release Date	**Contract or CDRL Number (if required)**

Authorization for Release

AUTHOR:	*1.1* [redacted] (b) (4)	AC-03-NF31 Org. Number	Date
APPROVAL:	*1.2* [redacted] (b) (4)	AE-H2-S250 Org. Number	Date
DOCUMENT RELEASE:		Org. Number	Date

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EXECUTIVE SUMMARY

The Boeing Company identified and analyzed multiple alternative launch vehicle (LV) architectures. The emphasis of this effort centered on the LVs which were in NASA Marshal Space Flight Center (MSFC) SLS trade-space while other expendable systems, reusable systems and combination systems were considered.

The analysis evaluated various upper stage, first stage and booster options. The evaluation begins with performance analysis including the predicted payload mass (delivered) to Low Earth Orbit (LEO) and/ or to Beyond Earth Orbit (BEO). These analyses address BEO performance requirements (50mT) and LEO performance requirements (130mT) derived from mission analysis and government requirements. Representative analysis cases are included in HLPT paper #1. Launch vehicle configurations with promising performance capabilities were analyzed against a broad range of programmatic and technical Figures of Merit including affordability, schedule, payload performance, industry base and others. The results of those trades are included in HLPT paper #2. Summary findings of both of these evaluations are summarized below:

General –

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1.0 INTRODUCTION

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4.0 LOX/LH2 CORE DESIGN CONCEPTS WITH RS-68B MAIN ENGINES

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9.0 SUMMARY COMPARISON OF RS25 AND RS68-BASED VEHICLES

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[redacted]

(b) (4), (b) (3)

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Acronyms

BEO Beyond Earth Orbit

BLEO Beyond Low Earth Orbit

CBC Common Booster Cores

FOM Figures Of Merit

HLLV Heavy Lift Launch Vehicle

HLV Heavy Lift Vehicle

HLPT Heavy Lift & Propulsion Technology

IMLEO Injected Mass into Low Earth Orbit

lbf Pounds Force

LEO Low Earth Orbit

LRB Liquid Rocket Boosters

LV Launch Vehicle

MECO Main Engine Cut-off

MSFC Marshal Space Flight Center

OML Outer Mold Line

NRV NASA Reference Vehicle

POR Point Of Reference (generally the Ares 1 US

SLS Space Launch System

SLV Space Launch Vehicle (as an element of the SLS)

SOW Statement of Work

SRB Solid Rocket Booster

STS Shuttle Transportation System

SSME Space Shuttle Main Engine

TVC Thrust Vector Control

VAB Vertical Assembly Building

References

Reference 1

Preliminary Report Regarding NASA's Space Launch System and Multi-Purpose Crew Vehicle Pursuant to Section 309 of the NASA Authorization Act of 2010 (P.L. 111-267), as presented to Congress, January 2011

Reference 2

PUBLIC LAW 111–267—OCT. 11, 2010 124 STAT. 2805
Public Law 111–267, 111th Congress
An Act to authorize the programs of the National Aeronautics and Space Administration for fiscal years 2011 through 2013, and for other purposes. AKA: NASA Authorization Act of 2010.

Reference 3

Consolidated Analysis - Kutter (ULA) open distribution Paper on ACES (12/2010)

Appendix A

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HLPT-2

Key Decision Attributes and Weighting Assessments

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0002	**June 2011**	
		For Document Release Use

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Space Exploration – Exploration Launch Systems

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National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama   Contract #NNM11AA09C: DRD #1376MA-003

Document Information

Original Release Date	Contract or CDRL Number (if required)
4-20-2011	

Authorization for Release

		Org. Number	Date
	[redacted]	AV-2J-AD0R	
APPROVAL:	[redacted]	AV-2J-AD0R	
DOCUMENT RELEASE:			

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Executive Summary

This paper summarizes the formal trade study process applied to the evaluation of Space Launch System heavy lift launch vehicles – the Space Launch Vehicle. Figures of Merit (FOMs) are based on NASA data from the Preliminary Report to Congress, dated 11 January 2011 and other public information. The category of FOMs are held constant, and three cases are studied utilizing different weighting factors for each FOM.

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1.0 Introduction and Scope

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2.0 Recommended List of Figures of Merit and Rationale

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5.0 Alternate FOM Weighting

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6.0 SUMMARY

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[redacted]

Acronyms

BAA Broad Agency Announcement
BEO Beyond Earth Orbit
BLEO Beyond Low Earth Orbit
CBC Common Booster Cores
FOM Figures Of Merit
HLLV Heavy Lift Launch Vehicle
HLV Heavy Lift Vehicle
HLPT Heavy Lift & Propulsion Technology
IMLEO Injected Mass into Low Earth Orbit
KDA Key Decision Attribute
lbf Pounds Force
LEO Low Earth Orbit
LRB Liquid Rocket Boosters
LV Launch Vehicle
MECO Main Engine Cut-off
MSFC Marshal Space Flight Center
OML Outer Mold Line
NRV NASA Reference Vehicle
POR Point Of Reference (generally the Ares 1 US
SLS Space Launch System
SLV Space Launch Vehicle (as an element of the SLS)
SOW Statement of Work
SRB Solid Rocket Booster
STS Shuttle Transportation System
SSME Space Shuttle Main Engine
TVC Thrust Vector Control
VAB Vertical Assembly Building

References

(1) **Preliminary Report Regarding NASA's Space Launch System and Multi-Purpose Crew Vehicle: Section 309 of the NASA Authorization Act of 2010 (P.L. 111-267), dated January 2011**

(2) **Space Launch System Goals for use by SLS Study Teams, Derived from HEFT SE Team by SLS Steering Committee, dated December 6, 2010**

(3) **Boeing Space Exploration Trade Study – Houston Supplemental Information HOU-EGP-005, dated September 25, 2008**

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HLPT-3

Heavy Lift Solutions and Architecture Assessments

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0003	**June 2011**	
		For Document Release Use

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Space Exploration – Exploration Launch Systems

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National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama Contract #NNM11AA09C: DRD #1376MA-003

Document Information

Original Release Date 4/21/2011	**Contract or CDRL Number (if required)**

Authorization for Release

AUTHOR:	[redacted]	AE-H2-S250 Org. Number	Date
APPROVAL:	[redacted]	AV-2J-AD0R Org. Number	Date
DOCUMENT RELEASE:		Org. Number	Date

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EXECUTIVE SUMMARY

The Heavy Lift and Propulsion Technology (HLPT) study provides consistent assessment of multiple trajectory runs based on a wide variety of engine, propellant, size and launch vehicle architecture options. For this study, those objectives were defined as a minimum of 70 metric tons initial capability, desired early service capability of 100 metric tons, and potential for growth to 130 metric tons for exploration. [redacted] (b) (4)

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1.0 INTRODUCTION

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[redacted]

2.0 DISCUSSION

(b) (4), (b) (3)

[redacted]

3.0 ARCHITECTURE ASSESSMENT AND DISCUSSION

(b) (4), (b) (3)

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(b) (4), (b) (3)

[redacted]

4.0 CONCLUSION

(b) (4), (b) (3)

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Acronyms

BEO	Beyond Earth Orbit
CONOP	Concept of Operations
DDT&E	Design, Development, Test & Evaluation
DRM	Design Reference Mission
FOM	Figures of Merit
HLPT	Heavy Lift and Propulsion Technology
L-1	Lagrange point 1
LEO	Low Earth Orbit
NEO	Near-Earth Object
SEP	Solar Electric Propulsion
SRB	Solid Rocket Booster
SSME	Shuttle System Main Engine
TLI	Trans-Lunar Injection

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HLPT-5

Design Reference Missions

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0005	**June 2011**	

For Document Release Use

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National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama   Contract #NNM11AA09C: DRD #1376MA-003

Document Information

Original Release Date	Contract or CDRL Number (if required)
4/20/2011	

Authorization for Release

AUTHOR: [redacted] AV-2C-AG0M
Org. Number Date

APPROVAL: [redacted] AV-2J-AD0R
Org. Number Date

DOCUMENT RELEASE:
Org. Number Date

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Executive Summary

This document defines the Design Reference Missions (DRMs) for the Heavy Lift and Propulsion Technology Broad Agency Announcement (BAA) activities. These DRMs have been divided into a set of primary and secondary DRMs. The primary DRMs are human missions to the moon, NEOs, and Mars. The set of secondary DRMs were developed to establish the in-space infrastruture to support the primary DRMs.

The primary Design Reference Missions selected for this study include a:

- Moon Landing in 2026
- Near Earth Object rendenzvous in 2028
- Mars Moon/Fly-by in 2032
- Mars Landing in 2040

These DRMs represent a series of missions which systematically expand human space exploration out from LEO to a neighboring planet. Executing these DRMs requires the use of a HLV described by NASA for the Space Launch System.

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These DRMs require a series of supporting development missions which also allow for a series of human mission opportunities as the various propulsive elements are developed and deployed. While the primary DRMs require the development of non-propulsion elements, these are not within HLPT study scope and are therefore treated only at a summary level.

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1. INTRODUCTION

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2. GROUNDRULES AND ASSUMPTIONS FOR DRM DEVELOPMENT

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3. DESIGN REFERENCE MISSIONS

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4. SUMMARY

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Acronyms

AR&PO	Autonomous Rendezvous & Proximity Operations
BAA	Broad Agency Announcement
BART	Built A Rocket Tool
BEO	Beyond Earth Orbit
CONOP	Concept of Operations
CCDev	Commercial Crew Development
CE&R	Concept Evolution and Refinement (2005 NASA contract)
CEV	Crew Exploration Vehicle
CG	Center of Gravity
CTV	Crew Transfer Vehicle
DRA	Design Reference Architecture
DRM	Design Reference Mission
EELV	Evolved Expendable Launch Vehicles (Delta IVs and Atlas Vs)
ESL2	Earth-Sun Lagrange point #2
EML-1	Earth Moon Lagrange point #1
FAST	Fast Access Spacecraft Testbed (Air Force Project to development technology)
FOM	Figure of Merit
G	Acceleration measured relative to Earth's
GEO	Geostationary Earth Orbit
GR&A	Groundrules and Assumptions
GTO	Geostationary Transfer Orbit
HE	High Energy
HEPT	Human Exploration Framework Team
HLPT	Heavy Lift and Propulsion Technology
HLLV	Heavy Lift Launch Vehicle
HLV	Heavy Lift Vehicle
HR	Human rated
IMLEO	Initial Mass to Low Earth Orbit
iMAT	Interactive Mission Analysis Tool
IOC	Initial Operational Capability
IRAD	Internal Research and Development
ISS	International Space Station
L-1	Earth Moon Lagrange point #1, AKA L-1 gateway
LCC	Life Cycle Cost
LEO	Low Earth Orbit
ME	Medium Energy
MPCV	Multi-Purpose Crew Vehicle
m	meter
mt	metric tons
MW	Megawatt
N/A	Not Applicable
NASA	National Aeronautics and Space Administration
NEO	Near Earth Object (Asteroids with orbits that approach Earth's orbit)

NEP	Nuclear Electric Propulsion
NES	Near Earth Space
NTP	Nuclear Thermal Propulsion
P/L	Payload
RCS	Reaction Control System (small thrusters to control vehicle direction, pitch, yaw and roll)
RFI	Request for Information
RTG	Radioisotope Thermoelectric Generator
SEL2	Sun-Earth Lagrange Point #2
SEP	Solar Electric Propulsion
SLS	Space Launch System
SLV	Space Launch Vehicle
SSME	Space Shuttle Main Engine (RS-25)
STCAEM	Space Transfer Concepts and Analyses for Exploration Missions
STS	Space Transportation System
t	tonne or metric ton
TBD	To Be Determined
TLI	Translunar Injection
VASIMR	Variable Specific Impulse Magnetoplasma Rocket

References

[1] Boeing CE&R Option Period Final Report, Doc. #NS05HB046, Contract No. NNT04AA00C, Deliverable 14, September 15, 2005

[2] NASA Design Reference Architecture (DRA) Approach NASA/TM—2010–216450.

[3] M. Raftery, G.R. Woodcock, International Space Station As A Platform For Exploration Beyond Low Earth Orbit, IAC-10-D9.2.8 , 61st International Astronautical Congress, 2010

[4] LeRoy, S., 2011, "Ground Rules and Assumptions," Boeing HLPT Study White Paper No. 6.

[5] LeRoy, S., 2011, "Architecture Functional Analysis," Boeing HLPT Study White Paper No. 7.

[6] Thomas, B.; Vaughan, D.; Drake, B; Griffin, B.; Woodcock, G; A Comparison of Transportation Systems for Human Missions to Mars, AIAA 2004-3834,

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HLPT 6

Enabling Architecture Elements
Gateways & In-Space Propulsion

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0006	**June 2011**	

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National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama Contract #NNM11AA09C: DRD #1376MA-003

Document Information

Original Release Date	Contract or CDRL Number (if required)
4/20/2011	

Authorization for Release

AUTHOR:	(b) (4) [redacted]	AV-2C-AG0M Org. Number	Date
AUTHOR:	(b) (4) [redacted]	AV-2C-AG0M Org. Number	Date
APPROVAL:	(b) (4) [redacted]	AV-2J-AD0R Org. Number	Date
DOCUMENT RELEASE:		Org. Number	Date

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Executive Summary

The HLPT study addressed how potential use of different gateways and in-space propulsion options would affect the requirements driving the design of a Heavy Lift Vehicle (HLV) (Shuttle Launch System (SLS)). Gateways considered herein include the following options: non-use of an in-space gateway, use of Low Earth Orbit (LEO) as a gateway and the use of the L1 point as a gateway. (b) (4) (b) (4) [redacted]

1.0 INTRODUCTION AND SCOPE

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2.0 IN-SPACE PROPULSION ELEMENTS

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3.0 GATEWAYS

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4.0 SUMMARY AND CONCLUSIONS

4.1 Summary

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Acronyms

BAA	Broad Agency Announcement
BEO	Beyond Earth Orbit
BPCS	Brayton Power Conversion System
CMB	Common Module Bay
CRV	Crew Return Vehicle
dV	Delta Velocity
DTS	Drop Tank System
EDS	Earth Departure Stage
FAST	Fast Access Spacecraft Testbed
GEO	Geosynchronous Orbit
HEO	Helio-centric Orbit
HLV	Heavy Lift Vehicle
HLPT	Heavy Lift & Propulsion Technology
IMLEO	Initial Mass in Low Earth Orbit
ISP	Specific Impulse
LEO	Low Earth Orbit
LV	Launch Vehicle
MPD	Magnetoplasmadynamic [thruster]
MMH	Mono-Methyl Hydrazine
NEP	Nuclear Electric Propulsion
NTO	Nitros-Tetroxide
NTP	Nuclear Thermal Propulsion
OTV	Orbital Transfer Vehicle
PIT	Pulsed Inductive Thrusters
POD	Point-of-Departure
SEP	Solar Electric Propulsion
SLS	Shuttle Launch System
SM	Service Module
SNAP	Space Nuclear Auxiliary Power
SOI	Sphere of Influence
TEI	Trans-Earth Injection
TMI	Trans-Mars Injection

References

Reference 1
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HLPT-7

Architecture Elements Functional Analysis

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0007	**June 2011**	
		For Document Release Use

CONTENT OWNER:
Space Exploration – Exploration Launch Systems

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National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama Contract #NNM11AA09C: DRD #1376MA-003

Document Information

Original Release Date 4/20/2011	**Contract or CDRL Number (if required)**

Authorization for Release

AUTHOR: [redacted] (b) (4) — AV-2C-AG0M (Org. Number) — Date

APPROVAL: [redacted] (b) (4) — AV-2J-AD0R (Org. Number) — Date

DOCUMENT RELEASE: ________ — Org. Number — Date

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Executive Summary

This white paper summarizes the architectural elements and the top-level functionality for the Heavy Lift and Propulsion Technology (HLPT) study. The functionality and performance allocation for the major elements have been documented to the level necessary to compare heavy lift vehicles and other in-space technologies for a variety of alternative architectures, Design Reference Missions (DRM) and ground rules and assumptions.

This paper also addresses the statement of work (SOW) section 4.3.c.1. "The Contractor shall allocate performance and other functions among the launch vehicle element, in-space vehicle element, and any other architectural elements (e.g., propellant depot) for each alternative architecture". This task is typically called functional analysis.

The top 17 functions have been mapped at the architecture level against 20 independent elements which comprise an architecture to meet the study DRMs. Each element has been defined and important performance allocations have been documented.

By packaging actual payloads into the HLV and in-space propulsion elements instead of idealized infinitely divisible masses, the advantage and disadvantages of various propulsion technologies can be better assessed. Addressing specific elements drives out key technology limitations and capability gaps.

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2.0 ELEMENT FUNCTIONS

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3.0 CONCLUSION

(b) (3)

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Acronyms

AR&PO	Autonomous Rendezvous & Proximity Operations
BAA	Broad Agency Announcement
BEO	Beyond Earth Orbit
CONOP	Concept of Operations
CCDev	Commercial Crew Development
CE&R	Concept Evolution and Refinement (2005 NASA contract)
CEV	Crew Exploration Vehicle
CG	Center of Gravity
CTV	Crew Transfer Vehicle
DRA	Design Reference Architecture
DRM	Design Reference Mission
EELV	Evolved Expendable Launch Vehicles (Delta IVs and Atlas Vs)
ESL2	Earth-Sun Lagrange point No.2
EML-1	Earth Moon Lagrange point No.1
FAST	Fast Access Spacecraft Testbed (Air Force Project to development technology)
FOM	Figure of Merit
FRAM	Flight Releasable Attachment Mechanism
G	Acceleration measured relative to Earth's
GEO	Geostationary Earth Orbit
GR&A	Ground Rules and Assumptions
GTO	Geostationary Transfer Orbit
HE	High Energy
HEPT	Human Exploration Framework Team
HLPT	Heavy Lift and Propulsion Technology
HLS	Heavy Lift System
HLV	Heavy Lift Vehicle
HR	Human rated
IMLEO	Initial Mass to Low Earth Orbit
IOC	Initial Operational Capability
IRAD	Internal Research and Development
ISS	International Space Station
L-1	Earth Moon Lagrange point No.1
LCC	Life Cycle Cost
LEO	Low Earth Orbit
ME	Medium Energy
MPCV	Multi-Purpose Crew Vehicle
m	meter
mt	metric tons
MW	Megawatt
N/A	Not Applicable
NASA	National Aeronautics and Space Administration
NEO	Near Earth Object (Asteroids with orbits that approach Earth's orbit)
NEP	Nuclear Electric Propulsion
NES	Near Earth Space

NTP	Nuclear Thermal Propulsion
ORU	Orbital Replacement Unit
P/L	Payload
RCS	Reaction Control System (small thrusters to control vehicle direction, pitch, yaw and roll)
RFI	Request for Information
RTG	Radioisotope Thermoelectric Generator
SEL2	Sun-Earth Lagrange Point No.2
SEP	Solar Electric Propulsion
SLS	Space Launch System
SLV	Space Launch Vehicle
SSME	Space Shuttle Main Engine (RS-25)
STCAEM	Space Transfer Concepts and Analyses for Exploration Missions
STS	Space Transportation System
t	tonne or metric ton
TBD	To Be Determined
TLI	Translunar Injection
VASIMR	Variable Specific Impulse Magnetoplasma Rocket

BOEING

HLPT-8

Heavy Lift Incremental Development for Launch Capability

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0008	**June 2011**	
		For Document Release Use

CONTENT OWNER:

Space Exploration – Exploration Launch Systems

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National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama Contract #NNM11AA09C: DRD #1376-003

Document Information

Original Release Date	**Contract or CDRL Number (if required)**
4-20-2011	

Authorization for Release

AUTHOR:	(b) (4) [redacted]	AE-H2-S250 Org. Number	Date
APPROVAL:	(b) (4) [redacted]	AV-2J-AD0R Org. Number	Date
DOCUMENT RELEASE:		Org. Number	Date

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3. Summary and Conclusions

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Acronyms

BEO	Beyond Earth Orbit
CBC	Common Booster Core (Delta IV Booster)
CCB	Common Core Booster (Atlas V Booster)
EELV	Evolved Expendable Launch Vehicles (Delta IVs and Atlas Vs)
GEO	Geostationary Earth Orbit
GR&A	Groundrules and Assumptions
GTO	Geostationary Transfer Orbit
HEFT	Human Exploration Framework Team
HLPT	Heavy Lift and Propulsion Technology
HLV	Heavy Lift Vehicle
IMLEO	Initial Mass to Low Earth Orbit
IRAD	Internal Research and Development
ISS	International Space Station
ITTB	Inter-tank Thrust Beam
L-1	Earth Moon Lagrange point #1, AKA L-1 gateway
LCC	Life Cycle Cost
LEO	Low Earth Orbit
MPCV	Multi-Purpose Crew Vehicle
m	meter
mT	metric tons
N/A	Not Applicable
NASA	National Aeronautics and Space Administration
NEO	Near Earth Object (Asteroids with orbits that approach Earth's orbit)
P/L	Payload
RP	RP-1 rocket fuel
SEL2	Sun-Earth Lagrange Point #2
SEP	Solar Electric Propulsion
SLS	Space Launch System
SLV	Space Launch Vehicle
SSME	Space Shuttle Main Engine (RS-25)
STS	Space Transportation System
TBD	To Be Determined
TLI	Translunar Injection
U.S.	United States of America

HLPT-9

Innovative and Non-Traditional Approaches

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0009	June 2011	

For Document Release Use

CONTENT OWNER:

Space Exploration – Exploration Launch Systems

All revisions to this document must be approved by the content owner before release.

National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama Contract #:NNM11AA09C DRD #1376MA-003

Document Information

Original Release Date	Contract or CDRL Number (if required) **NNM11AA09C**

Authorization for Release

AUTHOR: [redacted] (b) (4) — AV-2J-AD0R Org. Number — Date

APPROVAL: [redacted] (b) (4) — AV-2C-AG0M Org. Number — Date

DOCUMENT RELEASE: ________ Org. Number — Date

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Executive Summary

In the area of innovation, Boeing provides information from our technical experts and supplier base and we bring "reach back" from our large scale production, operations, upgrade history, and requirements and design experiences. Our study approach draws from our broad experience in the development, production and decades-long operation of space and launch systems vehicles. [redacted] This innovation is balanced with the rigor derived from our long experience in development, design and operation of human-rated systems, such as Space Shuttle and the International Space Station.

This report provides a survey of innovations from recent government and Boeing programs which are applicable to SLS heavy lift launch vehicle development and propulsion technology, and describe several specific technology innovations which represent opportunities for improvement of launch vehicle performance and cost.

The effective application of innovation and 'non-traditional' approaches in NASA represents a significant advantage for many activities considered required for HLPT. Specific innovations, demonstrated in recent programs within Boeing and the US government (many in partnership with MSFC) provide substantial advantage in development, production and operation of a Space Launch System.

Additionally, several promising technologies provide opportunity to improve launch vehicle performance, production efficiency and in-space propulsion. [redacted]

1.0 INTRODUCTION AND SCOPE

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3.0 CONCLUSION

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Acronyms

ATP	Authority To Proceed
BASP	Boeing Agile Software Process
BDS	Boeing Defense, Space & Security
RUBA	Roll On Beam Assembly (RUBA)
CAD	Computer Aided Design
CCM	Continuous Compression Molding
CCDev	Commercial Crew Development
CDR	Critical Design Review
CFD	Computational Fluid Dynamics
CLIN	Contract Line Item Number
DDT&E	Design, Development, Test, & Evaluation
DAC	Design Analysis Cycle
EAC	Estimate at Completion
EP	Electric Propulsion
HLPT	Heavy Lift & Propulsion Technology
HLV	Heavy Lift Vehicle
IMS	Integrated Master Schedule
ISS	International Space Station
IUAC	Instrument Unit Avionics Contract
LV	Launch Vehicle
MCA	Multifunctional Concentrator Assembly
NDT	NASA Design Team
PDR	Preliminary Design Review
PFT	Press Forming Technology
PMAD	Power Management and Distribution
SEP	Solar Electric Propulsion
SDR	System Definition Review
SLS	Space Launch System
SLV	Space Launch Vehicle
USG	United States Government
USPC	Upper Stage Production Contract
WIP	Work In Process

HLPT-10

Commonality Assessment

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0010	**June 2011**	

For Document Release Use

CONTENT OWNER:

Space Exploration – Exploration Launch Systems

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National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama   Contract #NNM11AA09C: DRD #1376MA-003

Document Information

Original Release Date	Contract or CDRL Number (if required)
04/20/2011	

Authorization for Release

AUTHOR: [redacted] AV-2J-AD0R
Org. Number Date

APPROVAL: [redacted] AE-H2-S250
Org. Number Date

DOCUMENT RELEASE:
Org. Number Date

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Executive Summary

This white paper presents the results of commonality assessments for the SLS Launch Vehicle. [redacted] Commonality is assessed for components, manufacturing, and analysis methods.

[redacted]

[redacted]

The approach to commonality must be tailored to reach the correct balance between performance, cost and schedule. The table below provides a listing of the types of commonality along with benefits and select examples of where it is applicable to our design.

[redacted]

[redacted]

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3. Summary & Conclusion

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Acronyms

BAA	Broad Agency Announcement
BDU	Brass-board Development Unit
BEO	Beyond Earth Orbit
CONOP	Concept of Operations
CCDev	Commercial Crew Development
CdA	discharge area
CG	Center of Gravity
Core Stage	First Stage of the SLS Vehicle
DRM	Design Reference Mission
DSH	Deep Space Hab
EELV	Evolved Expendable Launch Vehicles (Delta IVs and Atlas Vs)
FAST	Fast Access Spacecraft Test bed
FTINU	Fault-Tolerant Inertial Navigation Unit
G	Acceleration measured relative to Earth's
GR&A	Groundrules and Assumptions
HLPT	Heavy Lift and Propulsion Technology
HLV	Heavy Lift Vehicle
ISS	International Space Station
IUAC	Ares Instrument Unit Avionics Contract
LCC	Life Cycle Cost
LEO	Low Earth Orbit
LV	Launch Vehicle
MPCV	Multi-Purpose Crew Vehicle
mt	metric tone
N/A	Not Applicable
NASA	National Aeronautics and Space Administration
NDT	Ares NASA Design Team
NEO	Near Earth Object
P/L	Payload
PDR	Preliminary Design Review
QD	Quick Disconnect
RCS	Reaction Control System
RINU	Redundant Inertial Navigation Unit
RGA	Rate Gyro Assembly
SITF	Software Integration Test Facility
SLS	Space Launch System
SLV	Space Launch Vehicle
SRB	Solid Rocket Booster
SSME	Space Shuttle Main Engine (RS-25)
t	Metric tones

TBD	To Be Determined
TVC	Thrust Vector Control
USPC	Ares Upper Stage Production Contract
VRV	Vent Relief Valve

HLPT -11

Heavy Lift Vehicle Capability Gap Assessment

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0011	**June 2011**	

For Document Release Use

CONTENT OWNER:

Space Exploration – Exploration Launch Systems

National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama Contract #: NNM11AA09C DRD #1376MA-003

Document Information

Original Release Date	Contract or CDRL Number (if required)
04-21-2011	

Authorization for Release

AUTHOR:	[redacted]	AC-03-NF50	
		Org. Number	Date
APPROVAL:	/ [redacted]	AV-2J-AD0R	
		Org. Number	Date
DOCUMENT RELEASE:			
		Org. Number	Date

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Acronyms

AFP	Automated Fiber Placement
ATP	Authority to Proceed
BEO	Beyond Earth Orbit
DC-X	Delta Clipper Experimental
DRM	Design Reference Mission
EMA	Electromechanical actuation systems
FDIR	Fault Detection, Isolation and Recovery
GGC	Gas Generator Cycle
GH_2	Gaseous Hydrogen
GO_2	Gaseous Oxygen
HLV	Heavy Lift Vehicle
LCH_4	Liquid Methane
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
MPS	Main Propulsion System
MTTF	Mean Time to Failure
OoA	Out-of-Autoclave
SCC	Stage Combustion Cycle
SLS	Space Launch System
SOFI	Spray-On-Foam-Insulation

References

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Appendix A HLV Survey

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HLPT-12

Test and In-Space Demonstrations

DOCUMENT NUMBER:	RELEASE/REVISION:	RELEASE/REVISION DATE:
ITD-HLP-0012	June 2011	
		For Document Release Use

CONTENT OWNER:

Space Exploration – Exploration Launch Systems

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National Aeronautics and Space Administration
Marshall Space Flight Center
Huntsville, Alabama Contract #NNM11AA09C: DRD #1376MA-003

Document Information

Original Release Date	Contract or CDRL Number (if required)
04/22/2011	

Authorization for Release

AUTHOR:	[redacted] (b) (4)	9R-TE-8XXH	
		Org. Number	Date
APPROVAL:	[redacted] (b) (4)	AE-H2-S250	
		Org. Number	Date
DOCUMENT RELEASE:			
		Org. Number	Date

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EXECUTIVE SUMMARY

This paper discusses the incremental testing approach to the Space Launch System (SLS). This approach is directed by program-level requirements. Implementation will be based on a combination of structural, component and subsystem development activities. These will be followed by component, subsystems and structural qualification and integrated stage-level analyses, verification tests and finally acceptance tests. The key elements of this test approach include maximum reuse of test articles, and a “test early”- “test what you fly” philosophy.

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1.0 INTRODUCTION

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2.0 TEST PROGRAM OVERVIEW

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3.0 CONCLUSION

This paper summarizes the Boeing approach to SLS test and evaluation. The goal was to understand NASA's needs and the expected requirements. Early opportunities for improvement and efficiency gains have been identified. [redacted]

The SLS test approach provides an affordable, timely and evolvable solution for meeting NASA's requirements while minimizing life cycle cost. This will be accomplished by leveraging the test planning, facility upgrades and infrastructure improvements NASA made during the Ares I program and finding innovative ways to minimize the number of large scale articles through re-use and commonality.

Acronyms

A&S	Avionics and Software
A&S	Avionics and Software
CS	Core Stage
DDT&E	Design, Development, Test & Evaluation
EGSE	Electrical Ground Support Equipment
GFE	Government Furnished Equipment
GHe	Gaseous Helium
GN2	Gaseous Nitrogen
GVTGRC	Glenn Research Center
GVT	Ground Vibration Test
HDS	Hydrodynamic Support Systems
HFTA	Hot Fire Test Article
H/W	Hardware
HWIL	Hardware in the Loop
I&CO	Integration & Checkout
IMS	Integrated Master Schedule
IRR	Integration Readiness Review
IS	Interstage
IT	Intertank
IU	Instrument Unit
KSC	Kennedy Space Center
LH2	Liquid Hydrogen
LN2	Liquid Nitrogen

LO2	Liquid Oxygen
MAF	Michoud Assembly Facility
MPS	Main Propulsion System
MSFC	Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
PLF	Payload Fairing
PMTP	Program Master Test Plan
QTA	Qualification Test Article
RCS	Reaction Control System
SDRL	Supplier Deliverable Requirements List
SLS	Space Launch System
SLV	Space Launch Vehicle
SRB	Solid Rocket Booster
SS	Subsystem
SSC	Stennis Space Center
SSME	Space Shuttle Main Engine
STA	Structural Test Article
SWAIL	Software & Avionics Integration Lab
TBD	To Be Determined
TS	Thrust Structure
TSS	Test Summary Sheet
TVC	Thrust Vector Control
US	Upper Stage
USM	Ullage Settling Motors
USP	Upper Stage Production

WSMR	White Sands Missile Range

A&S	Avionics and Software
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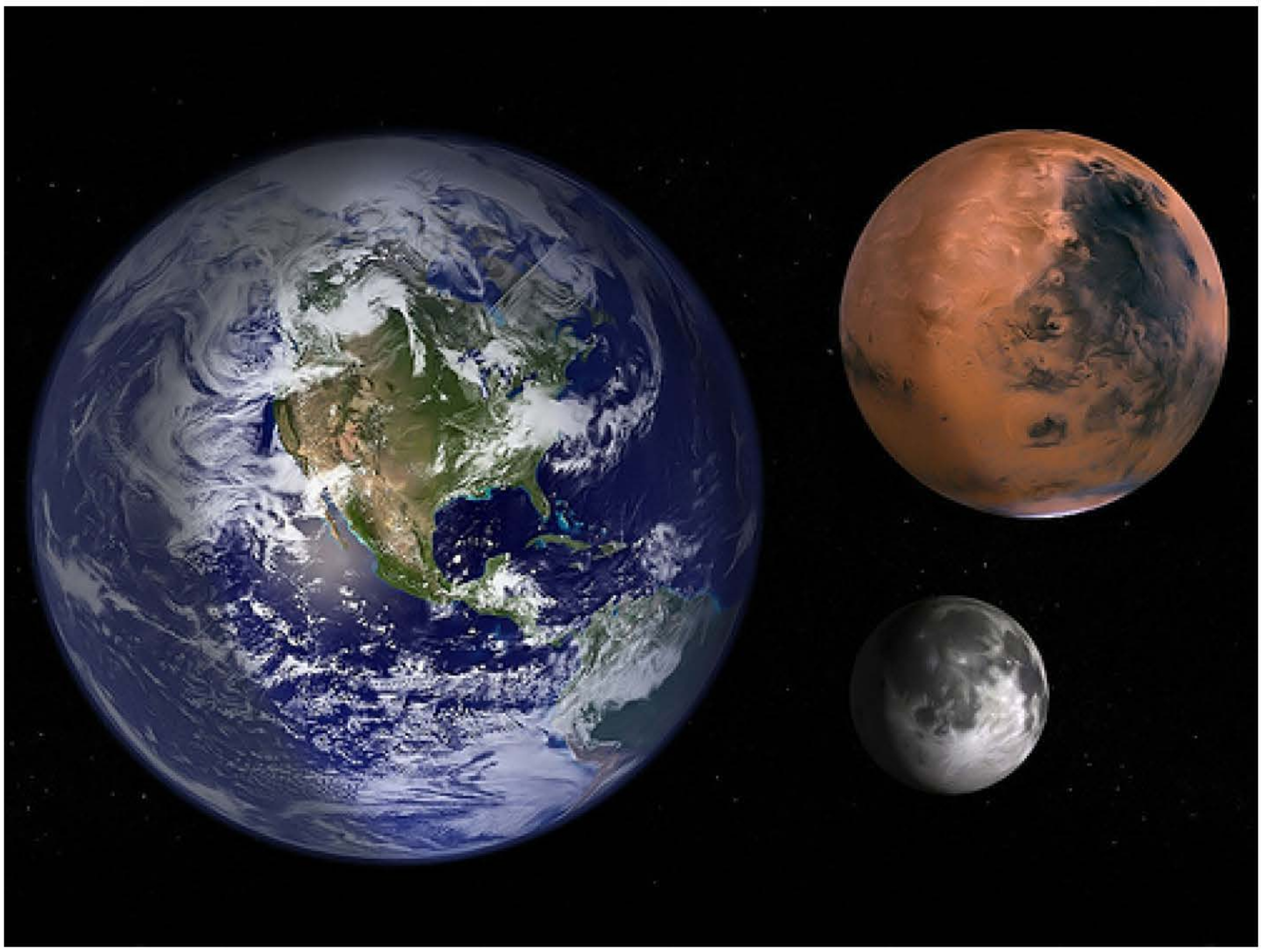
HLPA-1377MA003-11-06: Heavy Lift and Propulsion Technology BAA Final Report

Contract Number: NNM11AA10C 06/03/2011



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HLPA-1377MA003-11-06: Heavy Lift and Propulsion Technology BAA Final Report

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FOREWORD

This report is prepared and submitted in accordance with Data Requirement Document 1377MA-003 of the Heavy Lift and Propulsion Technology Systems Analysis and Trade Study Data Procurement Document, DPD 1377, dated November 22, 2010, to Contract NNM11AA10C.

1.0 EXECUTIVE SUMMARY

Lockheed Martin appreciated the opportunity to support NASA through their Heavy Lift Propulsion Trade (HLPT) Systems Analysis and Trade Study (SATS) Broad Agency Announcement (BAA). An outstanding interchange occurred, which provided meaningful and insightful information and data that will enable NASA to make informed decisions as we move forward together in our nation's human space exploration of Low Earth Orbit (LEO), Beyond Earth Orbit (BEO), our Moon and ultimately Mars.

This Final Report documents the approach followed to assess the critical areas required to establish the foundation for defining and assessing a "Building Block" or evolutionary path for the definition of the Heavy Lift Launch Vehicle (HLLV) configuration. We began by initially reviewing, ranking and assessing the NASA provided Figures of Merit (FOMs), Ground Rules & Assumptions (GR&As) and Goals. The results of this assessment showed that Affordability, Safety & Reliability and Schedule are the three key FOMs. The NASA provided FOMs and GR&A were found to be adequate and LM did not recommend any additions or changes.

The next step in the process was an assessment of the NASA provided Design Reference Missions (DRMs), which were reviewed and determined to be similar to LM developed DRMs. We then began an assessment of HLLV configurations against the FOMs, GR&A, Goals and DRMs. We leveraged previous HLLV studies that traded:

- 33 ft. and 27.6 ft. dia. and nominal (External Tank) and stretch length Core Stage
- LO_2/LH_2 (RS-68 and RS-25) and LO_2/RP-1 Core Stage engines
- LO_2/LH_2, (J-2X, SSME and XX-100), LO_2/RP-1 and LO_2/CH_4 2nd Stage engines
- Dedicated In-Space propulsion stages as a part of the payload
- 4 and 5 segment steel case/ PBAN boosters and 5 segment Advanced Composite Boosters (ACBs) with HTPB and LO_2/LH_2 (RS-68 and RS-25) and LO_2/RP-1 boosters.

Shuttle-derived HLLV configurations are an optimal solution for satisfying the FOMs, GR&A, Goals and DRMs provided by NASA. These configurations are based on a "Building Block" approach which can be evolved from an uncrewed test vehicle to HLLV configurations that can provide 64.5 mT to 135 mT to LEO. These configurations are 27.6 foot in diameter with nominal length and leverage existing Shuttle ET designs, certifications, tooling and facilities. The configurations also leverage Shuttle RS-25 engines for both the Core Stage and 2nd Stage and 4 and eventually 5 segment boosters. This approach achieves Affordability, reduces Program Risk and achieves schedule milestones, such as initial flight in 2016.

33 foot and stretch length Core Stage added significant cost and schedule and were not required to meet the requirements. LO_2/RP-1 introduced significant cost, risk and schedule impacts with little performance improvement. Use of LO_2/LH_2 for the 2nd Stage engine was logical with its high I_{sp} and the commonality of the Core Stage and 2nd Stage engines also became attractive. Multiple stage launch vehicles are very effective in providing the greatest performance. As the BAA progressed, developing a 2nd Stage configuration for optimal ascent and In-Space capability became difficult to achieve as the Stage would be large and burn half of its propellant

during ascent, making loiter a difficult design environment, especially for deep space missions. It became evident that a better solution is to design the 2nd Stage for ascent only and have a dedicated In-Space Stage be a part of the payload. This approach also provides affordability as it stretches out the near-term need for a 2nd Stage. This allows the 2nd Stage development to be spread over a greater period of time, which smoothes the funding profile requirements.

4 segment steel case booster HLLV configurations can put over 80 mT into LEO and Advanced Composite Booster (ACB) can achieve 135 mT to LEO. Liquid boosters were assessed and determined to have significant Affordability issues due to Design Development Test and Evaluation (DDT&E) costs or recurring costs due to the number of engines required and liquid boosters also increase the possibilities of launch scrubs and delays, which also impact Affordability.

Affordability discusses program cost associated with a 135 mT LEO HLLV. The Life Cycle Cost (LCC) addresses DDT&E, Program Management (PM) and Systems Engineering and Integration (SE&I), Launch Site activation, as well as the cost associated with production. This is based on 2 uncrewed test flights 2 crewed test flights and 3 crewed flights from the period of 2016 thru 2022. These costs are in the $12.1B to $14.5B range in 2011 dollars. These numbers are based on traditional LM program data and data from several studies of similar type vehicles. These numbers do not reflect potential savings in NASA over-site/insight as demonstrated on the Multi-Purpose Crew Vehicle (MPCV) program, which could potentially yield 10 to 15% savings. The manufacturing flow was assessed and an overall reduction of workstations (lean manufacturing approach) could be accomplished over the heritage ET manufacturing flow. This reduction in workstations and required facilities affects Affordability by reducing operations and maintenance (O&M) costs at the Michoud Assembly Facility (MAF). Commonality, System Engineering, KSC Ground and Launch Operations, and extensibility also provide ways of improving Affordability and reducing Program Risk.

No technical Capability Gaps were identified for configurations with DRMs in LEO, Lunar or several BEO DRMs. A Mars DRM identified the need to develop cryogenic fluid management due to mission duration. Non-technical Capability Gaps (i.e. reduced funding, schedule slide and loss of resources due to these gaps) require innovative ConOps and business models.

Innovations are proposed in several areas (i.e. avionics, TVC, alternative heat shield materials and manufacturing processes). Virtual manufacturing and digital inspection are highlighted as methods of offering additional Affordability and reducing Program Risk through leaner manufacturing approaches. A list of additional innovations that are not large contributors but which should be considered for the HLLV program is included. These innovations are based on lessons learned from the Shuttle ET and MPCV production at MAF.

A discussion of Incremental Testing and In-Space Demonstration and Certification is provided, with lessons learned from the Shuttle ET certification and testing programs. A detailed discussion on the Main Propulsion System (MPS) testing is provided with a testing methodology that eliminates the need for a Main Propulsion Test Article (MPTA). This approach reduces cost and provides the necessary confidence and certification of the MPS design and systems.

A compliance matrix to the original BAA Statement of Work (SOW) and a list of recommendations for further study and actions are provided as Appendix D.

2.0 INTRODUCTION

The LM effort on the NASA HLPT SATS BAA focused on providing recommendations in the area of configurations with an eye to Affordability, satisfying the FOMs, GR&A, Goals and DRMs and providing an HLLV by 2016.

The LM core team is located in Huntsville (HSV), AL and has been supporting HLLV studies since 2008 with IRAD funds. The IRAD activity complemented the BAA activities by providing generic vehicle studies/assessments that were enhanced and tailored to meet the more specific objectives of the HLPT BAA..

The LM HSV team consists of members previously on the Shuttle ET Project in the areas of Safety & Product Assurance, Management, Engineering Design Analysis, Systems Engineering, Manufacturing, Test and Checkout. The HSV team was supplemented by LM personnel at other sites for Mission, Performance, and LCC Analyses.

10% of the BAA contract value was allocated to local small businesses. These small businesses were selected because of their experience in the areas that were required for HLLV design trades. This effort allowed us to develop a working relationship that would be beneficial during the design and analysis phase of HLLV. These businesses were:

Small Business	**Task**
Ares Engineering	Configuration Probability Risk Assessment
KT Engineering	Thrust Vector Controllability Study
Victory Solution	System Engineering
Watring Engineering	Heat Shield Concept Study

3.0 FOMs, GR&A AND GOALS ASSESSMENT

3.1 Scope

The discussion in this section covers GR&A that were taken into consideration for this study and the approach that was used to determine the key FOMs and their weighting.

3.2 Approach

The LM team reviewed relevant documentation to capture applicable GR&A with respect to the development of a HLLV. The following list of NASA provided documentation containing HLLV GR&A was reviewed and assessed during this study:

- HLLV Architecture Study – ETO Launch Vehicle Team Ground Rules & Assumptions- Block 1 Vehicles
- HLLV Architecture Study – Safety and Reliability Ground Rules & Assumptions
- HLLV Architecture Study – Cost Team Preliminary Estimating WBS and Ground Rules & Assumptions
- HLLV Summary of NASA Studies – Overview to HLPT BAA Participants
- Kennedy Space Center Key Driving Constraints
- Preliminary Report Regarding NASA's Space Launch System and Multi-Purpose Crew Vehicle

In addition, applicable HLLV FOMs were captured and the following list of NASA provided documentation containing HLLV FOMs was reviewed and assessed during this study:

- Space Launch System Goals for use by HLLV Study Teams
- HLLV Summary of NASA Studies – Overview to HLPT BAA Participants
- Preliminary Report Regarding NASA's Space Launch System and Multi-Purpose Crew Vehicle

3.3 Discussion

3.3.1 FOMs

The following is a list of the FOMs, with definitions, used by LM and suggested for future use:

- Affordability – Focused on minimizing Life Cycle Cost.
    - Initial Human Flight (IHF) – The financial expenditures required to successfully complete the first flight of a crewed HLLV.
    - Recurring – The recurring costs required to operate and maintain a HLLV capability.
- Schedule – Focused on minimizing Cycle Times.
    - Initial Human Flight (IHF) – The time required to successfully complete the first flight of a crewed HLLV.

- Performance – Focused on meeting the required payload mass goals to accomplish a given mission while still meeting the Affordability and Schedule goals.
    - Payload – The payload required to be placed in LEO to achieve all mission goals.
- Safety and Reliability – Focused on maximizing HLLV safety and reliability while still meeting the Affordability and Schedule goals. Minimum safety and reliability requirements shall be met.
    - Loss of Crew (LOC) – The probability of a Loss of Crew for the HLLV.
    - Loss of Mission (LOM) – The probability of a Loss of Mission for the HLLV.
- Commonality – The ability to leverage hardware, software, workforce, etc. between the HLLV and other Government agencies, commercial partners and international partners, as well as, between the elements of the HLLV.
- Extensibility – The ability of the HLLV architecture to extend its capability through incremental upgrades to increase performance, enabling challenging BEO missions. In addition, the use of existing infrastructure such as MAF, MSFC, KSC, Ares I tooling/facilities and existing Constellation Program contracts.
- Operability – Focused on a Concept of Operations that meets all operability goals while still meeting the Affordability and Schedule goals.
    - Ground Operations – The resources required for ground operations in support of the HLLV.
    - Flight Operations – The resources and logistics required for flight operations in support of the HLLV.
    - Manufacturability – The resources and logistics required to manufacture the HLLV.
- Reusability – The ability of systems/subsystems/components of the HLLV to be reused for future missions.
- Industry Base – The impact of the HLLV design concept on the Industrial Base that would be needed to design, build and operate the HLLV.

3.3.2 GR&A and FOM Assessment and Recommendations

Once GR&A and FOMs were captured, an evaluation was performed to validate that they bounded and were within the scope of the study. The results of this evaluation did not show any gaps or inconsistencies in the GR&A or FOMs NASA had used in the HLLV trade studies and analysis tasks completed to date.

Following the previously described gap assessment, the recommended FOMs were ranked using the Pair-Wise Comparison tool, which is a part of the LM Rapid Affordability and CAIV Exploration (RACE) Toolkit.

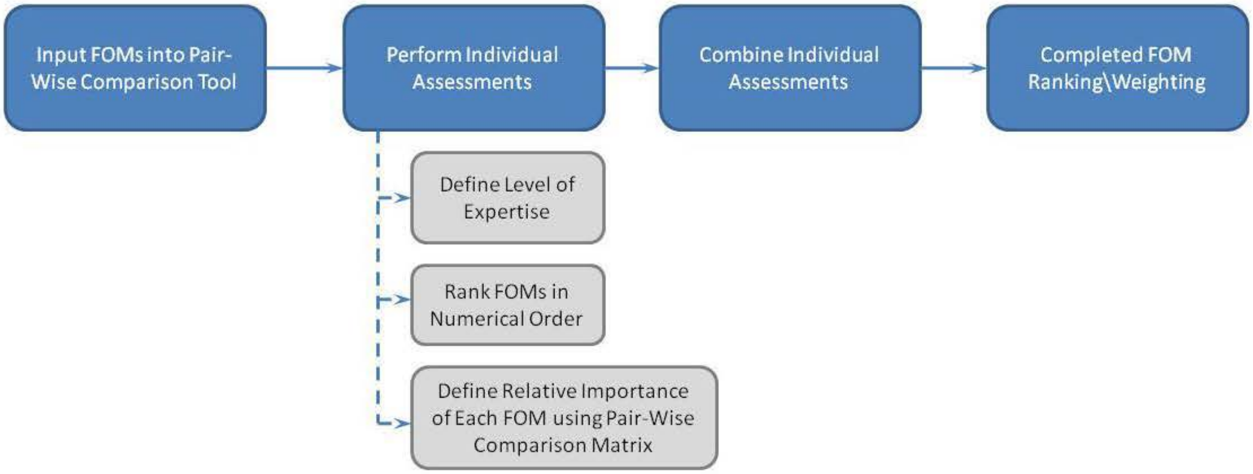

Figure 3-1: Figure of Merit Ranking/Weighting Process

The Pair-Wise Comparison tool provided a process to quantify the relative importance of the recommended HLLV FOMs based upon a diverse group of experts. The output of this process was a list of FOMs ranked and weighted in order of relative importance. Figure 3-1 describes the steps of this process:

The results of the FOMs Ranking are presented in Figures 3-2, 3-3 and 3-4:

	Overall Score			
	W/ Expertise		W/O Expertise	
Binned Metrics	Weight		Weight	
Affordability		**5.93**		**6.07**
├ Initial Human Flight (IHF)	1.03		1.09	
└ Recurring	1.53		1.49	
Schedule		**3.50**		**3.71**
└ Initial Human Flight (IHF)	1.00		1.00	
Performance		**3.16**		**3.14**
└ Payload	1.00		1.00	
Safety & Reliability		**4.27**		**4.34**
├ Loss of Crew (LOC)	2.20		2.28	
└ Loss of Mission (LOM)	1.00		1.00	
Commonality		**2.02**		**2.02**
Extensibility		**2.09**		**2.14**
Operability		**2.45**		**2.53**
├ Ground Operations	1.78		1.60	
├ Flight Operations	1.08		1.09	
└ Manufacturability	1.52		1.44	
Reusability		**1.13**		**1.13**
Industry Base		**1.37**		**1.37**

Figure 3-2: Figure of Merit Numerical Weighting Results

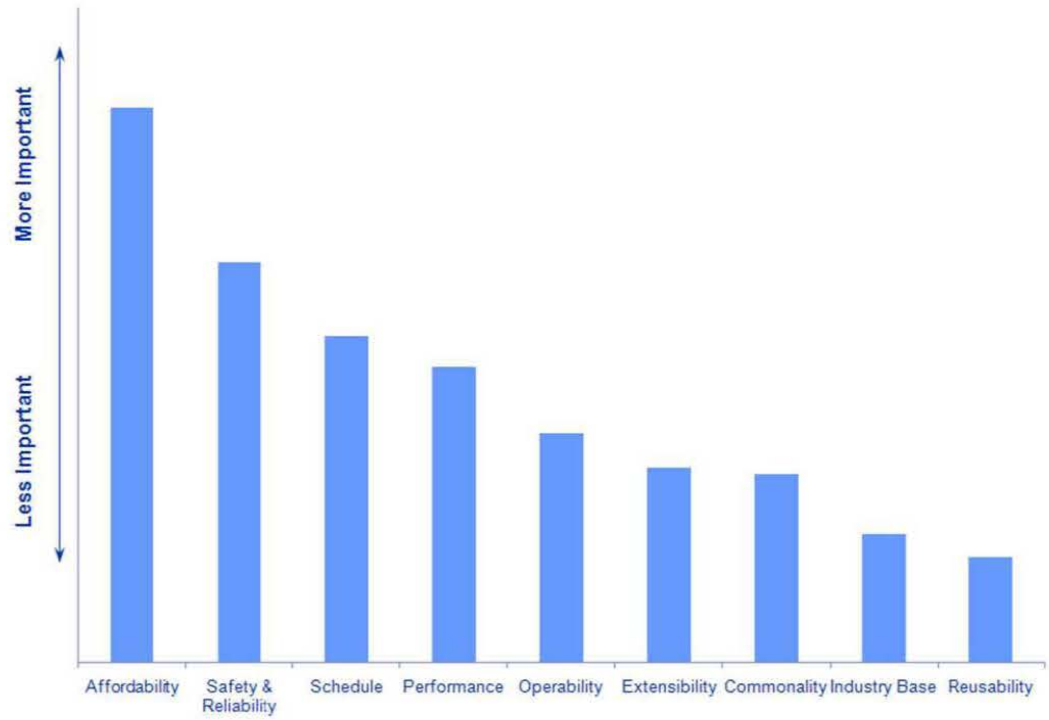

Figure 3-3: Top Tier Figures of Merit Relative Importance

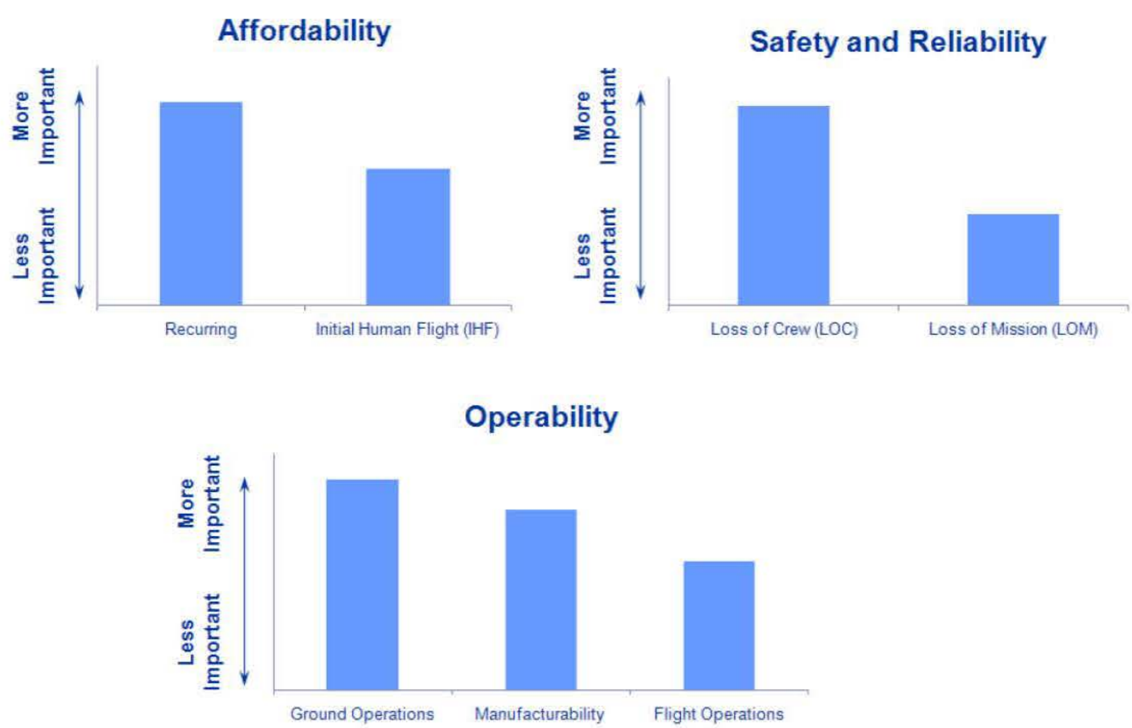

Figure 3-4: Sub-Tier Figures of Merit Relative Importance

The results, provided in Figure 3-4 above, show that the top three FOMs are:

1. Affordability
2. Safety and Reliability
3. Schedule

Performance was ranked No. 4 in this assessment, which is typically a major driver for Launch Vehicle designs. However, the LM team concluded that Affordability, Safety and Reliability and Schedule are the primary FOMs. An evolutionary "Building Block" approach can satisfy these FOMs and enable additional performance to be added later if required.

3.4 Summary

After a review of the available data, it was determined that the NASA provided FOMs and GR&A are complete and no gaps were identified. The recommended FOMs previously discussed in this section, regardless of GR&A, are sufficient for defining the criteria necessary to perform HLLV trade studies. The weighting provided in this assessment is highly dependent on the definition of the GR&A and therefore, this weighting should be repeated for any studies that consider any additional, deleted or changed GR&A. For this assessment the GR&A focused on Affordability, which heavily influenced the FOMs weighting and the results reflected this.

4.0 DESIGN REFERENCE MISSION (DRM) ASSESSMENT

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6.0 AFFORDABILITY

6.1 Introduction

This section discusses the Life Cycle Cost (LCC) of the proposed configurations, as well as other aspects of Affordability which include Operability, Extensibility, Commonality and Reusability.

6.2 Discussion

6.2.1 Life Cycle Cost

This section presents the life cycle cost (LCC) estimate for the 130 mT HLLV architecture. Section 6.2.1.2 provides an overview of the reference architecture and a detailed list of GR&A associated with the Life Cycle Cost Estimate (LCCE) estimate in Section 6.2.1.3. Section 6.2.1.2 also provides an overview of the cost estimating methodology used to develop the LCCE.

6.2.1.1 Approach

Cost Estimating Methodology

LM used a multitude of cost estimating approaches and data sources to develop the LCCE for HLLV. Figure 6-1 summarizes the various HLLV elements and specific cost estimating approaches utilized. We have consulted with element providers (i.e., boosters and liquid engines) and received inputs from ATK and PWR, respectively. For the Core Stage, 2nd Stage and shroud, we used NAFCOM, but calibrated the results to LM program historical data (i.e., ET, Atlas) and/or contractor inputs. Avionics was based on a combination of historical launch vehicle data, including our recent Ares 1 experience, as well as results from the 2005 Shuttle-derived Launch Vehicle (SDLV) study. Launch site activation and operations cost is based on Atlas experience, the SDLV study and engineering estimates developed by LM KSC personnel. SE&I/PM cost estimating relationship (CER) $-to-$ factors were derived from the SDLV study.

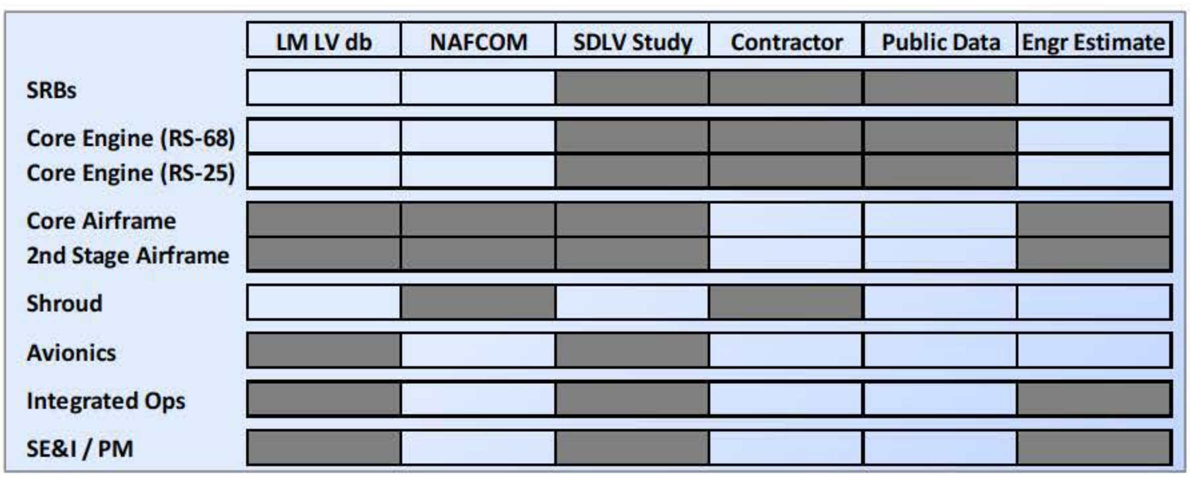
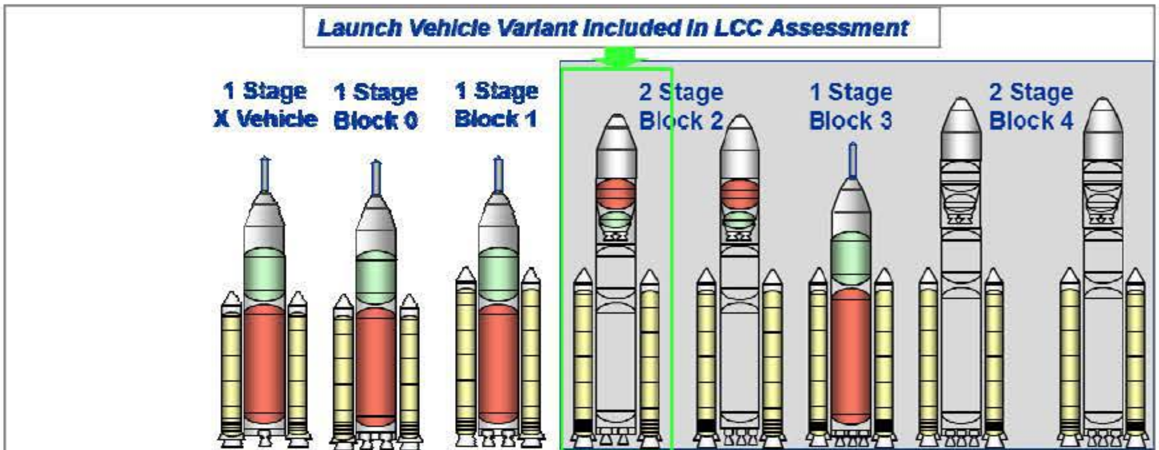


Figure 6-1: Cost Estimating Methodology per Element

Ground Rules & Assumptions

The LCCE is based on the Block 2 vehicle shown in Figure 6-2 below. The launch vehicle consists of the following elements:

- 5-segment SRBs
- 27.6' diameter ET-derived LO_2/LH_2 Core Stage
- 27.6' diameter LO_2/LH_2 2nd Stage
- RS-25 (SSME) used for both Core Stage (4) and 2nd Stage (1)
- 10m diameter PLF



Configuration	X–Nom Core	1–Nom Core	2–Nom Core	8c/8g–Nom Core	8e/8f–Nom Core	4–Stretch Core	8l–Stretch Core	8h–Stretch Core
Boosters	4 seg Steel/ PBAN	4 seg Steel/ PBAN	5 seg Steel/ PBAN	5 seg Steel/ PBAN or 5 seg Comp/ HTPB	5 seg Steel/ PBAN or 5 seg Comp/ HTPB	5 seg Steel/ PBAN	5 seg Comp/ HTPB	5 seg Comp/ HTPB
Core (27.6 ft dia.)	1 RS-68	4 SSME	4 SSME	4 SSME @ 113%	5 SSME	4 SSME	4 SSME @ 113%	5 SSME
2nd Stage	NA	NA	NA	1 SSME	1 SSME	NA	1 SSME	1 SSME
30x130nmi @28.5° LEO (mT)	26.0	64.5	82.5	124.2 135.1	132.9 142.3	104.5	148.4	160.2
130x130nmi @28.5° TI for C3=3 km^2/s^2	NA	NA	NA	42.6 45.9	44.8 48.5	NA	51.9	56.0
ATP 2011	NET 2013	NET 2015	NET 2016	NET 2016	NET 2018	NET 2018	NET 2018	NET 2018
Max Q (psf) Max g	771 3.86	676 3.90	791 3.99	800/793 3.07/2.94	798/797 3.24/3.06	800 3.98	785 2.62	786 2.83
Mission	LEO / Lunar			LEO / Lunar / NEO / Mars		LEO / Lunar	LEO / Lunar / NEO / Mars	

Figure 6-2: Reference HLLV Architecture (130 mT)

The LCCE mission model, shown in Figure 6-3, assumes 1 uncrewed test flight in 2016, followed by 1 flight/ year through 2022 (3 crewed, 3 uncrewed)- total of 6 operational flights.

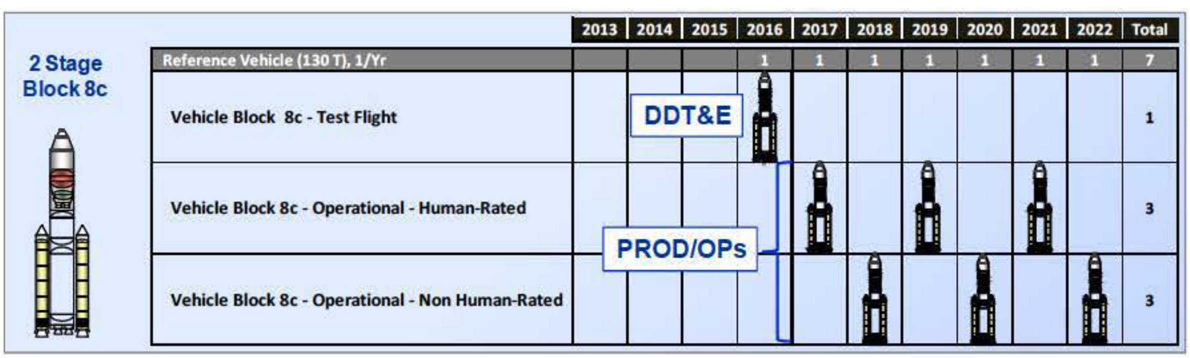


Figure 6-3: Reference Vehicle (130 mT) Mission Model

The following is a detailed list of additional GR&A used in the development of this Life Cycle Cost estimate.

1. LCCE is intended for architecture / trade study purposes only and should not be construed as an offer by Lockheed Martin.
2. LCCE includes DDT&E, Production and Operations phases.
3. Contractor costs only.
    - Includes applicable contractor burdens (through fee)
    - Excludes NASA Oversight.
4. LCC Analysis period of performance: ATP (10/2011) through 12/2022.
5. Costs provided in 2011 $'s and TY$'s
6. Launch Vehicle hardware quantities as indicated in Figure 6-4

	2013	2014	2015	2016	2017	2018	2019	2020	2021	2022	
Reference Vehicle (130T) @ 1/YR											
Element	DDT&E			Production							Total
Bstr - 4 Seg											0
Bstr - 5 Seg				2.0	2.0	2.0	2.0	2.0	2.0	2.0	14
RS-68											0
SSME				5.0	5.0	5.0	5.0	5.0	5.0	5.0	35
SSME - Core				4.0	4.0	4.0	4.0	4.0	4.0	4.0	28
SSME - 2nd Stage				1.0	1.0	1.0	1.0	1.0	1.0	1.0	7
Core				1.0	1.0	1.0	1.0	1.0	1.0	1.0	7
2nd Stg				1.0	1.0	1.0	1.0	1.0	1.0	1.0	7
Avionics				1.0	1.0	1.0	1.0	1.0	1.0	1.0	7
LVHM				0.7	0.7	0.7	0.7	0.7	0.7		4
Shroud						0.6	0.6	0.6	0.6	0.6	3

New Build Residual Asset Wtd Average

Figure 6-4: Time-phased Launch Vehicle Hardware Quantities

7. Assumes use of 12 SSME residual assets from STS program.
8. No dedicated industrial base protection funds included.
    - i.e., HLLV peculiar hardware and operations cost only.
9. Key program milestones assumptions
    - FSD ATP 2011
    - PDR 2012
    - CDR 2013
    - Test Flight 2016
    - IOC 2017

10. Launch hardware DDT&E and Production Costs derived using (as applicable):
    - LM historical cost analysis data base
    - NASA Air Force Cost Model (NAFCOM)
    - SDLV industry team study (2005)
    - Element contractor supplied Data – i.e., boosters, liquid rocket engines
    - Publically available industry data
    - Engineering estimates
11. Site Activation Costs (NRE) Based on Lockheed Martin Experience on Atlas / Titan Programs.
12. Operations Costs (REC) Based on SDLV Study and LM Atlas/Titan Experience
13. NASA new start inflation indices used for all program phases – i.e., DDT&E, Production and Operations
14. Life Cycle Cost risk range provided.
15. DDT&E Cost Includes:
    - Design and Development activities for the Launch Vehicle segment.
    - Ground and Infrastructure / Operations capability development activities.
    - Operations facilities and activation related to launch vehicle processing and checkout.
    - New or modified facilities required to produce and process HLLV system elements.
    - Checkout of the ground processing and flight operations systems.
    - Integration activities associated with test hardware & test flights.
    - Software development and verification.
    - Flight test cost including flight test article (1) and associated test operations.
    - Non-recurring production facilities, tooling and process development.
16. Recurring annual Operations Cost includes:Launch (ground) processing and operations
    - Flight and Crew operations
    - Replenishment spares
    - Facilities refurbishment and maintenance
    - Logistics
    - Depot maintenance
    - Sustaining engineering
17. Cost of unreliability (i.e., mission loss, failure investigation, stand down time, etc.) is excluded.

6.2.1.2 Discussion

As shown in Figure 6-5 below, the nominal life cycle cost estimate (LCCE) for the reference architecture is \$12.9 billion (B) in constant fiscal year 2011\$'s (FY11 \$'s). Design, Development, Test & Evaluation (DDT&E) – (\$8.3B, 2011\$'s) and Production/ Operations – (\$4.6B, 2011\$'s) constitute approximately 65% and 35%, respectively, of the total LCCE.

A cost risk range is also provided, based on the relative risk associated with each phase: 1) -5% to + 15%, or \$7.9B to \$9.5B (2011 \$'s) for the DDT&E phase and 2) -10% to + 10%, or \$4.2B to \$5.0B (2011 \$'s) for the Production/Operations phase. This equates to an overall LCCE range of -7% to +13%, or \$12.1B to \$14.5B (2011 \$'s)

Each phase has been further broken down by segment for each program phase and for the total LCCE. As indicated the Launch Vehicle segment constitutes the largest portion of the cost, making up 54%, 80% and 64% of DDT&E, Production/Operations and Total LCC, respectively. This segment includes all DDT&E and Production activities associated with the Launch Vehicle hardware elements – i.e., boosters, RS-25 engines, core vehicle, 2nd stage, avionics and shroud. The magnitude of the Launch Vehicle segment within the Production/Operations phase is driven by the operational mission quantity of 6 flights. Note that the launch vehicle hardware associated specifically with the Test Flight in 2016 is excluded from the Launch Vehicle segment – i.e., it is included under Test Flight.

Launch Site/Launch Operations makes up 19%, 11% and 16% of DDT&E, Production/Operations and Total LCC, respectively. It includes: 1) launch site infrastructure and site activation during the DDT&E phase and 2) the ongoing launch processing, operations, and maintenance associated with the 6 operational missions from 2017 through 2022.

The Test Flight (Launch Vehicle Hardware + Launch Operations) cost is \$0.6B (2011 \$'s), or approximately 7% of DDT&E. This segment includes both the launch vehicle hardware and launch operations required for the test flight in 2016.

System Engineering and Integration / Program Management (SE&I/PM) is approximately 27% of the base cost (i.e., Launch Vehicle + Launch Site + Test Flight) for DDT&E and 10% of the base cost (i.e., Launch Vehicle + Launch Operations) for Production / Operations.

Reference Vehicle (130T)

2 5-Seg SC PBAN SRBs
27.6'D Core, 4 SSMEs @ 113%
2nd Stage, 1 SSME

2016 - 1 Uncrewed Test Flight
2017 - 2022
3 Operational Crewed Flights
3 Operational Uncrewed Flights

Launch Vehicle DDT&E Funding
~ $6.0B to $7.2B Total (2011 – 2017)
~ $0.9B to $1.0B per Year (2011 $'s)

SLS DDT&E (NRE) - $B's

Description	2011 $B's
Launch Vehicle DDT&E (NRE)	
Launch Vehicle	$4.5
Test Flight - LV Hardware	$0.5
SE&I / PM	$1.3
Launch Vehicle DDT&E (NRE)	**$6.3**
Launch Vehicle DDT&E Range (-5% to +15%)	**$6.0 to $7.2**
Launch Site / Test Flight Ops DDT&E (NRE)	
Launch Site	$1.5
Test Flight - Launch Ops	$0.1
SE&I / PM	$0.4
Launch Site DDT&E (NRE)	**$2.0**
Launch Site DDT&E Range (-5% to +15%)	**$1.9 to $2.3**
Total DDT&E (NRE)	**$8.3**
DDT&E Range (-5% to +15%)	**$7.9 to $9.5**

SLS Production / Operations (REC) - $B's

Description	2011 $B's
Production / Operations (REC)	
Launch Vehicle	$3.7
Launch Operations	$0.5
SE&I / PM	$0.4
Total Production / Operations (REC)	**$4.6**
Prod / Ops Range (-10% to +10%)	**$4.2 to $5.0**

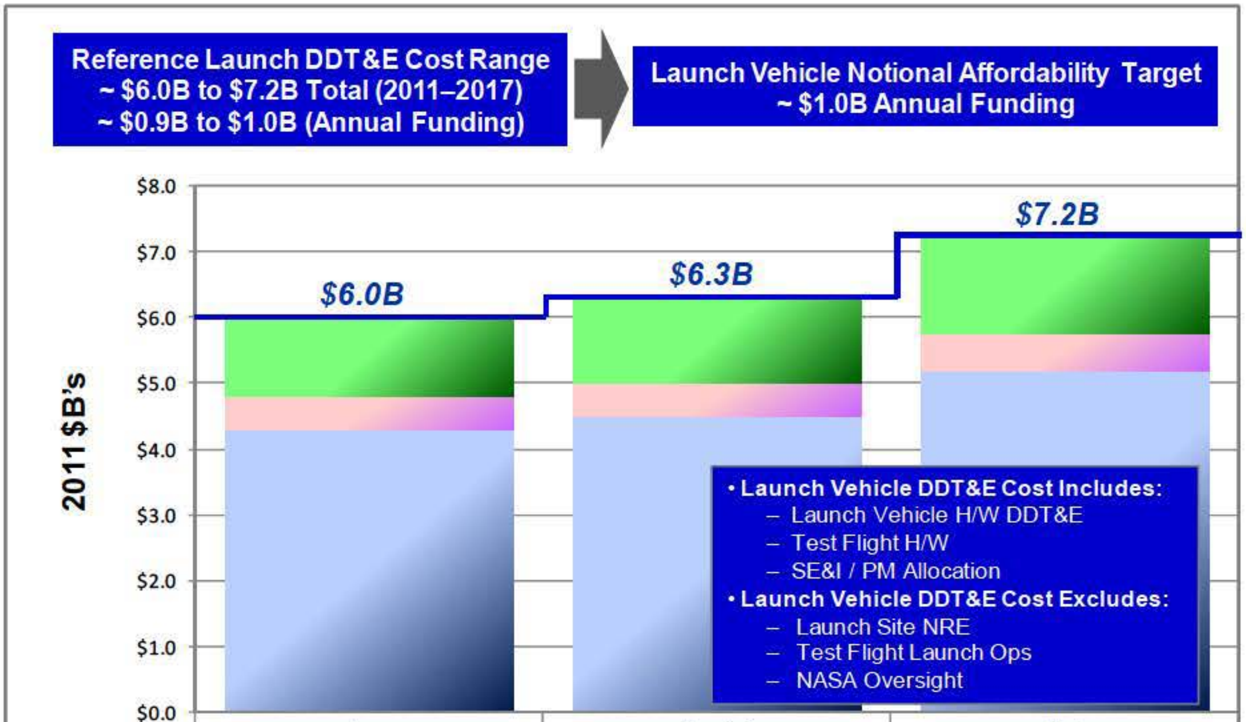
SLS Life Cycle Cost - $B's

Description	2011 $B's
Life Cycle Cost = DDT&E + Production / Ops	
Launch Vehicle	$8.2
Test Flight - LV Hardware + Launch Ops	$0.6
Launch Site / Launch Operations	$2.0
SE&I / PM	$2.1
Total Life Cycle Cost	**$12.9**
LCC Range (-7% to +13%)	**$12.1 to $14.5**

Figure 6-5: SLS Life Cycle Cost by Program Phase and Element

As shown above in Figure 6-5, the Launch Vehicle portion of the DDT&E cost estimate is $6.3B, bounded by a range of $6.0B to $7.2B, which equates to approximately $0.9B to $1.0B per year.

As indicated in Figure 6-6, the Launch Vehicle DDT&E cost of $6.3B includes only the Launch Vehicle development, Test Flight hardware and associated SE&I/PM and excludes Launch Site NRE, Test Flight Launch Ops and NASA Oversight. The annual funding estimate range of $0.9B to $1.0B, then, falls below the notional annual SLS affordability target of $1.0B.



	Low	Nominal	High
SE&I / PM	$1.2	$1.3	$1.5
Test Flight	$0.5	$0.5	$0.6
Launch Vehicle	$4.3	$4.5	$5.2
Total	$6.0	$6.3	$7.2

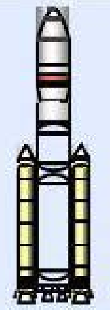
Figure 6-6: Reference Launch Vehicle Cost is Within Notional Annual Funding Limits

Figure 6-7 provides a summary of the SLS average unit cost for a human-rated and nonhuman-rated mission. A human-rated mission excludes the shroud and augments the basic launch vehicle avionics suite with additional launch vehicle health management (LVHM) avionics. Likewise, the nonhuman-rated mission includes the shroud but excludes the LVHM avionics. As indicated, given the addition and deletion of hardware for the 2 respective missions, the recurring unit cost for each mission type approximately converges on the same value. The recurring unit cost estimate for the launch vehicle hardware is $0.61B and $0.60B for a human-rated and nonhuman-rated mission, respectively. There is no measurable difference in the recurring Launch Operations ($0.09B) or SE&I/PM ($0.07B) for either type of mission. Thus, the total recurring unit cost for a human-rated mission is $0.77B (2011$'s). The total recurring unit cost for a nonhuman-rated mission is $0.76B (2011$'s). The extended Production/Operations cost, then, is $4.6B.

Reference Vehicle (130T)

2 5-SegSC PBAN SRBs
27.6'D Core, 4 SSMEs @ 113%
2nd Stage, 1 SSME

2016 - 1 Uncrewed Test Flight
2017 - 2022
3 Operational Crewed Flights
3 Operational Uncrewed Flights



Production / Operations (REC) Cost - $B's	
HumanRated REC Cost - $B's	
Description	**2011 $B's**
Launch Vehicle	$0.61
Launch Operations	$0.09
SE& I / PM	$0.07
HumanRated REC Unit Cost	**$0.77**
Cost Range (-10% to +10%)	**$0.69 to $0.89**
x Qty	**3**
Total REC HumatedRated Cost	**$2.3**
REC HumanRated Cost Range (-10% to +10%)	**$2.1 to $2.5**
Non-HumanRated REC Cost - $B's	
Description	**2011 $B's**
Launch Vehicle	$0.60
Launch Operations	$0.09
SE& I / PM	$0.07
HumanRated REC Unit Cost	**$0.76**
Cost Range (-10% to +10%)	**$0.68 to $0.84**
x Qty	**3**
Total REC HumatedRated Cost	**$2.3**
REC Non- HumanRated Cost Range (-10% to +10%)	**$2.1 to $2.5**
Total REC Cost	**$4.6**
REC Cost Range (-10% to +10%)	**$4.2 to $5.0**

Figure 6-7: SLS Recurring Unit Cost ($B's) LCC Summary

6.2.1.3 Life Cycle Cost Summary

In conclusion, the LCC data provided in this report demonstrates that the reference 130 mT Shuttle-Derived configuration for HLLV can be achieved within NASA budget guidelines – i.e., $1B per year. It should be noted that the values presented here represent a historically-based approach with respect to government acquisition and NASA insight/oversight. Lockheed Martin believes that a "minimal oversight" approach would yield at least 10% to 15% reduction in the overall LCC.

The LCCE values are based on leveraging much of the proven Shuttle design and infrastructure – e.g., SSME, SRB, ET. Use of this type of configuration reduces DDT&E costs thru leveraging

Shuttle design and a resulting reduction in certification testing. For example, utilizing an ET heritage structural design for the core stage would result in a reduced (i.e., low cost) approach to structural testing. A "lean" certification program for the main propulsion system (MPS) avoids the cost of a dedicated main propulsion test article (MPTA). Likewise, utilizing existing SRB and SSME designs will reduce overall DDT&E costs – i.e., vs. development of new liquid engines and/or liquid boosters.

6.2.2 Operability

As part of this BAA final report, two aspects of Operability are discussed these approaches to Operability would have an impact on affordability.

6.2.2.1 KSC Operations

6.2.2.1.1 Introduction

Lockheed Martin (LM) Kennedy Space Center (KSC) Operations personnel are supporting the NASA HLLV Analysis and Trade Study being performed under a BAA at Marshall Space Flight Center (MSFC).

The LM HLPT Study BAA focuses on HLLV configurations that meet NASA's FOMs, GR&A, Goals and Design Reference Mission (DRM) Models. The KSC Ops' focus has been on prospective launch site configurations, architectures and concept of operations (CONOPS) to support the HLLV.

Some background and history of this assessment is based on the KSC Ops personnel extensive experience in launch site requirements definition, design, development, construction, activation, testing, maintenance and operation, dating from the advent of the Space Shuttle Program through the current program-of-record. Some examples of the programs are listed below:

- LC-39 pads A & B and MLPs-1, 2 & 3 at KSC
- X-33 Single-Stage-To-Orbit Launch Site at Edwards Air Force Base (EAFB), CA
- Atlas V Evolved Expendable Launch Vehicle (EELV) Launch Site at Launch Complex 41 (LC-41), Cape Canaveral Air Force Station (CCAFS), FL
- West Coast Atlas V Evolved Expendable Launch Vehicle (EELV) Launch Site at Space Launch
- Complex 3 East (SLC-3E), Vandenberg Air Force Base (VAFB), CA
- MPCV Manufacturing Facility - Operations & Checkout (O&C) Building, KSC, FL
- Numerous others

The KSC Ops' intent is to demonstrate that it is both economically feasible and technically possible to build a HLLV launch site within the time and cost constraints identified by the NASA customer.

A key assumption of this assessment is for a first launch as early as 2013. The intent is to utilize launch site architectures, features & vehicle processing techniques from the above listed programs that are most applicable to HLLV.

6.2.2.1.2 Scope

The proposed Launch Site ConOps describes how LC-39A could be modified to efficiently and affordably process and launch the HLLV. Launch site modifications would support the potential early test flights (4 each) using baseline HLLV configuration, See Figure 5-1. The follow-on flights using larger 2-stage, Block 1 "evolved" HLLV.

6.2.2.1.3 Approach

The KSC Ops recommended HLLV launch site ConOps plan details are described below.

The GR&A used for this assessment are based on the HLLV configuration of Shuttle-derived components, including an ET-based Core Stage with an aft engine section (SSMEs or RS-68s). Vehicle Core will be processed horizontally at an existing KSC facility, probably the Vehicle Assembly Building (VAB) or Orbiter Processing Facility (OPF).

The HLLV also utilizes STS Solid Rocket Boosters (SRBs) consisting of 4 each Solid Rocket Motor (SRM) segments. The SRM segments will be processed in same manner and using the same facilities -- notably the Rotating Processing & Surge Facility (RPSF) -- as STS.

The SRM segments and Vehicle Core will be transported individually to LC-39A and assembled at the launch site, using Stack-On-Pad methodology. The "payload" components (MPCV/Orbital Test Flight-1 (OFT-1) assembly) will be assembled off-site, transported to LC-39A and integrated with the HLLV at the launch site.

The main emphasis of this assessment is the re-use of existing KSC facilities, including the Fixed Service Structure (FSS), launch platform support pedestals, LC-39A propellant, commodity & vehicle servicing systems and SRB processing facilities. *Note: SRB recovery and re-use is assumed to not be baselined for the HLLV program for this assessment.* This concept is readily adaptable to additional launch configurations (launch platform can be removed to allow access by other launch vehicles). It provides complete weather protection up to day of launch. There is an added benefit of rapid launch anomaly turnaround (complete vehicle access available at launch site; no rollback required to fix problems)

Depicted in Figure 6-8 and 6-9 is a Block diagram and time-line of a functional flow showing how the major elements of the HLLV (SRMs, Vehicle Core, interstage adaptor, OFT-1 assembly) are integrated at the launch site, over a thirty three day period culminating in HLLV launch on the last day.

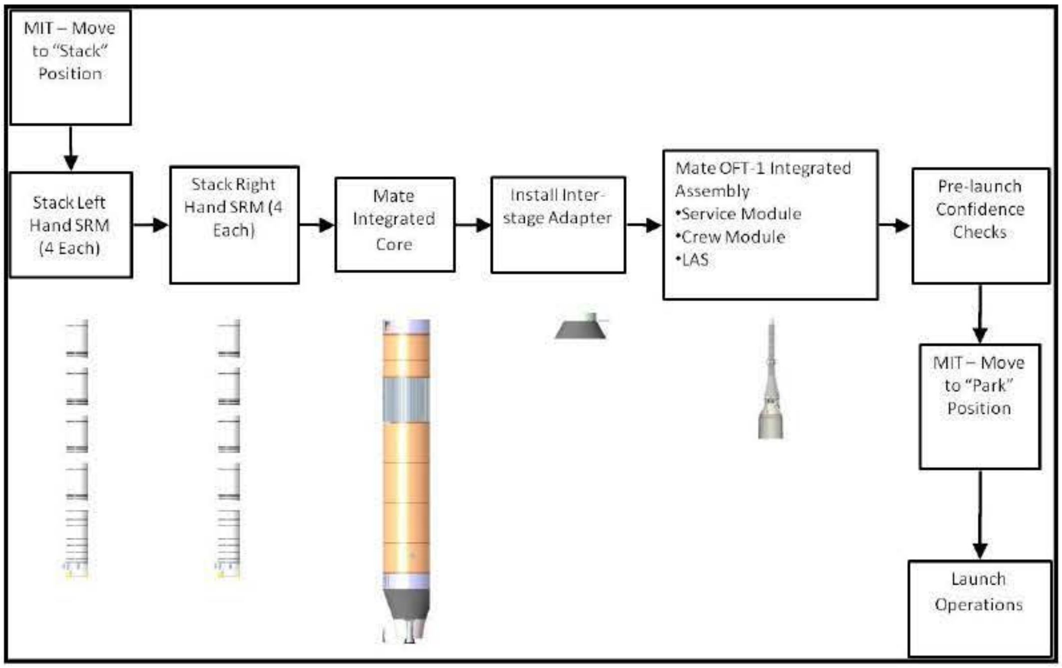


Figure 6-8: Block Diagram Functional Flow of Proposed Test Vehicle Integration

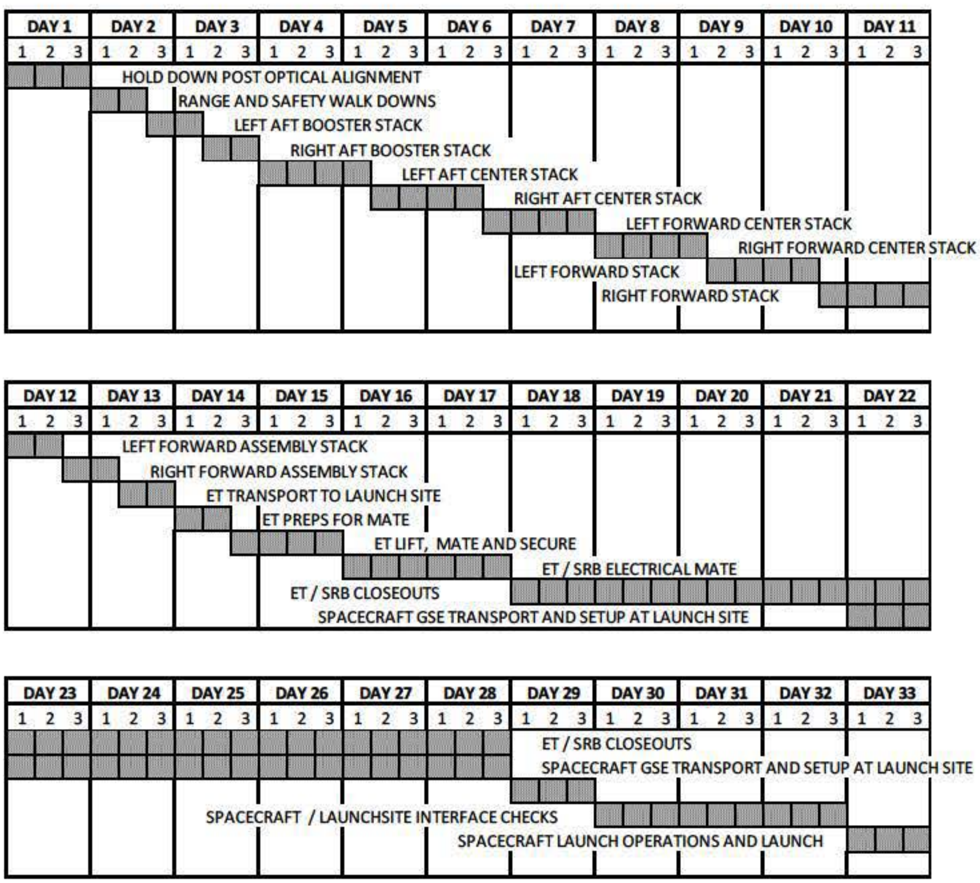


Figure 6-9: Time-line of Proposed Functional Flow of Proposed Test Vehicle

6.2.2.1.4 Discussion

6.2.2.1.4.1 Reconfigurable (or replaceable) Launch Platform (RLP)

This Platform will span the existing flame trench and be mounted on the 4 southernmost shuttle Mobile Launch Platform (MLP) support pedestals. The proposed RLP will be approximately 2/3 the length of the Shuttle MLP and will therefore not require the use of the northernmost support pedestals. The RLP will incorporate most of the functionality of the Shuttle MLP, including connection points for propellants, power, communication, HVAC, etc., as well as T-0 umbilicals for the HLLV MPS. The RLP is not intended to be a mobile launch platform, but will be removable to accommodate other launch vehicles & systems (RLP design will not preclude installation of other launcher configurations)

6.2.2.1.4.2 Mobile Integration Tower (MIT)

The tower in its extended position will also span the flame trench and cover the RLP and HLLV. The MIT will include a 200-ton bridge crane and cantilevered porch on the east side of the structure that will support stacking and assembly of the HLLV elements, including SRM segments, Core Stage, payload (and fairing if needed) and a 2nd Stage. The SRM segments are transported to LC-39A on existing transporters. The Vehicle Stage will be transported on new transporter, based on the current ET transporter, that is designed to allow Vehicle Core lift / break over using a single crane (similar to Atlas V). Elements will be lifted from the east side of the pad surface (adjacent to the flame trench) and stacked on the RLP.

In the extended position, the MIT will also provide weather protection for the HLLV. Retractable doors will be provided on both the east and west sides of the MIT to facilitate extension and retraction of the tower. Access to critical sections of the vehicle will be provided by moveable platforms within the MIT. In its retracted position the MIT will be parked behind the FSS. The MIT doors will be designed so that the tower can be retracted without disturbing umbilical connections between the FSS and HLLV. The MIT will move on trucks & rails (similar to the existing RSS). New bridge rails will need to be constructed to allow the MIT to span the flame trench.

6.2.2.1.4.3 Fixed Service Structure (FSS) Modifications

The FSS will need to be "stretched" to accommodate the additional height required for future HLLV Block 1 configurations. The existing umbilical arms (ET/IT & GO_2 Vent) will be re-utilized to the maximum extent possible.

6.2.2.1.4.4 Additional Modifications to LC-39A

The RSS and associated equipment (e.g., flame trench bridge rail) will need to be demolished. The North bridge and associated propellant piping will also need to be demolished. The new LO_2 and LH_2 propellant fill and vent lines will be installed and routed to the RLP connection points via new piping trenches on the pad surface. The propellant line RLP connection interfaces will be fully blast-protected by structural enclosures. Existing SRB / SSME flame deflectors may need to be relocated modified or replaced (depending on vehicle orientation on launch platform). Structural reinforcement on east pad surface will need to be added to accommodate SRM segments prior to stacking. Acoustic Suppression Water System (ASWS) modifications will be required.

The following figures illustrate of some of the features of the proposed Launch facility.

Figure 6-10 is an illustration of the launch site layout, showing the RLP installed on 4 of the existing STS launch platform support pedestals. In addition, the MIT is shown in the "Park" position (behind the existing FSS). Some dimensional data is also provided.

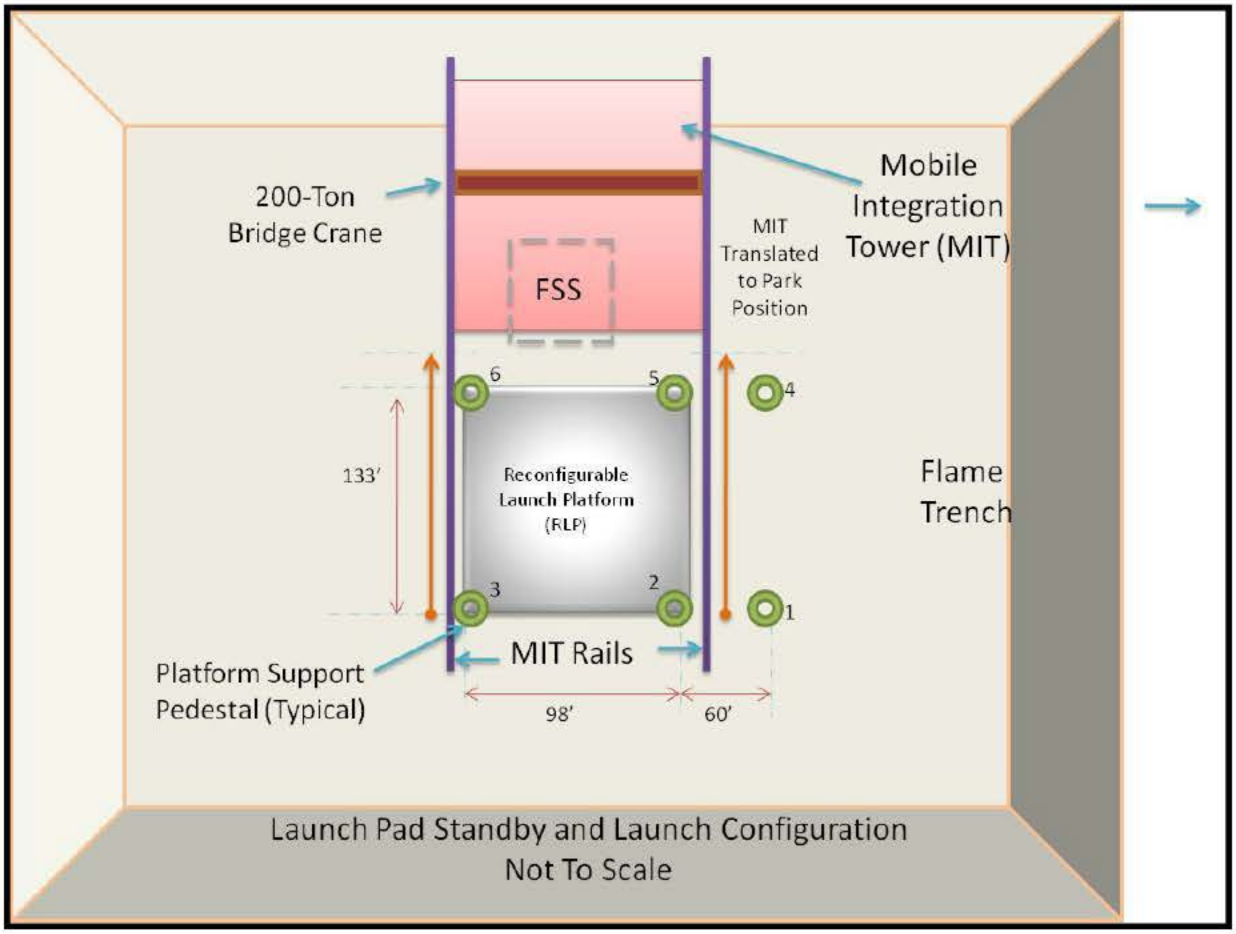


Figure 6-10: MIT Retracted over FSS with RLP on Pedestal

Figure 6-11 depicts the launch site showing the MIT translated to the "Stack" position, covering the RLP. It also shows the offload position on the east apron of the pad surface where the elements of the HLLV (SRM segments, Vehicle Core and OFT-1) will be removed from their transporters, lifted into the MIT and stacked on the RLP.

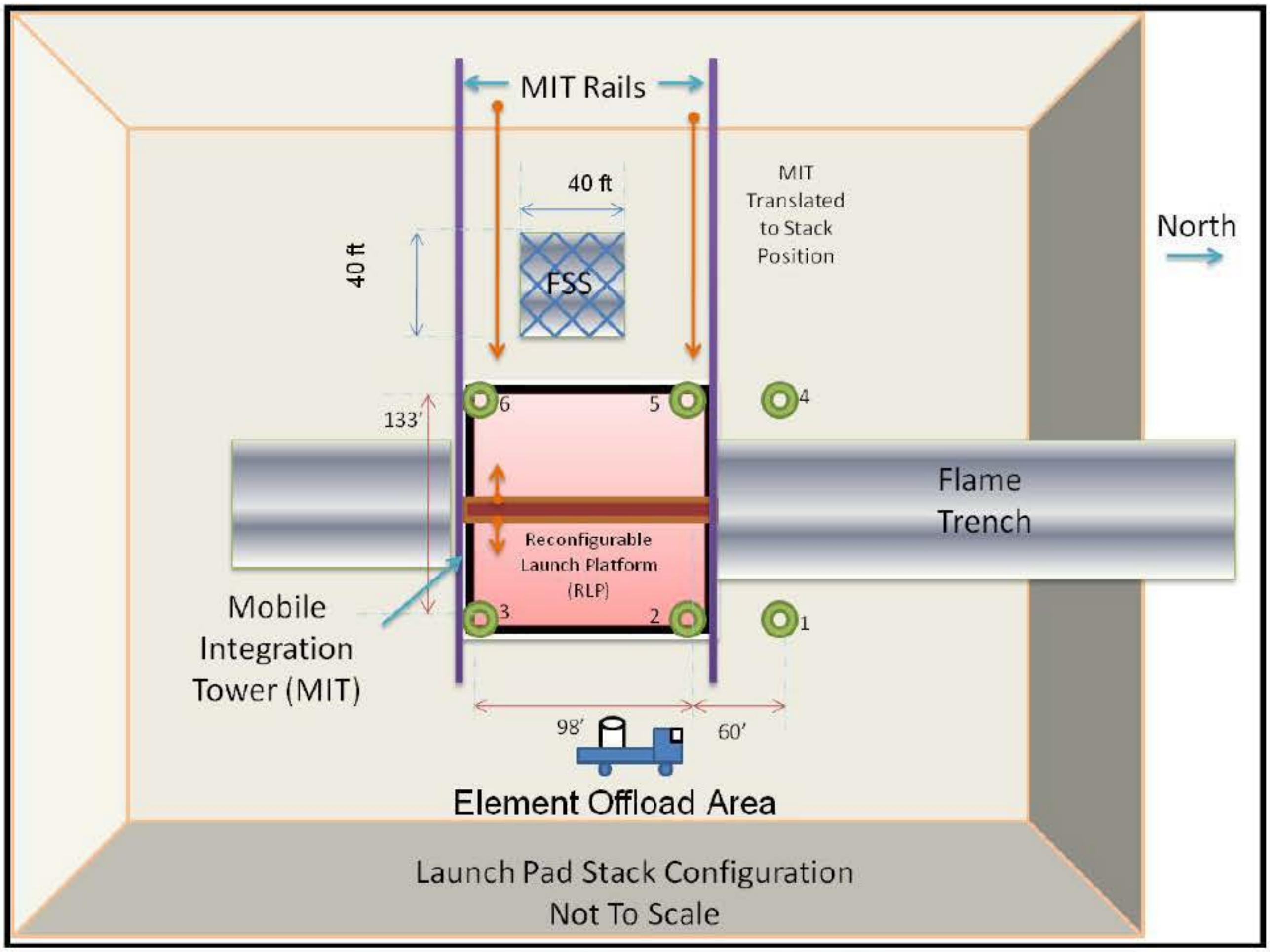


Figure 6-11: MIT Extended over the RLP

Figure 6-12 is an elevation view of the launch site looking east-to-west (into the MIT). Elevation dimensions of the MIT and a new 60' FSS extension are shown. Both the baseline and Block 1 HLLV configurations are depicted, which demonstrates that this launch site configuration is reusable over the entire life of the HLLV Program. *Note: for purposes of clarity, the HLLV stages are shown 90 degrees out-of-plane from their likely orientation (east/west) on the RLP.*

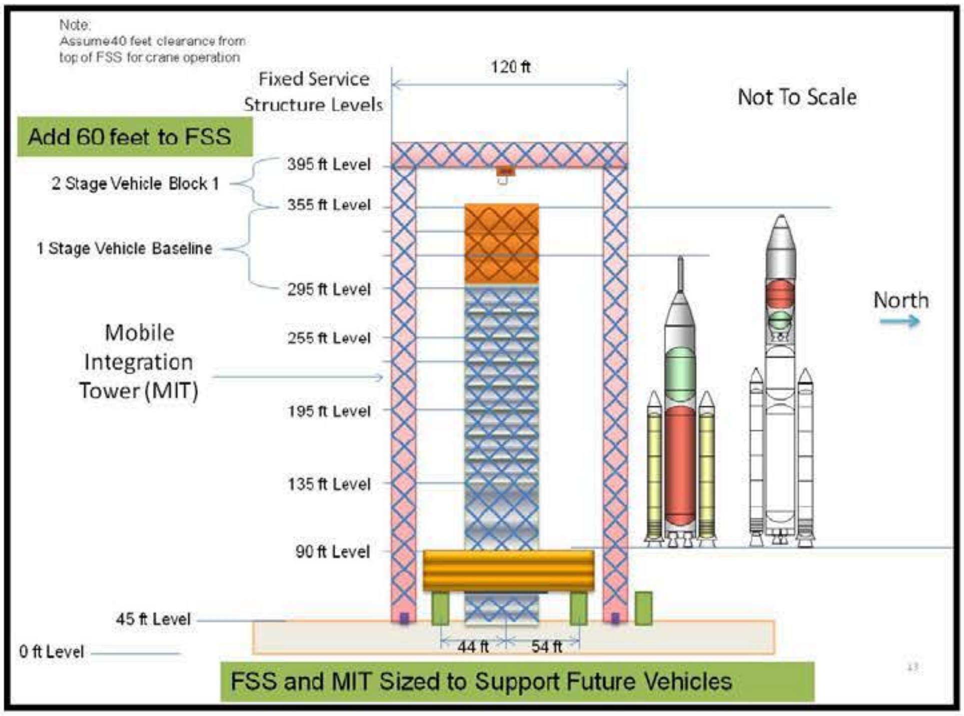


Figure 6-12: West View of Launch Pad

Figure 6-13 is an elevation view of the launch site looking north-to-south at the MIT with the tower in the Stack position. An SRM segment is shown on the pad east apron to depict a nominal offload position. *Note: the RLP is shown for reference, but it would be obscured by the north wall of the MIT in this position.*

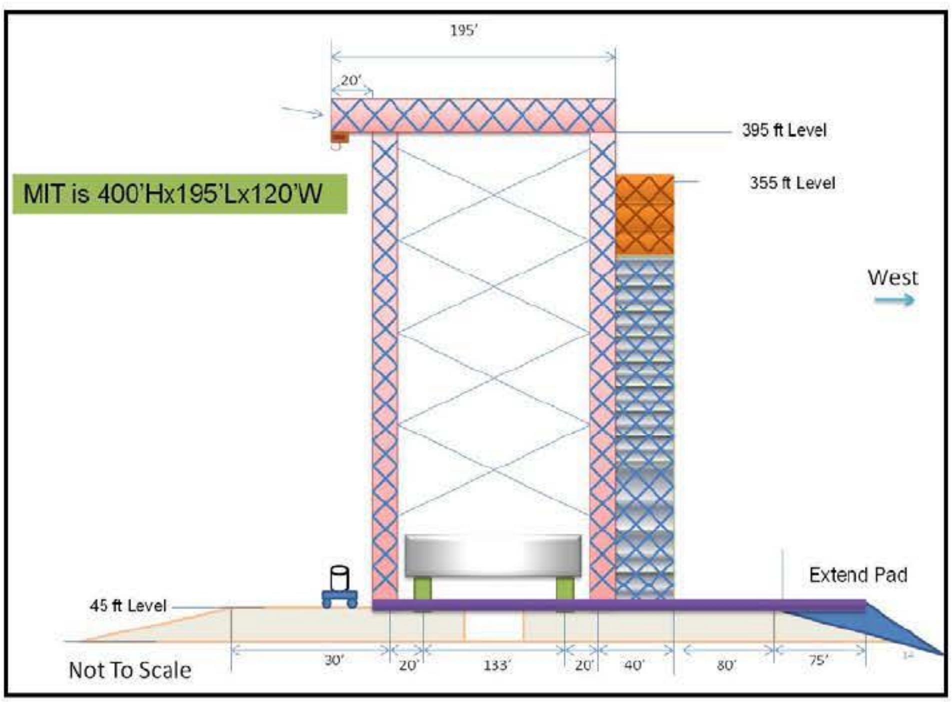


Figure 6-13: North View of Launch Pad, MIT Extended

Figure 6-14 depicts an elevation view of the launch site looking north-to-south at the MIT with the tower in the Park position behind the FSS. This view shows a required 75' extension of the west pad apron.

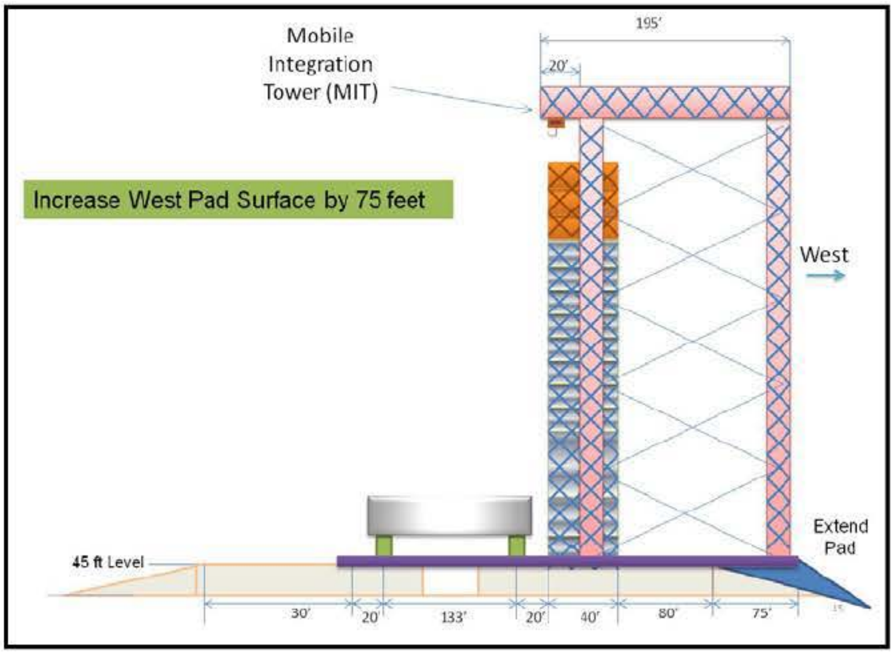


Figure 6-14: North View of Launch Pad, MIT Retracted

(b) (3)

[redacted]

(b) (3)

[redacted]

Figure 6-17 is an aerial view of LC-39, for reference. Note: LC-39B (prior to FSS/RSS demolition) is shown, as similar views of LC-39A are not readily available.

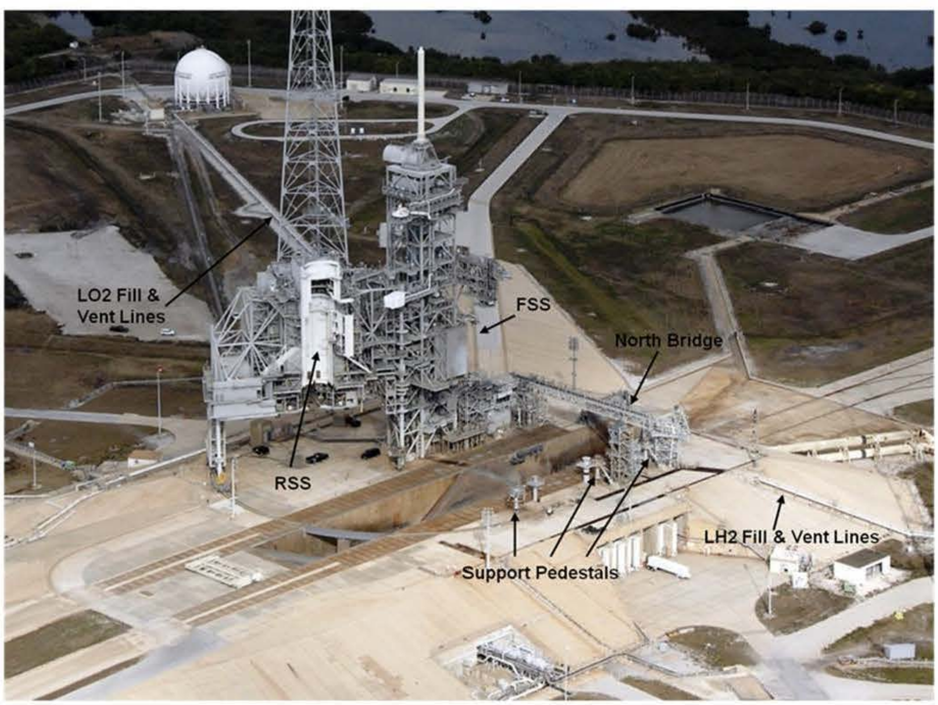


Figure 6-17: Launch Complex 39 Aerial View

6.2.2.1.4.5 Cost and Schedule

Lockheed Martin has led several similar launch facility projects, Figure 6-18, that involved revising, or building new facilities.

- MPCV Crew Exploration Vehicle Manufacturing Facility - Operations & Checkout (O&C) Building, KSC, FL
- West Coast Atlas V Evolved Expendable Launch Vehicle (EELV) Launch Site at Space Launch Complex 3 East (SLC-3E), Vandenberg Air Force Base (VAFB), CA
- Atlas V EELV Launch Site at Launch Complex 41 (LC-41), Cape Canaveral Air Force Station (CCAFS), FL
- X-33 Single-Stage-To-Orbit Launch Site at Haystack Butte, Edwards Air Force Base (EAFB), CA

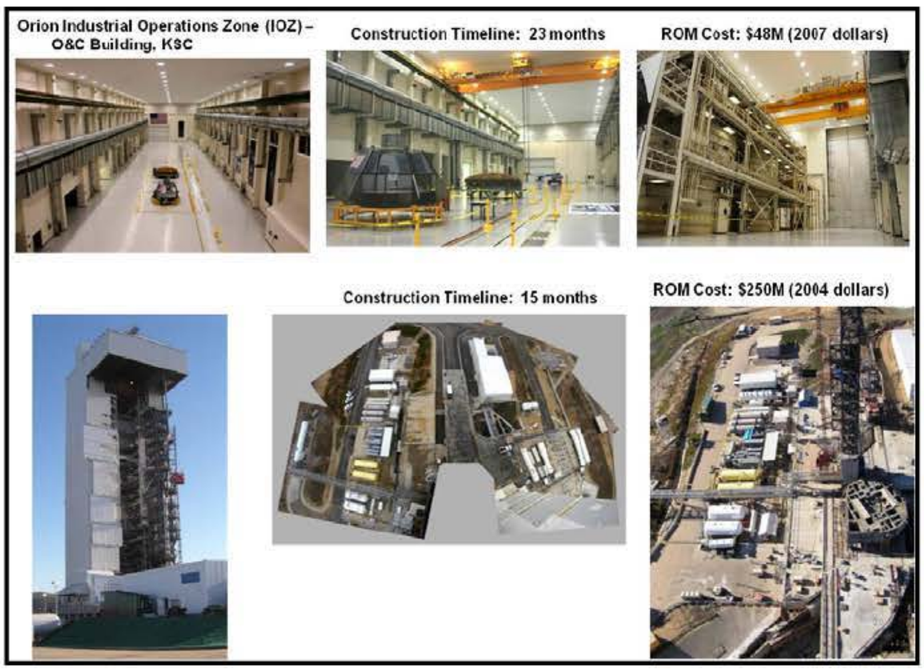


Figure 6-18: Examples of Recent Launch or Space Vehicles Processing Facilities

Compared to the listed projects, the proposed HLLV launch site modification technical complexity scope is midway between LC-41 & SLC-3E. The proposed HLLV launch site modification is comparable to the LC-41 and SLC-3E builds. Based on experience Lockheed Martin's assessment is that the proposed LC-39A build is economically and technically feasible.

Actual construction timelines and costs are shown for several recent launch site construction / modification projects led by Lockheed Martin, including SLC-3E, LC-41, the MPCV manufacturing facility (O&C building) and the X33 launch site. A graphic depiction of those costs & timelines is shown in Figure 6-19.

Facility	Dollars (Millions)	Months	2	3	4	6	8	10	12	14	16	18	20	22	24	26	28	30
X-33 Launch Site, Haystack Butte, EAFB	36	24																
Atlas V Launch Site, SLC-3E, VAFB	250	15																
Atlas V Launch Site, LC-41, CCAFS	450	30																
O&C Building, KSC	48	23																

Figure 6-19: Cost and Schedule for Several Lockheed Martin Launch Facilities Projects

6.2.2.1.5 Operability Summary

The Proposed CONOPS re-utilizes many of the existing STS processing and launch facilities and leverage vehicle processing and launch techniques that have been proven at other similar facilities in the recent past. The proposed CONOPS is economically feasible and technically possible within the time and cost constraints identified by the NASA customer for the HLLV Program.

In addition, the proposed LC-39A / HLLV ConOps provides the NASA customer with the best opportunity of achieving its goals of an functional and affordable HLLV launch capability as early as 2013.

6.2.2.2 Systems Engineering

Systems Engineering is defined as the collective set of methods, procedures, scientific and engineering skills applied to large and complex system development to achieve efficient and accurate translation of fundamental mission objectives into a system that best meets the objectives at minimum cost within the required schedule and at a minimum risk. Systems Engineering processes and tools, when properly implemented, can help meet affordability goals by providing a clear technical direction for a HLLV program.

The following key objectives of Systems Engineering that, if followed, can aid in meeting planned affordability goals.

- Assure that the definition of the system or component, to satisfy an established customer need, is conducted on a total system basis, reflecting hardware, facilities, personnel data, computer programs, and support requirements to achieve required effectiveness at minimum life cycle cost within the required schedule and at minimum risk.
- Provide a structured framework, with beginning-to-end traceability, of clear and concise system requirements under strict configuration management to serve as a basis for development plans, contract work statements, specifications, test plans, design drawings and other engineering documentation.
- Provide clear and concise requirements for making major technical decisions that optimize the total system to best meet the mission objectives.
- Integrate the design requirements and related efforts of reliability, maintainability, integrated logistics support, human factors engineering, safety, and other engineering specialties with respect to each other as well as into the mainstream of the engineering effort.

- Assure that the engineering effort is fully integrated, so that it reflects adequate and timely consideration of design, test and demonstration, production, operation, and support of the system/equipment.
- Assure compatibility of all interfaces within the system, including necessary supporting equipment and facilities; and to assure the compatibility and proper interface of the system with other systems and equipment that will be present in the operational environment.
- Provide means to establish and control the Work Breakdown Structure throughout the life of the system/project.
- Provide means for evaluation of changes that will reflect consideration of the effect of the change on overall system performance and effectiveness, schedule, and cost as well as assure that all affected activities participate in the evaluation of changes.
- Provide visibility to measure and judge technical performance status for timely identification of problems.

The most common pitfall that programs run into is insufficient requirement definition in the initial development phase which can force design engineers to design a system based on their assumptions and not a clear requirement. Once requirements are defined and the assumptions are found to be in conflict with the newly defined requirements the design must be reworked. This rework can lead to cost overruns and schedule slips. It is imperative that discipline in requirement baseline definition and requirement change management is exercised in all phases of a program to help ensure the planned affordability goals are met.

In addition, Victory Solutions, Inc., a small business located in Huntsville, AL, has authored a discussion that offers some practical and implementable suggestions and explores affordable SE&I approaches that can be put into practice. This discussion can be found in Appendix A of this report.

6.2.3 **Extensibility**

Extensibility has been a major theme throughout this BAA assessment. Utilizing existing assets, such as hardware, databases, tooling and production processes, will result in the savings of significant funds. We used this criteria in assessing the "Building Block" approach and the extensive leverage of the Shuttle ET assets from design through production.

6.2.4 **Commonality**

Maximizing commonality can have a significant impact on Affordability. Commonality can be applied within the design of a new vehicle, across previous and existing space programs, and across agencies (NASA, DoD, or Commercial Space interests). New vehicle designs (a similar forward dome design, for LO_2 and LH_2 tanks, is an example). The use of existing components, systems, processes, tooling, and facilities that were developed on other programs (Shuttle, MPCV, Ares, Atlas, Delta, or Falcon 9) can reduce the time and money spent generating new specifications and procuring new parts. This could also increase affordability by leveraging existing supply chains and resources. Commonality across agencies would increase reliability by allowing similar programs to share resources. For example, a cryogenic seal purchased to a common specification for DoD could be used on a NASA program in the event of a shortage or a supplier issue.

For the HLLV program, an effort should be made to review components, systems, processes, tooling, and facilities and to document areas where commonality makes sense. The effort could be lead by the NASA HLLV program and implemented through the vehicle integration prime contractor. The contract for each element would need to be structured to facilitate the use of common parts, specifications, tooling, and processes between sub-contractors. If the HLLV is Shuttle based, there is a large amount of components, processes, tooling, and facilities that are already available. For example, standard parts that are available are fasteners, seals, sensors, and connectors. Shuttle processes currently used for cleaning, coatings, TPS application, adhesive bonding, leak testing, NDE inspection, etc. could also be used on the new HLLV.

To share a specification, each element contractor could review the specification and identify what needed to be included. The responsible contractor could then revise the specification so it would be common across the elements.

This example of commonality would yield many benefits, such as, one supplier and the ability to share common hardware at the production facility or the launch site during processing or rework/mods. One common supplier could reduce the associated supplier documentation required to produce the seals. One supplier reduces the resources required for supplier surveillance. Impacts due to changes to the specification can be reduced by having one common specification and supplier.

Multiple specifications from similar parts or new specification for parts that may be used from previous programs drive cost. Every specification requires a process of selecting a supplier. The supplier generates internal documentation to meet and verify the specification. This documentation will then be required by the prime for approval and change management. Multiple suppliers of critical parts require surveillance of processes which requires additional resources and drive costs. Multiple specifications also affect logistics during production and at launch site. A common specification for a seal may allow sharing of identical size seals between Core Stage and 2nd Stage, thereby, reducing the amount of effort certifying that the parts are the same and the approval required to implement this part sharing.

For instance, if there is a 6 inch Liquid Oxygen Naflex seal used in both the MPS for the Core Stage and the 2nd Stage, commonality may have an impact if there is rework required at the launch site and the Core Stage does not have the part in-stock at the launch site and the 2nd Stage contractor does have this part. If a common specification for both program is used than the part can be transferred from one program to the other with little review since there are purchased to the same specification. If each contractor has a unique specification for the same size seal then a review is required to verification that the requirements and certification for the similar parts are acceptable before it can be used between programs. Common specification standard parts can also reduce logistics at the launch site by allowing common stores. If both vehicles are manufactured at the same facility, the facility management contractor can store the parts in a common area.

This approach, if use widely, could reduce cost and provide efficiency. This will take coordination and a change in business practices to setup up processes to allow this commonality. This approach could be extended to other NASA, DoD and Commercial program, where the use of these common items is applicable.

An approach for commonality may consist of the following:

1. Review the requirements for each component, standard parts, processes, tooling, facilities, (hereinafter referred to as common item) of the HLLV, i.e. Core Stage, 2nd Stage, boosters and liquid engines.

2. Identify area of commonality.

3. Identify owner of common item(s) requirements specification

4. Review, resolve and implement unique requirements of each HLLV element, i.e. size, inspection criteria, material, etc.

5. Provide contract direction to allow usage by multiple HLLV elements

Note: Items 4 and 5 could be accomplished thru a HLLV change management board early in the DDT & E phase of the program.

6.2.5 **Reusability**

Reusability of the Solid Rocket Boosters should be re-assessed. Several existing studies from the Shuttle Program may offer insight into the aspects of reusability of the SRBs. While total re-use may not be considered due to logistics and rework/refurbishment, it may be cost effective to reuse some avionics components. This would have to be traded against the cost associated with recovery parachutes and retrieval versus purchasing new avionics.

6.3 Affordability Summary

Many factors can affect affordability. The discussion for this report focuses on only a few topics. It is felt that leveraging existing designs, tooling, facilities, reductions in test programs and commonality will have a great impact. A very strong approach to requirements development and implementation will have many benefits to the new programs, applying lessons learned from MPCV and other Constellation and Shuttle programs will provide benefits. A strong contributor that should be considered is a different approach to launch operations at KSC and some recommendations are provided in this section.

7.0 CAPABILITY GAPS

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8.2.3 Carbon Foam Heat Shield Material

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9.0 INCREMENTAL TESTING & IN-SPACE DEMONSTRATION

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11.0 FINAL REPORT SUMMARY

The Lockheed Martin FOMs, GR&A, Goals and DRMs assessment performed as part of this BAA effort did not find any additional requirements. This assessment concluded that Affordability, Safety & Reliability and Schedule are the main FOMs.

NASA provided DRMs and LMSSC DRMs are similar and achievable with the LM assessed HLLV configurations.

The "Building Block" configuration approach is an Affordable way to meet NASA FOMs, GR&A, Goals, and DRMs. These configurations Shuttle-derived Element designs which further enhance affordability and reduce program risk.

Although further study is required, elimination of the Thrust Vector Control system on the SRBs for these configurations appears to be feasible and could enhance Affordability, reduce Program Risk and operational complexity.

The vehicle PRA numbers for the proposed vehicle configurations are driven by the Engineer PRA number and can be improved with assessment of Loss of Crew numbers and a more detailed FMEA study for these configurations. These PRA numbers are based on demonstrated probabilities on similar flight hardware on Shuttle and are not reliant on new or untested designs. These PRA numbers are in-line with current Shuttle goals.

The design of these configurations is well understood and will leverage Shuttle heritage, design, processes and certifications.

The manufacture of the Core Stage will benefit from being performed at MAF. This will reduce Program Risk and provide affordability by using well understood and existing facilities, tooling and infrastructure. Extensive knowledge of current ET production and "Lessons Learned" allows for streamline of the process, reducing workstations and building use at MAF which will benefit affordability and reduce overhead at MAF.

LCC cost analysis shows the HLLV program is achievable with existing NASA budget and schedule milestones. This LCC analysis is based on experience on existing systems, (Shuttle ET, Atlas V) LCC data.

The LCC data provided, related to affordability, in this report demonstrates that the Shuttle-derived configuration for HLLV can be achieved within NASA budget guidelines. Additional savings could be realized using program management techniques that are approved and in-place on the MPCV Program. These LCC figures are based on leveraging many of the Shuttle designs, i.e. SSME, SRB, similar Tank structure. Use of this type of configuration reduces DDT &E costs thru leveraging Shuttle design and a resulting reduction in Certification testing. Some examples would be reduced amount of structural testing on the Core Stage thru use of similar designs for the structure. A leaner certification program, is recommended, for the Main Propulsion System avoiding the cost of a full up dedicated Main propulsion Test Article. Leveraging SRB and SMME designs in this configuration will reduce overall program DDT&E costs over new liquid engine and/or liquid booster development.

No current technology gaps exist in the near term for LEO objectives, longer term DRMs will benefit from In-Space cryo-propellant management development.

Non-technical gaps do exist in delays in schedule and non-level funding to sustain the program and in the areas of resources, facilities and suppliers. The longer the program slips beyond end of Shuttle the more these resource gaps will become an issue.

Several innovations on Avionics, TVC, and potential heat shield materials require more investigation and can add to affordability of the HLLV program, thru "commonality" (Avionics and TVC), less complex systems (TVC) and potential cost and processing saving realized thru the development of the potential heat shield material. The extensive background of "Lessons Learned" from Shuttle program can be applied (potential innovations list in Section 8.2.6) to the HLLV to enhance affordability and reduce program risk.

A lower cost Main Propulsion Systems testing approach is achievable to reduce the requirement for a full up Main propulsion Test Article approach that was used on the Shuttle program. This is possible thru similarity of many components and systems in the MPS, delta component testing, sub-scale testing and limited full scale testing leveraging on test, analysis and operations data from similar shuttle systems. Similarity of the Core Stage to the Shuttle ET will reduce the requirement for structural and dynamics testing by leveraging Shuttle testing and analysis certifications.

APPENDIX A: System Requirements from an Affordability Perspective

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APPENDIX B: Recommendations & Forward Work

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APPENDIX C: Compliance Matrix

Scope

This appendix contains the compliance matrix which provides traceability from the SOW to the applicable sections of this report.

HLPT SATS Solicitation Technical Objectives Paragraph	Contract Task #	SOW Task Description	Final Report Section
Provide a recommended list of key decision attributes and rationale associated with each	**1**	**Define Heavy Lift Launch System Mission Models & Requirements**	**Section 4.0**
	1.1	Review Mission\Operational Requirements	Section 3.2
	1.1.1	Document Stakeholder Needs, expectations, constraints, system boundaries, interfaces and intended uses	Section 3.3
	1.1	Analyze and define rationale for stakeholder needs, expectations, constraints, system boundaries, interfaces and intended uses	Section 3.4
	1.3	Prioritize stakeholder needs, expectations, constraints, system boundaries, interfaces, and intended uses	Section 3.4
	1.4	Document mission models that meet stakeholder needs, expectations, constraints, system boundaries, interfaces, and intended uses	Section 4.2
	1.5	Define Mission Requirements, with rationale, based on prioritized stakeholder needs	Section 4.3
	2	**Define Figures of Merit (inc. Key Decision Attributes)**	**Section 3.0**
	2.1	Define relevant Figures of Merit, that reflect stakeholder needs, operational capabilities, and mission requirements	Section 3.2
	2.1.1	Define Key Decision Attributes	Section 3.3

HLPT SATS Solicitation Technical Objectives Paragraph	Contract Task #	SOW Task Description	Final Report Section
Provide a recommended list of key decision attributes and rationale associated with each Identify how changes to the weighting of key decision attributes affect the architectures	2.1.2	Define Figure of Merit Objective Values & Min/Max Threshold Values	Section 3.4
	2.2	Perform pair-wise comparison of Figures of Merit and incorporate previously defined stakeholder needs prioritization to determine weighting.	Section 3.4
	2.4	Create bins and populate metrics, within RACE tool, based on the previously defined Figures of Merit	Section 3.4
Identify how alternative ground rules and assumptions (Reference NASA HLLV Study) impact the identified alternative system solutions. For example, due to time and resource constraints, the NASA HLLV study could not address system alternatives associated with the number of launches, alternative LO_2/RP-1 1st stage main engine characteristics, evolutionary vehicle development, the use of propellant transfer or depot, the incorporation of international partner participation, the use of multiple crew spacecraft options, and the effect of technology development	**3**	**Define Current and Alternative Heavy Lift System Architectures**	**Section 5.2.7**
	3.1	Define current/target operational environment and existing operational/system capabilities	Section 4.2
	3.1.1	Perform POST simulation of POD Heavy Lift architecture per the previously defined mission models and requirements	Section 5.2.7
	3.1.2	Perform CAIV analysis of Heavy Lift Architecture	Section 6.0
	3.1.3	Perform manufacturability assessment of Heavy Lift Architecture	Section 5.2.14

HLPT SATS Solicitation Technical Objectives Paragraph	Contract Task #	SOW Task Description	Final Report Section
	3.1.3	Perform operability assessment of Heavy Lift Architecture	Section 6.2.1
	3.1.5	Perform reliability\safety assessment of Heavy Lift Architecture	Section 5.2.9
	3.1.6	Enter parameter values calculated in each analysis, for the current state, into the RACE tool	Section 5.2.7
	3.2	Define alternative Heavy Lift architectures that meet the previously defined Mission Requirements and mission models	Section 5.2.7
	3.2.1	Perform POST simulation of Heavy Lift architecture per the previously defined mission models and requirements	Section 5.2.7
	3.2.2	Perform CAIV analysis of Heavy Lift Architecture	Section 6.0
	3.2.3	Perform manufacturability assessment of Heavy Lift Architecture	Section 5.2.14
	3.2.3	Perform operability assessment of Heavy Lift Architecture	Section 6.2.1
	3.2.5	Perform reliability\safety assessment of Heavy Lift Architecture	Section 5.2.9
	3.2.6	Enter parameter values calculated in each analysis, for the current state, into the RACE tool & score\rank Heavy Lift Alternative Architectures	Section 5.2.7
	3.3	Determine how changes to the weighting of key decision attributes affect the architecture rankings by performing sensitivity analysis of Figure of Merit weighting	Section 3.4, 4.4
	3.3	Assess impact of alternative ground rules and assumptions on identified alternative heavy lift architectures' Figure of Merit objective and threshold values	Section 4.4

HLPT SATS Solicitation Technical Objectives Paragraph	Contract Task #	SOW Task Description	Final Report Section
Identify capability gaps associated with the Heavy Lift System, and for each capability gap identify specific areas where technology development may be needed. Items identified as requiring technology development shall be quantitatively evaluated using established metrics, e.g. NASA Technology Readiness Level (TRL), Capability Readiness Level (CRL), Manufacturing Readiness Level (MRL), Process Readiness Level (PRL).	**4**	**Perform Heavy Lift Technology Capability Gap Analysis**	**Section 7.0**
	4.1	Identify capability gaps in the POD and alternative architectures by assessing the results generated by the RACE tool	Section 7.3
Identify capability gaps associated with the first-stage main engine functional performance and programmatic characteristics required to support each Heavy Lift System studied. The minimum set of functional performance characteristics identified shall include engine thrust, specific impulse (Isp), mixture ratio, mass, throttle range, and physical envelope. This assessment shall include, but is not limited to, LOX/RP-1 main engine systems. The minimum set of programmatic characteristics identified shall include an estimated overall lifecycle cost (i.e., DDT&E, production and operations (fixed and variable) per engine cost), development schedule, and production rate. Identify any impacts to overall life cycle costs of the Heavy Lift System based on the engine studied.	4.2	Using the LM Risk Management Process, determine first stage main engine capability gaps and develop a mitigation plan for the gaps to meet the Mission Requirements	Section 7.3
	4.2.1	Identify potential capability gaps and enter into the Active Risk Manager (ARM) tool as an opportunity	Section 7.3
	4.2.2	Determine the probability of occurrence, based on the current environment & state of the art, of capability gaps	Section 7.4

HLPT SATS Solicitation Technical Objectives Paragraph	Contract Task #	SOW Task Description	Final Report Section
	4.2.3	Determine a priority for high probability capability gaps based on the TRL, LCC, Manufacturability & Operability analysis and probability	Section 7.4
	4.2.4	Determine actions required to mitigate capability gaps	Section 7.4
	4.2.4	Create a Risk and Opportunity Assessment Report for identified capability gaps	Section 7.4
Identify capability gaps associated with the upper-stage main engine functional performance and programmatic characteristics required to support each heavy lift system studied. The minimum set of functional performance characteristics identified shall include engine propellants, thrust, specific impulse (Isp), mixture ratio, mass, throttle range, and physical envelope. The minimum set of programmatic characteristics identified shall include an estimated overall lifecycle cost (i.e., DDT&E, production and operations (fixed and variable) per engine cost), development schedule, and production rate. Identify any impacts to overall life cycle costs of the Heavy Lift System based on the engine studied	4.3	For upper stage main engine capability gaps, develop justification for change or improvement of existing capabilities to meet the Mission Requirements	Section 7.4
	4.3.1	Identify potential capability gaps and enter into the Active Risk Manager (ARM) tool as an opportunity	Section 7.4
	4.3.2	Determine the probability of occurrence, based on the current environment & state of the art, of capability gaps	Section 7.4
	4.3.3	Determine a priority for high probability capability gaps based on the TRL, LCC, Manufacturability & Operability analysis and probability	Section 7.4
	4.3.4	Determine actions required to mitigate capability gaps	Section 7.4

HLPT SATS Solicitation Technical Objectives Paragraph	Contract Task #	SOW Task Description	Final Report Section
	4.3.4	Create a Risk and Opportunity Assessment Report for identified capability gaps	Section 7.4
Identify capability gaps associated with the In-Space space propulsion elements functional performance and programmatic characteristics required to support each Heavy Lift System studied. This assessment shall include, but is not limited to, LO_2/LH_2 and LO_2/CH_4 propulsion systems. The minimum set of functional performance characteristics identified shall include propellant definition, thrust, specific impulse (Isp), mixture ratio, mass, throttle range (if any), and physical envelope. The minimum set of programmatic characteristics identified shall include an estimated overall lifecycle cost (i.e., DDT&E, production and operations (fixed and variable) per engine cost), development schedule, and production rate. Identify any impacts to overall life cycle costs of the Heavy Lift System based on the engines studied	4.4	For In-Space propulsion capability gaps, develop justification for change or improvement of existing capabilities to meet the Mission Requirements	Section 7.4
	4.4.1	Identify potential capability gaps and enter into the Active Risk Manager (ARM) tool as an opportunity	Section 7.4
	4.4.2	Determine the probability of occurrence, based on the current environment & state of the art, of capability gaps	Section 7.4
	4.4.3	Determine a priority for high probability capability gaps based on the TRL, LCC, Manufacturability & Operability analysis and probability	Section 7.4
	4.4.4	Determine actions required to mitigate capability gaps	Section 7.3
	4.4.4	Create a Risk and Opportunity Assessment Report for identified capability gaps	Section 7.3

HLPT SATS Solicitation Technical Objectives Paragraph	Contract Task #	SOW Task Description	Final Report Section
Identify capability gaps associated with all other technical aspects of heavy lift system, e.g. tanks, propellant and pressurization systems, integrated system health management, auxiliary propulsion systems, avionics and control systems, structures. Identify test and integrated demonstrations to mitigate risk associated with the gaps. Identify capability gaps associated with all other technical elements of the In-Space space propulsion element, e.g. tanks, propellant and pressurization systems, cryogenic fluid management, integrated system health management, auxiliary propulsion systems, avionics and control systems, structures, autonomous rendezvous and docking. Identify test and integrated demonstrations to mitigate risk associated with the gaps.	4.4	For other HLS technology capability gaps, develop justification for change or improvement of existing capabilities to meet the Mission Requirements	Section 7.4
	4.4.1	Identify potential capability gaps and enter into the Active Risk Manager (ARM) tool as an opportunity	Section 7.4
	4.4.2	Determine the probability of occurrence, based on the current environment & state of the art, of capability gaps	Section 7.4
	4.4.3	Determine a priority for high probability capability gaps based on the TRL, LCC, Mfg. & Operability analysis and probability	Section 7.4
	4.4.4	Determine actions required to mitigate capability gaps	Section 7.4
	4.4.4	Create a Risk and Opportunity Assessment Report for identified capability gaps	Section 7.4

HLPT SATS Solicitation Technical Objectives Paragraph	Contract Task #	SOW Task Description	Final Report Section
Identify how innovative or non-traditional processes or technologies can be applied to the Heavy Lift Systems to dramatically improve its affordability and sustainability.	**5**	**Perform Heavy Lift Innovative\Non-Traditional Technology Assessment**	**Section 8.0**
	5.1	Using the LM Opportunity Management Process, determine how innovative or non-traditional processes or technologies can be applied to the Heavy Lift Systems to dramatically improve the affordability & sustainability	Section 8.2
	5.1.1	Identify potential innovative\non-traditional processes and\or technologies and enter into the Active Risk Manager (ARM) tool as an opportunity	Section 8.2 - 8.3.5
	5.1.2	Determine the Technology Readiness Level (TRL) of innovative\non-traditional process and\or technology	Section 8.2 - 8.3.5
	5.1.3	For innovative\non-traditional process and\or technology identified, perform LCC, manufacturability & operability analysis to determine potential cost & schedule savings	Section 8.2 - 8.3.5
	5.1.4	Determine the probability, based on the current environment & state of the art, of innovative\nontraditional process or technology being ready for implementation in a Heavy Lift System	Section 8.2 - 8.3.5
	5.1.5	Determine a priority for identified innovative\non-traditional process and\or technology based on the TRL, LCC, Manufacturability & Operability analysis and probability	Section 8.2 - 8.3.5
	5.1.5	Determine actions required to make innovative\non-traditional process or technology ready for implementation in a Heavy Lift System	Section 8.2 - 8.3.5

HLPT SATS Solicitation Technical Objectives Paragraph	Contract Task #	SOW Task Description	Final Report Section
	5.1.7	Determine any risks that may be introduced by each innovative\non-traditional process and\or technology and perform an assessment of impact and probability	Section 8.2 - 8.3.5
	5.1.8	Create a Risk and Opportunity Assessment Report for identified innovative\non-traditional process and\or technology	Section 8.2 - 8.3.5
Identify how aspects of a Heavy Lift System (including stages, subsystems, and major components) could have commonality with other user applications, including NASA, DoD, commercial, and international partners.	**6**	**Perform Heavy Lift Technology Commonality Assessment**	**Section 6.2.3**
	6.1	Perform assessment to determine how elements of a Heavy Lift Launch System could have commonality with other user applications, including NASA, DoD, commercial and international partners	Section 6.2.3
Identify how incremental development testing, including ground and flight testing, of Heavy Lift System elements can enhance the heavy lift system development.	**7**	**Perform Heavy Lift System Incremental & Demonstration Testing Assessment**	**Section 9.0**
	7.1	Using the LM Opportunity Management Process, determine how incremental development testing, including ground and flight testing, of Heavy Lift System elements can enhance the heavy lift system development	Section 9.2
	7.1.1	Identify potential Incremental Development Tests and enter into the Active Risk Manager (ARM) tool as an opportunity	Section 9.2
	7.1.2	Determine the Technology Readiness Level (TRL) of any supporting technologies that may be needed to perform Incremental Development Test	Section 9.2

HLPT SATS Solicitation Technical Objectives Paragraph	Contract Task #	SOW Task Description	Final Report Section
	7.1.3	For Incremental Development Tests identified perform analysis to determine potential cost & schedule impacts	Section 9.2
	7.1.4	Determine the probability, based on the current environment & state of the art, of success for Incremental Development Test	Section 9.2
	7.1.5	Determine a priority for identified Incremental Development Tests based on the LCC, Manufacturability & Operability analysis and probability of success	Section 9.2
	7.1.6	Determine actions required to make Incremental Development Tests ready for implementation in a Heavy Lift System	Section 9.2
	7.1.7	Determine any risks that may be introduced by Incremental Development Tests and perform an assessment of impact and probability for each one	Section 9.2
	7.1.7	Create a Risk and Opportunity Assessment Report for identified Incremental Development Tests	Section 9.2
Identify what In-Space space propulsion elements, if any, which should be demonstrated via space flight experiments.	7.2	Using the LM Opportunity Management Process, determine what In-Space propulsion elements, if any, should be demonstrated via space flight experiments	Section 9.2
	7.2.1	Identify potential In-Space Propulsion Space Flight Demonstration Experiments and enter into the Active Risk Manager (ARM) tool as an opportunity	Section 9.2
	7.2.2	Determine the Technology Readiness Level (TRL) of any supporting technologies that may be needed to perform In-Space Propulsion Space Flight Demonstration Experiments	Section 9.2
	7.2.3	For In-Space Propulsion Space Flight Demonstration Experiments identified, perform LCC, Manufacturability & Operability analysis to determine potential cost & schedule savings	Section 9.2

HLPT SATS Solicitation Technical Objectives Paragraph	Contract Task #	SOW Task Description	Final Report Section
	7.2.4	Determine the probability, based on the current environment & state of the art, of success for In-Space Propulsion Space Flight Demonstration Experiments	Section 9.2
	7.2.5	Determine a priority for identified In-Space Propulsion Space Flight Demonstration Experiments based on the LCC, Manufacturability & Operability analysis and probability of success	Section 9.2
	7.2.6	Determine a priority for identified In-Space Propulsion Space Flight Demonstration Experiments based on the LCC, Manufacturability & Operability analysis and probability of success	Section 9.2
	7.2.7	Determine any risks that may be introduced by In-Space Propulsion Space Flight Demonstration Experiments and perform an assessment of impact and probability for each one	Section 9.2
	7.2.7	Create a Risk and Opportunity Assessment Report for identified In-Space Propulsion Space Flight Demonstration Experiments	Section 9.2

APPENDIX D: Acronyms List

ACRONYM LIST

2GRLV	2nd Generation Return to Launch Vehicle
ACB	Advanced Composite Boosters
ALETP	Advanced Liquid Engine Test Program
ALS	Advance Launch System
ALTA	Aluminum-Lithium Test Article
AOA	Angle of Attack
APU	Auxiliary Power Unit
AR&D	Automated Rendezvous and Docking
ARC	Ames Research Center
ARM	Active Risk Manager
ASTM	American Standards for Testing and Materials
ASWS	Acoustic Suppression Water System
ATK	Alliant Techsystems Inc.
ATP	Authority To Proceed
BAA	Broad Agency Announcement
BAC	Broad Area Cooling
BATC	Ball Aerospace and Technologies Corporation
BEO	Beyond Earth Orbit
BLEO	Beyond Low Earth Orbit
BSTRA	Ball Strut Attachment
C&DM	Configuration and Data Management
CAD	Computer Aided Design
CAIT	Constellation Analysis Integration Tool
CAIV	Cost As An Independent Variable
CCAFS	Cape Canaveral Air Force Station
CDR	Critical Design Review
CER	Cost Estimating Relationship
CFD	Computational Fluid Dynamics
CFM	Cryogenic Fluid Management
CG	Center of Gravity
CHIL	Collaborative Human Immersive Laboratory
CIL	Critical Items List
CM	Configuration Management
CO_2	Carbon Dioxide
COBE	Cosmic Background Explorer
CONOPS	Concept of Operations
COQ	Certificate Of Qualification
COTS	Commercial Off The Shelf
CPST	Cryogenic Propellant Storage and Transfer
CRES	Corrosion-Resistant Steel
CRL	Capability Readiness Level
CRYOTE	Cryogenic Orbital Test
CSCA	Core Stage Intertank Carrier Plate Assembly
CTE	Critical Technology Element
CY	Calendar Year

DAC	Design Analysis Cycles
DAGGR	Data Aggregator
DAISI	Data Analysis Integration and Systems Implementation
DCS	Design Certification Sheet
DCSS	Delta Cryogenic Second Stage
DDMS	Design Data Management System
DDT&E	Design Development Test and Evaluation
DFI	Development Flight Instrumentation
DFMA	Design for Manufacture and Assembly
DOD	Department of Defense
DPD	Data Procurement Document
DPM	Data Parts Management
DRD	Data Requirement Document
DRM	Design Reference Mission
EAFB	Edwards Air Force Base
EBOM	Engineering Bill of Material
ECO	Engine Cut-Off
EELV	Evolved Expendable Launch Vehicle
EIS	End Item Specification
EMRL	Engineering and Manufacturing Readiness Level
EPDM	Enterprise Product Data Management
ERP	Engineering Release for Procurement and Production
ET	External Tank
ETO	Earth-to-Orbit
EVM	Earned Value Management
FASTPASS	Flexible Analysis for Synthesis, Trajectory and Performance for Advanced Space Systems
FCV	Flow Control Valve
FEM	Finite Element Method
FFBD	Functional Flow Block Diagram
FMEA	Failure Mode and Effects Analysis
FOD	Foreign Object Debris
FOM(s)	Figure(s) of Merit
FRF	Flight Readiness Firing
FRR	Flight Readiness Review
FSD	Full Scale Development
FSS	Fixed Service Structure
FSW	Friction Stir Weld
FTB5	SRB Interface Load Indicator
FTB6	SRB Interface Load Indicator
FY	Fiscal Year
GEO	Geostationary Orbit
GFE	Government Furnished Equipment
GH_2	Gaseous Hydrogen
GHSV	Gas Hourly Space Velocity
GO_2	Gaseous Oxygen

GR&A	Ground Rules and Assumptions
GRC	Glenn Research Center
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
GUCA	Ground Umbilical Carrier Assembly
GUI	Graphical User Interface
GVTA	Ground Vibration Test Article
HALE	High Altitude Long Endurance
HCS	Hardware Certification Sheet
HEFT	Human Exploration Framework Team
HIL	Human Immersive Laboratory
HLLV	Heavy Lift Launch Vehicle
HLPT	Heavy Lift and Propulsion Technology
HLS	Heavy Lift System
HTPB	Hydroxyl-Terminated Polybutadiene
HTTA	Hydrogen Thermal Test Article
HVAC	Heating, Ventilation, and Air Conditioning
IES	Innovative Engineering Solutions
IHF	Initial Human Flight
IMLI	Integrated Multi-Layer Insulation
IMS	Information Management System
IOC	Initial Operational Capability
IPT	Integrated Product Team
IRAD	Independent Research and Development
IRAS	Infrared Astronomical Satellite
IRMA	Integrated Risk Management Application
ISCPD	In-Space Cryogenic Propellant Depot
ISO	International Organization for Standardization
ISS	International Space Station
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
JSF	Joint Strike Fighter
KSC	Kennedy Space Center
LaRC	Langley Research Center
LC	Launch Complex
LCC	Launch Commit Criteria
LCCE	Life Cycle Cost Estimate
LCROSS	Lunar Crater Observation and Sensing Satellite
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LLC	Long Life Cooler
LM	Lockheed Martin
LMC	Lockheed Martin Corporation
LMSSC	Lockheed Martin Space Systems Company
LN	Liquid Nitrogen
LO_2	Liquid Oxygen

LOC	Loss of Crew
LOM	Loss of Mission
LOX	Liquid Oxygen
LSP	Launch Services Program
LVHN	Launch Vehicle Health Management
LWT	Light Weight Tank
MAF	Michoud Assembly Facility
MAVERIC	Marshall Aerospace Vehicle Representation in C
MBD	Model Based Design
MBE	Model Based Enterprise
MECO	Main Engine Cut-Off
MEI	Main Engine Ignition
MEKM	Marshall Engineering Knowledge Management
MIT	Mobile Integration Tower
MLI	Multi-Layer Insulation
MLP	Mobile Launch Platform
MMOD	Micro-Meteoroid and Orbital Debris
MPCV	Multi-Purpose Crew Vehicle
MPS	Main Propulsion System
MPTA	Main Propulsion Test Article
MRB	Material Review Board
MRL	Manufacturing Readiness Level
MSFC	Marshall Space Flight Center
MVP	Master Verification Plan
NASTRAN	NASA Structural Analysis
NASA	National Aeronautics and Space Administration
NCAM	National Center for Advanced manufacturing
NDE	Non-Destructive Examination
NEO	Near Earth Object
NGLT	Next Generation Launch Technology
NLS	National Launch System
NPSP	Net Positive Suction Pressure
NRE	Non-Recurring Expenditures
OFI	Operational Flight Instrumentation
OFT	Orbital Flight Test
OJT	On the Job Training
OML	Outer Mold Line
OPF	Orbiter Processing Facility
OSP	Orbital Space Plane
OTTA	Oxygen Thermal Test Article
OTV	Orbital Transfer Vehicle
PAUT	Phased Array Ultrasonic Testing
PBAN	Polybutadiene Acrylonitrile
PDK	Product Development Kaizens
PDR	Preliminary Design Review
PLF	

PLM	Product Lifecycle Management
PM	Program Manager
PMS	Production Master Schedule
POD	Point of Departure
POST	Program to Optimize Simulated Trajectories
PRA	Probabilistic Risk Analysis
PRL	Process Readiness Level
PRSA	Power Reactance Storage Assembly
PTSD	Propellant Transfer and Storage Demonstration
PU	Propellant Utilization
PWR	Pratt & Whitney Rocketdyne
RAC	Requirement Analysis Cycle
RACE	Rapid Affordability and CAIV Exploration
REC	Recurring
RFI	Request for Information
RFID	Radio Frequency Identification
RFQ	Request for Quote
RLP	Reconfigurable Launch Platform
RP	Rocket Propellant
RS	Rocket System
RTF	Return To Flight
SATS	Systems Analysis and Trade Study
SB	Small Business
SDB	Small Disadvantaged Businesses
SDLV	Shuttle Derived Launch Vehicle
SDR	System Design Review
SEA	Statistical Energy Methods
SFHe	Super Fluid Helium
SHE	Safety Health and Environmental
SIA	Structured Improvement Activities
SIL	System Integration Lab
S-IVB	Saturn V Rocket Third Stage
SLC	Space Launch Complex
SLS	Space Launch Systems
SLWT	Super Light Weight Tank
SMD	Science Mission Dewar
SME	Subject matter Expert
SOFI	Spray On Foam Insulation
SOW	Statement of Work
SPIRIT	Spatial Infrared Imaging Telescope
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
SSME	Space Shuttle Main Engine
SSP	Space Shuttle Program
STA	Structural Test Article
STS	Space Transport System

SWT	Standard Weight Tank
TC	Titan/Centaur
TCS	Thermodynamic Cryogen Subcooler
TIM	Technical Interchange Meeting
TPOC	Technical Point of Contact
TPS	Task Description Sheet
TPS	Thermal Protection System
TREP	Technical Representative
TRL	Technology Readiness Level
TVC	Thrust Vector Control
TY$	Then-Year Dollars
ULA	United Launch Alliance
VAB	Vertical Assembly Building
VAC	Verification, Acceptance and Certification
VAFB	Vandenberg Air Force Base
VCS	Vapor Cooled Shields
VMEDP	Virtual Manufacturing Engineering Development Platform
WIRE	Wide-field Infrared Explorer

NASA HEAVY LIFT and PROPULSION TECHNOLOGY SYSTEMS ANALYSIS and TRADE STUDY

FINAL STUDY REPORT

CONTRACT NUMBER: NNM11AA13C

DRD NUMBER: 1380MA-003

PREPARED BY

Orbital Sciences Corporation
Launch Systems Group
CAGE Code 27555

ORIGINAL ISSUE DATE

TM-22899	REV -
CODE A1-216-C7	

NASA HEAVY LIFT STUDY

FINAL REPORT

CONTRACT NUMBER: NNM11AA13C

DRD NUMBER: 1380MA-003

APPROVED BY [redacted] 6-2-11
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REVISION SUMMARY

REV	DATE	CHANGE	PAGE

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1. INTRODUCTION

1.1. Purpose

The purpose of this document is to describe the results from the NASA Heavy Lift and Propulsion Systems Analysis and Trade Study (HLS Study) conducted by the Orbital Sciences Corporation (Orbital). The principal objective of the study was to support NASA with their goal of defining a new launch vehicle and the propulsion technology that enables NASA to meet the Nation's Exploration goals and objectives over the next two to three decades. Orbital's approach for this study was to define a safe, affordable and realistic Heavy Lift System (HLS) architecture to support implementation of the Vision for Space Exploration.

1.2. Task Description

The HLS Study task description is provided through a Statement of Work (SOW) which describes Orbital's effort to define a technical and affordable solution for accomplishing the NASA Heavy Lift and Propulsion Systems Analysis and Trade Study as defined in the Broad Agency Announcement (BAA NNM10ZDA001K). One primary goal of the study was to analyze multiple HLS architectures and make recommendations on how to develop a system capable of affordably conducting the NASA Design Reference Missions (DRMs) for low Earth orbit, lunar exploration, Near Earth Objects (NEOs), and Mars exploration. Starting with the President's Vision for Space Exploration and the Level 0 and Draft Level 1 requirements supplied by NASA, the study team utilized a system of systems approach to derive scientific, economic, and security goals and objectives for a HLS architecture that supports evolutionary human space exploration activities, with destinations including the Moon, Mars and its environs, near-earth asteroids, and Earth-Moon Lagrange points.

Given Orbital's desire to focus on specific, near-term technology implementation, our study shows a clear connection between NASA's Exploration system requirements and optimal launch vehicle sizing and performance. This connection is particularly strong for the lift-off and injection portions of Heavy Lift missions. As recorded in this document, Orbital's study results emphasize the broader applications of the propulsion and heavy lift technology elements, and identifies the benefits of their incorporation into other U.S. launch systems and for launch applications by other potential government and commercial users.

As discussed in this report, Orbital's study results support NASA in seeking an innovative evolutionary approach that enables human space exploration and the capability to extend human and robotic presence throughout the solar system. In this regard, Orbital examined the trade space of potential heavy lift launch and space transfer vehicle elements with a focus on affordability, operability, reliability, and commonality with multiple end users. A major thrust of our proposed approach is the identification and development of affordable space launch propulsion technologies that will stimulate a more robust exploration program, while also supporting NASA, Department of Defense (DoD), commercial, science, and international partner ventures and related national security needs.

Orbital's study addresses NASA's need for a viable Heavy Lift System (HLS) in a systematic but direct manner. The critical trade studies performed can directly lead to actionable plans with a clear path of execution for propulsion technology and HLS elements development within the 5 year time frame dictated by Congress. The Orbital HLS study results also support the goal of maximizing the use of existing U.S. resources, technology, hardware, and launch vehicles. In support of this study, the collective and demonstrated expertise of Orbital's Advanced Programs Group (APG) and Launch Systems Group (LSG) were combined to converge on a recommended HLS Exploration architecture and to define an affordable and sustainable path to technology development, demonstration, and implementation.

1.3. Scope

This report documents the results from Orbital's HLS Study, as defined in the Statement of Work (SOW), attached as Appendix B.

1.4. Summary of Accomplishments and Work to Date

Included in this Final Report are the results of the activities performed during the course of the HLS study. A summary of the major areas of study emphasis are included below.

HLS Architecture Development

Section 3 of this report outlines the process Orbital implemented in the conduct of the study. The process is based on systems engineering best practices and includes emphasis on:

1. Establishing the Key Decision Attributes
2. Identifying alternative Ground Rules and Assumptions
3. Developing Requirements
4. Formulating candidate Heavy Lift Launch Vehicle architecture concepts
5. Performing trade studies, technology assessments, reliability assessments, performance assessments, operational assessments, and cost assessments

Technology Assessment

Section 4 provides an assessment of key HLS technologies. The technologies assessed include:

- Main engine technologies, propulsion systems, and associated propulsion system elements
- Upper stage engine technologies, propulsion systems, and associated propulsion system elements

- In-space systems and elements including in-space engines, transfer stages, etc.
- Heavy lift launch system elements and hardware including tanks and structures, propellant and pressurization systems, auxiliary propulsion systems, avionics and control systems, and payload shrouds/fairings.
- Ground and launch operations and infrastructure

Orbital also assessed the Technology Development necessary to meet identified preferred HLS configuration and propulsion systems requirements, and quantitatively evaluated their technology status using established metrics.

Performance Assessment

Section 5 provides an assessment of required HLS performance. Each of the Heavy Lift Launch Vehicle (HLLV) concepts was assessed and traded to ensure that the final recommended HLLV concepts were capable of meeting the stated goals of 100 tons to LEO, evolvable to 130 mT.

Reliability Assessment

Section 6 provides an assessment of launch vehicle reliability. Each of the HLLV concepts was assessed to ensure that the final recommendation was capable of meeting the stated Loss of Crew goal of less than 1 in 700.

In-Space Module Assessment

Section 7 provides an assessment of the required In-Space elements. Each of the In-Space Module concepts was assessed to ensure that the final recommended concepts were capable of meeting the NASA Design Reference Missions for low Earth orbit, lunar exploration, Near Earth Objects (NEOs), and Mars.

DRM Assessment

Section 8 provides an assessment of the ability of the candidate HLLVs to meet the NASA Design Reference Missions (DRMs). Each of the HLLV concepts was assessed to ensure that the final recommended concepts were capable of efficiently performing the NASA DRMs for low Earth orbit missions (e.g.; ISS and LEO transfer), Lunar exploration, missions to Near Earth Objects (NEOs), and Mars exploration.

Recommended HLS Architecture

The final recommended Heavy Lift System Architecture is provided in Section 9 of this report. The Heavy Lift System Architecture recommended by Orbital is capable of meeting not only the NASA Design Reference Missions for low Earth orbit, Lunar exploration, Near Earth Objects (NEOs), and Mars explorations, but also meets the schedule and affordability goals established by NASA.

Cost and Affordability Assessment

The Heavy Lift System Architecture cost and affordability assessment is provided in Section 10 of this report. The cost and affordability assessment provides background on the cost estimating approaches utilized by Orbital to predict the non-recurring and recurring costs for the candidate and recommended HLS architectures.

1.5. Applicable Documents

The following documents are applicable to the extent specified herein:

- BROAD AGENCY ANNOUNCEMENT NNM10ZDA001K Amendment 2, Heavy Lift & Propulsion Technology Systems Analysis and Trade Study, Release Date: June 29, 2010.
- Orbital Sciences Corporation Response to BAA (proposal), July 29, 2010.
- Attachment J-1 to Contract NNM11AA13C, Heavy Lift & Propulsion Technology (HLPT) Systems Analysis and Trade Study STATEMENT OF WORK (SOW) for Orbital Sciences Corporation, November 18, 2010.
- Attachment J-2 to Contract NNM11AA13C, Heavy Lift & Propulsion Technology (HLPT) Systems Analysis and Trade Study Data Procurement Document (DPD) 1380 for Orbital Sciences Corporation, November 16, 2010.
- Orbital's Briefing Package for the NASA Heavy Lift Study Technical Interchange Meeting (TIM) #1 (*NASA HLS TIM 1_ORBITAL TM22690_MASTER_FINAL.pptx*), DRD Number: 1380MA-002, Orbital Technical Memo (TM) 22690, posted to NASA PBMA SLS System Analysis and Trade Studies Contracts sharepoint February 22, 2010.
- Orbital's Briefing Package for the NASA Heavy Lift Study Technical Interchange Meeting (TIM) #2 (*NASA HLS TIM 2_ORBITAL TM22690_FINAL_R1_Apr 26 2011.pptx*), DRD Number: 1380MA-002 - Revision 01, Orbital TM-22810, posted to NASA PBMA SLS System Analysis and Trade Studies Contracts sharepoint April 26, 2011.

2. OVERVIEW

Orbital Overview

- Leading Developer and Manufacturer of Small- and Medium-Class Space Systems
  - Three-Decade Record of Reliable, Rapid and Affordable Development and Production
  - Serving Customers in Commercial, National Security and Civil Government Markets
- About 1,000 Satellites and Launch Vehicles Built or Under Contract for Customers
  - 198 Satellites and Space Systems
  - 165 Space and Strategic Launch Vehicles
  - 631 Target Vehicles and Sounding Rockets
- 3,700 Employees and 1.7 Million Square Feet of State-of-the-Art Facilities
- Over $5.6 Billion Total Contract Backlog With Premier Customers
- Revenues of About $1.35 Billion Expected in 2011
- Conservative Balance Sheet With Strong Liquidity

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The Orbital Sciences Corporation (Orbital) is a leading developer and manufacturer of small and medium class space systems. Orbital has three decades of demonstrated reliable, rapid and affordable development and production experience, serving customers in Commercial, National Security and Civil Government markets. Over the course of these three decades Orbital has approximately 1,000 satellites and launch vehicles either built or under contract for future delivery. Of those 1,000 systems, 198 are satellites and space systems, 165 space and strategic launch vehicles and 631 target vehicles and sounding rockets. Orbital employs a high-caliber engineering workforce with half of its 3700 person workforce made up of Engineers and Scientists and 1.7million square feet of state-of-the-art R&D and Production facilities located in Virginia, Arizona and Maryland.

2.1. Study Team

Orbital Launch Systems Group

John Steinmeyer
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Aerojet
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SpaceWorks Engineering
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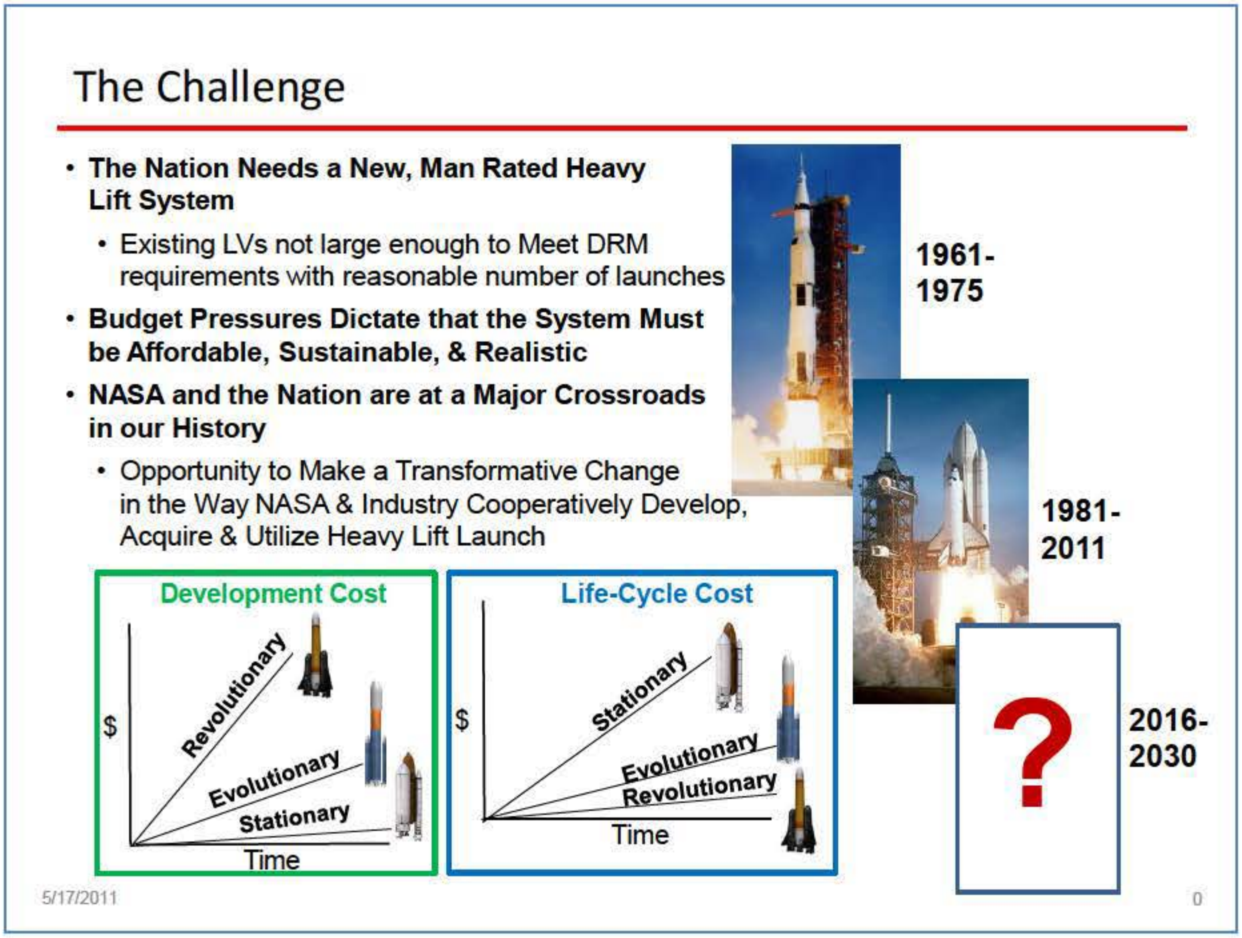
Spincraft

2.2. The Heavy Lift Challenge and Solution



Orbital supports NASA's intention of developing a new man-rated Heavy Lift Launch Vehicle (HLLV) and a Heavy Lift System (HLS) that satisfies a broad range of providers and users both inside and outside of NASA. Orbital also strongly concurs with NASA's desire to advance liquid chemical propulsion technologies to support a more affordable and robust U.S. space transportation industry. Orbital recognizes the budgetary pressures that dictate that the HLS be Affordable, Sustainable, and Realistic in the time frame dictated by Congress. Orbital believes that a transformative change to a more cooperative NASA and Industry paradigm is essential to affordably develop a heavy lift capability necessary for extending the U.S. presence within the solar system, regaining the preeminence in space launch that our Nation once commanded.

The Solution

- **Flexible Heavy Lift Architecture Configuration**
  - Intermediate performance to meet Lunar and most NEO requirements (consistent with Congressionally Mandated guidelines)
    - Affordably & rapidly developed
    - Lower performance configuration for MPCV and large ISS Cargo
    - Derivative Growth variants to better meet Mars Exploration DRMs (~2025)
  - Right-sized, Affordable, Realistic configurations for specific missions
- **Capitalize and Leverage Industry-Wide Advances in Launch Vehicle Technology**
  - Targeted Investment in specific areas to "close gaps"
- **HLS Configurations that support an Affordable and Sustainable Life Cycle Approach**
  - Constrain HLLV parameters to largely Utilize Existing Infrastructure
  - Leverage industry advances in Lower-Cost Technology
  - Commonality of components and elements to Leverage Economies of Scale
  - Applicability of elements to Other Vehicles in the U.S. Fleet
- **Innovative, Commercial Contracting and Program Management**

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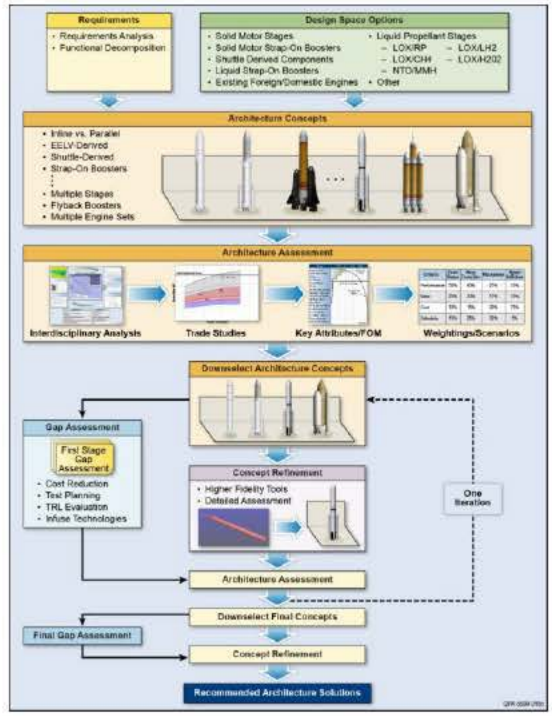
After considerable analysis and study, Orbital believes that the best solution for the Heavy Lift System is to develop an architecture that can be affordably and rapidly developed with adequate performance to meet NASA's near term needs for LEO missions. Furthermore, this can be done with a design that is sufficiently flexible to meet the mid-term Lunar mission up-mass requirements, and eventually evolvable to meet the most demanding Near Earth Object and Mars exploration missions in the 2025 or later time frame. This approach provides a system that is right-sized, affordable and realistic for the current NASA budget, yet enables NASA to continue to develop the HLS capabilities and on-ramp new technologies in a sustainable manner. This approach also ensures that the HLS employs common technologies and elements that can be used by multiple launch vehicle providers, and Government entities, further reducing life-cycle costs.

3. HLS ARCHITECTURE DEVELOPMENT

Orbital HLS Architecture Study Process

- **Orbital follows a Top-down, Goal-driven Study Approach to Develop Optimal, Robust Heavy Lift System Architectures**
  - Capture and synthesize requirements
  - Explore design space for architecture concepts
  - Perform assessments and sensitivities against weighted system attributes
  - Define candidate architecture concepts for refinement and gap assessment
- **Concept Evaluation Utilizes Orbital's Integrated Systems Analysis Processes and Leverages Government, Industry, and Proprietary Tools**
- **Existing and Developing Technologies Incorporated into Trade Space to Supplement Capability Gaps and Reduce Risks Associated with Schedule and Cost**



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Orbital implemented a goal-driven process to develop optimal, robust HLS architectures. A top-level roadmap of Orbital's approach to performing the HLS architecture study is provided in the figure above. This architecture development process has been successfully proven on previous NASA studies. Requirements analysis and functional decomposition initiated the process, in parallel with identifying options for each of the HLS elements. After the driving requirements were synthesized, various vehicle configurations, technologies, propellants, and components were assessed and assembled into candidate architectures. Each of these architectures was sized to meet the system requirements and then assessed against weighted system attributes to determine the most effective architectures. Gap assessments were performed to determine technologies that must be developed and to identify areas where significant cost reductions can be realized. While the gap assessments were being performed, the selected architectures were refined and further optimized. These two efforts converged at the second downselect where the best architectures were selected based on the evaluation of system attributes and an assessment of the technology and cost gaps. A final refinement of the best architectures was performed along with a final gap assessment.

3.1. Key Decision Attributes and Alternative Ground Rules

Recommended Key Decision Attributes

Orbital

1. **Affordable:**
   - HLS DDT&E Costs - must be within the budget limits set by Congress (i.e.; $11.5B)
2. **Sustainable:**
   - HLS Recurring Launch Costs - must be within the annual budget limits (i.e.; $1.8B)
   - HLS Life Cycle Costs must be Sustainable over the next 20 years
3. **Realistic:**
   - HLS designs, systems, subsystems, components, and operations must be realistically achievable in the time 2016 frame and not rely on budget increases, unplanned technology advances, or impractical programmatic shenanigans to succeed
4. **Performance:**
   - HLS Initial MTO – *Initial* performance of Heavy Lift Launch Vehicle must support reasonable and actionable set of near-term (i.e.; 2016-2020) DRMs (i.e.; Crew to ISS, GTO missions, Heavy cargo to ISS and LEO, etc.)
   - HLS Evolved MTO – *Evolved* performance of Heavy Lift Launch Vehicle must support reasonable and actionable set of mid-term (i.e.; 2021-2029) DRMs (i.e.; Lunar Flyby, Lunar Cargo, Lunar Crew Landing, NEO Low, etc.)
   - HLS Mature MTO – *Mature* performance of Heavy Lift Launch Vehicle must support reasonable and actionable set of long-term (i.e.; 2021-2029) DRMs (i.e.; NEO High, Mars Flyby, Mars Cargo, Mars Crew Landing, etc.)

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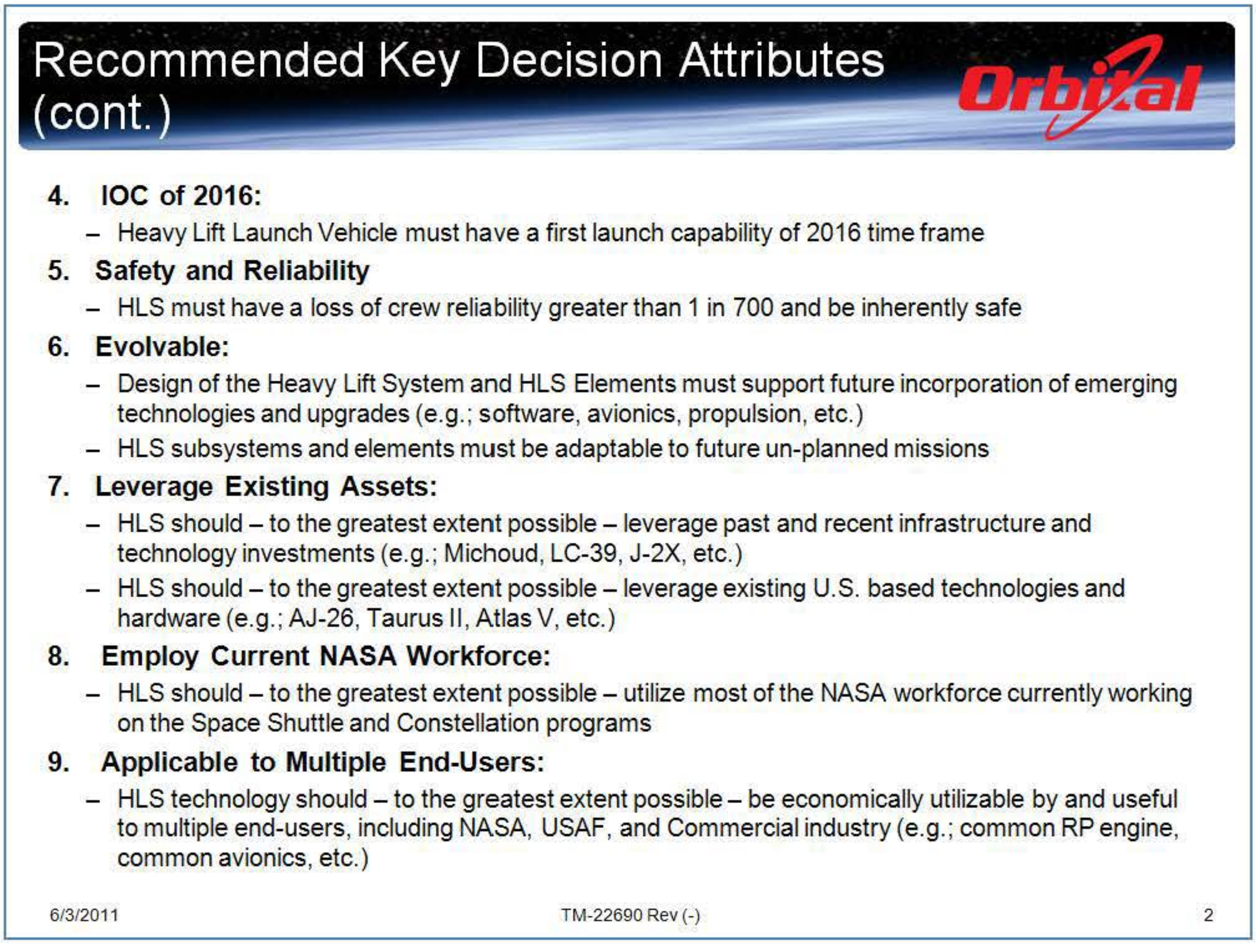
As defined in the SOW, Orbital developed a set of Recommended Key Decision Attributes (KDA) that was utilized to evaluate each of the candidate HLS architectures and Heavy Lift Launch Vehicle (HLLV) concepts. Affordability was identified as the primary KDA since it is a both a driving requirement and a practical constraint due to the funding limits imposed by Congress. Affordability was followed closely by sustainability in terms of annual and life cycle costs, and ensuring that the HLS architecture was realistic in terms of its achievability without additional infusions of funds or reliance on unrealistic expectations of future technical breakthroughs or program "sleight of hand." KDAs were also established to meet the performance goals of 100 tons to LEO, evolvable to 130 mT. After affordability and performance, schedule is a key driving requirement and therefore any architecture that can meet the IOC of 2016 should score higher than those that do not. Another important KDA that was used to evaluate the various candidate HLS architectures was ensuring that the resulting system was safe and reliable.



Additional HLS KDAs identified by the Orbital study team included meeting the schedule for an initial launch within the 2016 time frame; ensuring that the system be inherently safe and reliable; and, making certain that the HLS was evolvable since any architecture must by definition provide NASA with the flexibility to accommodate a broad range of DRMs. This sixth KDA also encompassed the need for the HLS system to have the ability to adapt to future un-planned missions, unknown schedules, and funding profiles over the next 25 years or longer. Maximum leveraging of existing assets both owned by NASA (e.g.; Michoud, KSC, J-2x, Shuttle heritage infrastructures and hardware, etc.), and technology currently in use by U.S. government and commercial launch industry (e.g.; AJ-26 engines, existing launch vehicles and hardware, etc.), employment of the NASA workforce, and applicability of the HLS to multiple end users were also considered to be important KDAs. The last KDA is supportive of the first KDA (affordability) since commonality leads to lower overall life cycle costs as a result of economies of scale amortizing development and procurement costs among several end-users.

Alternative Ground Rules & Assumptions

1. **Affordability:**
   - HLS DDT&E Costs must conform to NASA's proposed $1.8B annual funding profile from 2012-2016
   - Alternate funding scenarios can be considered to test sensitivity of funding on affordability and other KDAs (e.g.; performance, schedule, etc.)
2. **Design for Evolution:**
   - The reasonable and actionable set of near-term (i.e.; 2016-2020) DRMs (i.e.; Crew to ISS, GTO missions, Heavy cargo to ISS and LEO, etc.) require 50 tons to LEO or less
   - The reasonable and actionable set of mid-term (i.e.: 2021-2029) DRMs (i.e.; Lunar Flyby, Lunar Cargo, Lunar Crew Landing, NEO Low, etc.) require 100 – 120 tons to LEO or less
   - A reasonable and actionable set of long-term (i.e.; 2021-2029) DRMs (i.e.; NEO High, Mars Flyby, Mars Cargo, Mars Crew Landing, etc.) needs to be developed that doesn't require a huge leap in performance from 120 mT to 160 mT and greater
3. **Reliability:**
   - Single engine out capability is a viable method to achieve LOC goal for clustered engine configurations

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Similar to the KDAs, the HLS Alternative Ground Rules and Assumptions (GR&A) identified by the Orbital study team put Affordability at the top of the list, because all things being equal, establishing a ground rule that ensured the heavy lift launch system was affordable and could be procured within the available annual funding became the single greatest constraint to any architecture conceived. Although various funding scenarios were considered as an alternative ground rule, as the study progressed and the budgetary battles increased in Washington D.C. it became obvious that Congress would not likely approve any funding increases, and might even reduce funding (which actually happened before the study was complete). Although Design for Evolution has a companion KDA (evolvable), it really is an alternative ground rule since evolution was never specifically identified as a requirement by NASA. This last alternative ground rule also allowed Orbital to consider HLS approaches that might not initially meet the performance KDA. Adding engine out capability was alternative assumption that increased reliability of the multi-engine first stage concepts. Further, because no existing engine technology is capable of lifting the required mass, engine out became one way to ensure the loss of crew requirement could be met with the multi-engine concepts.

Alternative Ground Rules & Assumptions (Cont.)

6. **Domestic System**
   - The primary elements and components of the HLS must be domestically developed and produced
7. **Foreign Participation :**
   - Foreign participation & “like-kind” exchanges are a viable method of achieving capabilities that are not critical path
8. **Commercial Contracting:**
   - Commercial contracting approaches (e.g.; COTS like) are a viable method of procuring elements of the HLS
9. **Cost Estimating**:
   - NAFCOM is not necessarily the best tool to perform cost estimates for a modern, commercially procured launch system
   - Alternate costing methods can be used that are based on known best commercial practices so long as they are correlated to an actual LV development and production program

(b) (4) [redacted]

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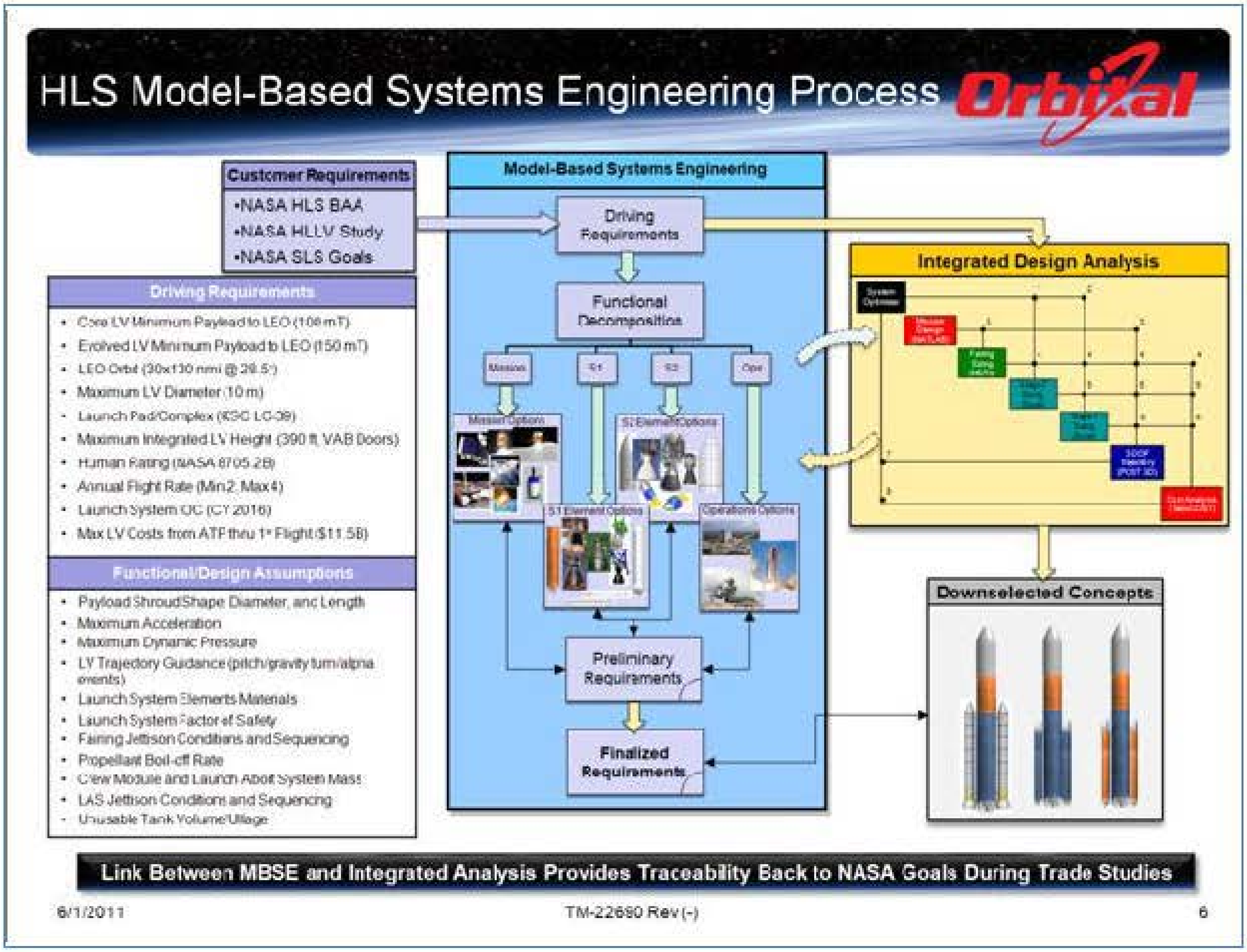
Additional HLS alternative Ground Rules and Assumptions (GR&A) identified by the Orbital study team included Domestic production of critical systems and components, as well as the ability to utilize foreign partnerships for hardware and capabilities not on the HLS critical path.

Commercial Contracting and procurement approaches and alternate Cost Estimating approaches and tools were also identified by the Orbital team.

3.2. Requirements Development Using MBSE



Requirements and objectives provided from NASA served as the basis for the development of the driving requirements and preliminary functional decomposition to each of the HLS elements. Functional/design assumptions serving as inputs into the integrated design analysis were iterated upon during the architecture development to refine the requirements and produce design solutions that satisfied the NASA HLS Goals. Model-Based Systems Engineering (MBSE), the formalized application of modeling to support system requirements, design, analysis, verification and validation activities,[1] was used at the beginning in the conceptual phase of the Heavy Lift study. The motivation for using an MBSE approach was to avoid the typical document-centric systems engineering approach and employ one that was more model-centric, more useful, and more efficient. Requirements were entered directly into the modeling tool and functional flow block diagram models were generated in the tool and linked to the requirements. Finally representative architecture diagrams and system models were developed and also linked back to the functional blocks which automatically linked the associated architecture elements to the appropriate requirement(s), thereby integrating the requirements with the end-to-end system design. Appendix F documents the results of using this methodology on the HLS study, identifies the pros and cons of employing MBSE vs. a more traditional systems engineering approach, and provides examples of the HLS system models developed in the tool.

[1] INCOSE SE Vision, INCOSE-TP-2004-00402, Sep 07

3.3. HLS Architecture Development Approach

HLS Integrated Systems Analysis Model

- **Orbital's Integrated Systems Analysis Model within Phoenix Integration ModelCenter Collaborative Design Environment**
  - Model provides data transfer between tools & system-level MDO, and enforces constraints
- **Preliminary Reliability and Cost Analyses Performed Outside Integrated Model**
  - Refined cost and reliability analyses included into integrated analysis during concept refinement

(b) (4)

[redacted]

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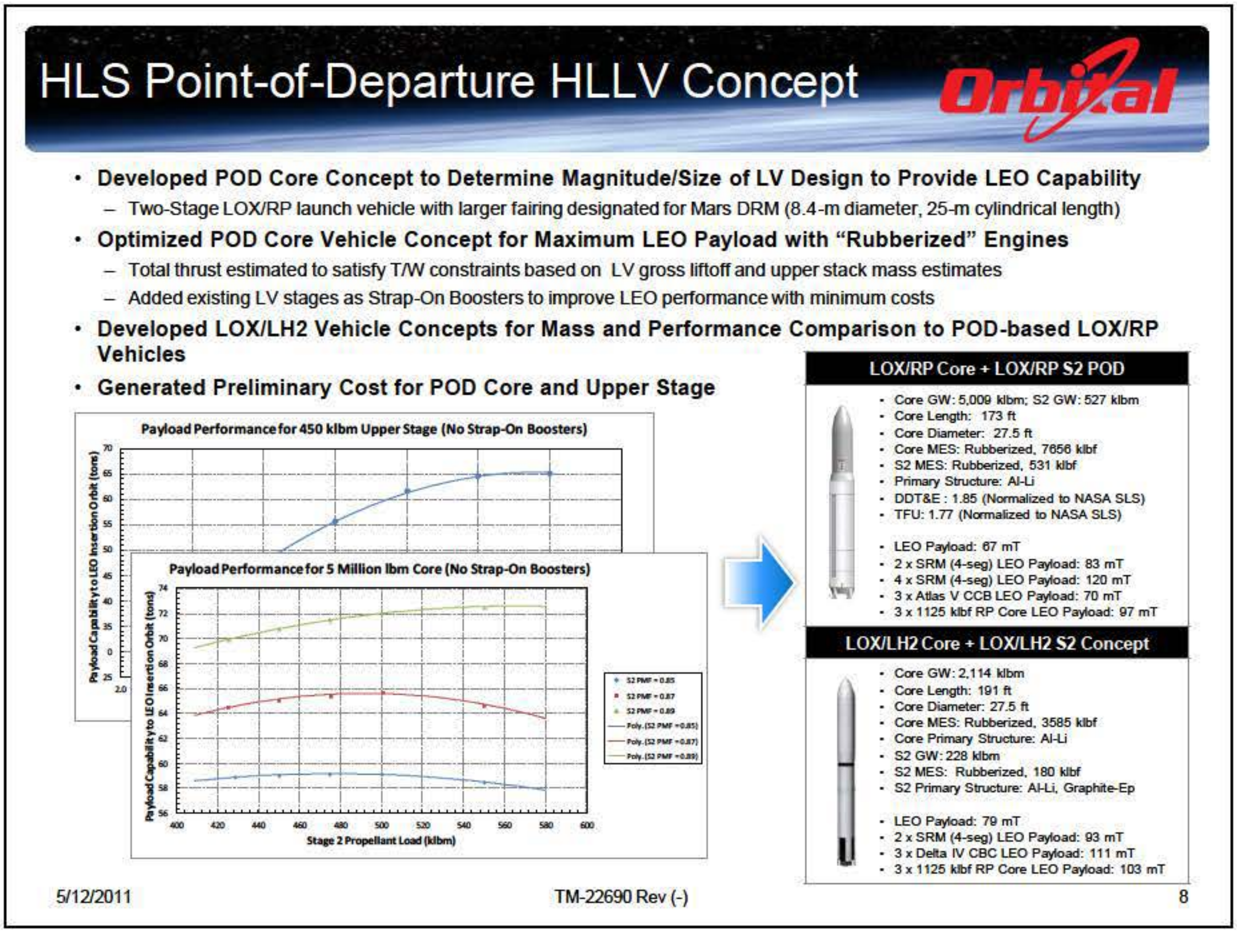
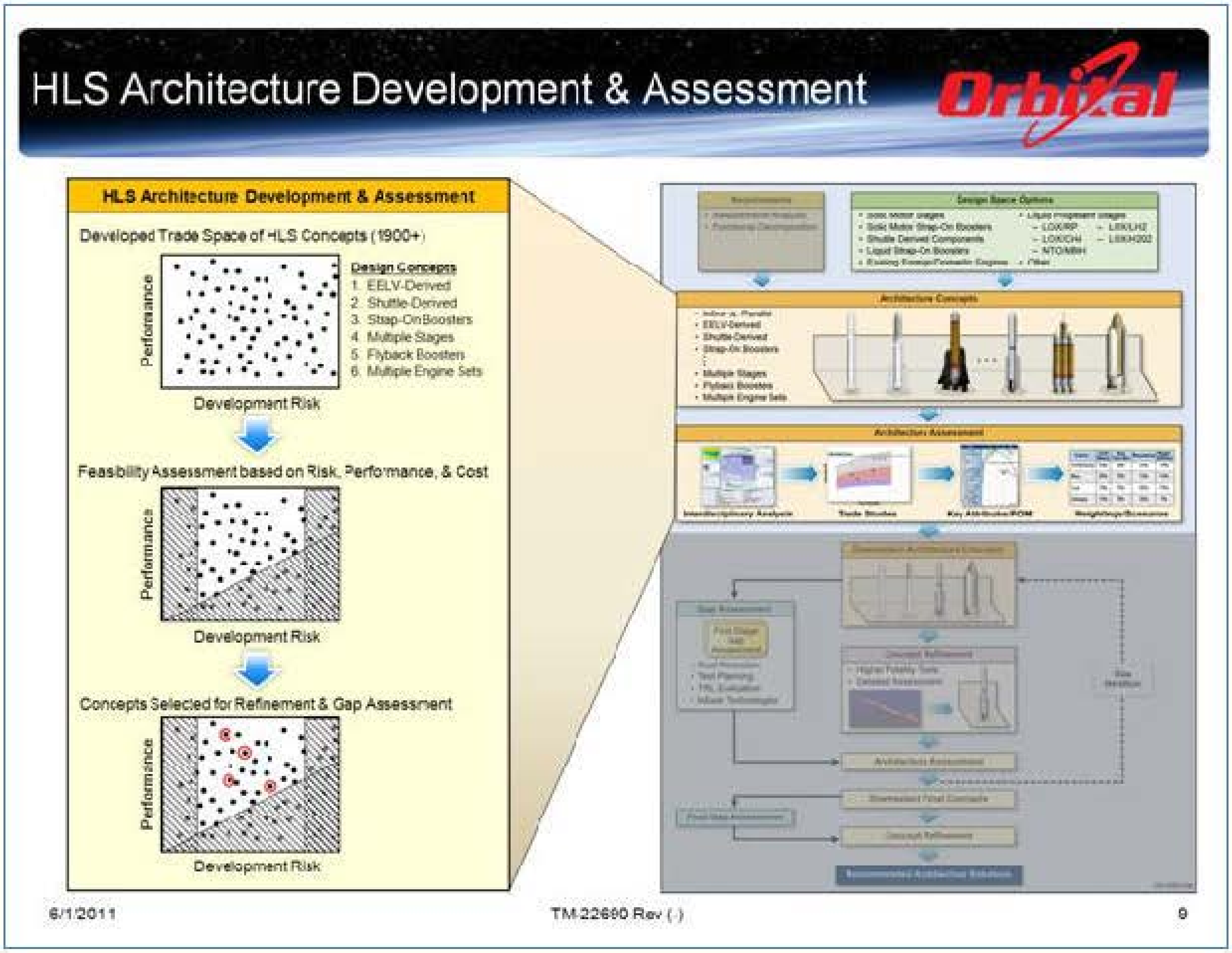
Following the requirements development, Orbital conducted a comprehensive architecture study that explored the entire Heavy Lift System trade space. The study relied primarily on quantifiable results for decision making rather than qualitative judgments. In doing so, several top level objectives related to risk, reliability, affordability, and performance were kept at the forefront during trade study execution. The system trade studies and analyses leveraged Orbital's Integrated Systems Analysis Process, which connects Orbital's existing suite of analysis tools inside an integrated design environment to apply multidisciplinary design optimization techniques necessary to fully explore the design space. Orbital has extensive experience using these analysis tools on architecture study programs such as Space Transportation Architecture Studies (STAS), Space Launch Initiative (SLI), Orbital Space Plane (OSP), Concept Evaluation and Refinement (CE&R), Hybrid Launch Vehicle, and similar activities. Many of the government-developed tools (e.g., POST, NAFCOM) have a long and successful history and provide accurate results sufficient for rapid exploration of the design space. Where no government or commercial code exists, Orbital has developed its own line of tools (e.g., Sizing, Mass Properties, and

Reliability) that have been validated over years of successful use on other programs. To facilitate data transfer between tools, decrease iteration times, and provide system-level optimization functionality, Orbital integrated the individual analyses inside the Phoenix Integration ModelCenter® design environment. In addition to providing multidisciplinary design optimization capability, ModelCenter® enforces constraints on the candidate architectures and ensures the final candidates meet the criteria specified in the requirements.



At the start of the HLS study, a Point-of-Departure (POD) launch vehicle was developed to estimate the overall magnitude of size for a launch vehicle that provided an extremely large payload capability to LEO. The POD was a two-stage LOx/RP launch vehicle sized to maximize LEO payload capability without the use of strap-on boosters. Conceptual "rubberized" engines were used with the POD as initial engine estimates based on the average thrust and specific impulse of similar LOx/RP engines. Total thrust of the POD was estimated to satisfy the thrust-to-weight ratio (T/W) constraint of 1.2 based on the launch stack gross mass at liftoff. Several trade studies were conducted for the POD vehicle to examine the sensitivity of the LEO payload performance with stage propellant mass fraction (PMF), stage propellant load, and stage gross mass. Results of the Stage 1 Core gross mass trade study indicate the maximum LEO performance peaks at a gross mass of approximately five million pounds-mass (lbm). Assuming a Stage 2 PMF of 0.85 and Stage 1 Core gross mass of five million lbm, the maximum LEO performance LEO performance peaks at a Stage 2 propellant load of approximately 175,000 lbm. Preliminary development (DDT&E) and first-unit production (TFU) costs were estimated for the POD launch vehicle. These costs were then normalized to the NASA SLS costs approximated from the NAFCOM based cost model. A LOx/LH2-based launch vehicle was also developed for mass and performance comparison to the LOx/RP-based POD launch vehicles.



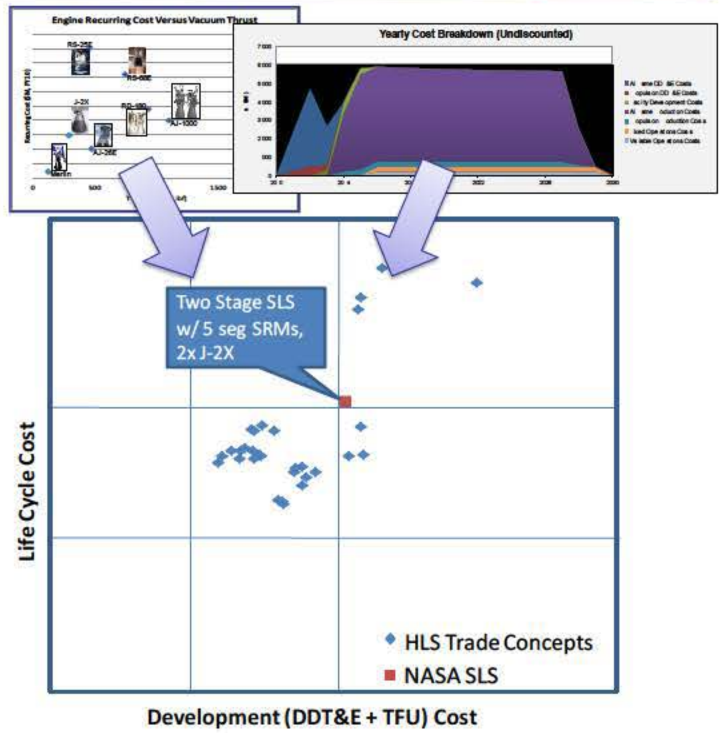
Various vehicle configurations, technologies, propellants, and components were assessed and assembled into candidate architectures. Each of these architectures was sized to meet the system requirements and then assessed against weighted system attributes to determine the most effective architectures. To fully explore the design space, Orbital traded characteristics such as, but not limited to, the following:

- Liquid vs. Solid vs. No Strap-On Boosters
- Number of Boost Vehicle Stages
- Parallel vs. Serial Staging
- Reusability and Commonality
- Domestic vs. Foreign Design Heritage

Over 1,900 concepts were developed based on the available options within the trade space. Assessments based on risk, performance, reliability, and affordability (development cost, production cost) were conducted to reduce the trade space to the feasible concepts with regards to the defined requirements and study objectives. Robust concepts were selected from the narrowed trade space for further refinement and gap assessment.

Cost Analysis Approach

- **Orbital's Cost Analysis based on Development and Production Costs for a Wide Range of Existing and To-be-Developed Elements**
- Two Key Aspects To Affordability
  - Affordability of developing the system now
  - Sustainability of the system throughout its lifecycle
- **Each Concept Under Consideration was Assessed with a Three-pronged Approach to Cost Estimation**
  - Validation of historical cost data
  - Refinement of parametric cost tools
  - ROM estimates from technology vendors
- **Available Cost Data Provides Basis for Cost Comparison**
  - Total Development Cost = DDT&E + TFU (Congressionally constrained)
  - Life Cycle Cost – seek to minimize



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Orbital's initial cost analysis approach based on looking at concepts that developed a system within the cost constraints provided by the NASA funding availability, as well as minimizing lifecycle costs to ensure the long-term affordability of the system. Cost modeling is based on comparison to applicable historical systems, refinement of available and new cost modeling tools, and the incorporation of estimates from vendors.

3.4. HLS Architecture Exploration and Downselect

HLS Lowest Risk Trades Assumptions

- **Trade Space Reduced to 30+ Concepts Based on Engineering Judgment and Qualitative Risk Assessment**
- **Eliminating :**
  - EELV-class cores stages as core options due to cost and performance
  - Un-stretched ET due to performance
  - Options without boosters due to the large number of main engines required to achieve sufficient T/W
  - All but the lowest risk booster options
- **Higher Risk Option Added:**
  - RS-68 (w/ air-light capability) Upper Stage added to compare against J-2X
- **Further trade space options under consideration:**
  - RS-25E w/ air-light capability
  - AJ-26 & AJ-26X
  - New 1M lbf LOX/RP-1 Engine
  - Core Stage Diameter
  - Upper Stage Diameter

LOX / LH2 Core

Concept	Core	Core Engine	Booster	Second Stage Engine	Risk Score
512	Stretched ET	RS-68	4 Seg SRM	J-2X (x1)	6
518	Stretched ET	SSME (Expendable)	4 Seg SRM	J-2X (x1)	6
602	Stretched ET	RS-68	Delta IV CBC	J-2X (x1)	6
608	Stretched ET	SSME (Expendable)	Delta IV CBC	J-2X (x1)	6
632	Stretched ET	RS-68	Atlas V CBC	J-2X (x1)	6
638	Stretched ET	SSME (Expendable)	Atlas V CBC	J-2X (x1)	6
752	Stretched ET	RS-68	4 Seg SRM	J-2X (x2)	6
758	Stretched ET	SSME (Expendable)	4 Seg SRM	J-2X (x2)	6
842	Stretched ET	RS-68	Delta IV CBC	J-2X (x2)	6
848	Stretched ET	SSME (Expendable)	Delta IV CBC	J-2X (x2)	6
872	Stretched ET	RS-68	Atlas V CBC	J-2X (x2)	6
878	Stretched ET	SSME (Expendable)	Atlas V CBC	J-2X (x2)	6
32	Stretched ET	RS-68	4 Seg SRM	RS-68	7
38	Stretched ET	SSME (Expendable)	4 Seg SRM	RS-68	7
122	Stretched ET	RS-68	Delta IV CBC	RS-68	7
128	Stretched ET	SSME (Expendable)	Delta IV CBC	RS-68	7
152	Stretched ET	RS-68	Atlas V CBC	RS-68	7
158	Stretched ET	SSME (Expendable)	Atlas V CBC	RS-68	7

LOX / RP Core

Concept	Core	Core Engine	Booster	Second Stage Engine	Risk Score
530	Stretched ET	RD-180	4 Seg SRM	J-2X (x1)	6
620	Stretched ET	RD-180	Delta IV CBC	J-2X (x1)	6
650	Stretched ET	RD-180	Atlas V CBC	J-2X (x1)	6
770	Stretched ET	RD-180	4 Seg SRM	J-2X (x2)	6
860	Stretched ET	RD-180	Delta IV CBC	J-2X (x2)	6
890	Stretched ET	RD-180	Atlas V CBC	J-2X (x2)	6
50	Stretched ET	RD-180	4 Seg SRM	RS-68	7
140	Stretched ET	RD-180	Delta IV CBC	RS-68	7
170	Stretched ET	RD-180	Atlas V CBC	RS-68	7
1490	Stretched ET	RD-180	4 Seg SRM	AJ-26	7
1580	Stretched ET	RD-180	Delta IV CBC	AJ-26	7
1610	Stretched ET	RD-180	Atlas V CBC	AJ-26	7

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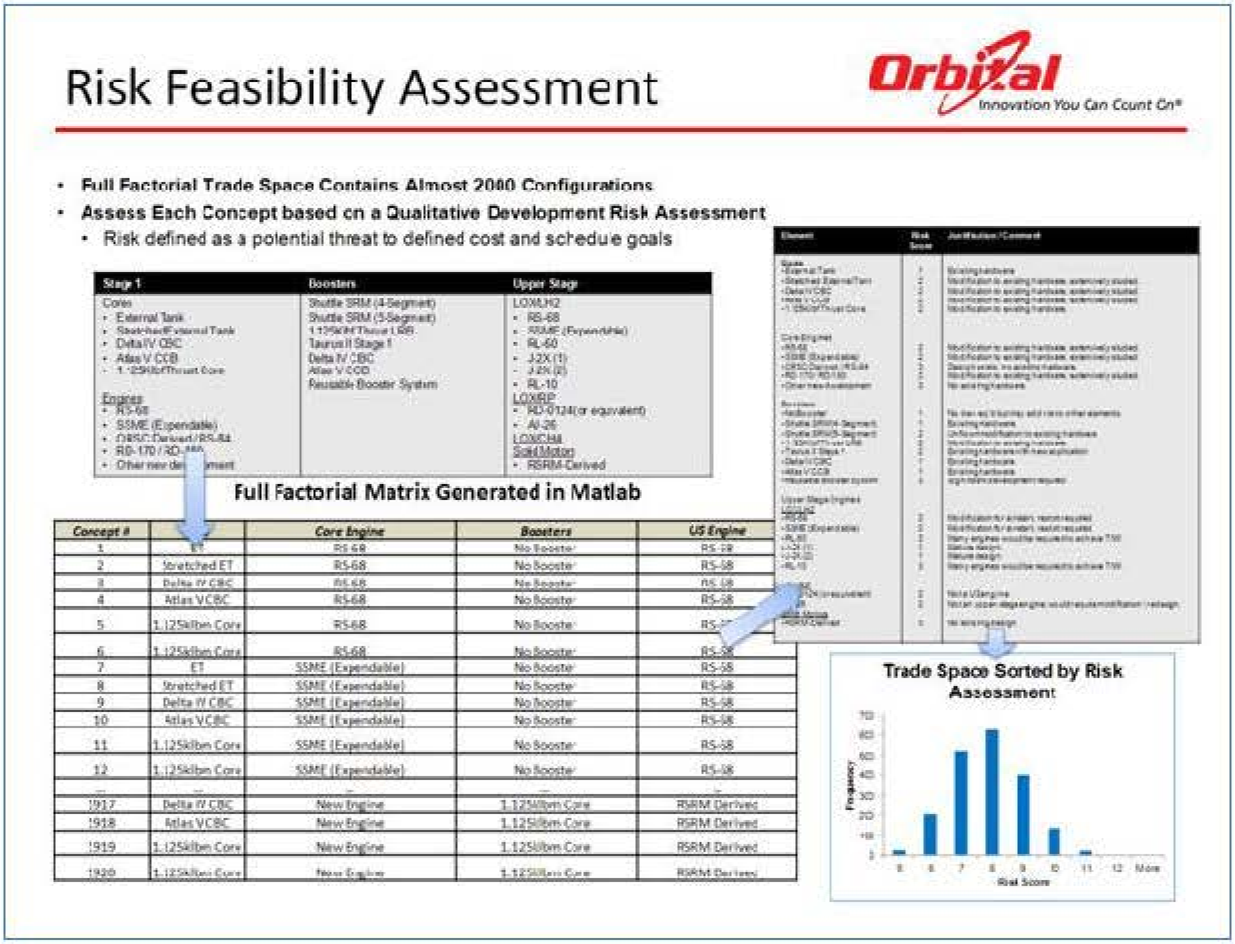
A preliminary assessment of a large design space was conducted. The configurations that were perceived as lowest risk were selected for further study. These systems are viewed as lower risk in this context because they make maximal use of existing structures and propulsion system elements. All concepts that do not utilize boosters were eliminated due to the large number of main engines required for sufficient liftoff T/W. Other options that were eliminated from the design space include those utilizing clusters of EELV cores on the core stages due to their relative cost, and the overall vehicle performance. The RS-68 with air-light capability was considered as an alternative to the J-2X for an upper stage engine. While this would have the advantage of engine commonality with the first stage for concepts utilizing that engine on the first stage, it has the disadvantage of additional development cost and risk, as well as an impact to upper stage performance due to the additional weight of that engine.



Based on the defined configuration options, a full factorial matrix of design solutions was created. Each option in this matrix was then mapped through risk scores (high, medium, and low) for each element of the configuration (core engine, upper stage engine, booster, core structure). The risk scores of the entire trade space are shown. The lowest concepts in the trade space were then brought forward for further analysis and refinement.

Feasibility Assessment Approach

Orbital

- **Feasibility Assessment Conducted for 30+ Low Risk Trade Concepts based on Performance, Development Cost, Life Cycle Cost, & Propulsion System Reliability**

Attribute	Figures of Merit	Feasibility Assessment
Performance	Payload Mass	• Sized Core and Upper Stage based on payload mass, propulsion parameters, and thrust-to-weight (T/W) – T/W at Liftoff: 1.2; T/W at Upper Stage Ignition: 0.8 – Validated Number of Engines Required to Match Total Thrust at Liftoff Fits on Core 27.5-ft Diameter • Performed POST 3DOF Trajectory Analysis to Estimate HLS Concept Payload Capability to LEO Orbit – LEO Orbit: 30 x 130 nmi @ 51.6°; Orbit Insertion at 130 nmi • Iterated on Payload Mass Between Sizing and Trajectory Analysis to Obtain Converged HLS Solution
Development Cost	DDT&E, TFU	• Development Cost estimated for Trade Space Concepts using NAFCOM08 • DDT&E + TFU Costs based on subsystem weights, booster/engine actual costs, and programmatic wraps • Programmatic wraps assumptions: Fee = 12%; Program Support = 12%, Contingency = 30%; Vehicle Level Integration = 8%
Life Cycle Cost	LCC, $/kg	• Spreads development costs and manufacturing costs as appropriate • Includes facility modification costs • Includes fixed annual operations costs and variable (per flight) operations costs
Reliability	LOM, LOV, LOC	• Reliability Block Diagram (RBD) based analysis of propulsion system elements to estimate propulsion system reliability • Examined sensitivity of single engine-out capability on propulsion system reliability

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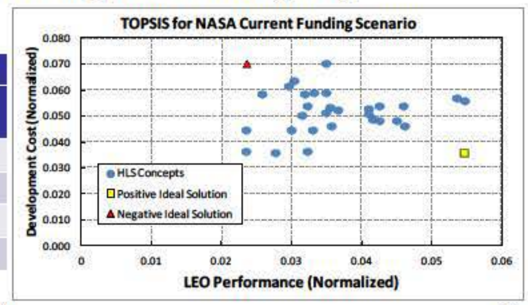
After reducing the trade space to 30 architecture concepts based on the qualitative risk assessment, a feasibility assessment determined how each concept compared to the rest based on performance, development cost, life cycle cost, and propulsion system reliability. The results of the feasibility assessment also provided a basis for the relative comparison of the key decision attributes between the low risk concepts.

HLS Concept Downselect Process: Preliminary TOPSIS Results

Orbital

- **HLS concepts downselected based on technical, cost, and programmatic FOMs/attributes with weightings**
- **Downselect performed by ranking HLS concepts using Technique for Order Preference by Similarity to Ideal Solution (TOPSIS)**
  - TOPSIS is Multi-Attribute Decision Making (MADM) technique that ranks concepts based on their distance to FOM-based ideal solution
  - Normalized FOM Values Generated By TOPSIS Provide Relative Comparison Of Each HLS Concept To The Positive/Negative Ideal Solution
- **Downselect based on Technical and Cost FOMs since generated "Low Risk" Concepts most capable of meeting NASA programmatic/schedule constraints**
- **Attribute weightings show sensitivity of concepts rankings under different scenarios (political, economic, technical)**
- **Selected HLS robust concepts that remain top ranked regardless of weighting**

	Weighting Scenarios for NASA Funding 2012-2016			
Scenarios	Current Funding	Current Funding +10%	Current Funding -5%	Current Funding -10%
Performance	20%	20%	20%	14%
Reliability	25%	25%	25%	25%
Development Cost	28%	32%	23%	23%
Life Cycle Cost	27%	23%	32%	38%



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The HLS concepts were assessed against sets of key decision attributes. For each set of weighted attributes, individual optimum HLS architecture concepts emerged and were selected for additional refinement. The concept downselect was performed by ranking the HLS concepts using the Technique for Order Preference by Similarity to Ideal Solution (TOPSIS) method. TOPSIS is a Multi-Attribute Decision Making (MADM) technique that ranks concepts based on their distance to the attribute-based positive ideal solution. The closer the concept is to the positive ideal solution, the higher the ranking. TOPSIS was performed using all of the HLS low risk architecture concepts for each set of weighted attributes. Competing concepts were evaluated by ranking the key decision attributes with weighting factors to indicate the importance each attribute had on the overall system architecture. Alternative weighting scenarios were developed to examine the sensitivity of the architecture concepts to variations in the relative attribute importance based on NASA Funding. The weightings used for the architecture assessment were based on a survey of experts within Orbital in relevant technical, programmatic, and cost disciplines. The importance of reliability remained the same for each scenario in that, regardless of the funding scenario, the reliability required (Loss of Mission, Loss of Crew) was not compromised. For the current and +10%/-5% funding scenarios, the objective was to maintain the necessary LEO performance (i.e. payload capability) with adjustments in funding. Funding was a primary concern in the scenario of funding -10%, and performance was slightly reduced to accommodate the increased focus on affordability. For the current and +10%/-5% funding scenarios, the combined percentage of the development and life cycle cost was 55% to show the importance of system costs. Current Funding development costs are slightly above life cycle cost. As funding was increased, the importance of development cost increased. As funding was reduced, the importance of life cycle cost to operate, produce, and maintain the system increased. For the current -10% scenario, the increased focus on affordability increased the importance of life cycle costs (while sacrificing some of performance). Each qualitative NASA Funding weighting scenario tested the sensitivity of the architecture concepts to the variations in ranking. Evaluation of the architecture concepts with the different weightings scenarios resulted in alternate top-ranked candidate architectures as shown on the next page.

HLS Concept Downselect Process: Preliminary TOPSIS Results

Orbital

- **Top HLS LOX/RP And LOX/LH2 Core Concepts Remain Top Ranked Across Qualitative NASA Funding Weighting Scenarios**
- **Probabilistic Variation Of FOM Weightings In Work To Validate Qualitative Weighting Scenarios**
  - Quantitative Approach Examines Sensitivity Of Attributes Weightings On Concept Rankings
  - Performed With Uniform Distributions Over Limited FOM Ranges

HLS Rankings	NASA Current Funding	NASA Current Funding +10%	NASA Current Funding -5%	NASA Current Funding -10%
1	RS-68 (x4) Core J-2X (x2) S2 4 seg RSRM (x2)	RS-68 (x4) Core J-2X (x2) S2 4 seg RSRM (x2)	RS-68 (x4) Core J-2X (x2) S2 4 seg RSRM (x2)	RD-180 (x6) Core J-2X (x2) S2 Atlas V CCB (x2)
2	RD-180 (x6) Core J-2X (x2) S2 Atlas V CCB (x2)	RD-180 (x6) Core J-2X (x2) S2 Atlas V CCB (x2)	RD-180 (x6) Core J-2X (x2) S2 Atlas V CCB (x2)	RS-68 (x4) Core J-2X (x2) S2 4 seg RSRM (x2)
3	RD-180 (x6) Core J-2X (x2) S2 Delta IV CBC (x2)	RD-180 (x6) Core J-2X (x2) S2 Delta IV CBC (x2)	RD-180 (x6) Core J-2X (x2) S2 Delta IV CBC (x2)	RS-68 (x4) Core J-2X (x2) S2 Atlas V CCB (x2)
4	RS-68 (x5) Core J-2X (x2) S2 Delta IV CBC (x2)	RS-68 (x5) Core J-2X (x2) S2 Delta IV CBC (x2)	RS-68 (x5) Core J-2X (x2) S2 Delta IV CBC (x2)	RS-68 (x5) Core J-2X (x2) S2 Delta IV CBC (x2)
5	RD-180 Core RS-68 S2 Atlas V CCB (x2)	RD-180 Core RS-68 S2 Atlas V CCB (x2)	RD-180 Core RS-68 S2 Atlas V CCB (x2)	RD-180 (x6) Core J-2X (x2) S2 Delta IV CBC (x2)

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Several of the top-ranked LOx/RP and LOx/LH2 Core concepts remained top-ranked across the qualitative NASA Funding weighting scenarios:

- The LOx/LH2 RS-68 based core + four-segment RSRM ranked #1 for three of the four scenarios, and second in the -10% funding scenario.
- The LOx/RP RD-180 based core + RP LRBs ranked #2 for three of the four scenarios, and first in the -10% funding scenario.
- The LOx/RP RD-180 based core + LH2 LRBs ranked third for the current, +10%, and -5% funding scenarios.

(b) (4)

[redacted]

Orbital also used a quantitative approach to examine the sensitivity of the attribute weighting on the concept rankings. Attribute weightings were varied probabilistically via Monte Carlo simulation to comprehensively explore the design space. The Monte Carlo analysis was performed with uniform distributions on the attribute weightings. Ranges of the attribute weightings were limited to explore more feasible weighting scenarios within the trade study design space. The LOx/LH2 RS-68 + four-segment RSRM ranked #1 for approximately 73% of the 1,000 Monte Carlo runs. The LOx/RP RD-180 based core + RP LRBs ranked #1 for approximately 9% and ranked #2 for approximately 74% of the 1,000 Monte Carlo runs. The LOx/LH2 RS-68 core + RSRM concept and LOx/RP RD-180 core + RP LRB concept remained top ranked across both qualitative and quantitative attribute weighting assessments. The quantitative probabilistic variation of the attribute weightings validated the results of the qualitative NASA Funding weighting scenarios approach.

4. TECHNOLOGY ASSESSMENT

Orbital performed a capability gap and readiness assessment for major HLS elements, and identified the functional performance characteristics required (i.e., thrust, Isp, mixture ratios, mass, throttle range, physical envelope, life-cycle costs, development schedules, and production rates) for:

- Main engine technologies, propulsion systems, and associated propulsion system elements
- Upper stage engines technologies, propulsion systems, and associated propulsion system elements
- In-space systems and elements including in-space engines, transfer stages, etc.
- Heavy lift launch system elements and hardware including tanks and structures, propellant and pressurization systems, auxiliary propulsion systems, avionics and control systems, and payload shrouds/fairings.
- Ground and launch operations and infrastructure

Technology Assessment

- **Directly and Indirectly Surveyed Launch Vehicle Subsystem Vendors to Discover Innovative Technologies that are Applicable to HLS Concepts**
  - Struggling but innovative U.S. industrial base has achieved many advances
  - Advances not necessarily reflected in cost models
  - Foreign industrial partners may also contribute through commercial partnerships or as mission participants (e.g.; for specific In-Space elements)
- **Objective is to Quantify the Impacts of the Innovative Technologies**
  - Performance, Mass, Reliability, and/or Costs
  - Will apply directly to down-selected HLS concepts
  - Includes innovative ground operations approaches

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Orbital worked with vendors and conducted surveys to identify and determine both existing and emerging technologies that could have application to a Heavy Lift Launch System. Foreign participation was also considered for elements of the HLS that were not on the critical path. Each technology was evaluated for applicability to HLS in terms of form/fit/function, performance, reliability, mass, availability, sustainability, and costs. In addition to the hardware technologies considered, ground operations approaches were evaluated against the KDAs and GR&As.

HLLV Technology Assessment

Orbital

- Assessed Key Technologies with Four Readiness Levels producing Composite Score
  1) Technology Readiness Level
  2) Cost Readiness Level
  3) Manufacturing Readiness Level
  4) System Readiness Level

Core Structures & Tankage				
8.4 m	Cylinders	TRL	8	6
		MRL	6	
		CRL	7	
		SRL	3	
	Domes	TRL	8	6
		MRL	6	
		CRL	7	
		SRL	3	
	Tankage	TRL	7	5
		MRL	6	
		CRL	7	
		SRL	3	

Upper Stage Structures				
8.4 m	Cylinders	TRL	8	6
		MRL	6	
		CRL	7	
		SRL	3	
	Domes	TRL	8	6
		MRL	6	
		CRL	7	
		SRL	3	
	Tankage	TRL	7	5
		MRL	6	
		CRL	7	
		SRL	3	

Fairing/Shroud Diameter x Length (m)			
5.4 x 25	TRL	9	8
	MRL	9	
	CRL	8	
	SRL	8	

Fairing/Shroud Diameter x Length (m)			
8.4 x 25	TRL	3	4
	MRL	5	
	CRL	6	
	SRL	2	

Core Propulsion Systems			
RP & LOX	TRL	5	4
	MRL	3	
	CRL	6	
	SRL	3	

Upper Stage Propulsion Systems			
RP & LOX	TRL	6	5
	MRL	3	
	CRL	6	
	SRL	4	

Large Scale Separation Systems			
5.4	TRL	9	8
	MRL	9	
	CRL	9	
	SRL	7	

Strap On Boosters			
RP & LOX	TRL	5	4
	MRL	3	
	CRL	6	
	SRL	3	

Strap On Boosters				
HTPB	5 Seg SRM	TRL	6	6
		MRL	7	
		CRL	6	
		SRL	6	

Large Scale Separation Systems			
8.4	TRL	4	4
	MRL	5	
	CRL	6	
	SRL	2	

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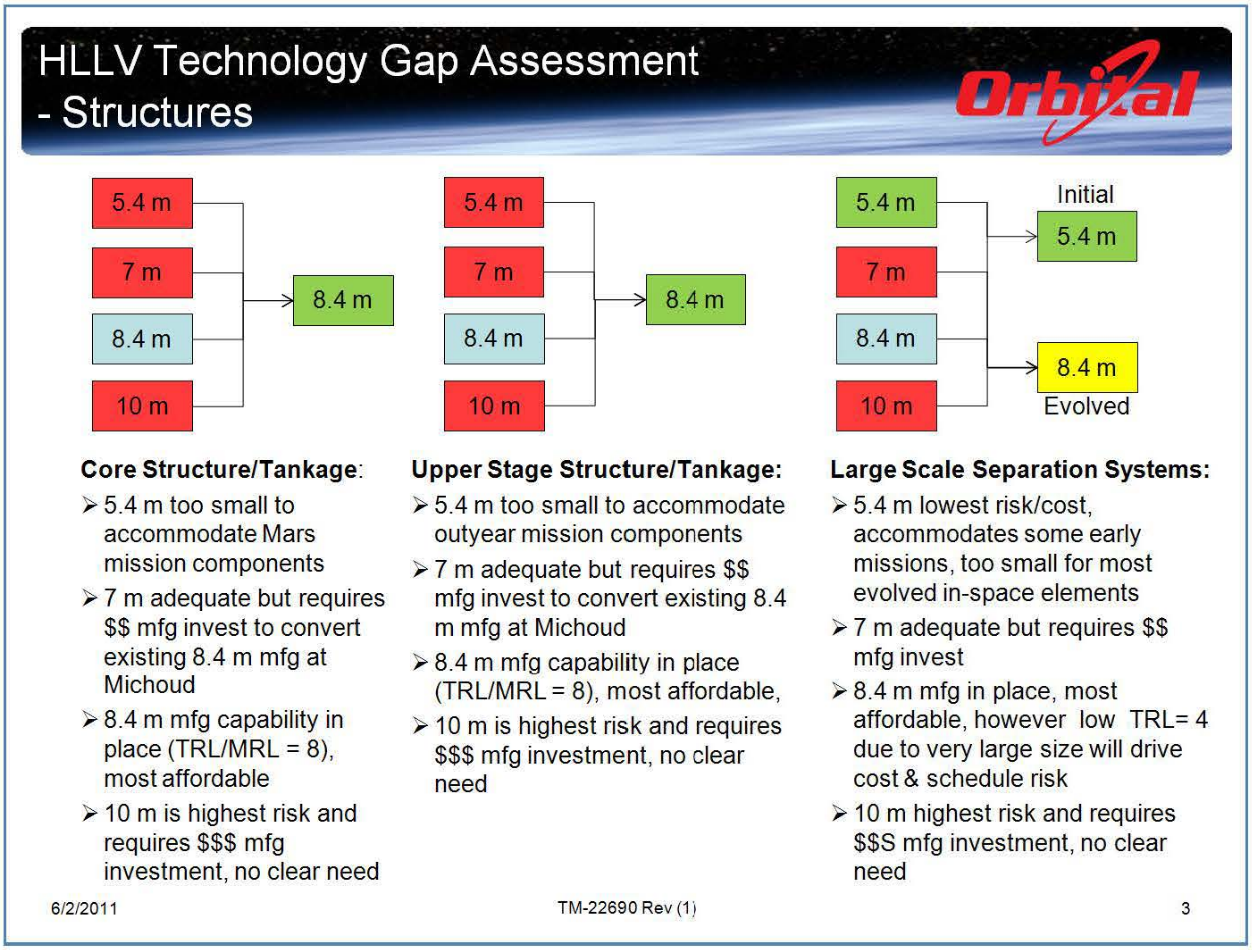
Orbital also assessed the Technology Development necessary to meet identified preferred HLS configuration and propulsion systems requirements, and quantitatively evaluated their technology status using established metrics, specifically NASA Technology Readiness Level (TRL), Manufacturing Readiness Level (MRL), Cost Readiness Level (CRL), and System Readiness Level (SRL). Technology gaps were identified for core propulsion, strap on boosters, upper stage propulsion, fairing/shroud diameter, and large scale (i.e.; greater than 5.4 m diameter) separation systems. Although technologies were identified that could eventually meet the requirements of HLS, a gap existed that would need additional development in order to increase the technologies to acceptable readiness levels.

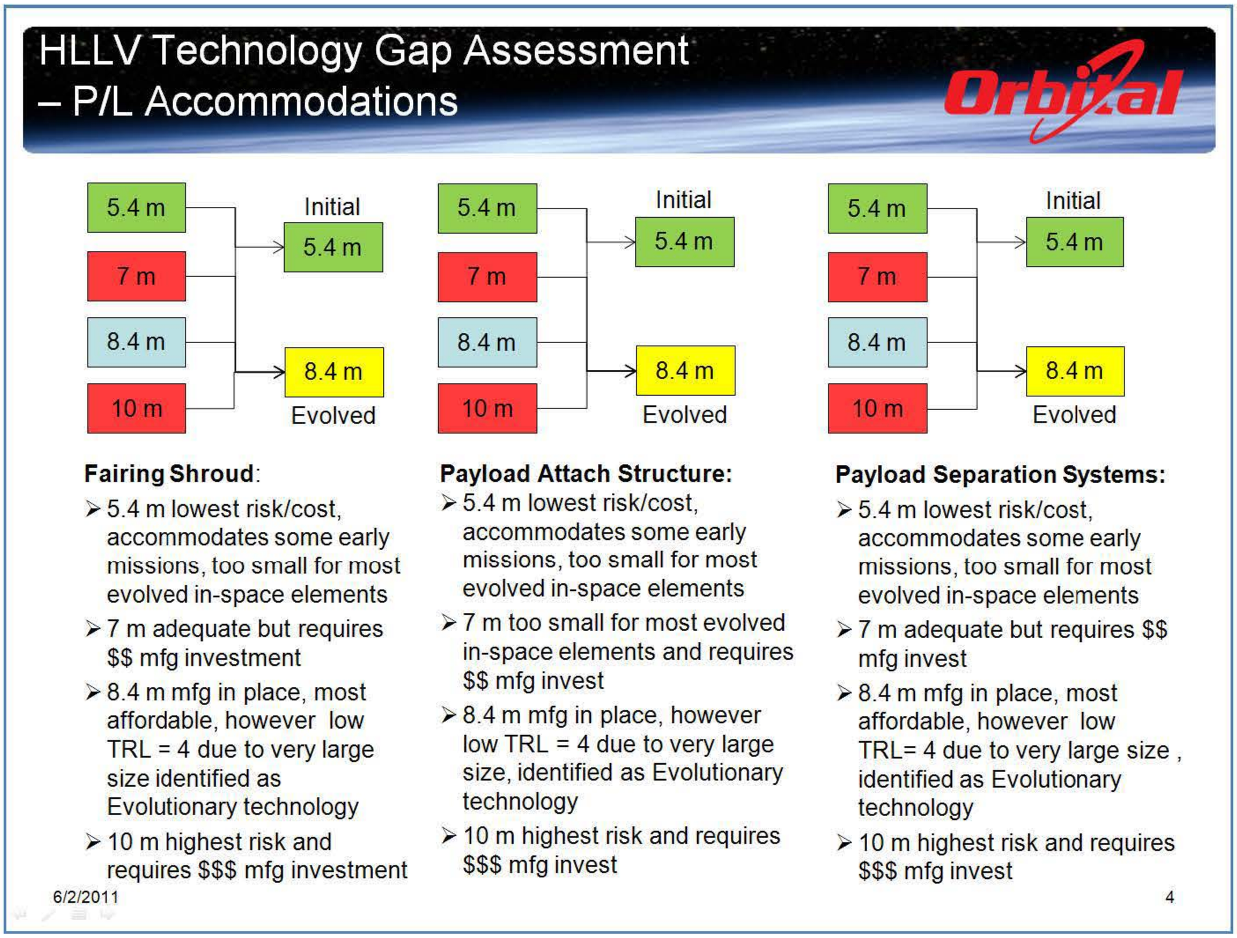
(b) (4)

[redacted]

Once the necessary TRL, MRL, CRL, and SRL values were determined for each technology area, Orbital evaluated each technology to identify those with the highest readiness levels. This approach was implemented due to the driving KDAs of affordability and schedule because if a technology area had a low readiness level then cost and schedule would be driven potentially to the point where the HLS could not be affordably developed in a reasonable time frame. The chosen technologies for HLS are identified by the green boxes in the above chart, with yellow boxes identifying evolutionary technologies, and red indicating those technologies that could not meet one or more of the KDAs.



The chosen technologies for HLS structures and tankage are identified by the green boxes in the above chart, with yellow boxes identifying evolutionary technologies, and red indicating those technologies that could not meet the KDAs. As shown, the Core Structure and Tanks would be based on existing 8.4 m technologies formerly developed for and utilized on the Shuttle External Tank. Similarly, the HLS Upper Stage structures and tanks would also be based on 8.4 m technology. Materials were also considered, with 2219 Aluminum and 2195 Aluminum-Lithium technology being chosen. Although Al-Li technology is proven and well known it is also some 20% more costly than 2219, so a cost-benefit trade might be in order to determine if the 10% performance increase is worth the additional material costs. State of the art for large scale separation systems are currently at 5.4 m. A technology development program would need to be implemented to evolve to the much larger (1.55x) 8.4 m separation systems required of the Mars exploration missions.

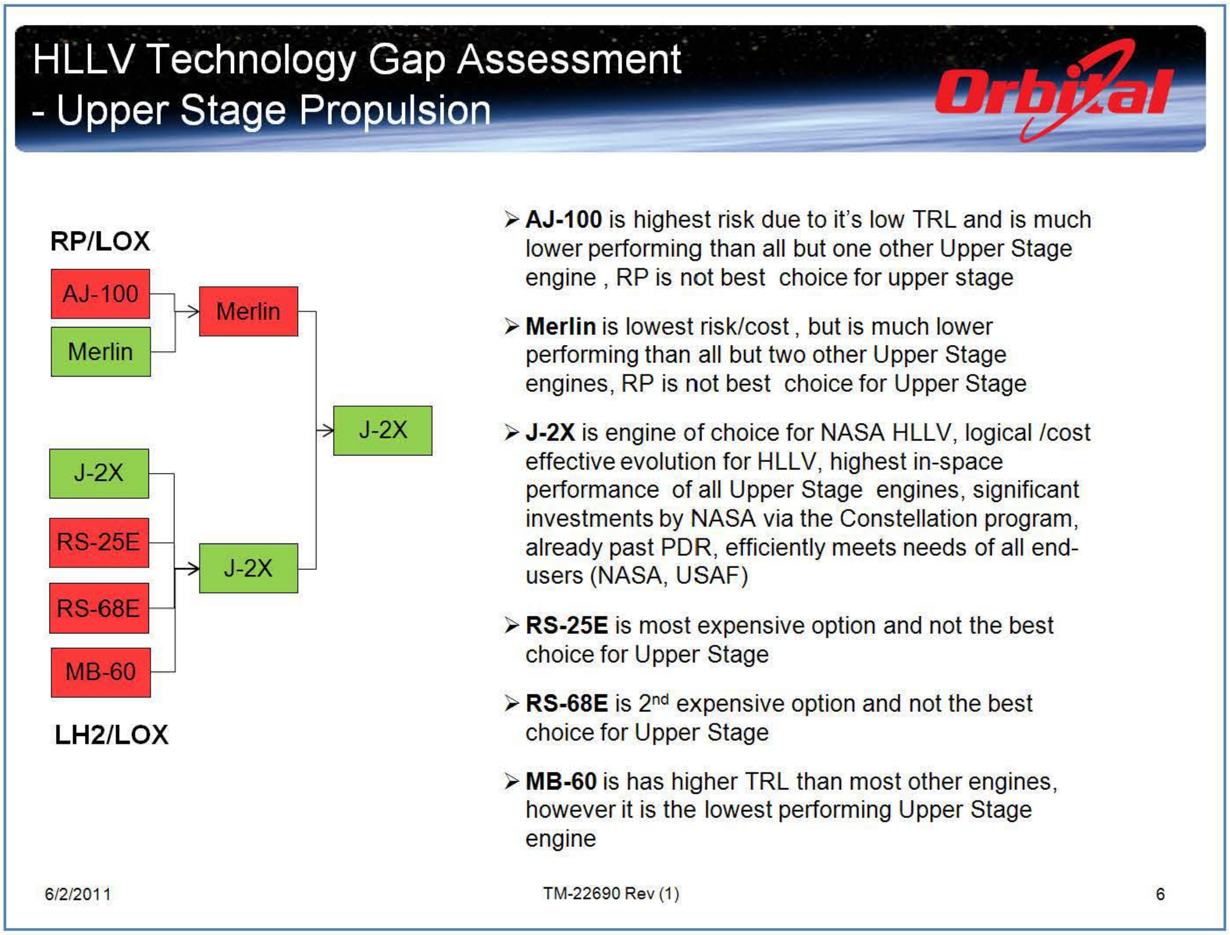


The chosen technologies for HLS payload accommodations are identified by the green boxes in the above chart, with yellow boxes identifying evolutionary technologies, and red indicating those technologies that could not meet the KDAs. As shown, the state of the art for Payload Fairing/Shrouds is currently at 5.4 m. A technology development program would need to be implemented to evolve to the much larger (1.55x) 8.4 m Fairing/Shrouds required to accommodate the Mars exploration elements. Similarly, state-of-the-art for payload attach systems and large scale payload separation systems are currently at 5.4 m. A technology development program would need to be implemented to evolve to the much larger (1.55x) 8.4 m payload attach and separation systems required of the Mars exploration missions.

(b) (4)

[redacted]

The chosen propulsion technologies for the HLS Core Stage are identified by the green boxes in the above chart, with yellow boxes identifying evolutionary technologies, and red indicating those technologies that could not meet the KDAs. A Technology Gap was identified for a domestically produced Oxygen-Rich Staged Combustion (ORSC) LOx/RP engine with performance to sustain evolving HLS architecture. The assessment indicated that ORSC technology available in U.S. could be used to develop a domestic engine through AJ-26 production and up-rating. A technology development program would need to be implemented to evolve the engine to 1 Mlb class. The LOx/LH2, while high performing, fell out due to cost and complexity concerns that greatly exceeded the relatively low development and recurring costs and relative simplicity of the AJ-26E option. And although the RD-180 is a good candidate, these engines would need to be domestically produced to meet HLS quantity requirements, and it was not clear that domestic production of the Russian engine was a viable option in the time frame under consideration for HLS.



The chosen propulsion technologies for the HLS Upper Stage are identified by the green boxes in the above chart, with the red boxes indicating those technologies that could not meet the KDAs. The J-2X was determined to be the best engine for NASA HLS Upper Stage propulsion since it provides a logical /cost effective solution for HLLV, has the highest in-space performance of all Upper Stage engines considered, maximizes the significant investments and technical progress made by NASA via the Constellation program (past PDR), and can efficiently meet the needs of other end-users (NASA, USAF).

(b) (4)

[redacted]

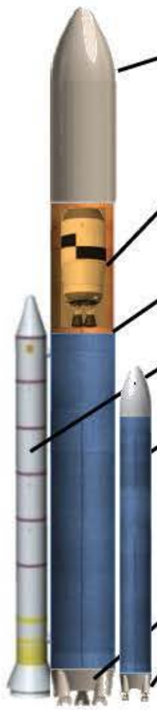
The chosen propulsion technology for the HLS Strap-On Boosters is identified by the green box in the above chart, with the red boxes indicating those technologies that fell short primarily due to high recurring costs. Orbital evaluated various options relative to supplemental propulsion for the Heavy Lift System and both solid and liquid strap on booster options were evaluated. While Solid boosters were evaluated as "stand alone" options, liquid booster propulsion was considered in the context of an integrated liquid booster solution which would be sized and optimized to the performance of the vehicle. Certain propulsion system options were eliminated due to lack of adequate performance in this application or lack of technical maturity. Viable propulsion solutions were then evaluated in the context of benefit, system complexity, and cost.

The five segment solid boosters currently in development for NASA's Constellation program were studied extensively. While they provide an efficient supplemental propulsion solution, they were seen to be the most extensive of all possible options. Given investments already made in development, only the Alliant Segmented Motors were evaluated. Other options exist, specifically the Aerojet solid motor used on the Atlas V launch vehicle, but these are not sized effectively for a vehicle of the size required for HLS, and development costs for a suitable motor were seen to be prohibitive. The Alliant Segmented motors recurring costs and sustaining costs associated with the segmented motor production, were assessed and found to be significantly higher than possible liquid strap-on booster options. [redacted] Moreover, a sustaining manufacturing infrastructure would need to be funded that would support only production for the NASA heavy lift vehicle. At the low launch rates envisioned for HLS, the costs to sustain this infrastructure would seem prohibitive from a life cycle perspective.

Although clearly modification would be required, liquid boosters tailored specifically to meet the requirements of the HLLV can be developed rapidly from existing liquid launch vehicles currently in production. Liquid propulsion solutions for these boosters were evaluated using the same aforementioned criteria. LOx/Hydrogen solutions were seen to be underperforming in this application. Of the LOx/ RP solutions evaluated, the AJ-26 derived solutions seem to be the most promising. Development and evolution of this engine system has been discussed elsewhere in this report. Using this engine system on the liquid boosters as well would further enhance the economic benefits resulting from production economies of scale. As a result, a liquid rocket booster based on the AJ-26E was determined to be the best option for NASA HLS booster propulsion since it provides a logical/cost effective solution in a reasonable time frame, that can also efficiently meet the needs of other end-users (NASA, USAF, and Commercial) thereby reducing life-cycle costs by amortizing the boosters over multiple programs and users.

Technology Gap Assessment

Orbital



- **Fairing/Separation Systems -** 8.4m dia. is 50% larger than current state-of-the-art, low TRL= 4 due to very large size, will drive cost & schedule risk
- **Upper Stage Engines –** the J-2X likely engine of choice for HLLV, logical /cost effective evolution for HLS, highest in-space performance of all Upper Stage engines, significant investments by NASA via the Constellation program, already past PDR, efficiently meets needs of all end-users (NASA, USAF)
- **Core/Upper Stage Structures -** Common structure/tankage manufacturing capability in place (TRL/MRL = 8), 2219 most affordable w/least risk
- **5 Segment SRBs –** High performance. Significant investment via the Constellation program, high fixed sustaining/recurring costs, NASA HLS is only user
- **Liquid RBs –** More cost effective than SRBs. Launch Vehicle physics dictate Strap-On performance must be greater than Core propulsion system to provide significant supplemental performance, no current U.S. LV "Cores" are optimal, however either a 4m or 5m Liquid Strap-On could be affordably developed using existing technology
- **Booster Core Propulsion-** Lox/ RP ORSC provides logical /cost effective evolution for HLS core and strap-on booster, common core/strap-on engine increases performance and reduces risk/costs, usable by multiple LVs, efficiently meets needs of all end-users (NASA, USAF, Atlas, Taurus II), significant cost savings with common U.S. engine across multiple end-users and launch systems

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The above chart summarizes the primary conclusions derived from the technology assessment, and identifies areas where significant technology gaps exist.

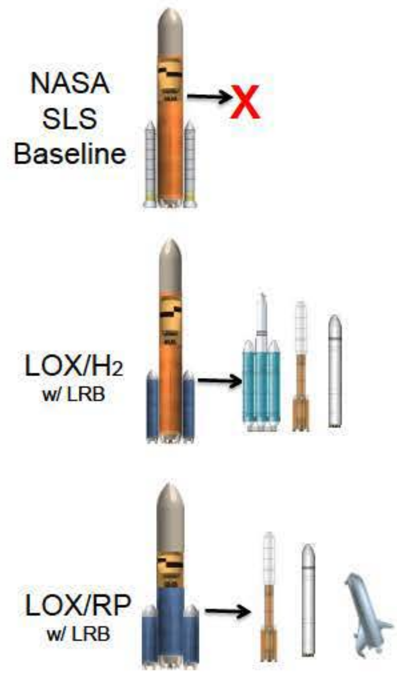
Specific gaps occur with 8.4 m diameter fairing/shroud structures and separation systems which will require significant R&D and technology investment to attain.

The other technology areas identified above are either near or at a level of readiness that can be directly applied to a Heavy Lift System.

4.1. Main Engine Technology Assessment

Main Engine Technology Summary

- **No Currently Available Propulsion System is Ideal for HLS**
  - Cost, Performance, and Availability "Challenged"
- **Near Term HLV Funding and Schedule Constrained**
  - Insufficient to support "Start from Scratch" propulsion system development program with IOC of 2016
- **Economies of Scale Can Positively Influence Recurring Costs**
  - Use engines and propulsion systems with extensibility to other vehicles in the U.S. fleet
- **Existing Engines / Technologies Can be Leveraged**
  - Existing engines can be Human Rated or Upgraded to meet HLS requirements



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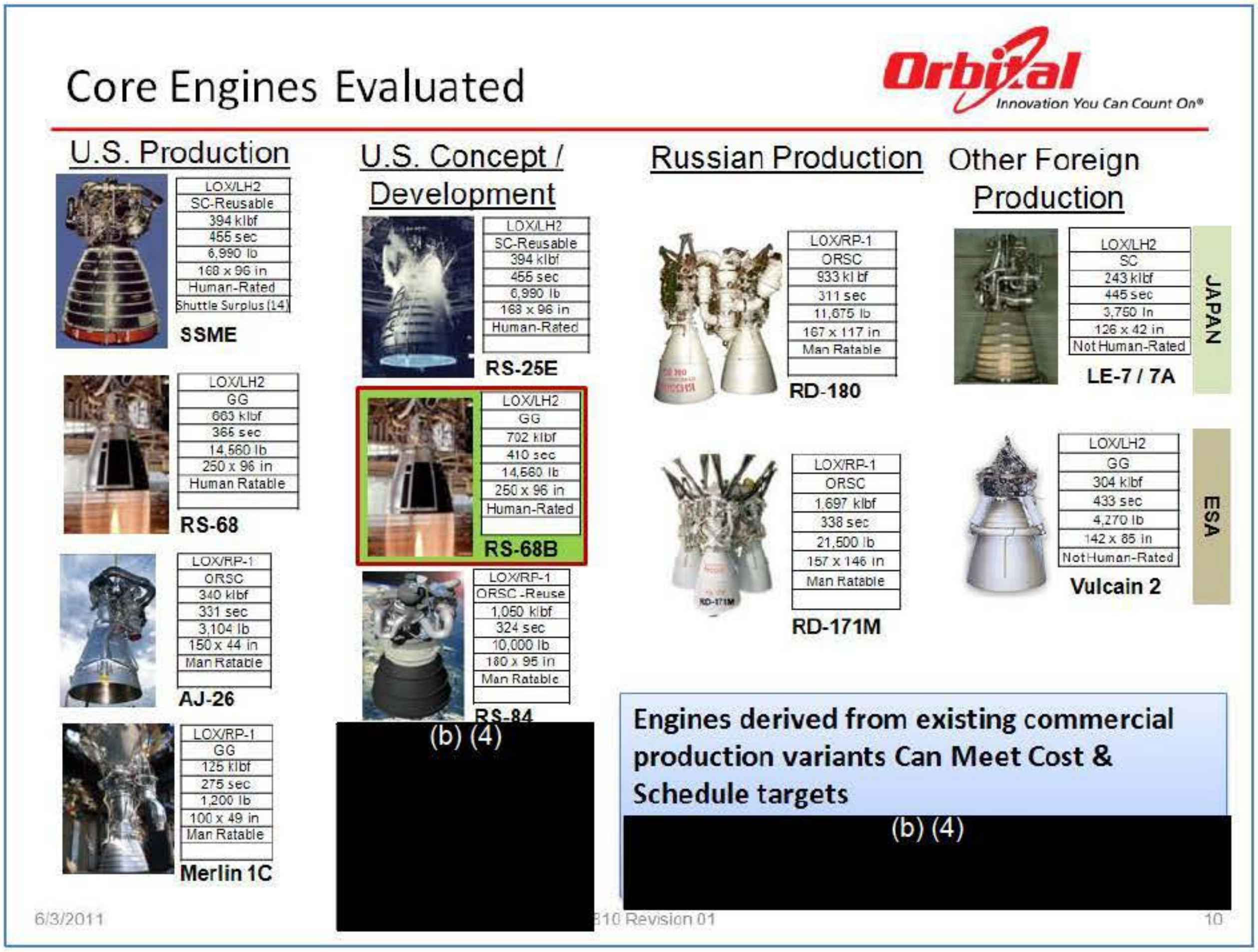
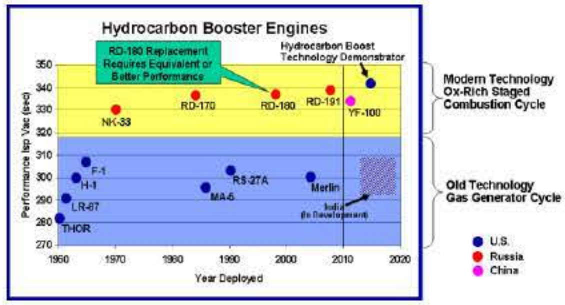
Orbital's review indicates that there is no currently available main engine system technology that can ideally meet the cost and performance requirements for NASA's Heavy Lift Launch System. Certain propulsion systems are available internationally, but they face potential restrictions and would not be available in quantities to sustain NASA's requirements. Insufficient time and funding are available to develop an ideal propulsion system from the ground up. To solve this problem, NASA should look to invest in systems that have broad extensibility to other systems and vehicles in the U.S. fleet as this can have a positive effect on the cost of these systems due to economies of scale; the broader this applicability, the greater the potential cost advantages. Moreover, while no currently existing system meets all requirements, existing domestically available engine technology can be leveraged to incrementally develop larger, cost effective propulsion systems.



Currently available and developing main engine systems were extensively studied for applicability to NASA's Heavy Lift System. As previously stated, it was found that no commercially available system meets all requirements. Two promising developing systems were identified however, the incremental development of which was deemed to not be cost prohibitive. The RS-68B, a human-rated variant of the RS-68 engine currently employed on the Delta IV launch system, affords the best solution for a LOx/Hydrogen core booster solution. [redacted]

Domestic vs. Foreign Engine Gap

- **Oxygen-Rich Staged Combustion (ORSC) LOX/RP engines Offer Required High Performance in Reasonable Size**
- **Downselected LOX/RP HLS Concepts Could Incorporate RD-180 or AJ-26X Engines to Achieve Necessary LEO Performance**
  - RD-180 is Highest Performing ORSC engine currently in use by a U.S. company but only available to ULA/Atlas
  - However, Domestically Produced Engine will be Required to meet NASA & U.S. Government demands



- **Technology Gap: Domestically Produced ORSC LOX/RP Engine with Performance to Sustain Evolving HLS Architecture**
  - Domestic LOX/RP engine supports HLS core and Liquid Rocket Booster (LRB)
  - Multiple Government and Commercial users (NASA, USAF, Atlas V, Taurus II)
- **Solution: Use ORSC Technology Available in U.S. to Develop Domestic Engine**
  - Two possible paths – Domestic RD-180 co-production or AJ-26 production and up-rating

AJ-26 Up-Rating Path Clearly Defined to Orbital

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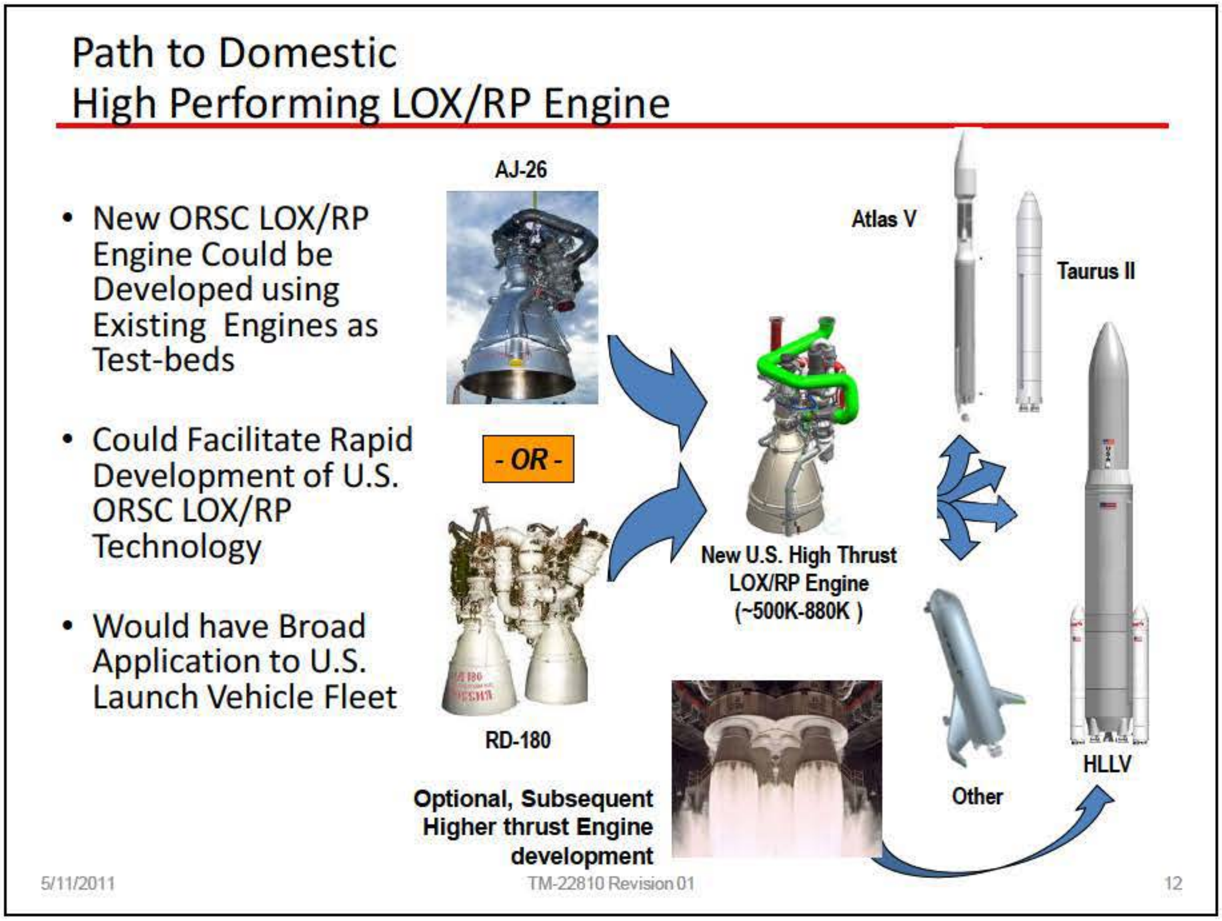
After surveying the U.S. launch vehicle fleet and available and potential engine solutions, Orbital believes that LOx/RP main engine systems afford the most cost effective solution to meet NASA's heavy lift launch vehicle requirements. As stated, some further development of these systems will be required to meet all requirements. If investment is to be made, it should be made in the highest performing LOx/RP technology – ORSC. As shown in the graph above, Oxygen-Rich Staged Combustion engines have significant performance advantages versus more simplified Gas Generator cycle engines. While a LOx/RP ORSC engine which meets all of NASA's requirements is not currently available in the US, the technology is. This technology can be leveraged to develop a domestically produced high performing LOx/RP engine that meets NASA's requirements in a cost effective manner.



The path to develop a domestically produced, high performing LOx/RP Oxygen Rich Stage Combustion (ORSC) engine is relatively straightforward. The basic technology behind LOx/RP ORSC engines is fairly well understood, however such engines have never been domestically produced. Further testing and characterization of the complexities associated with high pressure oxygen rich staged combustion is required. Two sources of this technology are available and in use in the US today – the Aerojet AJ-26 engine and the RD-180 engine. Using either of these engines, the testing and modeling required to fully characterize the combustion cycles of these engines can be conducted. This testing can be expeditiously performed in a cost effective manner. Insights gained for testing and analysis can be rapidly incorporated into a domestically produced LOx/RP ORSC engine. Once such an engine is available in the US, this engine would have broad applicability to multiple vehicles in the US fleet, including NASA's Heavy Lift Launch Vehicle, current commercial launch vehicles, and possibly even advanced reusable vehicles currently being proposed by the USAF and others. Lastly, once such an engine is in production in the US, higher performance variants can be developed and produced to further enhance the performance of NASA's HLLV.

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[redacted]

(b) (4)

[redacted]

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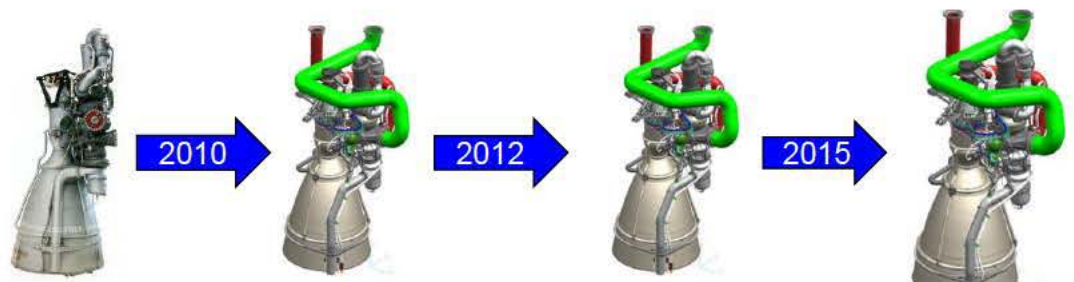

[redacted]

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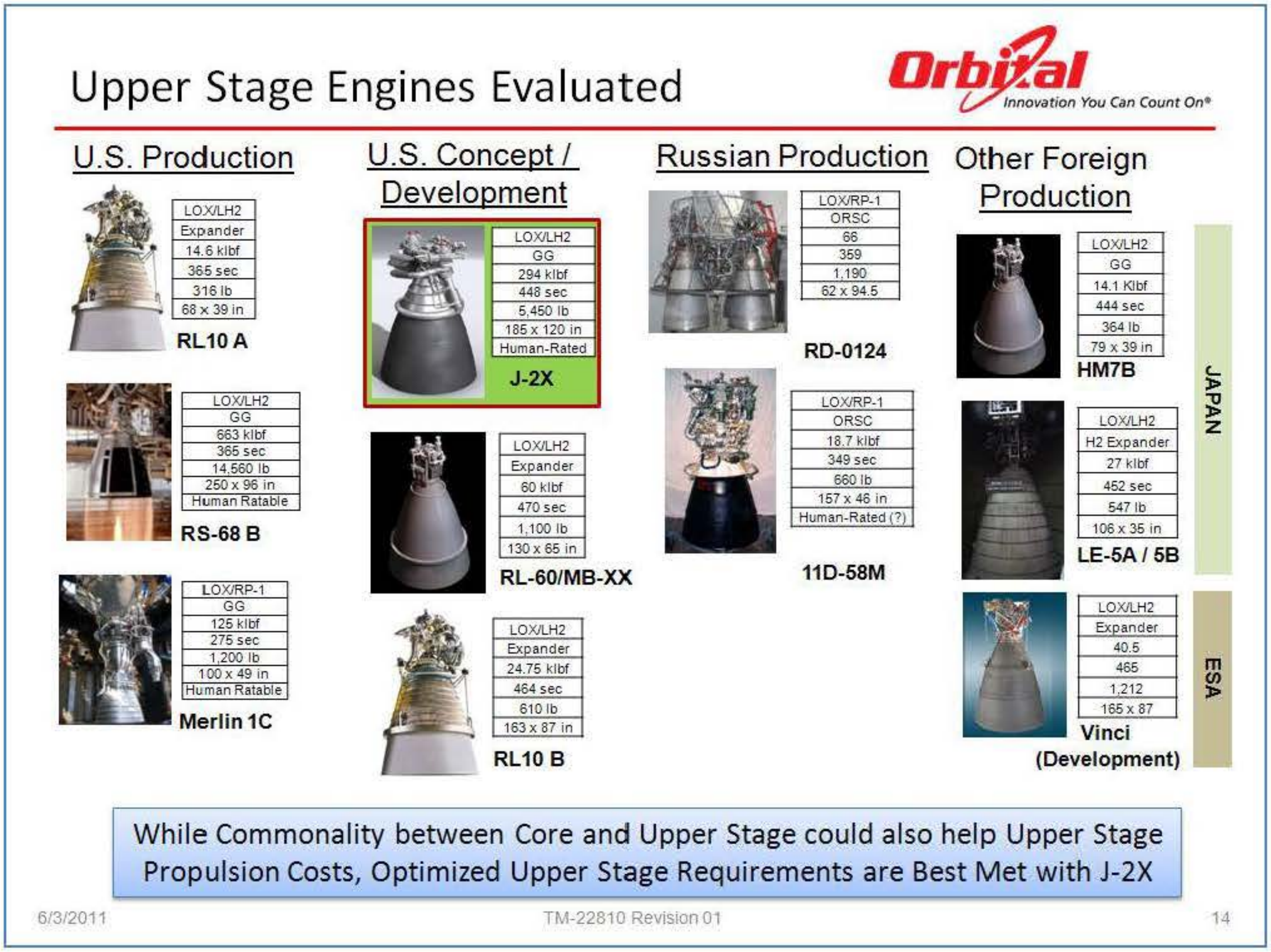
Orbital has worked in conjunction with Aerojet and NASA Marshall Space Flight Center representatives to outline a systematic approach to testing, modeling and analysis which would translate directly to upgrades and domestic production of the [redacted] This low-risk, phased approached is outlined above and could be executed inside of four years which would support HLLV ILC in the 2016 timeframe. Specific details of the AJ-26 enhancement program have been discussed with NASA are being coordinated with MSFC representatives.

(b) (4)

[redacted]

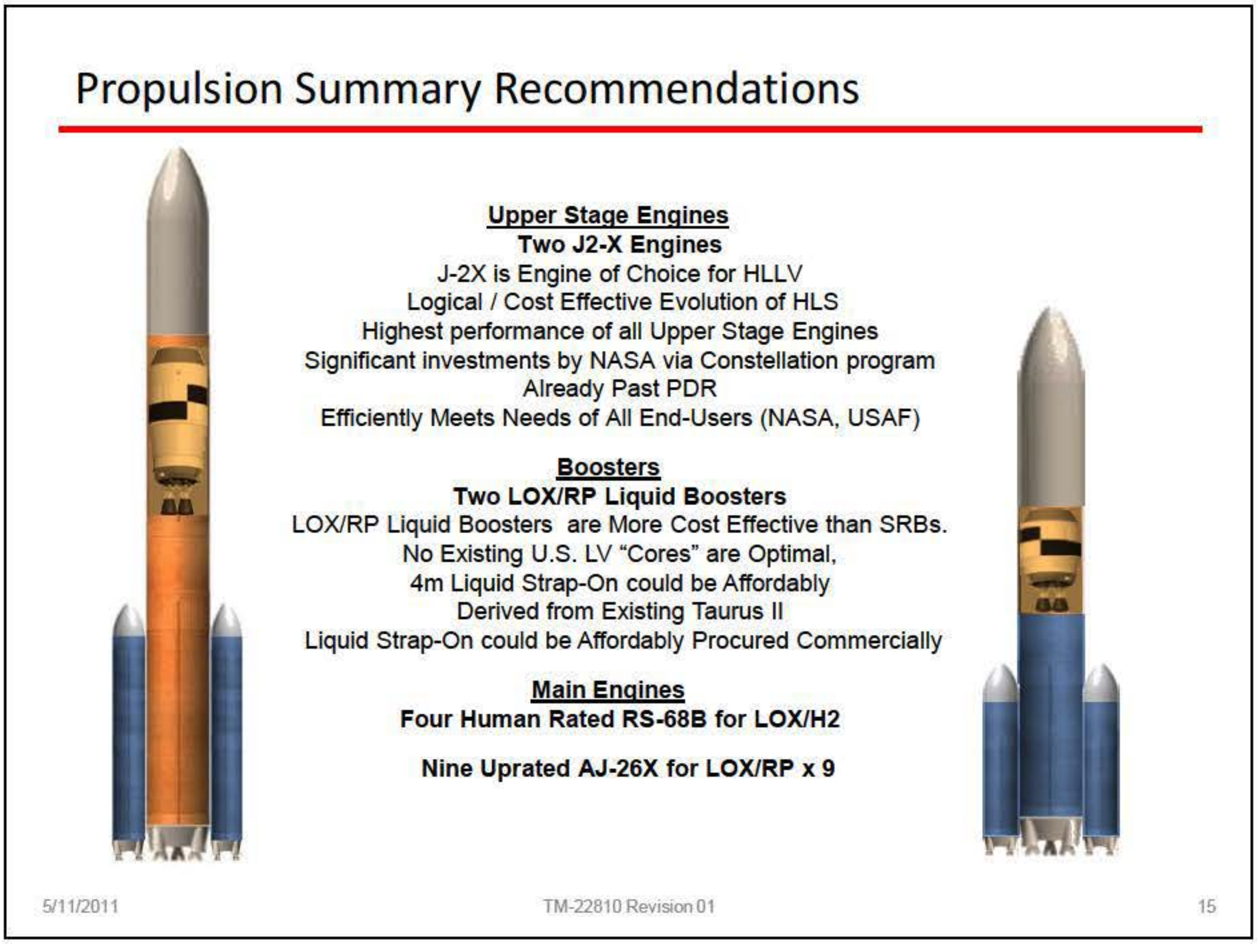
Orbital performed an initial assessment of performance associated with various possible HLLV configurations incorporating the upgraded AJ-26 engine. These initial assessments indicated that such an engine could work well, in a clustered configuration for the main engine system and / or in tandem for use on a liquid strap-on booster, for vehicles in this performance class.

4.2. Upper Stage Engine Technology Assessment



Orbital performed a similar assessment of currently available and developing upper stage engine systems for applicability to NASA's Heavy Lift System. As with the main engine systems, it was found that no commercially available system meets all requirements. Promising developing systems were identified however, the incremental development of which was deemed to not be cost prohibitive. The J-2X, a new human-rated LOx/Hydrogen engine, affords the best solution for a LOx/Hydrogen upper stage solution. While the concept of commonality and broad applicability of engine systems could also extend to the upper stage, it was concluded that optimized upper stage requirements as dictated by NASA as well as by Orbital's preferred Heavy Lift Launch Vehicle configurations, could best be met with the J-2X engine.

4.3. Propulsion Technology Summary



Orbital's extensive survey of engine systems in support of an optimal, affordable, reliable Heavy Lift Launch Vehicle resulted in the following conclusions:

- Main Engine Systems:
  - Four Human Rated RS-68 Engines for a LOx/Hydrogen Vehicle
  - Nine Uprated AJ-26X Engines for a LOx/RP based vehicle

- Boosters:
  - Two LOx/RP Liquid Boosters incorporating two AJ-26X engines

- Upper Stage:
  - Two J2-X Upper Stage Engines

4.4. Other Technologies Assessed

In addition to the detailed propulsion technology assessment, Orbital also assessed the readiness of other technologies necessary to meet identified preferred HLS configuration and requirements, and quantitatively evaluated their technology status using established metrics.

Booster Core and LRB Sizing and Manufacturing

- **Structural, Performance and Manufacturing Efficiencies Dictate Large, Single Core Configuration for HLLV**
- **U.S. has Two Large Scale Booster Manufacturing/Integration Facilities**
  - ULA Decatur: Sized for 5m (& 4M) diameter structures
  - Michoud Assembly Facility (MAF): Sized for 8.4m and larger structures
  - Both have excess capacity
  - Both have made investments in tooling and welding equipment
- **However, MAF has a Low Cost Center Commercial Option and also has Capability to Manufacture 4 m to 8.4 m diameter Structures**
- **While the 2195 Technology Gap is largely closed, an Affordability and Availability Gap Still Exists that Favors 2219 to Reduce Costs – However:**
  - LOX/RP Booster Tankage (Concept 2A) is only ~157 ft long (including the upper stage), which is approximately the same size as the Shuttle ET (~154 ft long), so availability of 2195 should not be an issue
  - Although using 2219 would save ~20% material costs on core structures/tankage , this is only a Minor Savings (~$40M NRe and $15M Recurring for entire HLV), and there would be a performance penalty

CONCLUSIONS
Large 8.4M Diameter HLLV Core and 4-5M Diameter LRB Cores will be Most Cost Effectively Manufactured and Integrated at MAF Using a Commercial Procurement Approach

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Orbital believes that structural, performance and manufacturing efficiencies dictate that a large, single core HLLV vehicle configuration is optimal. Given that existing infrastructure, advanced manufacturing equipment, and tooling that was developed for the shuttle external tank program is available, sized to 8.4M (27.6') diameter, this can be cost effectively utilized to develop and manufacture the large structural components of the HLLV. While two large-scale manufacturing facilities of this nature exist in the U.S. – ULA's Decatur factory and NASA's Marshall Michoud Assembly Facility (MAF), the MAF is more optimal for HLLV manufacture. Tooling and advanced friction stir welding equipment is in place at MAF to manufacture both large core and liquid strap-on booster structures. Manufacturing these pressurized structures from 2195 Aluminum Lithium would add cost, but it is believed that the mass efficiency gained justifies the added costs. Moreover, tankage for Orbital's LOx/RP configuration is only slightly larger than that for the Shuttle External Tank, mitigating concerns regarding material availability.

At the request of NASA, Orbital verified that the MAF manufacturing costs as quoted by Vivace and incorporated into our study are correct and reflective of the anticipated costs of using the Michoud Manufacturing Facility (MAF). Orbital conferred with representatives of Vivace and MAF who confirmed that the fixed costs of operating the facility are built into the rates incorporated into the manufacturing quotes that were generated. These rates are what Orbital (or any other commercial contractor) would pay for the facility and are based on how much space would be occupied, what services would be used, etc. Those assumptions were built into the pricing that was part of the study.

(b) (4)

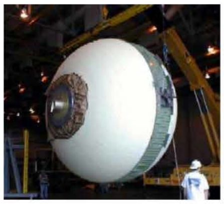
[redacted]

Like any other occupancy arrangement, the commercial contractor would pay only a portion of facility costs that is in relation to how much of the building's footprint they would be using. There is a complexity with MAF in that there are different prices for different types of areas, but that's taken into account as well. Orbital would employ further strategies to reduce costs by wisely using the resources - for example spending as little time in the VAB (some of the most expensive space there) as possible during proof tests and application of insulation, and then moving back into the "low" bay portion of the facility for most manufacturing.

As far as work force availability, Orbital has been assured that many of the technicians are still available and, while the fortunate ones have moved on to other employment, most would be anxious to return to work on such an ambitious program.

Detailed costs will be specifically provided in a supplementary proprietary package that will be submitted separately from this report.

Examples of Technology Development

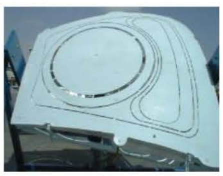


22' Dia LOX Tank

14' Dia LOX Tank

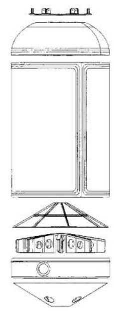
22' Dia Toroidal RP Tank (FWD Dome Pictured)

Assembly of 22' Dia Composite/Metallic Launch Vehicle Segment

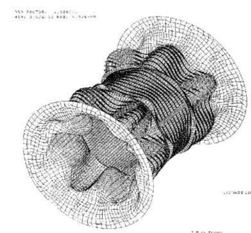


Gore Panel Chem Milling Tool

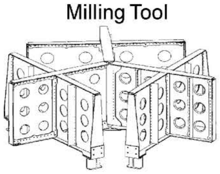
"Soft" Tooling Weld Trim Tool



77" Dia Secondary LOX Tank with Retention Screen



Analysis of Toroidal Tank "Tunnel"



Triple Anti-Vortex Baffle Design

8" Bellows Assembly

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Large, commercial structures have been successfully fabricated at the MAF. In attrition, manufacturing of other critical structures and components has been demonstrated, including large composite structures, tanks of complex shapes, and internal anti-vortex baffle systems.

(b) (3)

[redacted]

Orbital engaged engineering partners to develop detailed stage structures designs sufficient to provide high-fidelity manufacturing cost estimates. CAD models of both pressurized and unpressurized structures were developed and optimized for both LOx/Hydrogen and LOx/RP configurations. This provided detailed information on overall tankage volume, length and mass facilitating detailed cost estimated. Stage structural designs were further optimized to evaluate common bulkhead configurations. Even though they are more complex, these structures yield significant length and mass reductions.

Orbital further engaged large structures manufacturer Spincraft to provide cost estimates for large, spun-formed domes. Spincraft currently provides these structures to multiple launch vehicles, both domestically and internationally.

(b) (3)

[redacted]

Optimized structural elements and stages of the proposed HLLV were assessed in detail. Structures were sized based upon the propellant loading requirements dictated by the vehicle performance calculations. These structures we analyzed by our contributing partner Vivace, who made manufacturing assessments using the structural element sizes dictated. As indicated, they assessed both separate and common bulkhead tankage configurations and provided detailed cost quotes for all configurations. Shown here are the optimized, common bulkhead configurations. While higher cost, the mass and overall vehicle height savings realized warrant the additional development costs.

Orbital and our partners also evaluated several tank and structural configurations for the HLLV core, upper stage, and liquid rocket boosters. Two fuel systems were also considered (LOx/RP-1 and LOx/LH2) as well as 2 Stage 1 configurations, and 4 Stage 2 configurations.

For the final recommended LOx/RP systems, another option was added to show the benefits of a common bulkhead design approach (shown above). This is a low risk approach for LOx/RP cores and can save significant manufacturing time and money, while simultaneously improving performance and mass fractions of the HLLV and LRBs. It's a common misconception that separate tanks carry lower risk with the complexity of Centaur tanks being cited, but for LOx/RP a double (vacuum jacketed) bulkhead is not required between the propellants since foam insulation can be submerged in the RP (this has been done and qualified for single use so far).

Common Assumptions used for Tank and Structures:

- All tanks and other structures manufactured at the MAF utilizing their current commercial rate structure
- MAF has capacity and unique capabilities that will significantly reduce cost and risk
- All welds use Friction Stir Welding (FSW) technology
  - Michoud is only place where this technology is in regular, large scale use and not just in development mode
- Tank estimates include ROM for internal equipment (slosh baffles, anti-vortex, etc.)
- Tank estimates include manufacture, proof, clean, installation of internal components, and application of external insulation
- Thrust structure estimate includes fabrication of components and subassembly of the thrust structure
- Interstage/aft skirt/unpressurized structures estimated based on a typical skin/stringer/frame semi-monocoque aluminum construction
- All tank domes were assumed to be 0.707 domes except when noted as spherical
- Integration of tanks, unpressurized structure, and thrust structure included separately (including other elements as defined with the estimate)
- All tanks are constructed from 2195

MAJOR COST DRIVERS TO TANKS and STRUCTURES:

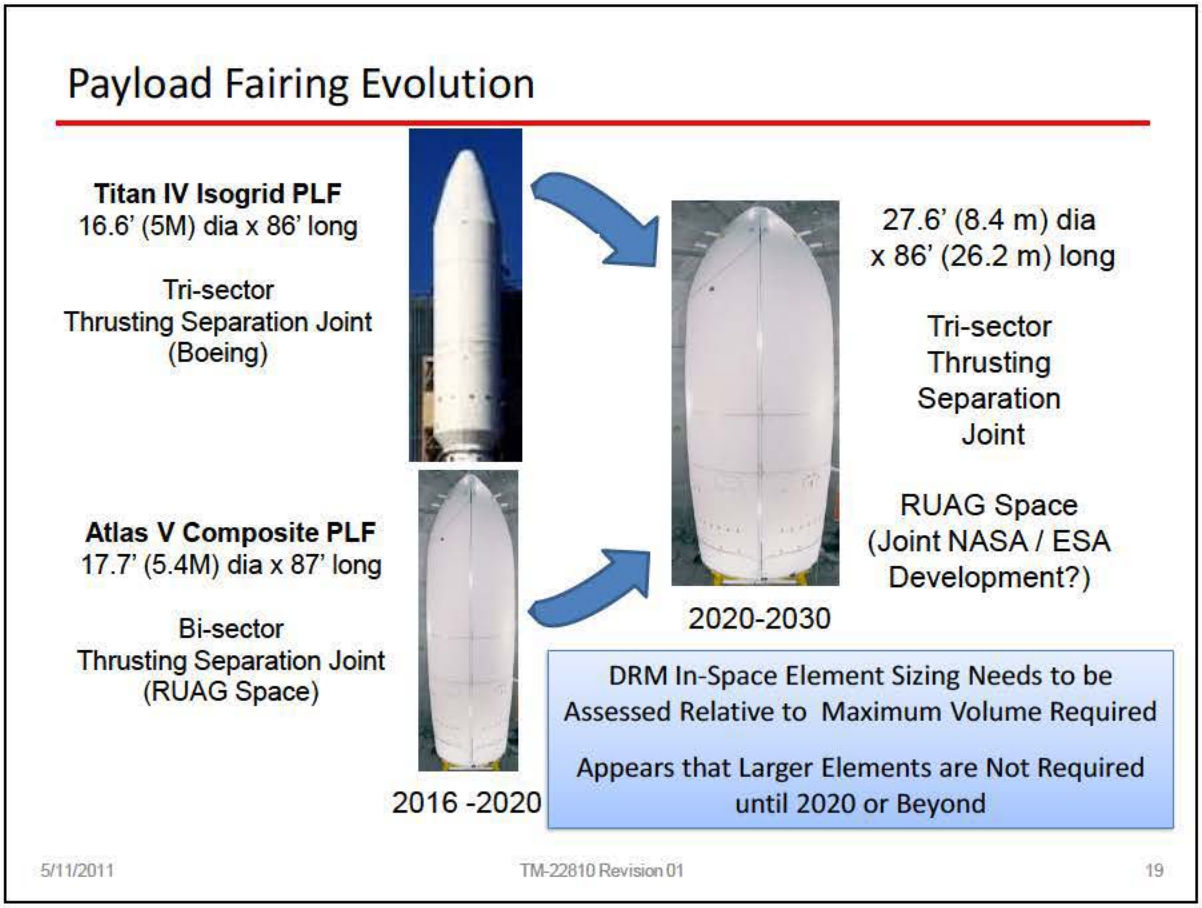
Of the factors driving the costs of the tanks and structures for a Heavy Lift Launch Vehicle, the following were found to be the most significant:

- Stage 1
  - Large diameter drives multi-piece domes meaning more welds and inspections
  - Thrust structure is very complex with 9 engines (has significant impact on following integration costs as well)
- Stage 2
  - Vacuum jacketed double common bulkhead to adequately insulate LH_2 and LOx
  - Integral thrust cone and aft LOx bulkhead, including thermal design of the thrust structure (however, this Centaur-like configuration would make the most sense from a performance standpoint because it would be lighter than other options)
  - Large diameter drives multi-piece domes
  - Large amount of insulation per tank surface area
- TII Core Booster
  - Redesign of structure to accommodate use as a strap on booster
  - New unpressurized structure, including forward aerodynamic fairing

HLLV and LRB INTEGRATION:

In addition to the tank and structural manufacturing, Orbital engaged engineering partners to develop high-fidelity cost estimates for integrating the HLLV core and upper stage, and the LRBs at Michoud, and ship the fully tested and integrated assemblies to KSC for final stacking prior to roll-out to the pad. The following assumptions were used to develop the integration approach and cost estimates:

- Final Assembly and Integration estimates based on a similar program being conducted at the Michoud Assembly Facility
- In addition to the elements estimated on the preceding charts, stage feed systems, engines, avionics, pressurization, and transport elements (small fluid lines and wiring) are integrated into the stages
- Stage level testing is included, but any testing of individual elements is not (tank testing already included in separate tank estimates)
- Assembly of unpressurized structure (interstages, intertanks, etc.) are included in integration
    - These structures are usually not assembled until final vehicle assembly to ensure proper aligiment and fit to tankage
- Commercial practices, private company, common facility



Orbital also worked to assess technology required to produce very large diameter payload fairings. Current technology available domestically and internationally is limited to 5.4m in diameter. Producing payload fairings of significantly larger diameters, specifically on the order of 8.4M, is significantly more complex than their simple structures might indicate. Fairing structural loads and separation dynamics would dictate that an 8.4M fairing development program would require substantial significant investment. In assessing the DRMs in the near term years, Orbital could not identify specific payloads that would require an 8.4M diameter. Further, in assessing out-year DRMs, specifically Mars DRMs that would require assembly of large structures, and for which such a large payload fairing would be optimal, Orbital believes that these might be better optimized and reduced to be compatible with fairing volumes of more readily achievable size. For this reason, Orbital advocates the use of commercially available payload fairings of 17.7' in diameter and up to 87' in length in the near term to meet the ILC objective of 2016. Presuming available funding, a larger scale fairing development program could be executed to bring this capability on line in the 2020 time frame. Moreover, it is possible that such a development program could be a cooperative effort with assistance in design, manufacturing and cost from foreign organizations such as ESA.

Avionics & Software

- **Extend the Constellation and Taurus II Supply Chains to Include Avionics for HLS Core and LRB**
  - Over 80% of Constellation avionics component designs have applicability to HLS
  - Almost 100% of Taurus II core avionics component designs have applicability to a Liquid Rocket Booster (LRB)
- **Leverage Existing Constellation and Taurus II Avionics Infrastructure**
  - Constellation SDF/SIL/SITF laboratory infrastructure and on-going BDU integration effort in SITF for HLS
  - Taurus II Avionics for LRBs
- **Capitalize on Constellation and Taurus II Flight Software Development**
  - Constellation flight software requirements specifications at PDR maturity
  - Taurus II flight software will be flight proven for at least 5 years and 8 launches by 2016
- **Extend and Leverage Constellation and Taurus II EGSE**
  - Constellation EGSE can incorporate HLS Core Stage functionality
  - Existing and proven Taurus II EGSE and GSE for LRB
- **Adapt Taurus II Architecture to Provide Human-Rated Avionics and Software Solution for HLS LRBs**

Extending and Leveraging the Constellation and Taurus II Avionics & Software presents a Technically Sound, Affordable solution for HLS and Supports Early Launch Capability

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Orbital conducted an avionics technology assessment and concluded that the most logical approach would be to leverage existing avionics development performed for the Ares I and V on the Constellation program. Since the LRB is based on a modified Taurus II launch vehicle, virtually all of the avionics and much of the control software could be adapted from existing designs or used in its current form.

Similarly, Electrical Ground Support Equipment (EGSE) developed for the Constellation and Taurus II programs could be adapted for use on the HLS program.

This approach will provide an affordable solution for the HLS program that conforms to NASA's preferred initial launch capability of 2016.

Other Vehicle Subsystems

- **Majority of HLS Required Sub-systems are Commercially Available with Technology at a High Technology Readiness Level**
- **Use HLS Sub-Systems Common with Other LVs in the U.S. Fleet (To the Maximum Extent Possible)**
  - Valves
  - TVC Actuators
  - Controllers
  - Sensors
  - Separation Systems
  - Thermal Protection
  - Ground Support Sub-systems
    - Fueling Systems
    - Hydraulic System s
    - LV Transportation
    - GSE

Uniqueness Drives Costs
Commonality and Re-Use Leads to Economies of Scale

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In assessing the other subsystems required to support the Heavy Lift Launch Vehicle, Orbital has largely found that such subsystems are either readily available, or could be readily produced by suppliers who are currently manufacturing similar systems with high reliability for other vehicles in the U.S. fleet and internationally. This includes such subsystems as valves, TVC actuators, controllers, sensors, separation systems and thermal protection. In addition, requisite elements of the ground support system infrastructure could also be readily sourced from available (primarily domestic) suppliers.

Orbital strongly advocates sourcing such components from existing suppliers manufacturing similar, if not identical, components for other launch vehicles to realize economies of scale and minimize both development and recurring costs.

4.5. Ground Operations and Infrastructure Assessment

Ground Operations and Infrastructure

- Analyzed Ground Operations Required for HLS:
  - Documented Baseline approach based on typical NASA launch operations and facilities (e.g.; Saturn, Shuttle, Constellation)
  - Use of existing NASA KSC facilities and GSE maximized to extent that it makes sense, is affordable, and sustainable:
    - Baseline approach has Highest recurring and fixed costs of all approaches considered
    - Requires maximum number of KSC personnel (“standing army”) that is significantly underutilized for launch rate of ~2 HLLVs per year
    - Requires maintaining facilities and infrastructure (e.g.; VAB, SSPF, O&C, SLC-39A, etc.) that are only used intermittently
  - Examined alternative methods to achieve integration, processing, roll-out to pad, and launch ops
  - Evaluated existing facilities and infrastructure, and developed cost estimates for required improvements to accommodate each of the downselected Heavy Lift architecture concepts

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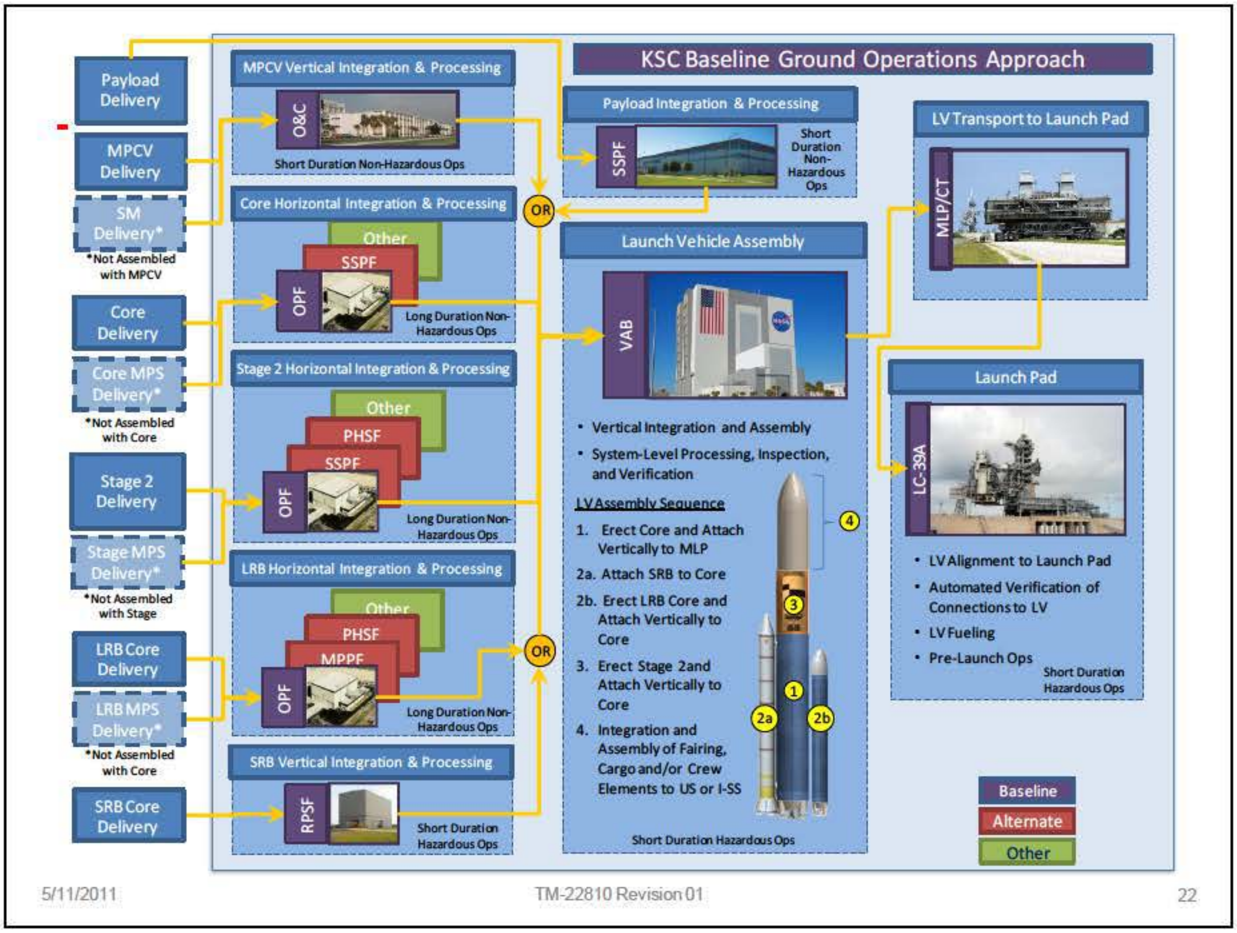
The ground Concept of Operations (CONOPS) for the Heavy Lift System was examined in some detail by the Orbital team. The assumption was that existing facilities, infrastructure, and Ground Support Equipment (GSE) at KSC and Stennis be utilized on the HLS program to the greatest extent – with the overriding ground rules of affordability and sustainability.

Three approaches were assessed:

- **Baseline:** this approach essentially followed the Constellation ground CONOPS of performing all integration, checkout, and launch operations at KSC using existing facilities.
- **Commercialized:** this approach followed most of the Constellation ground CONOPS, but considered commercial facilities – such as Astrotech – to perform some of the integration.
- **Commercial:** this approach created a ground CONOPS based on commercial best practices and campaign mode launch operations.



The Baseline ground CONOPS is based on the Constellation Approach:

- PROS:
  - Least aggressive of the approaches considered
  - Well known/understood processes, facilities, etc.
  - Employs maximum number of KSC personnel
- CONS:
  - Likely the highest recurring Life Cycle Costs of all approaches considered
  - Employs maximum number of KSC personnel ("standing army") that is not needed for ~2 launches/year
  - SLS Program must shoulder costs for utilizing/maintaining facilities (e.g.; VAB, SSPF, O&C, SLC-39A, etc.) that are used intermittently

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This CONOPS was designed to "Commercialize" Constellation Approach:

- PROS:
  - Moderate approach
  - Utilizes commercial facilities where it makes sense (e.g.; Astrotech for Stage 2 processing and ARF for commercial LRB processing)
  - Utilizes KSC processes, facilities, etc. in a "campaign style" launch approach
  - Employs "clean pad" approach to reduce recurring launch ops costs (reduces overall Life Cycle Costs)
  - Reduces number of KSC personnel
- CONS:

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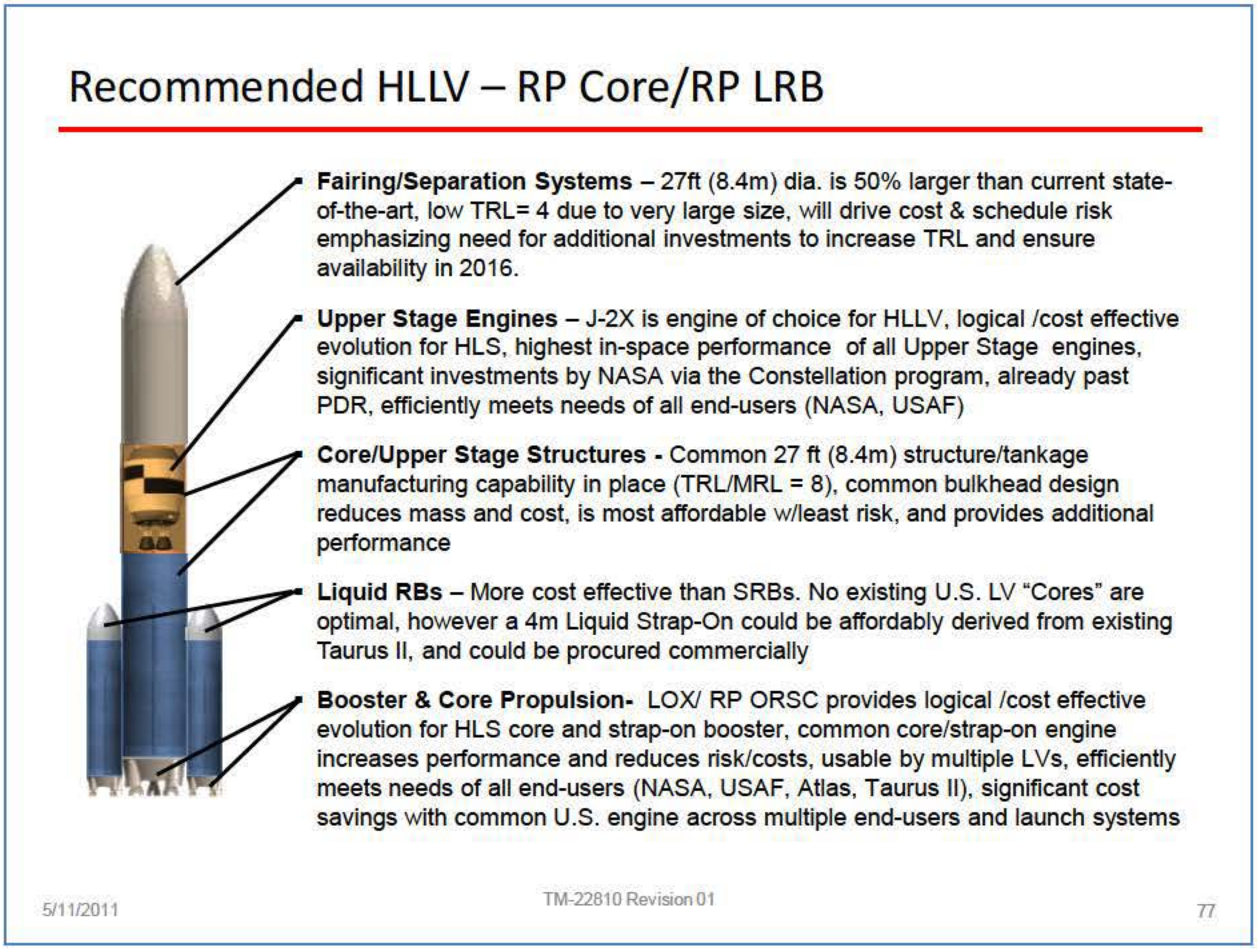
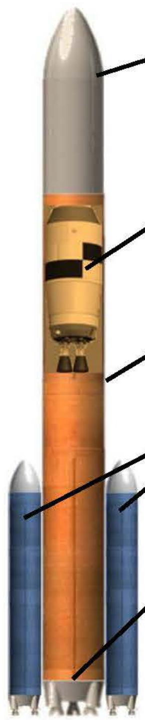
4.6. Technology Summary



Orbital's recommended Heavy Lift Launch Vehicle is based on an 8.4 m LOx/RP core stage and 3.9 m LOx/RP strap on Liquid Rocket Boosters. Other technologies for the HLS that Orbital recommends include initially utilizing essentially off the shelf 5.4 m diameter fairings/shrouds and separation systems, and then establishing a technology development program that evolves the fairings/shrouds and separation systems to 8.4 m. Upper stage engines are based on the new J-2X currently under development by NASA, as discussed previously.

Alternate HLLV – LH2 Core/RP LRB



- **Fairing/Separation Systems –** 27ft (8.4m) dia. is 50% larger than current state-of-the-art, low TRL= 4 due to very large size, will drive cost & schedule risk emphasizing need for additional investments to increase TRL and ensure availability in 2016.
- **Upper Stage Engines –** J-2X is engine of choice for HLLV, logical /cost effective evolution for HLS, highest in-space performance of all Upper Stage engines, significant investments by NASA via the Constellation program, already past PDR, efficiently meets needs of all end-users (NASA, USAF)
- **Core/Upper Stage Structures -** Common 27 ft (8.4m) structure/tankage manufacturing capability in place (TRL/MRL = 8), 2219 most affordable w/least risk.
- **Liquid RBs –** More cost effective than SRBs. No existing U.S. LV "Cores" are optimal, however a 4m Liquid Strap-On could be affordably derived from existing Taurus II, and could be procured commercially
- **Core Propulsion-** LOX/ LH2 has highest performance for HLS core, however this drives development and Life Cycle costs and cannot be developed/procured commercially, engines not usable by multiple LVs so does not meet needs of anyone but NASA

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Although industry experts interviewed by the study team indicated that there is considerable additional complexity and cost associated with liquid hydrogen systems, Orbital suggested an alternate recommended Heavy Lift Launch Vehicle based on an 8.4 m LOx/LH2 core stage. The other major elements and systems of this alternate HLS concept are identical to the primary recommended system, including a 3.9 m LOx/RP strap on Liquid Rocket Boosters, initially utilizing essentially off the shelf 5.4 m diameter fairings/shrouds and separation systems and evolving to 8.4 m fairings/shrouds and separation systems, and using upper stage engines based on the J-2X.

Other Technology Insertion

Fairing/Separation Systems
Suggest family of fairings to meet specific DRM requirements, with sizes from 5M (existing)to 27ft (8.4m) dia.

Core/Upper Stage Structures
Common 27 ft (8.4m) structure/tankage manufacturing capability in place (TRL/MRL = 8), common bulkhead design using 2219 reduces mass and is most affordable w/least risk, additional mass reduction & performance increase possible with upgrade to AlLi (2195) but at higher cost

Avionics
Complex core vehicle requires advanced avionics. Utilize Ares avionics for core with Orbital T-II derived for LRBs

Advanced Thermal Insulation

Streamlined Operations
Commercially derived vehicle elements lend themselves to streamlined processing and co-sharing of resources. Ageing infrastructure elements afford opportunity to invest in modern systems to take full advantage of clean pad approach

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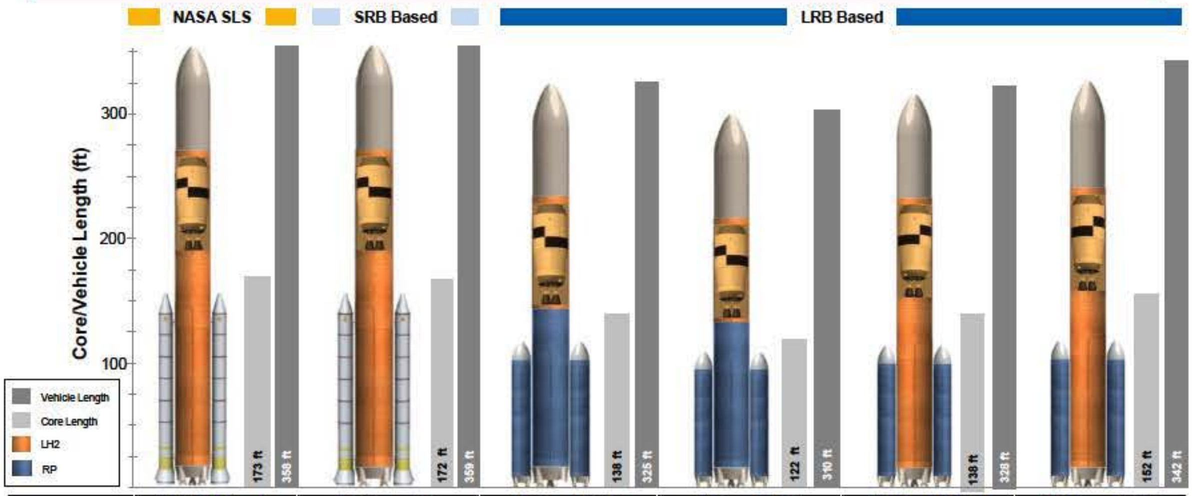
Additional technologies recommended by Orbital include fairing and separation systems tailored to specific DRMs, common structures and tankage, utilizing largely existing avionics from the Constellation and Taurus II programs, employing advanced thermal insulation, and embracing streamlined commercially derived integration and launch operations.

5. PERFORMANCE ASSESSMENT

Each of the Heavy Lift Launch Vehicle (HLLV) concepts were assessed and traded to ensure that the final recommended HLLV concepts were capable of meeting the stated goals of 100 tons to LEO with the ability to evolve to 130 mT. This section details the results of the performance assessment.

HLS Refined Concepts Evaluated



	NASA SLS (Apprx)	LH2 Core + SRB	RP Core + RP LRB	RP Core + RP LRB	LH2 Core + RP LRB	LH2 Core + RP LRB
Concept #	0	1	2	2A(4)	3	3A(4)
Core(3)	5 x RS-25E	4 x RS-68	6 x RD-180	9 x AJ-26X	4 x RS-68	4 x RS-68
Stage 2	2 x J-2X	2 x J-2X	2 x J-2X	2 x J-2X	2 x J-2X	2 x J-2X
Booster (ea)	5 seg SRB	5 seg SRB	1 Mlbf LRB (1 x RD-180)	1 Mlbf LRB (2 x AJ-26X)	1 Mlbf LRB (1 x RD-180)	1 Mlbf LRB (2 x AJ-26X)
Performance (1) (2)	120 mT	107 mT	104 mT	94 mT	69 mT	75 mT

Notes:

(1) Total Mass (mT) inserted into 30 x 130 nmi orbit at 285° inclination with insertion at 130 nmi and fairing jettisoned during Stage 2 burn.
(2) Fairing Cylindrical Envelope length and OML diameter are 25 m and 8.4 m, respectively.
(3) Stage 1 Core diameter is 8.4 m.
(4) Alternative concepts that replaced RD-180 engines with number of uprated AJ-26X engines that provide comparable total thrust.

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At the completion of the trade study downselection process, five Heavy Lift Launch Vehicle concepts were identified that met all of the KDAs. The NASA SLS is included as a baseline concept, and was by approximated Orbital using the same analysis tools and methods used for the other concepts in the trade study.

- Concept #1 has a LOx/LH2 based core stage with SRM boosters. During the refinement, the four-segment RSRMs on Concept #1 were replaced with five-segment RSRMs for additional performance capability.
- Concepts #2 and #2A have LOx/RP based core stages with LOx/RP LRBs.
- Concepts #3 and #3A have LOx/LH2 based core stages with LOx/RP LRBs.
- The LOx/RP LRBs for Concepts 2, 2A, 3, and 3A are based on the propellant capacity and structural mass of the Atlas V Common Booster Core (CBC).
- Stage 2 for all of the refined downselected concepts is a LOx/LH2 based upper stage powered by a J-2X propulsion system.

These five concepts and the SLS baseline were then subjected to the final sizing & performance, assessment.

HLLV Sizing & Performance Assessment

- Integrated Systems Analysis Model Developed within ModelCenter to Manage Data Transfer between Tools and Iteration Between Sizing and Performance
  - Vehicle Sizing: Orbital developed spreadsheets based on Heritage subsystem sizing
  - Trajectory/Performance: POST 3-DOF
- Vehicle Sizing and Performance Refinements
  - Aerodynamics updated for Launch Vehicle + Strap-On Booster Configuration
  - Atmospheric Models updated for KSC (GRAM99)
  - Strap-On Boosters based on Existing Elements
    - SRB-based Concepts utilize 5-Segment Shuttle RSRM
    - RP/LOX Liquid Rocket Boosters based on Atlas V Core with Foreign (RD-180) and Domestic (AJ-26X) Propulsion Systems

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ModelCenter was used to perform the final sizing and performance requirements, and detailed trades were conducted to determine the final concepts to be costed.

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All of the HLS Downselected Concepts comply with the KSC VAB and Crawler constraints defined by the NASA Kennedy Space Center Key Driving Constraints document (KSC_KDC_v1_0.pdf)

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6. RELIABILITY ASSESSMENT

An assessment of launch vehicle reliability was performed for each of the Heavy Lift Launch Vehicle (HLLV) concepts to ensure that the final recommended HLLV concepts were capable of meeting the stated Loss of Crew goal of less than 1 in 700.

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HLLV Reliability Assessment

- System Reliability Assessment was Conducted in Order to Compare Impacts of Propulsion System Trades on the Overall System Reliability
- In General Larger Engine Clusters Reduce System Reliability
  - Engine-out capability helps to Mitigate Risk of LOM event
  - Even with Engine Out, TVC can be a significant driver of Overall Reliability in concepts that use large engine clusters
- Engine Reliabilities Reflect Vendor Predicted Goals

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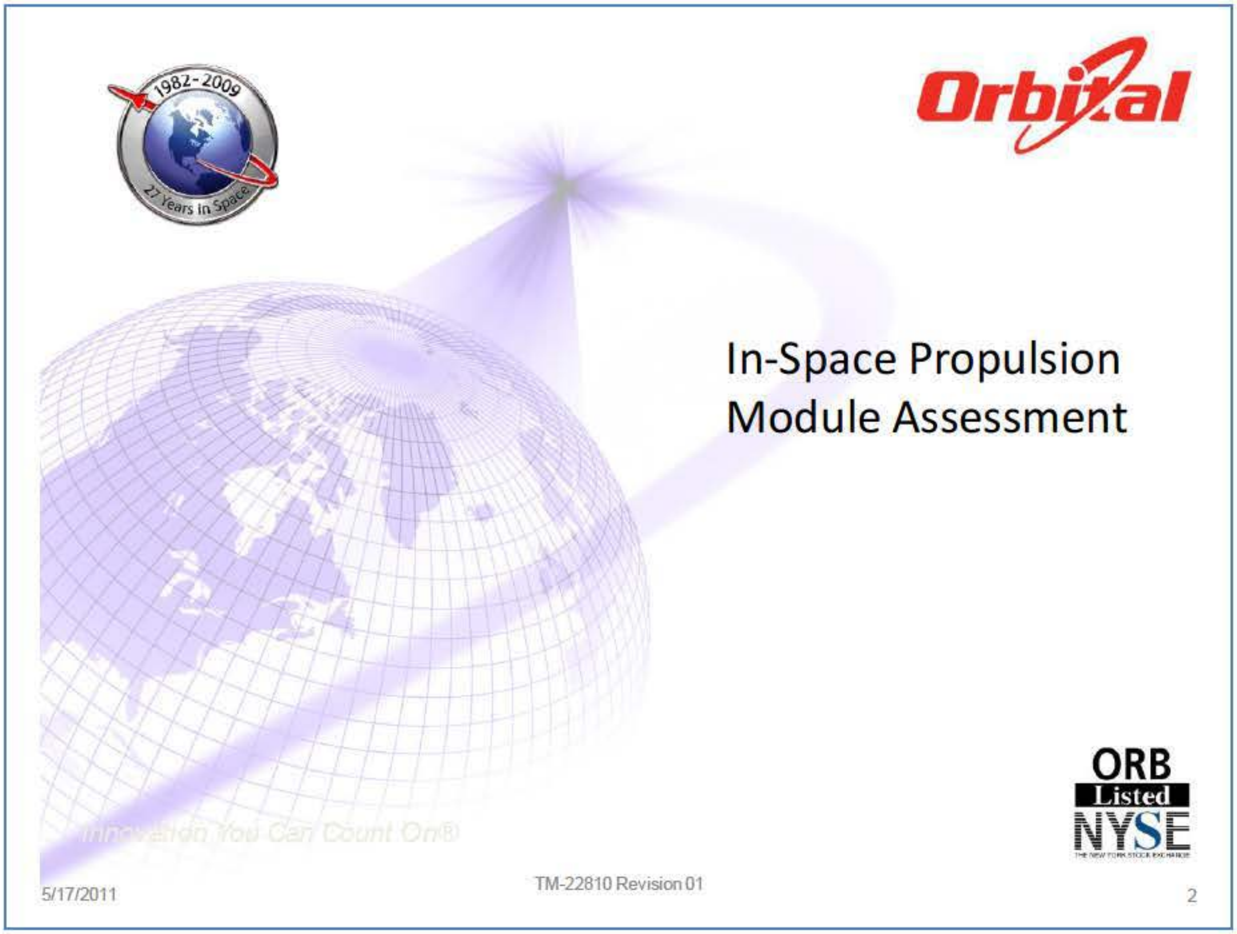
A complete system reliability assessment was conducted for each vehicle in the trade space, both with and without the capability for engine out on the first stage. With engine out enabled, however, there is a drop in payload performance to orbit. This is noted in the table and plot in the upper right of the above figure. For many concepts, it is possible to achieve the loss of crew target of 1 in 700 for the ascent portion (launch vehicle only). The sensitivity of this parameter to abort system reliability is shown in the lower left, indicating that a modest improvement in abort system reliability can make up the difference between a concept with engine out capability and one without. This potentially eliminates the need for reduced payload performance associated with engine out.

7. IN-SPACE MODULE ASSESSMENT



An assessment of the required In-Space elements was performed to ensure that the final recommended concepts were capable of meeting the NASA Design Reference Missions for low Earth orbit, lunar exploration, Near Earth Objects (NEOs), and Mars.

HLS In-Space Propulsion Module (ISPM)

Innovation You Can Count On®

- HLS In-Space Propulsion Module (ISPM) Developed to Transport Lunar Crew and Cargo from LEO to LLO
  - ISPM Provided Ascent/EOI and In-Space (TLI, LOI) Propulsion Capability
  - Propulsion Provided by either J-2X or RL-10s
- Examined Three Approaches Using the ISPM During the DRM Assessment
  - ISPM Upper/In-Space Stage + DRM Payload
    - Utilize ISPM as Ascent and In-Space Stage
    - Payload + ISPM on Same LV
  - Upper Stage + ISPM + DRM Payload
    - ISPM Sized for DRM Capability Only
    - Propulsion Provided by Multiple RL-10 Engines
    - Payload + ISPM on Same LV
  - **Upper Stage + ISPM (No Payload)**
    - **ISPM Sized for Maximum DRM Payload Capability without exceeding LV Capability**
    - **Propulsion Provided by Multiple RL-10 Engines**
    - **ISPM and DRM Payload Launched Separately**
- Assessed ISPM DRM Performance Capability and Development Cost for the HLS Downselected Architectures

Upper Stage + ISPM (No Payload) Approach:
Utilizes Upper Stage Developed for HLS Concepts and Maximizes DRM Payload Capability

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The HLS In-Space Propulsion Module (ISPM) was initially developed to transport lunar crew and cargo from Low-Earth Orbit (LEO) to Low-Lunar Orbit (LLO). During the assessment of the HLS Design Reference Missions (DRM), three ISPM variants were examined to determine which of the three would maximize performance while minimizing development cost: ISPM Upper/In-Space Stage with DRM Payload, Upper Stage + ISPM + DRM Payload, and Upper Stage + ISPM (without DRM Payload):

- ***ISPM Upper/In-Space Stage with DRM Payload:*** This approach utilizes the ISPM as both the ascent upper stage and the in-space stage to perform the DRM. ISPM propulsion is provided by a single J-2X engine since a high thrust is necessary for the ascent trajectory. Both the payload and ISPM are launched on the same launch vehicle.
- ***Upper Stage + ISPM + DRM Payload:*** The remainder of the ascent trajectory after Stage 1 separation is provided by the upper stage, and the in-space operations for the DRM are provided by the ISPM. The ISPM is sized to provide the necessary ΔV capability for the DRM. Propulsion for the ISPM is provided by multiple RL-10 engines that maintain a minimum T/W

of 0.25 at ISPM ignition. Both the DRM payload and ISPM are launched on the same launch vehicle.

- ***Upper Stage + ISPM (without DRM Payload):*** The upper stage provides the remaining propulsive capability required for the ascent trajectory. The ISPM is sized to maximize DRM payload capability without exceeding the launch vehicle capability. Propulsion for the ISPM is provided by multiple RL-10 engines that maintain a minimum T/W of 0.25 at ISPM ignition. Since the ISPM gross mass is equivalent to the launch vehicle capability, the DRM has to be launched separately.

The ISPM performance capability and development cost was assessed for HLS downselected architecture concepts 1, 2A, and 3A. The Upper Stage + ISPM (without DRM Payload) approach was selected as the baseline for the downselected concepts because it utilizes the upper stage developed for the HLS concepts and maximizes the DRM payload capability.

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The *ISPM Upper/In-Space Stage* approach sized the ISPM to provide the propulsive capability necessary to perform the post-Stage 1 separation portion of the ascent trajectory and the in-space DRM. The ΔV for the upper stage portion of the trajectory was provided from POST 3-DOF trajectory simulations. Delta-V values for each DRM were based on the associated DRM references. The ISPM for this approach was limited to a gross mass of approximately 340,000 lbm based on the results of propulsion module optimizing exercises conducted under previous Orbital architecture studies.

The upper stage and ISPM have the same on-orbit functionality such as on-orbit maneuvering, orbital maintenance, proximity operations, and deorbit/disposal. The upper stage provides propulsive capability for the ascent trajectory. DDT&E cost and first unit production (TFU) cost are normalized with respect to the development and first unit production cost of the Concept 2A upper stage. The ISPM development cost is approximately 58% of the Concept 2A upper stage development cost because of the increases in vehicle structure, propellant tankage, and non-engine main propulsion elements for the larger size upper stage. Avionics, electrical power & distribution, and RCS subsystems were the same between the ISPM and upper stage.

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The *Upper Stage + ISPM + DRM Payload* approach sized the ISPM to provide the propulsive capability to perform the DRM. Delta-V values for each DRM were provided from the associated DRM references. Sizing of the ISPM for this approach was limited by the payload capability of the two-stage launch vehicle. The combined mass of the ISPM and the DRM payload was equivalent to the launch vehicle payload capability.

The upper stage and ISPM have the same on-orbit functionality such as on-orbit maneuvering, orbital maintenance, proximity operations, and deorbit/disposal. The upper stage provides propulsive capability for the ascent trajectory. Development cost and first unit production cost are normalized with respect to the development and first unit production cost of the Concept 2A upper stage. Development costs for the ISPM concepts are approximately 52-53% of the Concept 2A upper stage development cost because of the increases in vehicle structure, propellant tankage, and non-engine main propulsion elements for the larger size upper stage. Avionics, electrical power & distribution, and RCS subsystems were the same between the ISPM and upper stage.

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The *Upper Stage + ISPM (without DRM Payload)* approach sized the ISPM to provide the propulsive capability to perform the DRM. Delta-V values for each DRM were provided from the associated DRM references. The ISPM is sized to maximize DRM payload capability without exceeding the launch vehicle capability. Since the ISPM gross mass is equivalent to the launch vehicle capability, the DRM has to be launched separately. The upper stage and ISPM have the same on-orbit functionality such as on-orbit maneuvering, orbital maintenance, proximity operations, and deorbit/disposal. The upper stage provides propulsive capability for the ascent trajectory. Development cost and first unit production cost are normalized with respect to the development and first unit production cost of the Concept 2A upper stage. (b) (4) [redacted] Avionics, electrical power & distribution, and RCS subsystems were the same between the ISPM and upper stage.

ISPM DRM Assessment Observations

Orbital *Innovation You Can Count On®*

- Performance Observations
  - US + ISPM (No P/L) Approach Provides the Most DRM Payload Capability, But Requires Separate Launch for DRM Payload
  - ISPM US/I-SS Approach Provides the Least DRM Payload Capability, But Eliminates the Need for Dedicated Upper Stage and Carries the DRM Payload on the Same LV
- Development Cost Observations
  - Both US and ISPM Equipped to Provide On-Orbit Functionality (Orbit Maneuvering, Orbital Maintenance, Proximity Operations, and Deorbit/Disposal)
  - ISPM Development Cost Is Approximately 52-60% of the US Development Cost due to Increases in Vehicle Structure, Propellant Tankage, and Non-Engine Main Propulsion Elements for the Larger Size Upper Stage
    - Avionics, Electrical Power & Distribution, and RCS Subsystems the Same Between the ISPM and US
  - ISPM US/I-SS Approach Provides Least Development Cost Since US Development is Not Required
  - US + ISPM (No P/L) Approach Has Highest Development Cost
    - Total Development Cost Includes Development of US and ISPM
  - Development Cost Does Not Include Additional Launch for DRM Payload
- Mission/Launch Related Cost Not Included in Cost Assessment

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The *Upper Stage + ISPM (without DRM Payload)* approach provides the most payload capability among the three ISPM approaches because it is sized to maximize the DRM payload capability based on the launch vehicle payload capability. Since the ISPM gross mass is equivalent to the launch vehicle capability, the DRM has to be launched separately.

The *ISPM Upper/In-Space Stage* approach provides the least payload capability among the three approaches. Though this approach provides the least payload performance, it eliminates the need for a dedicated upper stage since the ISPM provides ascent capability and the DRM payload is carried on the same launch vehicle. The size of the ISPM for the ISPM Upper/In-Space Stage approach was limited based on sizing from previous Orbital architecture studies. Increasing the size of the ISPM could result in additional payload capability for this approach.

The upper stage and ISPM have the same on-orbit functionality such as on-orbit maneuvering, orbital maintenance, proximity operations, and deorbit/disposal. The upper stage provides propulsive capability for the ascent trajectory. Development cost and first unit production cost are normalized with respect to the development and first unit production cost of the Concept 2A upper stage. Development costs for the ISPM concepts are approximately 52-60% of the Concept 2A upper stage development cost because of the increases in vehicle structure, propellant tankage, and non-engine main propulsion elements for the larger size upper stage. Avionics, electrical power & distribution, and RCS subsystems were the same between the ISPM and upper stage.

The *ISPM Upper/In-Space Stage* approach provides the least development cost among the three approaches because the development of a separate upper stage is not required. The Upper Stage + ISPM (without DRM Payload) approach has the highest development cost. Total development cost for this approach includes the development of the upper stage and the ISPM. The development cost for this approach is slightly higher than the Upper Stage + ISPM + DRM Payload approach because the ISPM is larger in order to maximize the payload capability.

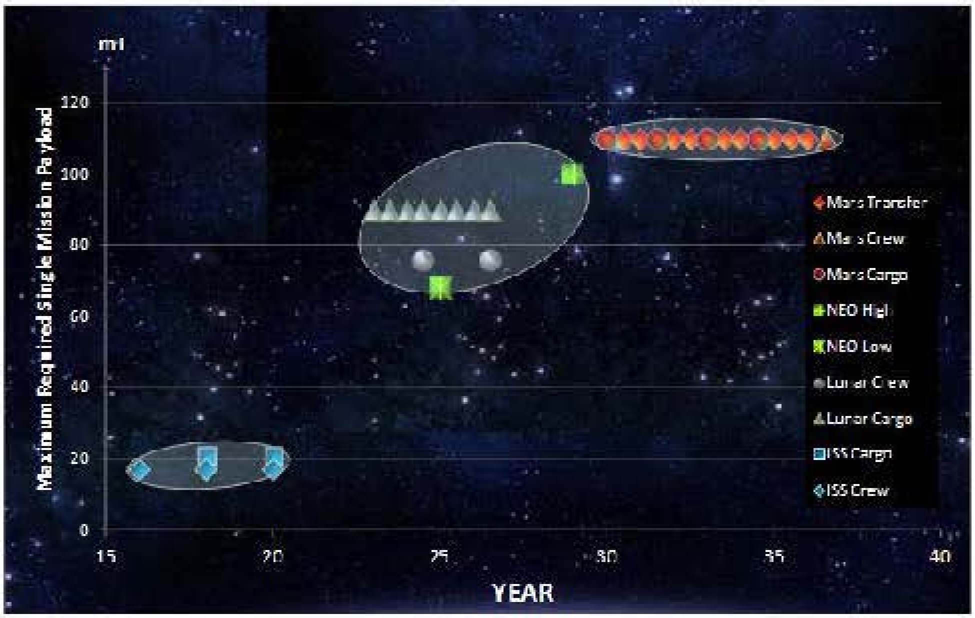
Development cost for the ISPM configurations does not include cost associated with the additional launch for the DRM payload. Mission and launch related cost were not included in the ISPM cost assessment.

8. DRM ASSESSMENT

An assessment of the ability of the candidate Heavy Lift Launch Vehicles to meet the NASA Design Reference Missions (DRMs) was performed to ensure that the final recommended concepts were capable of efficiently performing the NASA DRMs for low Earth orbit missions (e.g.; ISS and LEO transfer), Lunar exploration, missions to Near Earth Objects (NEOs), and Mars exploration.

HLS Design Reference Missions

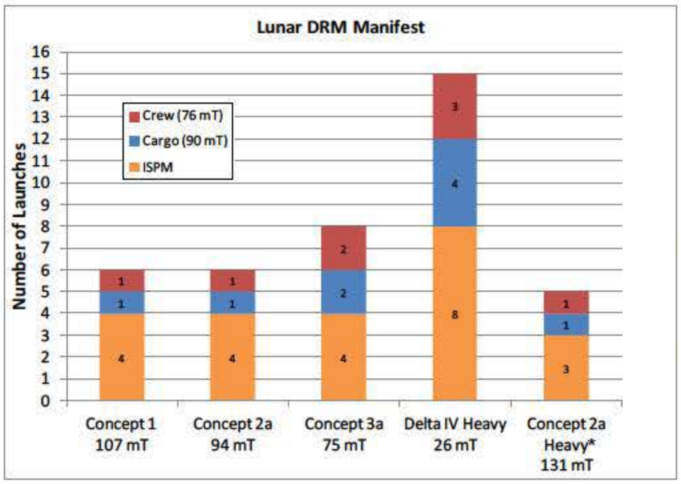
- HLS Baseline DRMs Support an Evolutionary Growth Path
  - **Lunar and Mars DRMs assume Concept 2A Heavy Lift Launch Vehicle**



Incremental Performance Increases of Payload Capability is Achievable, Affordable, & Sustainable

The DRM architecture recommended by Orbital supports an evolutionary growth path for the HLS. The first series of missions, beginning in 2016 are assumed to support ISS cargo and crew transportation needs. As the payload capability of the HLS concept 2A grows with the development of new engines, a series of lunar and Mars mission architectures is shown in the 2020 and 2030 decades respectively.

Lunar DRM Manifest



*Concept 2a Heavy Includes 2 Additional LRBs (4 Total)

- Assessment of Representative Lunar DRM Launch and Payload Manifest
  - LEO Insertion at 241 km Circular
  - Ref: Orion and LaRC Architecture Studies Lunar Missions
- Assumes Payload Launched Separately from Orbital ISPM
- Current EELV Heavy (represented by Delta IV Heavy) requires 2-3 Times as many Launches as Orbital HLS Concepts to Complete Lunar DRMs

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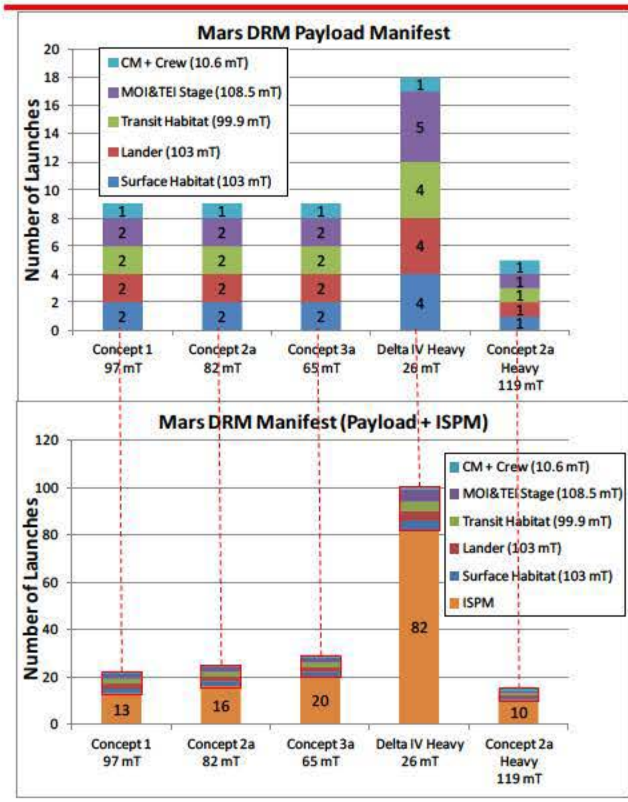
A sample lunar mission manifest shows a comparison between the number of launches required by the Orbital downselected concepts and a current evolved expendable launch vehicle (represented by the Delta IV Heavy). The mission architecture assumes that the payload is launched separately from the Orbital ISPM. The payload and the ISPM rendezvous in LEO before continuing on to the lunar destination. Given the assumed payload capabilities shown, the Orbital concept 2A heavy could reduce the number of launches required to carry crew and cargo to the Moon for an extended mission by a factor of three.

Mars DRM Manifest



- Assessment of Representative Mars DRM Launch and Payload Manifest
  - LEO Insertion at 407 km Circular
  - Ref: NASA-SP-2009-566 "Human Exploration of Mars Design Reference Architecture 5.0" July 2009
- Assumes Payload Launched Separately from Orbital ISPM
- Current EELV Heavy (represented by Delta IV Heavy) Insufficient to meet Payload Requirements of Mars Mission Architecture
- Concept 2A Heavy Results in > 80% Fewer Launches required for Mars Mission Architecture vs. Delta IV-H

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A sample Mars mission manifest shows a comparison between the number of launches required by the Orbital downselected concepts and a current evolved expendable launch vehicle (represented by the Delta IV Heavy). The mission architecture assumes that the payload is launched separately from the Orbital ISPM. The payload and the ISPM rendezvous in LEO before continuing on to the Mars destination. The top graph is an exploded view to provide more details on the non-ISPM payloads from the bottom graph. Given the assumed payload capabilities shown, the Orbital concept 2A heavy could reduce the number of launches required to carry crew and cargo to Mars for an extended mission by greater than 80%.

9. RECOMMENDED HLS ARCHITECTURE

The Heavy Lift System Architecture recommended by Orbital was thoroughly assessed to ensure that it is capable of meeting not only the NASA Design Reference Missions for low Earth orbit, Lunar exploration, Near Earth Objects (NEOs), and Mars explorations, but also meets the schedule and affordability goals established by NASA.

Recommended HLLV Architectures

	NASA SLS (Apprx)	LH2 Core + SRB	RP Core + RP LRB	RP Core + RP LRB	LH2 Core + RP LRB	LH2 Core + RP LRB
Concept #	0	1	2	2A(4)	3	3A(4)
Core(3)	5 x RS-25E	4 x RS-68	6 x RD-180	9 x AJ-26X	4 x RS-68	4 x RS-68
Stage 2	2 x J-2X	2 x J-2X	2 x J-2X	2 x J-2X	2 x J-2X	2 x J-2X
Booster (ea)	5 seg SRB	5 seg SRB	1 Mlbf LRB (1 x RD-180)	1 Mlbf LRB (2 x AJ-26X)	1 Mlbf LRB (1 x RD-180)	1 Mlbf LRB (2 x AJ-26X)
Performance (1) (2)	120 mT	107 mT	104 mT	96 mT(5)	69 mT	75 mT

Notes:

(1) Total Mass (mT) inserted into 30 x 130 nmi orbit at 285° inclination with insertion at 130 nmi and fairing jettisoned during Stage 2 burn.
(2) Fairing Cylindrical Envelope length and OML diameter are 25 m and 8.4 m, respectively.
(3) Stage 1 Core diameter is 8.4 m.
(4) Alternative concepts that replaced RD-180 engines with number of uprated AJ-26X engines that provide comparable total thrust.
(5) Commercial version of Concept 2A includes dual-bulkhead tankage to reduce structural mass and increase performance.

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Of the many concepts considered by Orbital during the course of the HLS study, two were determined to best meet the Key Decision Attributes established by NASA and the study team:

- **Concept 2A**
  - Based on an 8.4 m LOx/RP core stage and 4 m LOx/RP strap on Liquid Rocket Boosters
  - All stages (core, upper, and boosters) utilize common bulkhead tankage configurations to increase performance and reduce mass.

- **Concept 3A**
  - Based on an 8.4 m LOx/ LH_2 core stage and 4 m LOx/RP strap on Liquid Rocket Boosters
  - Only the boosters utilize common bulkhead tankage configurations to increase performance and reduce mass in this alternative option (it is possible to develop common bulkhead core, but that configuration was not evaluated by Orbital due to its relative complexity) .

Recommended HLS Architectures

- PREFERRED: Concept 2A with Commercial Approach (COTS Program Model):
  - RP/LOX Core with RP Liquid Strap-On Boosters
  - Lower DDT&E, Facility, and Operations Costs
  - Average Recurring LCC of the approaches considered
  - Utilizes Commercial Facilities where it makes sense (e.g.; Astrotech for Stage 2 processing and ARF for commercial LRB processing)
  - Utilizes KSC Processes, Facilities, etc. in a "Campaign Style" launch approach
  - Employs "Clean Pad" approach to Reduce Recurring Launch Ops Costs (reduces overall Life Cycle Costs)

- ALTERNATE: Concept 3A:
  - LH2/LOX Core with RP Liquid Strap-On Boosters
  - Higher DDT&E, Facility, and Operations Costs due to LH2 Core Complexity
  - Average Recurring LCC of the approaches considered
  - Utilizes Commercial Facilities where it makes sense (e.g.; Astrotech for Stage 2 processing and ARF for commercial LRB processing)
  - Utilizes KSC Processes, Facilities, etc. in a "Campaign Style" launch approach
  - Employs "clean pad" approach to Reduce Recurring Launch Ops costs (reduces overall Life Cycle Costs)

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As discussed in the previous page, Orbital chose a primary and an alternate configuration for the recommended HLS architectures, which are detailed in this chart.

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10. COST AND AFFORDABILITY ASSESSMENT

The Heavy Lift System Architecture cost and affordability assessment provides background on the cost estimating approaches utilized by Orbital to predict the non-recurring and recurring costs for the candidate and recommended HLS architectures.

10.1. NAFCOM vs. Orbital Custom Costing Approach

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10.2. Cost Refinement

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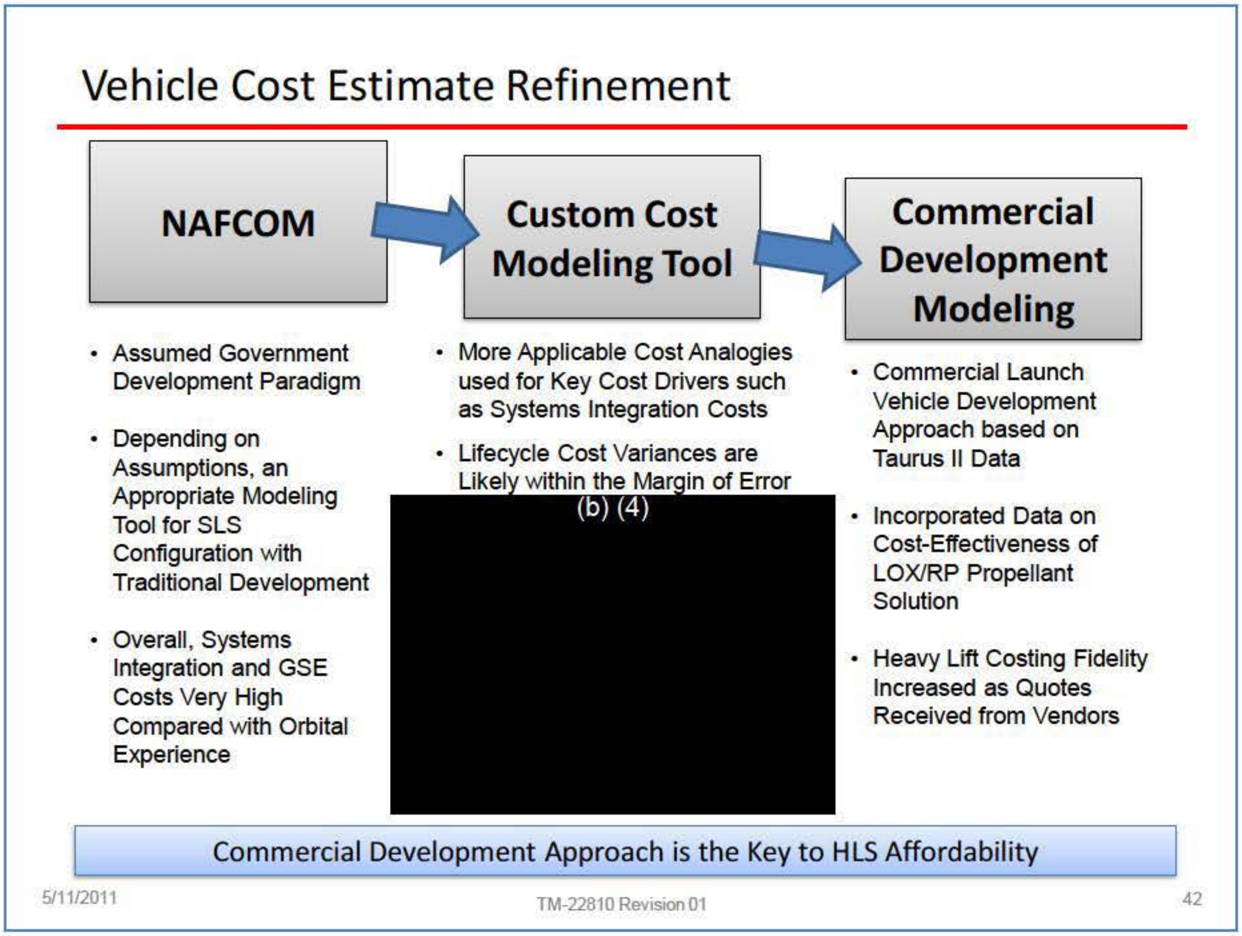
(b) (4)

[redacted]


Vehicle Cost Estimate Refinement
NAFCOM
Custom Cost Modeling Tool
Commercial Development Modeling
• Assumed Government Development Paradigm
• Depending on Assumptions, an Appropriate Modeling Tool for SLS Configuration with Traditional Development
• Overall, Systems Integration and GSE Costs Very High Compared with Orbital Experience
• More Applicable Cost Analogies used for Key Cost Drivers such as Systems Integration Costs
• Lifecycle Cost Variances are Likely within the Margin of Error
(b) (4)
[redacted]
• Commercial Launch Vehicle Development Approach based on Taurus II Data
• Incorporated Data on Cost-Effectiveness of LOX/RP Propellant Solution
• Heavy Lift Costing Fidelity Increased as Quotes Received from Vendors
Commercial Development Approach is the Key to HLS Affordability
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42

(b) (4)
[redacted]

10.3. Facility Modifications and Operations Costs

Facility ROM Cost Assessment

- **Analyzed Ground Facilities and Infrastructure to Identify Required Improvements to Accommodate each of the Heavy Lift Architecture Concepts:**
  - Stennis: Engine testing infrastructure (e.g.; new/upgraded test stands)
  - KSC Facilities: VAB, LCC, OPF, PHSF, O&C, SSPF, RPSF
  - KSC Infrastructure: fueling systems, fuel tanks, Launch Pad 39A/39B, Mobile Transporter, Mobile Launch Platform
- **Assumes that Existing KSC Facilities and Infrastructure can be Modified and Upgraded**
  - Stennis costs assume full up integrated engine tests (i.e.; costs driven by number of engines – no effort to determine costs for additional complexity of LH2 engines)
  - Shuttle derived concepts assume that existing infrastructure (e.g.; VAB, Crawler, pad fueling systems and tanks, etc.) can be used with moderate modifications/upgrades
  - Assumes that some new infrastructure must be developed (e.g.; new MLP, New Crawlers, new tanks, and new fueling systems for RP, etc.)
  - These ROM cost estimates are parametric, and actual bottoms up cost estimates need to be worked in much more detail with NASA KSC

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Orbital analyzed ground facilities and infrastructures at Stennis (for engine and stage testing) and Kennedy Space Center (launch operations) to identify the required improvements to accommodate a heavy lift launch vehicle. Parametric methods were utilized to develop these cost estimates.

(b) (4)

[redacted]

(b) (4)

[redacted]

(b) (4)

[redacted]

(b) (4)

[redacted]

Cost Breakdown By System

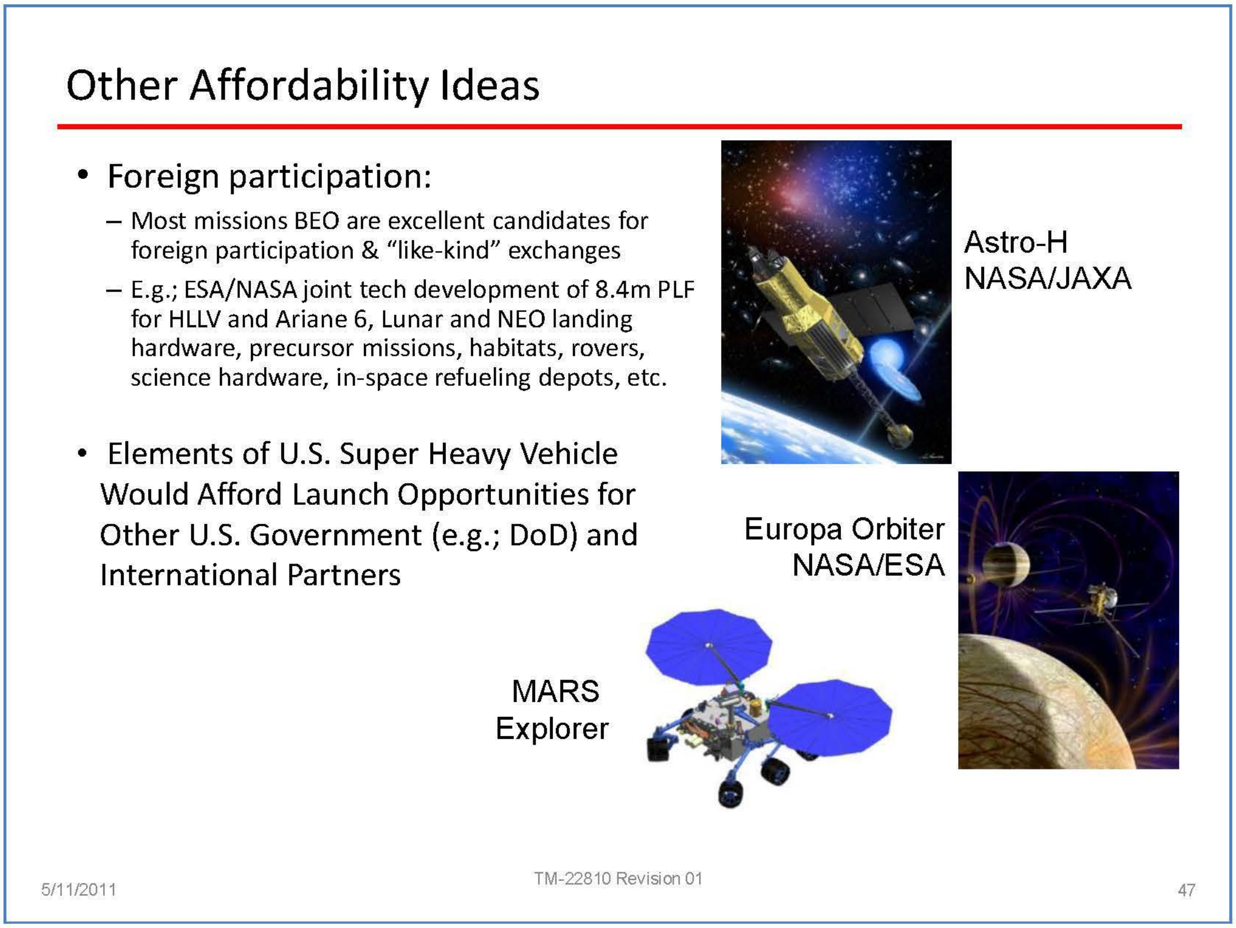
	CS Dev	CS Rec	US Dev	US Rec	B Dev	B Rec	CE Method
Vehicle Structures and Mechanisms	X	X	X	X	TII Analogy		Weight Based CER (Anchored to Vendor Quote)
Tank Structures and Mechanisms	X	X	X	X			Weight Based CER (Anchored to Vendor Quote)
Separation Structures and Mechanisms	X	X					Weight Based CER
Main Propulsion System (less Engines)	X	X	X	X			Weight Based CER
Thrust Vector Control	X	X	X	X			Weight Based CER
Thermal Control	X	X	X	X			Weight Based CER
Electrical Power and Distribution	X	X	X	X			Weight Based CER
Range Safety	X	X					Weight Based CER
Avionics			X	X			Weight Based CER
Shroud			X	X			Weight Based CER
Liquid Rocket Engine	[redacted] (b) (4)						
Systems Integration Costs	X	X	X	X	X	X	TII Analogy (% HW Costs)
Vehicle Integration	X		X		X		% HW Costs
Infrastructure and Facilities	X		X		X		Parametric Analysis
Wraps and Fees	X	X	X	X	X	X	% HW Costs

The cost estimate for Concept 2A is a rollup of development and recurring costs for the core stage, upper stage and boosters. The core and upper stage costs are broken down by subsystem. The subsystems included in the development and recurring costs for each subsystem are shown above. The boosters used a Taurus II analogy for subsystem costs. The cost estimation methods are a combination of weight-based cost estimating relationships, vendor quotes and parametric analysis based on existing hardware, infrastructure and facilities. The vehicle integration and program wraps and fees are a percentage of the hardware costs.

(b) (4)

[redacted]

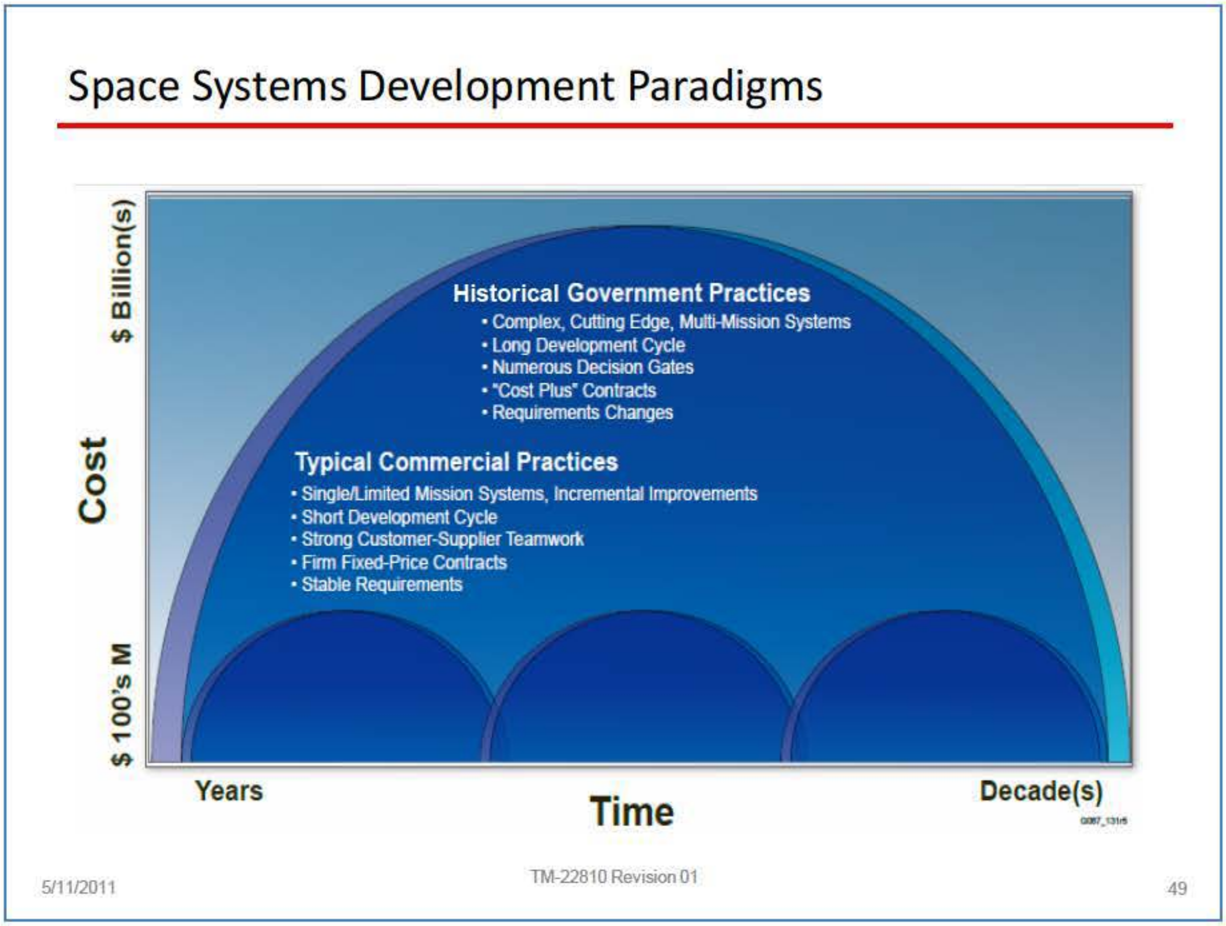
10.5. Other Affordability Ideas



As NASA looks to develop the heavy lift system, and examines the missions that such vehicles will serve, they could look to other countries and space organizations for collaborative and cooperative opportunities. Moreover, it is possible that the HLLV, or elements thereof, could be made available to other foreign space organizations to cooperatively launch their own space exploration elements in an effort to increase launch rates and offset program costs.

10.6. Commercial Path to an Affordable Heavy Lift System

Orbital proposes a Commercial procurement and contracting approach in order to meet the affordability and sustainability KDAs established by NASA, as discussed in this section.



In order to meet the ambitious schedule for development that Congress has dictated, and to minimize development costs, Orbital advocates that NASA look to adopt a commercialized development and contracting approach. The varied branches of the U.S. Government, NASA included, have an impressive record of bringing large, ambitious, complex systems to fruition. These programs however have spanned long development cycles, further impacted by the hierarchy of decision making in this large organizations. Moreover, the cost-plus contracting methodology traditionally employed tends to lead to significant cost escalation as such programs mature, due in part to evolving requirements over this long development period. In contrast, typical commercial practices such as those employed by Orbital are characterized by an evolutionary, incremental development approach facilitating shorter development cycles. A strong customer focused teamwork approach on such programs helps stabilize requirements and enables low-risk firm fixed price contracting. While the magnitude of the HLS program dictates a NASA managed development, Orbital strongly advocates that NASA adopt elements of a commercial approach in this ambitious development program, and maintain them through the sustainment of launch services for the heavy lift vehicles.

Commercial Approach to Affordable HLS

Concept 2A with "Commercial" Approach

- System or specific elements are contracted similar to COTS Program Model
- Lowest DDT&E and LCC of all approaches evaluated
  - RP/LOX Core with RP Liquid Strap-On Boosters are developed and procured commercially
  - Maximum use of off-site, shared Commercial Facilities (e.g.; Jacobs engineering and ULA) and Commercial Contracting (e.g.; FFP for recurring cores and LRBs)
  - "Just-in-time" integration approach Minimizes Personnel, Processes, Facilities, etc.
  - Core and LRB processed and integrated offsite, with final integration at VAB (i.e.; true "Campaign Style" launch approach)

  - Employs "Clean Pad" approach to reduce recurring launch ops costs (minimal time on pad, reduces overall Life Cycle Costs)
  - No Standing Army (majority of integration /launch team travels with launch vehicle)
  - Spreads the work across Multiple States /maximizes congressional support

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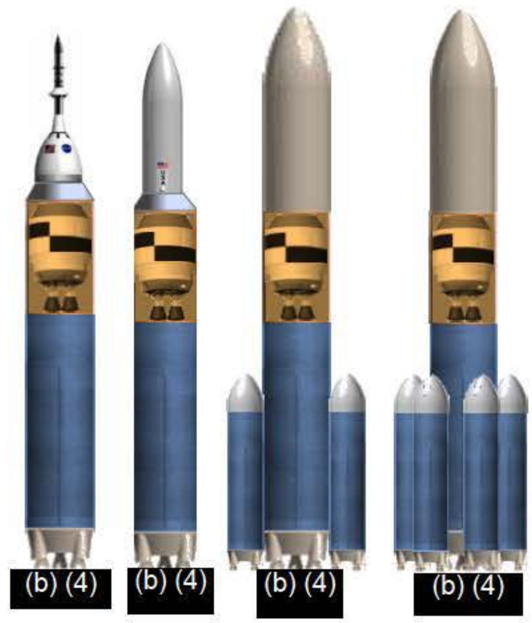
Orbital maintains that a development approach employing commercial aspects will enable NASA to successfully develop and operate a HLS system in the most efficient and cost effective manner. Orbital suggests that NASA adopt an approach similar to that currently being demonstrated in the NASA CRS program; a successful NASA / Industry partnership. By setting only overarching requirements, and subcontracting elements of the vehicle directly to industry, low DDT&E and Life Cycle Costs (LCC) can be achieved. Other aspects of such a commercial approach include maximum use of shared facilities, "Just in Time" contracting and integration approaches, and a "clean pad" approach to vehicle launch processing.

Commercially Procured RP Core/RP LRB

- **Fairing –** Use Commercially Available 5.4m PLF initially, Evolve to Larger Fairings as Required
- **Upper Stage –** J-2X based Concept with 27 ft (8.4m) Structure / Tankage Manufactured on Current Shuttle ET line in Michoud
- **Common 27 ft (8.4m) Structure/Tankage**
  - Commercially procured & manufactured on current Shuttle ET line in Michoud
- **Liquid RBs**
  - 4m Liquid Rocket Booster (LRB) Strap-Ons affordably derived from existing Taurus II
  - Procured commercially
- **Booster & Core Propulsion**
  - AJ-26X provides logical /cost effective evolution for HLS core and LRB strap-on
  - Common HLS core/LRB strap-on engine increases performance and reduces risk/costs
  - Usable by multiple LVs
    - ➢ Efficiently meets needs of all end-users (NASA, USAF, Atlas, Taurus II)
    - ➢ Significant cost savings with common U.S. engine across multiple end-users/launch systems



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Integrating Orbital's suggested launch vehicle configuration with the technology development plan and commercial contracting approach suggested yields a family of vehicle configurations to meet the full spectrum of NASA HLS requirements; all based upon a common LOx/RP fueled, AJ-26X powered booster core. On the low end, a variant with just this core booster and the dual engine J-2X based upper stage with a nominal performance of (b) (4) and an "enhanced reliability" or engine out performance of (b) (4) would meet NASA Crew requirements. This same vehicle utilizing a commercially available 5.4M diameter payload fairing would meet large ISS cargo and lunar vehicle requirements. Adding a pair of L O2/RP liquid strap-ons, utilizing two AJ-26X engines each, increases the nominal performance to (b) (4). Lastly, by adding an additional pair of liquid strap-ons, four total, the vehicle capability is increased to over (b) (4) Employing this evolutionary performance approach enables this HLS system to meet all of NASA's primary DRMs.

11. HEAVY LIFT ROADMAP

Further integrating Orbital's proposed HLS vehicle configurations into the NASA DRM manifest results in a road map for meeting NASA's Heavy Lift System requirements, as shown above. Starting with the AJ-26X engine development and an ATP in the early 2012 timeframe, the initial crewed vehicle configuration could be developed to meet the 2016 ILC deadline. The cargo variant could also be used in this same timeframe. A straightforward upgrade to add the liquid strap-on boosters could be executed rapidly to meet heavier lift requirements when required, in 2023 or earlier. A larger payload fairing could also be developed in this timeframe. [redacted] (b) (4) (b) (4)

Two other elements were considered that could further influence and augment this plan. To increase overall vehicle reliability predictions and to augment performance, the higher thrust 1M lbf variant of the AJ-26X engine could be developed. Cost savings depicted earlier possibly fund this development. Incorporating higher thrust variants of these main engines would be relatively straightforward, and could also possible lead to still lower recurring and operating costs. Another element considered was the incorporation of liquid fly-back strap-on boosters. This technology, currently being studied aggressively by the USAF, could be adapted to the HLLV and might further reduce recurring costs. While the maturity of this technology was not considered sufficient to warrant direct incorporation into our selected vehicle configurations, it does show promise in reducing recurring costs, and variants of the fly-back or rocket-back boosters currently being considered incorporate the same AJ-26X engine advocated by Orbital in this study.

12. FINDINGS AND RECOMMENDATIONS

Findings and Recommendations

- **Affordable HLS Vehicles can be Derived from Existing Technology with Targeted Investment in Selected Key Areas**
  - RP ORSC Engine Development and Production in U.S. (e.g.; AJ-26)
  - Liquid RP Strap-On Boosters derived directly from existing launch vehicle technology with limited investment utilizing COTS model (e.g.; Taurus II)
- **DRM Assessment Supports Evolutionary Vehicle Development**
  - Intermediate Performance capability with a Flexible Configuration tailorable to Specific DRMS
  - **HLLV evolves to meet DRM requirements *When Needed***
- Candidate HLS Booster Configurations can be Either LOX/RP or LOX/H2
  - **LOX/RP Solutions have Lower Life-Cycle Costs and Greater Applicability Across U.S. Vehicle Base and Multiple End-Users** (i.e.; NASA, Air Force, Commercial)
- Propulsion System Costs Dominate Recurring and Life-Cycle Costs
  - **Payback for limited development is quickly realized**

Evolutionary Commercial Procurement Approach Provides an Affordable and Sustainable HLS program

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Orbital's study conclusively shows that affordable HLS vehicles can be developed and derived from existing technology with target, limited investment in selected key technology areas. NASA's Design Reference Mission manifest supports an evolutionary vehicle development approach which fields a initial launch vehicle capable of reliably meeting initial DRM requirements by the 2016 deadline, and evolves that basic vehicle in incremental steps to meet heavier lift requirements when needed. Orbital's study further concludes that while configurations to effectively meet NASA's requirements can incorporate either LOx\Hydrogen or LOx\RP engines, LOx\RP solutions will have lower life cycle costs and afford greater applicability across the U.S. launch vehicle base facilitating further cost reductions due to economies of scale.

The Orbital Sciences Corporation has an extensive history of providing reliable space systems solutions to both government and commercial customers. Our comprehensive product base spans all segments of the launch vehicle, satellite and space infrastructure industry. Orbital stands ready to bring these resources and experience to bear, and to work with NASA to develop a reliable, sustainable, cost-effective heavy lift launch system to meet NASA and the Nation's ambitious space exploration goals.

APPENDIX

APPENDIX A – ORBITAL HLS STUDY COMPLIANCE MATRIX

Heavy Lift and Propulsion Technology Systems Analysis and Trade Study BAA Technical Objectives Compliance Matrix				
ORBITAL SOW PARAGRAPH		Section III Technical Objective	COMPLIANT?	REFERENCE
1.4.1	**Heavy Lift System Architecture Study**	Identify how alternative heavy lift system solutions address key decision attributes/figures of merit/measures of effectiveness.	Y	Section 3.1, Pages 11-12, Section 3.3, Pages 16-28,
1.4.2	**Architecture Development**	Identify and analyze multiple alternative architectures (expendable, reusable, or some combination) on which a Heavy Lift System addressing the objectives can be based.	Y	Section 3.3, Pages 16-28
1.4.3	**Architecture Assessment**	Provide a recommended list of key decision attributes and rationale associated with each.	Y	Section 3.1, Pages 11-12
1.4.3	**Architecture Assessment**	Provide a recommendation for the weighting of the recommended key decision attributes.	Y	Section 3.3, Pages 16-28,
1.4.4	**Architecture Sensitivity Assessment**	Identify how changes to the weighting of key decision attributes affect the architectures.	Y	Section 3.3, Pages 16-28
1.4.4	**Architecture Sensitivity Assessment**	Identify how alternative ground rules and assumptions (Reference NASA HLLV Study) impact the identified alternative system solutions.	Y	Section 3.1, Pages 13-14 Section 3.3, Pages 16-28,
1.4.4	**Architecture Sensitivity Assessment**	Identify how changes to the weighting of key decision attributes affect the architectures.	Y	Section 3.3, Pages 16-28,
1.4.5	**Capability Gap Assessment**	Identify how innovative or non-traditional processes or technologies can be applied to the Heavy Lift Systems to dramatically improve its affordability and sustainability.	Y	Section 4., Pages 29-60 Section 4.6, Pages 65-67
1.4.7	**Cost Reduction Strategies**		Y	Section 10.0, Pages 93-106 Section 10.6, Pages 107-110
1.4.5	**Capability Gap Assessment**	Identify capability gaps associated with the first-stage main engine functional performance and programmatic characteristics required to support each Heavy Lift System studied. The minimum set of functional performance characteristics identified shall include engine thrust, specific impulse (Isp), mixture ratio, mass, throttle range, and physical envelope.	Y	Section 4.1, Pages 40-46
1.4.7	**Cost Reduction Strategies**	The minimum set of programmatic characteristics identified for the First Stage propulsion engine shall include an estimated overall lifecycle cost (i.e., DDT&E, production and operations (fixed and variable) per engine cost), development schedule, and production rate.	Y	Section 10.0, Pages 93-105
1.4.7	**Cost Reduction Strategies**	Identify any impacts to overall life cycle costs of the Heavy Lift System based on the First-Stage Main engine studied.	Y	Section 10.0, Pages 93-105
1.4.5	**Capability Gap Assessment**	Identify capability gaps associated with the upper-stage main engine functional performance and programmatic characteristics required to support each heavy lift system studied. The minimum set of functional performance characteristics identified shall include engine propellants, thrust, specific impulse (Isp), mixture ratio, mass, throttle range, and physical envelope.	Y	Section 4.2, Pages 47 Section 9, Pages 89-92
1.4.7	**Cost Reduction Strategies**	The minimum set of programmatic characteristics identified for the Upper Stage propulsion engine shall include an estimated overall lifecycle cost (i.e., DDT&E, production and operations (fixed and variable) per engine cost), development schedule, and production rate.	Y	Section 10.0, Pages 93-105
1.4.7	**Cost Reduction Strategies**	Identify any impacts to overall life cycle costs of the Heavy Lift System based on the Upper-Stage Main engine studied.	Y	Section 10.0, Pages 93-105

Heavy Lift and Propulsion Technology Systems Analysis and Trade Study BAA Technical Objectives Compliance Matrix				
ORBITAL SOW PARAGRAPH		Section III Technical Objective	COMPLIANT?	REFERENCE
1.4.5	Capability Gap Assessment	Identify capability gaps associated with all other technical aspects of heavy lift system, e.g. tanks, propellant and pressurization systems, integrated system health management, auxiliary propulsion systems, avionics and control systems, structures. Identify test and integrated demonstrations to mitigate risk associated with the gaps.	Y	Section 4.4, Pages 49-60 Section 9, Pages 89-92
1.4.5	Capability Gap Assessment	Identify capability gaps associated with the in-space space propulsion elements functional performance and programmatic characteristics required to support each Heavy Lift System studied. This assessment shall include, but is not limited to, LOX/H2 and LOX/CH4 propulsion systems. The minimum set of functional performance characteristics identified shall include propellant definition, thrust, specific impulse (Isp), mixture ratio, mass, throttle range (if any), and physical envelope.	Y	Section 7.0, Pages 75-84
1.4.7	Cost Reduction Strategies	The minimum set of programmatic characteristics identified for the In-Space propulsion engine shall include an estimated overall lifecycle cost (i.e., DDT&E, production and operations (fixed and variable) per engine cost), development schedule, and production rate.	Y	Section 10.0, Pages 93-105
1.4.7	Cost Reduction Strategies	Identify any impacts to overall life cycle costs of the Heavy Lift System based on the In-Space propulsion engine studied.	Y	Section 7.0, Pages 75-84 Section 10.0, Pages 93-105
1.4.5	Capability Gap Assessment	Identify capability gaps associated with all other technical elements of the in-space space propulsion element, e.g. tanks, propellant and pressurization systems, cryogenic fluid management, integrated system health management, auxiliary propulsion systems, avionics and control systems, structures, autonomous rendezvous and docking. Identify test and integrated demonstrations to mitigate risk associated with the gaps.	Y	Section 4.4, Pages 49-60 Section 7.0, Pages 75-84
1.4.6	Test Planning Strategies	Identify what in-space space propulsion elements, if any, which should be demonstrated via space flight experiments.	?	Section 7.0, Pages 75-84
1.4.5	Capability Gap Assessment	Identify capability gaps associated with the Heavy Lift System, and for each capability gap identify specific areas where technology development may be needed. Items identified as requiring technology development shall be quantitatively evaluated using established metrics, e.g. NASA Technology Readiness Level (TRL), Cost Readiness Level (CRL), Manufacturing Readiness Level (MRL), System Readiness Level (SRL).	Y	Section 4., Pages 29-39
1.4.6	Test Planning Strategies	Identify how incremental development testing, including ground and flight testing, of Heavy Lift System elements can enhance the heavy lift system development.	Y	Section 4.1, Pages 40-46
1.4.6	Technology RoadMap	Develop proposed technology development road maps inclusive of test strategies consistent with gap assessments to mitigate development risks.	Y	Section 11.0, Pages 111-112
1.4.7	Commonality with Other User Applications	Identify how aspects of a Heavy Lift System (including stages, subsystems, and major components) could have commonality with other user applications, including NASA, DoD, commercial, and international partners.	Y	Section 4.6, Pages 65-67 Section 10.0, Pages 93-105

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APPENDIX B – STATEMENT OF WORK (SOW)

Heavy Lift & Propulsion Technology (HLPT) Systems Analysis and Trade Study

Statement Of Work (SOW) Rev. A

Revised 11 November 2010

1. The Contractor Statement of Work (SOW)

The Orbital Sciences Corporation ["the contractor"] proposes the following SOW which is suitable for incorporation into a contract. Included in this proposed SOW is our approach to completing the contract deliverables, as stated in Section VIII, Paragraph 9.0 of the Heavy Lift and Propulsion Systems Analysis and Trade Study Broad Agency Announcement (BAA).

1.1 Management

The contractor shall manage, control, and approve all work performed to accomplish the requirements of the SOW.

The contractor shall monitor, control, and report project budgets and schedules; define, monitor, and control the project contract.

The contractor shall provide a single point of contractual authority, communications, and contractual definition for the project to National Aeronautics and Space Administration (NASA).

The contractor shall manage, control, and approve all work performed by subcontractors of the program.

The contractor shall evaluate and report potential risks to accomplishing project objectives to NASA as they are identified.

1.2 Reviews

The contractor shall support weekly Telecons with NASA to status progress on study, issues, and action items.

The contractor shall conduct a one-day Technical Interchange Meeting #1 to status accomplishments, planned work, identifying issues and resolution plans, and reviewing action items at NASA. Track any action items to closure.

The contractor shall conduct a one-day Technical Interchange Meeting #2 to status accomplishments, planned work, identifying issues and resolution plans, and reviewing action items at NASA. Track any action items to closure.

1.3 Reporting

The Contractor shall report and document this work and fulfill the requirements of associated Data Requirement Descriptions (DRD's) as outlined in Data Procurement Document (DPD) 1380 (Attachment J- 2). The contractor shall determine the data restriction that applies to each data deliverable and mark or transmit the data restriction in accordance with section 2.3.3 of the Data Procurement Document.

The contractor shall prepare and deliver a Technical Interchange Meeting #1 and #2 Briefing Packages that provide a status of accomplishments, planned work, identifying issues and resolution plans, and reviewing action items in accordance with DRD 1380MA-002. [Total of two (2)

Briefing Packages]

The contractor shall prepare and deliver a Final Scientific and Technical Report that complies with the requirements of NFS 1852.235-73 and includes all of the research, assessments, trade study results, system architecture assessment, gap-analyses, vehicle sizing assessment, technology roadmap, and the draft of an executable engine test plan in accordance with DRD 1380MA-001. [Total of one (1) Final Report with supporting data]

1.4 Systems Engineering

The contractor shall provide a single point of decision authority and communication for Heavy Lift System (HLS) Program systems engineering.

1.4.1 Heavy Lift System Architecture Study

The contractor shall evaluate the overall systems architecture to meet NASA HLS mission objectives:

- Starting from the President's Vision for Space Exploration, Level 0 requirements, and Draft Level 1 requirements supplied by NASA; the contractor shall derive scientific, economic, and security goals and objectives for a HLS architecture that supports evolutionary human space exploration activities, with destinations including the Moon, Mars and its environs, near-earth asteroids, and Lagrange points.
- The contractor shall perform system-level trade studies in support of requirements development activities, determine trade study decision criteria, select preferred options, and record results.

1.4.2 Architecture Development

The contractor shall develop candidate architectures that satisfy the mission requirements and objectives.

1.4.3 Architecture Assessment

The contractor shall assess the overall systems architecture chosen to meet NASA HLS mission objectives.

The contractor shall provide a recommended list of HLS key decision attributes and rationale associated with each.

The contractor shall provide a recommendation for the weighting of the recommended HLS key decision attributes.

1.4.4 Architecture Sensitivity Assessment

The contractor shall identify how changes to the weighting of key decision attributes affect the HLS architectures.

The contractor shall identify how alternative ground rules and assumptions impact the identified alternative HLS solutions.

The contractor shall analyze any variations in weighting and evaluate their impact on the HLS architecture rating.

The contractor shall identify how innovative or non-traditional processes, tools, and technologies can be applied the HLS to improve affordability and sustainability.

1.4.5 Capability Gap Assessment

The contractor shall perform a capability gap assessment for major HLS elements, and identify the functional performance characteristics required (i.e., thrust, Isp, mixture ratios, mass, throttle range, physical envelope, life-cycle costs, development schedules, and production rates), including:

- First and Second stage engines (specifically including liquid oxygen/kerosene (LOx/RP-1) first and second stage propulsion systems and associated propulsion system elements)
- In-space systems and elements (e.g.; upper-stage engines, fuel depots, transfer stages, etc.)
- Heavy lift launch system elements consistent with identified preferred configuration(s) (e.g., tanks, propellant and pressurization systems, integrated system health management, auxiliary propulsion systems, avionics and control systems, structures).

The contractor shall assess the Technology Development necessary to meet identified preferred HLS configuration and propulsion systems requirements, and quantitatively evaluate technology status using established metrics, specifically NASA Technology Readiness Level (TRL), ~~Capability~~ Cost Readiness Level (CRL), Manufacturing Readiness Level (MRL), and ~~Process~~ System Readiness Level (SRL).

The contractor shall identify technology capability gaps in the HLS architecture and identify specific areas where technology development is needed.

The contractor shall identify impacts to the HLS life-cycle costs resulting from the architecture elements studied.

1.4.6 Test Planning Strategies

- The contractor shall selectively identify suggested incremental development testing, including ground and flight testing, of architecture study identified HLS elements, including first, second and upper stage propulsion elements, to enhance the Heavy Lift System development.
- The contractor shall develop proposed technology development road maps inclusive of test strategies consistent with gap assessments to mitigate development risks:
  - As an example, if supported by the architecture study, the contractor shall provide a roadmap for developing modular liquid oxygen/kerosene (LOx/RP-1) launch vehicle engine propulsion systems that shall support implementation of a propulsion system development program in support of Heavy Lift.
  - Further, if supported by the architecture study the contractor shall define a plan to utilize an available AJ26 engine to obtain significant insight and data relative to the performance and behavior of Oxygen-rich, staged combustion systems.

1.4.7 Cost Reduction Strategies and Commonality with Other User Applications

- The contractor shall identify aspects of a HLS (including stages, subsystems, and major components) with commonality for other user applications, including NASA, Department of Defense (DoD), commercial, and international partners, including:

- If supported by the architecture study the contractor shall define incremental approaches that lead from technology characterization to new engine system development in a short period of time.
- As an example, if supported by the architecture study, the contractor shall study and define the parameters and performance characteristics for a scalable main engine system that shall have broad application not only to NASA Heavy Lift, but to other launch vehicles and user applications, including broader NASA missions, DoD, commercial, and international partners.
- If supported by the architecture study, the contractor shall also study and propose a plan to adapt this new engine system to existing vehicles in the U.S. fleet, specifically the Taurus II and Atlas V.
- Further, if supported by the architecture study, the contractor shall also evaluate and assess the potential economic benefits of a widely utilized engine system and how, once in production, the use of these engine systems in existing vehicles could reduce life cycle costs.

- The contractor shall identify how innovative or non-traditional processes or technologies can be applied to the HLS to improve affordability and sustainability.
  - The contractor shall assess the applicability and benefits of selective innovative technologies implemented by the contractor in other areas.
  - The contractor shall assess the applicability and benefits of reusable system elements.
  - The contractor shall assess the applicability of technology elements from foreign partners.
  - The contractor shall discuss how commercial practices might be applied to accelerate development of Heavy Lift System elements.

2.0 Additional Data Deliverables

The contractor shall provide technical information concerning any invention, discovery, improvement, or innovation made by the contractor in the performance of work under this contract. Technology Reports shall be prepared in accordance with DRD 1380CD-001.

The contractor shall prepare and submit a Final Study Report in accordance with DRD 1380MA-003.

The contractor shall prepare and submit an Organizational Conflict of Interest (OCI) Mitigation Plan in accordance with DRD 1380MA-004.

The contractor shall report mishaps and safety statistics to the MSFC Industrial Safety Branch in accordance with DRD 1380SA-001. The contractor shall submit directly into the NASA Incident Reporting Information System (IRIS) or shall use the forms listed in section 15.4 of DRD 1380SA-001 or electronic equivalent to report mishaps and related information required to produce the safety metrics.

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APPENDIX C – HEAVY LIFT SYSTEM REQUIREMENTS AND CONSTRAINTS

SECTION#	SECTION TITLE	SECTION TEXT	Requirement#	Requirement Tittle	Requirement Text
HLS-1	SCOPE	This specification establishes the requirements for the NASA Heavy Lift System (HLS).			
HLS-2	APPLICABLE DOCUMENTS				
HLS-2.1	Government Documents				
HLS-2.2	Non-Government Documents				
HLS-3	REQUIREMENTS	This section contains all of the HLS requirements.			
HLS-3.1	System Definition	The Heavy Lift System (HLS) includes all requirements associated with developing the NASA HLS, including but not limited to the Ground Segment, the Launch Segment, and the In-Space Segment.			
HLS-3.2	System Characteristics	The Heavy Lift System (HLS) ...			
HLS-3.3	Ground Segment	This section contains all of the Ground Segment requirements for the Heavy Lift System (HLS).			
HLS-3.3.1	Integration	This section contains all of the Integration requirements for the Heavy Lift System (HLS).	HLS-057	Integration Approach	The HLS Integration Approach shall be Vertical.
HLS-3.3.1	Integration	This section contains all of the Integration requirements for the Heavy Lift System (HLS).	HLS-058	Launch Vehicle Stack Integrated Height Constraint	The HLS Launch Vehicle Stack Integrated Height Constraint shall be 390 ft.
HLS-3.3.2	Launch Pad	This section contains all of the Launch Pad requirements for the Heavy Lift System (HLS).	HLS-056	KSC Space Launch Complex 39 (SLC-39)	The HLS shall be Launchable from KSC Space Launch Complex 39 (SLC-39).
HLS-3.3.3	Transportation and Handling	This section contains all of the Transportation and Handling requirements for the Heavy Lift System (HLS).			
HLS-3.4	Launch Segment	This section contains all of the Launch Segment requirements for the Heavy Lift System (HLS).			
HLS-3.4.1	HLLV Performance	This section contains all of the Performance requirements for developing the Heavy Lift Launch Vehicle (HLLV).	HLS-001	Minimum Performance to LEO	The minimum HLS performance to LEO shall be 100 mt.
HLS-3.4.1	HLLV Performance	This section contains all of the Performance requirements for developing the Heavy Lift Launch Vehicle (HLLV).	HLS-002	Flight Performance Reserve	The Flight Performance Reserve (FPR) of total ideal dV for the mission shall be a minimum of 1%.
HLS-3.4.1	HLLV Performance	This section contains all of the Performance requirements for developing the Heavy Lift Launch Vehicle (HLLV).	HLS-024	Max Acceleration: Inline Cargo only Missions (g)	The HLS Max acceleration for Inline Cargo only Missions shall not exceed 5g.
HLS-3.4.1	HLLV Performance	This section contains all of the Performance requirements for developing the Heavy Lift Launch Vehicle (HLLV).	HLS-025	Max Acceleration Inline Crew Missions	The HLS Max acceleration for Inline Crew Missions shall not exceed 4g.

SECTION#	SECTION TITLE	SECTION TEXT	Requirement#	Requirement Tittle	Requirement Text
HLS-3.4.1	HLLV Performance	This section contains all of the Performance requirements for developing the Heavy Lift Launch Vehicle (HLLV).	HLS-026	Max Acceleration: Sidemount All Missions (g)	The HLS Max acceleration for all Sidemount Missions shall not exceed 3g.
HLS-3.4.1	HLLV Performance	This section contains all of the Performance requirements for developing the Heavy Lift Launch Vehicle (HLLV).	HLS-027	Max Dynamic Pressure: Inline	The HLS Max dynamic pressure for Sidemount designs shall not exceed 650 psf.
HLS-3.4.1	HLLV Performance	This section contains all of the Performance requirements for developing the Heavy Lift Launch Vehicle (HLLV).	HLS-028	Max Dynamic Pressure: Sidemount	The HLS Max dynamic pressure for Inline designs shall not exceed 800 psf.
HLS-3.4.1	HLLV Performance	This section contains all of the Performance requirements for developing the Heavy Lift Launch Vehicle (HLLV).	HLS-034	MECO Altitude	The HLS MECO altitude shall occur at 75 nmi.
HLS-3.4.1	HLLV Performance	This section contains all of the Performance requirements for developing the Heavy Lift Launch Vehicle (HLLV).	HLS-035	TLI(LOR) dV	The HLS TLI(LOR) dV from 130 nmi circ shall occur at 3165 m/s.
HLS-3.4.1	HLLV Performance	This section contains all of the Performance requirements for developing the Heavy Lift Launch Vehicle (HLLV).	HLS-049	Launch Vehicle Ultimate Strength Safety Factor	The HLS Launch vehicle ultimate strength safety factor for new stages shall be 1.4.
HLS-3.4.2	HLLV Engines	This section contains all of the requirements for developing the Heavy Lift Launch Vehicle (HLLV) engines.			
HLS-3.4.3	HLLV Core	This section contains all of the requirements for developing the Heavy Lift Launch Vehicle (HLLV) core.	HLS-003	Maximum Core Stage Length	The Maximum Core Stage Length shall be 71.3 m (234 ft)
HLS-3.4.3	HLLV Core	This section contains all of the requirements for developing the Heavy Lift Launch Vehicle (HLLV) core.	HLS-004	Maximum Diameter Constraint	The core maximum diameter shall be constrained to 10 m (33 ft)
HLS-3.4.4	HLLV Fairing	This section contains all of the requirements for developing the Heavy Lift Launch Vehicle (HLLV) fairing/payload shroud.	HLS-006	LEO 8.4 m (27.5 ft) Fairing length	The minimum LEO 8.4 m (27.5 ft) fairing dynamic envelope length shall be 25 m (82 ft).
HLS-3.4.4	HLLV Fairing	This section contains all of the requirements for developing the Heavy Lift Launch Vehicle (HLLV) fairing/payload shroud.	HLS-007	LEO 8.4 m (27.5 ft) Fairing diameter	The minimum LEO 8.4 m (27.5 ft) fairing dynamic envelope diameter shall be 7.5 m (24 ft).
HLS-3.4.4	HLLV Fairing	This section contains all of the requirements for developing the Heavy Lift Launch Vehicle (HLLV) fairing/payload shroud.	HLS-008	LEO 10 m (33 ft) Fairing length	The minimum LEO 10 m (33 ft) fairing dynamic envelope length shall be 20 m (65 ft).
HLS-3.4.4	HLLV Fairing	This section contains all of the requirements for developing the Heavy Lift Launch Vehicle (HLLV) fairing/payload shroud.	HLS-009	LEO 10 m (33 ft) Fairing diameter	The minimum LEO 10 m (33 ft) fairing dynamic envelope diameter shall be 9.1 m (30 ft).
HLS-3.4.4	HLLV Fairing	This section contains all of the requirements for developing the Heavy Lift Launch Vehicle (HLLV) fairing/payload shroud.	HLS-036	Fairing Jettison	The HLS Fairing shall be jettisoned when 3-sigma Free Molecular Heating Rate is less than 0.01 BTU/ftÂ²-sec.
HLS-3.4.5	HLLV Strap On Booster	This section contains all of the requirements for developing the Heavy Lift Launch Vehicle (HLLV) strap on booster.			

SECTION#	SECTION TITLE	SECTION TEXT	Requirement#	Requirement Tittle	Requirement Text
HLS-3.5	In-Space Segment	This section contains all of the In-Space Segment requirements for the Heavy Lift System (HLS).			
HLS-3.5.1	Upper Stage	This section contains all of the requirements for developing the Heavy Lift System (HLS) Upper Stage.			
HLS-3.5.2	Transfer Stage	This section contains all of the requirements for developing the Heavy Lift System (HLS) Transfer Stage.			
HLS-3.5.3	Orion Capsule	This section contains all of the requirements for ensuring that the Heavy Lift System (HLS) is compatible with the Orion Capsule.	HLS-005	Payload: Accommodate Orion MPCV	The HLS LV shall accommodate the Orion MPCV.
HLS-3.5.3	Orion Capsule	This section contains all of the requirements for ensuring that the Heavy Lift System (HLS) is compatible with the Orion Capsule.	HLS-037	Launch Abort System (LAS) Mass	The Launch Abort System (LAS) mass shall be defined as 16,005 lbm.
HLS-3.5.3	Orion Capsule	This section contains all of the requirements for ensuring that the Heavy Lift System (HLS) is compatible with the Orion Capsule.	HLS-038	Boost Protect Cover (BPC) Mass	The Boost Protect Cover (BPC) mass shall be defined as 2331 lbm.
HLS-3.5.3	Orion Capsule	This section contains all of the requirements for ensuring that the Heavy Lift System (HLS) is compatible with the Orion Capsule.	HLS-039	BPC Jettison	The BPC jettison shall occur 27 seconds after Upper Stage ignition.
HLS-3.5.3	Orion Capsule	This section contains all of the requirements for ensuring that the Heavy Lift System (HLS) is compatible with the Orion Capsule.	HLS-040	LAS Jettison	The LAS jettison shall occur 30 seconds after Upper Stage ignition.
HLS-3.5.3	Orion Capsule	This section contains all of the requirements for ensuring that the Heavy Lift System (HLS) is compatible with the Orion Capsule.	HLS-041	LAS+CM+SM+Vehicle Adapter Length	The combined LAS+CM+SM+vehicle adapter length shall be defined as 70.7 ft.

SECTION#	SECTION TITLE	SECTION TEXT	Requirement#	Requirement Tittle	Requirement Text
HLS-3.6	Programmatic	This section contains all of the Programmatic requirements for the Heavy Lift System (HLS).	HLS-019	Govâ€™t Oversight/Insight	The HLS shall ensure that the U.S. Government (NASA) has oversight/insight into the program.
HLS-3.6.1	Cost	This section contains all of the Cost requirements for the Heavy Lift System (HLS).	HLS-017	SLC Costs from ATP through 1st Flight	SLC Costs from ATP through 1st flight shall not exceed $11.5 B.
HLS-3.6.1	Cost	This section contains all of the Cost requirements for the Heavy Lift System (HLS).	HLS-018	Combined Annual SLS LV and KSC Ground Processing Costs	The HLS combined SLS LV and KSC Ground Processing Costs shall not exceed $1.2B annually.
HLS-3.6.2	Schedule	This section contains all of the Schedule requirements for the Heavy Lift System (HLS).	HLS-014	Minimum Flight Rate	The HLS shall have a Minimum Flight Rate of 2 flights per year.
HLS-3.6.2	Schedule	This section contains all of the Schedule requirements for the Heavy Lift System (HLS).	HLS-015	ATP Vehicle Development	The HLS shall have an ATP Vehicle development (Phase A) start of FY2011.
HLS-3.6.2	Schedule	This section contains all of the Schedule requirements for the Heavy Lift System (HLS).	HLS-016	Launch System Operational Capability Readiness	The HLS shall have a Launch System Operational Capability Readiness (a.k.a.; IOC) by CY2016
HLS-3.6.2	Schedule	This section contains all of the Schedule requirements for the Heavy Lift System (HLS).	HLS-055	Maximum Flight Rate	The HLS Maximum Flight Rate shall be 4 flights per year.
HLS-3.7	Human Rating	This section contains all of the Human Rating requirements for the Heavy Lift System (HLS).	HLS-010	Human Rated	The HLS shall be Human Rated.
HLS-3.7	Human Rating	This section contains all of the Human Rating requirements for the Heavy Lift System (HLS).	HLS-011	Safety & Reliability	The HLS Safety & Reliability compared to Shuttle shall be a minimum of 10 times better.
HLS-3.7	Human Rating	This section contains all of the Human Rating requirements for the Heavy Lift System (HLS).	HLS-012	Anytime Abort capability	The HLS shall have Anytime Abort capability.
HLS-3.7	Human Rating	This section contains all of the Human Rating requirements for the Heavy Lift System (HLS).	HLS-013	Single Failure Tolerance	The HLS shall be Single Failure tolerant to catastrophic event or utilize some other approved DFMR approach.
HLS-3.8	Evolved HLS	This section contains all of the requirements for Evolving the Heavy Lift System (HLS) beyond the initial operational capability.	HLS-050	Evolved Performance to LEO	The HLS Evolved Performance to LEO shall be 150 mt.
HLS-3.8	Evolved HLS	This section contains all of the requirements for Evolving the Heavy Lift System (HLS) beyond the initial operational capability.	HLS-051	Beyond LEO 8.4 m (27.5 ft) Fairing length	The minimum beyond LEO 8.4 m (27.5 ft) fairing dynamic envelope length shall be 9 m (30 ft).
HLS-3.8	Evolved HLS	This section contains all of the requirements for Evolving the Heavy Lift System (HLS) beyond the initial operational capability.	HLS-052	Beyond LEO 8.4 m (27.5 ft) Fairing Diameter	The minimum beyond LEO 8.4 m (27.5 ft) fairing dynamic envelope diameter shall be 7.5 m (24 ft).
HLS-3.8	Evolved HLS	This section contains all of the requirements for Evolving the Heavy Lift System (HLS) beyond the initial operational capability.	HLS-053	Beyond LEO 10 m (33 ft) Fairing Length	The minimum beyond LEO 10 m (33 ft) fairing dynamic envelope length shall be 9 m (82 ft).
HLS-3.8	Evolved HLS	This section contains all of the requirements for Evolving the Heavy Lift System (HLS) beyond the initial operational capability.	HLS-054	Beyond LEO 10 m (33 ft) Fairing Diameter	The minimum beyond LEO 10 m (33 ft) fairing dynamic envelope diameter shall be 9.1 m (29 ft).

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APPENDIX D – TECHNOLOGY READINESS ASSESSMENT

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Technology Readiness Levels (TRL) Color Coded

TRL	Definition	Description
9	Actual system 'flight proven' through successful mission operations	In almost all cases, the end of last 'bug fixing' aspects of true 'system development'. This might include integration of new technology into an existing system. This TRL does *not* include planned product improvement of ongoing or reusable
8	Actual system completed and 'flight qualified' through test and demonstration (ground or space)	In almost all cases, this level is the end of true 'system development' for most technology elements. This might include integration of new technology into an existing system.
7	System prototype demonstration in a space environment	TRL 7 is a significant step beyond TRL 6, requiring an actual system prototype demonstration in a space environment. The prototype should be near or at the scale of the planned operational system and the demonstration must take place in space.
6	System/subsystem model or prototype demonstration in a relevant environment (ground or space)	A major step in the level of fidelity of the technology demonstration follows the completion of TRL 5. At TRL 6, a representative model or prototype system or system - which would go well beyond ad hoc, 'patch-cord' or discrete component level breadboarding - would be tested in a relevant environment. At this level, if the only 'relevant environment' is the environment of space, then the model/prototype must be demonstrated in space.
5	Component and/or breadboard validation in relevant environment	At this level, the fidelity of the component and/or breadboard being tested has to increase significantly. The basic technological elements must be integrated with reasonably realistic supporting elements so that the total applications (component-level, sub-system level, or system-level) can be tested in a 'simulated' or somewhat realistic environment.
4	Component and/or breadboard validation in laboratory environment	Following successful "proof-of-concept" work, basic technological elements must be integrated to establish that the "pieces" will work together to achieve concept-enabling levels of performance for a component and/or breadboard. This validation must be devised to support the concept that was formulated earlier, and should also be consistent with the requirements of potential system applications. The validation is "low-fidelity" compared to the eventual system: it could be composed of ad hoc discrete components in a laboratory.
3	Analytical and experimental critical function and/or characteristic proof of concept	At this step in the maturation process, active research and development (R&D) is initiated. This must include both analytical studies to set the technology into an appropriate context and laboratory-based studies to physically validate that the analytical predictions are correct. These studies and experiments should constitute "proof-of-concept" validation of the applications/concepts formulated at TRL 2.
2	Technology concept and/or application formulated	Once basic physical principles are observed, then at the next level of maturation, practical applications of those characteristics can be 'invented' or identified. At this level, the application is still speculative: there is not experimental proof or detailed analysis to support the conjecture.
1	Basic principles observed and reported	This is the lowest "level" of technology maturation. At this level, scientific research begins to be translated into applied research and development.

Manufacturing Readiness Levels (MRL) Color Coded

MRL	Definition	Description	Phase
9	Full Rate Production demonstrated and lean production practices in place.	This is the highest level of production readiness. Engineering/design changes are few and generally limited to quality and cost improvements. System, components or items are in production and meet all engineering, performance, quality and reliability requirements. Materials, manufacturing processes and procedures, inspection and test equipment are controlled to six-sigma or some other appropriate quality level. FRP unit cost meets goal, funding sufficient for production at required rates.	Full Rate Production/ Sustainment
8	Low Rate Production demonstrated. Capability in place to begin Full Rate Production.	Major system design features are stable and proven in test and evaluation. Materials are available to meet planned rate production schedules. Manufacturing processes and procedures are established and controlled to three-sigma or some other appropriate quality level to meet a low rate production environment. Production risk monitoring ongoing. LRIP cost goals met, learning curve validated. Actual cost model developed for FRP environment, with impact of Continuous improvement.	Production & Deployment leading to a Full Rate Production (FRP) decision.
7	Pilot line capability demonstrated. Ready to begin low rate production.	Detailed system design essentially complete and sufficiently stable to enter low rate production. All materials are available to meet planned low rate production schedule. Manufacturing and quality processes and procedures proven in a pilot line environment, under	Engineering & Manufacturing Development (EMD) leading to a Milestone C decision.
6	Capability to produce systems, subsystems or components in a production representative environment.	Detailed design is underway. Material specifications are approved. Materials available to meet planned pilot line build schedule. Manufacturing processes and procedures demonstrated in a production representative environment. Detailed producbility trade studies and risk	Engineering & Manufacturing Development(EMD) leading to Post CDR Assessment
5	Capability to produce a prototype system or subsystem in a production relevant environment.	Initial mfg approach developed. Majority of manufacturing processes have been defined and characterized, but there are still significant engineering/design changes. Preliminary design of critical components completed. Producbility assessments of key technologies complete. Prototype materials, tooling and test equipment, as well as personnel skills have been demonstrated on subsystems/systems in a production relevant environment. Cost targets allocated. Long lead and key supply chain elements identified. Industrial Capabilities Assessment (ICA) for MS B completed. Mfg strategy refined and integrated with Risk Mgt Plan. Cost model based upon detailed end-to-end value stream map.	Technology Development (TD) phase leading to a Milestone B decision.
4	Capability to produce prototype components in a laboratory environment.	Required investments, such as manufacturing technology development identified. Processes to ensure manufacturability, producbility and quality are in place and are sufficient to produce technology demonstrators. Manufacturing risks identified for prototype build. Manufacturing cost drivers identified. Producibility assessments of design concepts have been completed. Key design performance parameters identified. Special needs identified for tooling, facilities, material handling and skills.	Material Solution Analysis (MSA) leading to a Milestone A decision.
3	Manufacturing Proof of Concept Developed	Conduct analytical or laboratory experiments to validate paper studies. Experimental hardware or processes have been created, but are not yet integrated or representative. Materials and/or processes have been characterized for manufacturability and availability but further evaluation and demonstration is required.	Pre-Material Solution Analysis
2	Manufacturing Concepts Identified	Invention begins. Manufacturing science and/or concept described in application context. Identification of material and process approaches are limited to paper studies and analysis. Initial manufacturing feasibility and issues are emerging.	Pre Material Solution Analysis
1	Basic Manufacturing Concepts Identified	This is the lowest level of manufacturing readiness. Basic research expands scientific principles that may have manufacturing implications. The focus is on a high level assessment of manufacturing opportunities. The research is unfettered.	Pre Material Solution Analysis

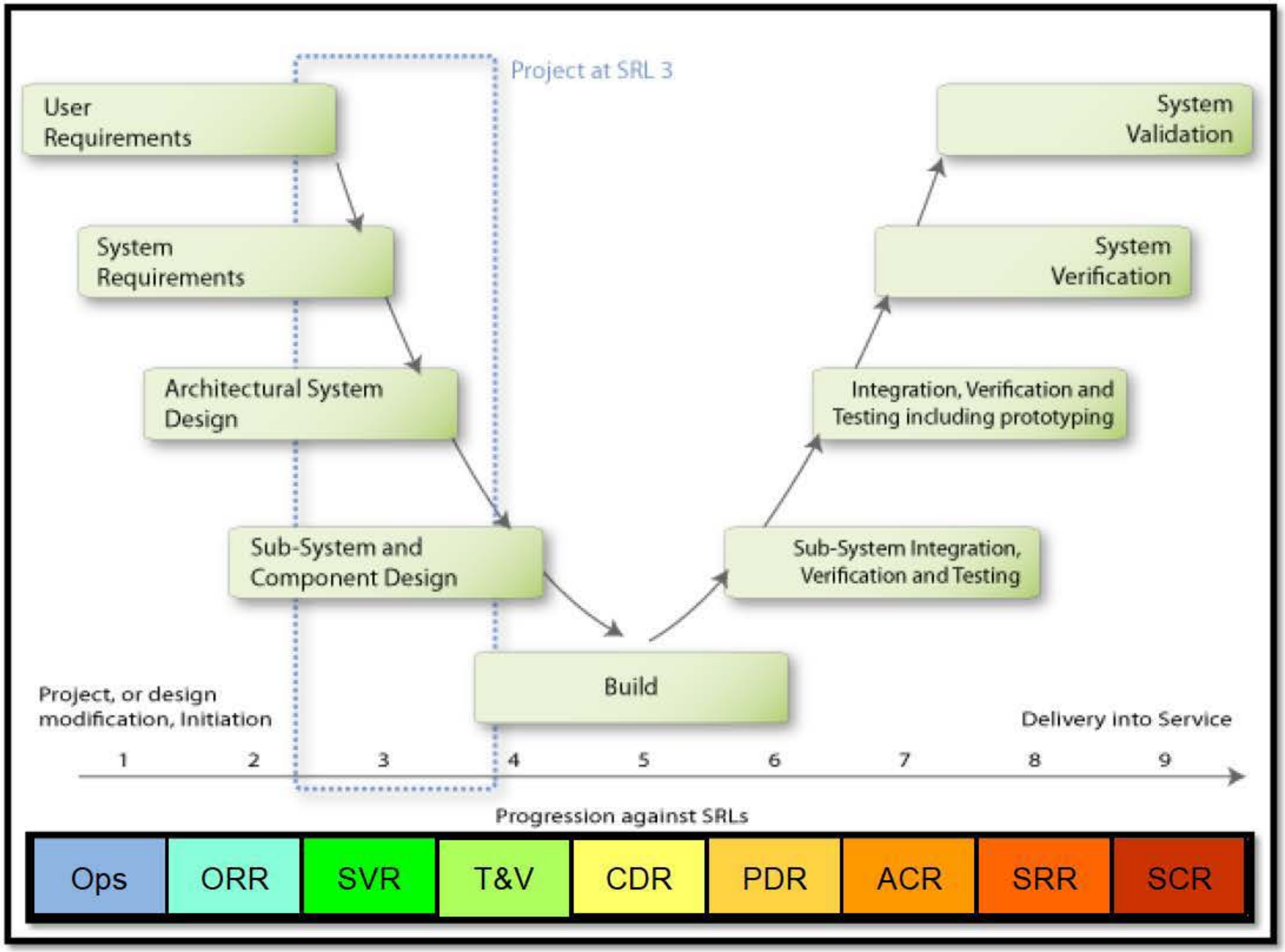
Cost Readiness Levels (CRL) Color Coded

CRL	Description
9	End of project actual cost
8	Cost fit for very firm engineering decisions and very firm budget commitments (+/- 5%)
7	Cost fit for firm engineering decisions and firm budget commitments (+/- 15%)
6	Cost fit for PDR engineering decisions and PDR budget use (+/- 25%)
5	Cost fit for preliminary engineering decisions and preliminary budget use (+/- 35%)
4	Cost fit for very preliminary engineering decisions and very preliminary budget use (+/- 45%)
1-3	Cost not fit for even very preliminary engineering decisions and very preliminary budget use (+/- 50%)

System Readiness Levels (SRL) Color Coded

9	System Deployment	Ops
8	System Verification and Validation	ORR
7	System Integration and Testing	SVR
6	Sub-System Integration and Testing	T&V
5	System Final Design and Start of Manufacturing	CDR
4	Sub-Systems and Componets have been designed.	PDR
3	System Architecture has been designed.	ACR
2	System Requirements have been developed.	SRR
1	User Requirements have been developed.	SCR

System Readiness Levels (SRL) Mapped to Program Phases



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APPENDIX E – HEAVY LIFT SYSTEM CONFIGURATION AND PERFORMANCE SUMMARY

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APPENDIX F – HEAVY LIFT SYSTEM GROUND FACILITIES ROM COST ESTIMATES

HLS Facility ROM Cost Assessment

The Orbital team analyzed the existing KSC and Stennis ground facilities and infrastructure to identify required improvements to accommodate each of the heavy lift architecture concepts.

Stennis: Engine testing infrastructure (e.g.; new/upgraded test stands)

KSC Facilities: VAB, LCC, OPF, PHSF, O&C, SSPF, RPSF

KSC Infrastructure: fueling systems, fuel tanks, Launch Pad 39A/39B, Mobile Transporter, Mobile Launch Platform

Assumes that Existing KSC Facilities and Infrastructure can be Modified and Upgraded

Stennis costs assume full up integrated engine tests (i.e.; costs driven by number of engines – no effort to determine costs for additional complexity of LH2 engines)

Shuttle derived concepts assume that existing infrastructure (e.g.; VAB, Crawler, pad fueling systems and tanks, etc.) can be used with moderate modifications/upgrades

Assumes that some new infrastructure must be developed (e.g.; new MLP, New Crawlers, new tanks, and new fueling systems for RP, etc.)

These ROM cost estimates are parametric, and actual bottoms up cost estimates need to be worked in much more detail with NASA KSC

The data provided in this Appendix was used by Orbital for reference in order to compare the costs of using existing vs. new MLPs and tank farm.

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APPENDIX G – HEAVY LIFT SYSTEM MODEL BASED SYSTEMS ENGINEERING RESULTS

Model Based Systems Engineering Approach Applied to HLS Study

Although not identified as a task or deliverable in the Statement of Work, Orbital management approved the use of applying Model Based Systems Engineering (MBSE) tools and techniques on the Heavy Lift study. The motivation for using an MBSE approach was a desire to avoid the typical document-centric systems engineering approach, and employ one that was more model-centric, more useful, and (hopefully) more efficient.

Since the MBSE effort was conducted by one of the study team-mates as an independent research project as part of a Masters of Systems Engineering research project, the effort associated with applying MBSE to the HLS study was performed in parallel with the HLS study outside of work, at no cost to NASA. Appendix F documents the results of using this methodology on the HLS study, and identifies the pros and cons of employing MBSE vs. a more traditional systems engineering approach.

Background

Currently systems engineering in most aerospace companies is performed using antiquated processes (e.g.; desktop spreadsheets, databases, analyses, and documents). System tasks are parsed out to the engineering team who do their best to communicate and coordinate the development of a complex system. However, as complexity of the system and the number of individuals involved increases, so does the likelihood of errors, miss-communication, and wasted effort. This inherent challenge increases dramically for large programs like the HLS, where many contractors are employed at multiple geographically distributed sites around the country. As a result, systems engineering oversight has to be increased, adding cost and schedule to the program.

In contrast, a promising technology called Model Based Systems Engineering (MBSE) utilizes an integrated desktop based tool to perform and manage the entire system engineering life-cycle process via an integrated model-based environment. What was really attractive about using this technology on the HLS study was that Orbital already has an MBSE tool but was not using it on any of their programs. This provided for a research opportunity to explore how well MBSE could be utilized to execute a short duration program.

The project sought to explore whether an opportunity existed to advance the state-of-the-art of Orbital's systems engineering processes to one that was more model-centric by exploiting an existing Orbital capability that is currently not being utilized. The goal was to employ MBSE as a research project, evaluate the costs and benefits, and determine whether a process improvement would benefit Orbital.

MBSE vs. Traditional Systems Engineering

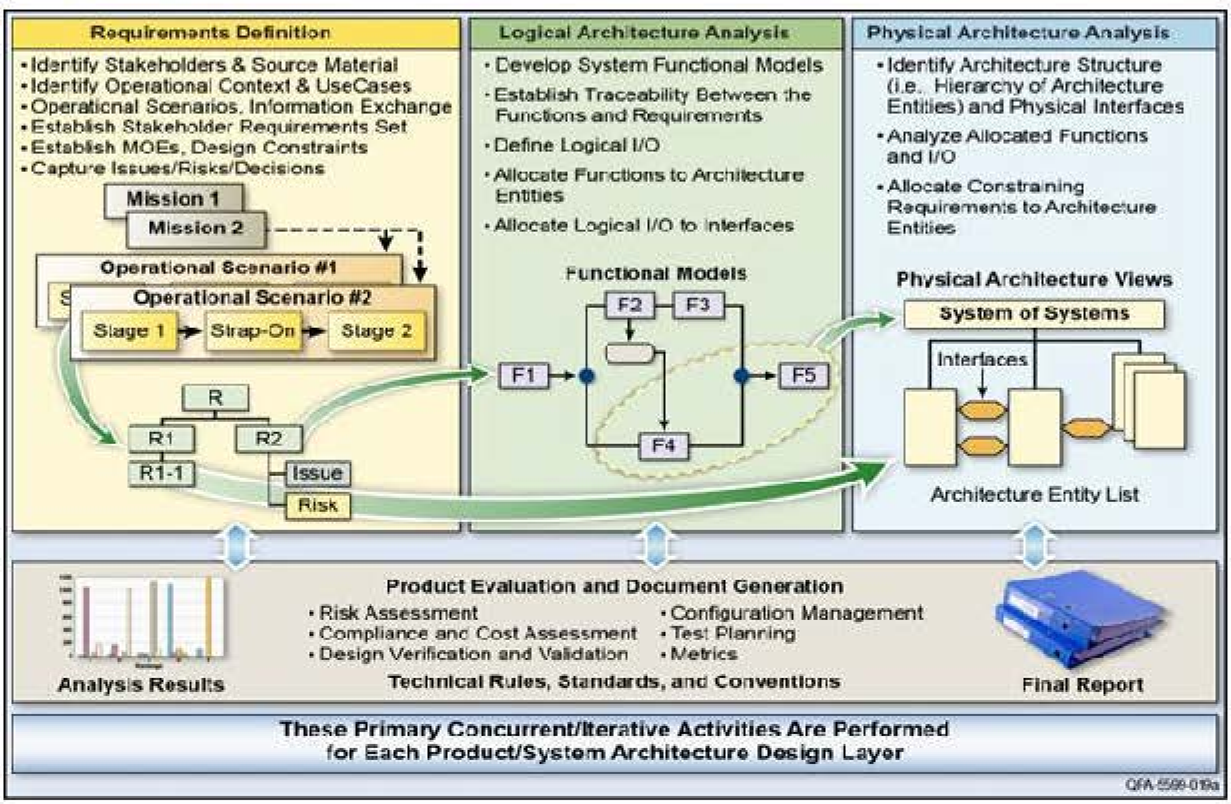
Typically in aerospace companies, systems are developed independent of MBSE techniques or based on "quasi" MBSE approaches that do not use the industry established guidelines for MBSE (e.g.; INCOSE's "4-Pillars of MBSE"). When actual MBSE tools (e.g.; System Modeling Language (SysML)) and techniques are used, it is usually as something of an "afterthought" and either poorly integrated into the program, or the tools and techniques are miss-applied due to a lack of knowledge and/or experience with MBSE.

For example, the traditional systems engineering approach would be to document system requirements in a specification (e.g.; Word® document), then once the specification was approved the requirements document is either uploaded into a database, or the requirements are entered into a database by hand. The system architecture is developed independently of the requirements development process – often in parallel – causing additional re-work due to the iterative nature of system development. Once the high level requirements and system architecture have been developed, the process is repeated for the major segments, elements, sub-systems, and components. If representative SysML-like models are produced they are generated in a viewgraph program (e.g.; PowerPoint®) primarily for communication purposes. Typically, no attempt is made to electronically link requirements to the stagnant model, nor are any of the more useful SysML features enjoyed (e.g.; integrating model elements, behavior, and requirements together).

Results

In contrast, the approach used during this project was to develop the system architecture directly in the SysML tool, thereby maximizing the potential advantages of employing MBSE early in the system engineering process. The generic systems modeling approach shown in the figure below illustrates how requirements definition leads to the logical architecture analysis, which in turn leads to the physical architecture analysis. Requirements were entered directly into the SysML tool, and functional flow block diagram models were generated in the tool and linked to the requirements. Finally representative architecture diagrams and system models were developed, and also linked back to the functional blocks which automatically linked the associated architecture elements to the appropriate requirement(s), thereby integrating the requirements with the end-to-end system design.

As the architectures and system requirements matured, the SysML models evolved, eventually producing all views (i.e.; requirements, behavioral, configuration, and physical) necessary to fully



Generic Systems Modeling Approach

describe the resultant system architecture. By employing this model-centric approach from the beginning, the research project succeeded in avoiding many of the drawbacks associated with a typical document-centric systems engineering approach, and realized some of the efficiencies and benefits offered by MBSE on a real-world program that was of a short enough duration to complete the project within the confines of a few months.

The MBSE techniques that were utilized facilitated development of a system-of-systems architecture in support of the NASA HLS launch vehicle architecture study. The project explored the effectiveness of employing MBSE to more efficiently define requirements, develop architectures, and document systems.

Examples of the system models as developed in the tool are provided, including physical architecture diagrams, use cases, and functional flow block diagrams are provided in this appendix.

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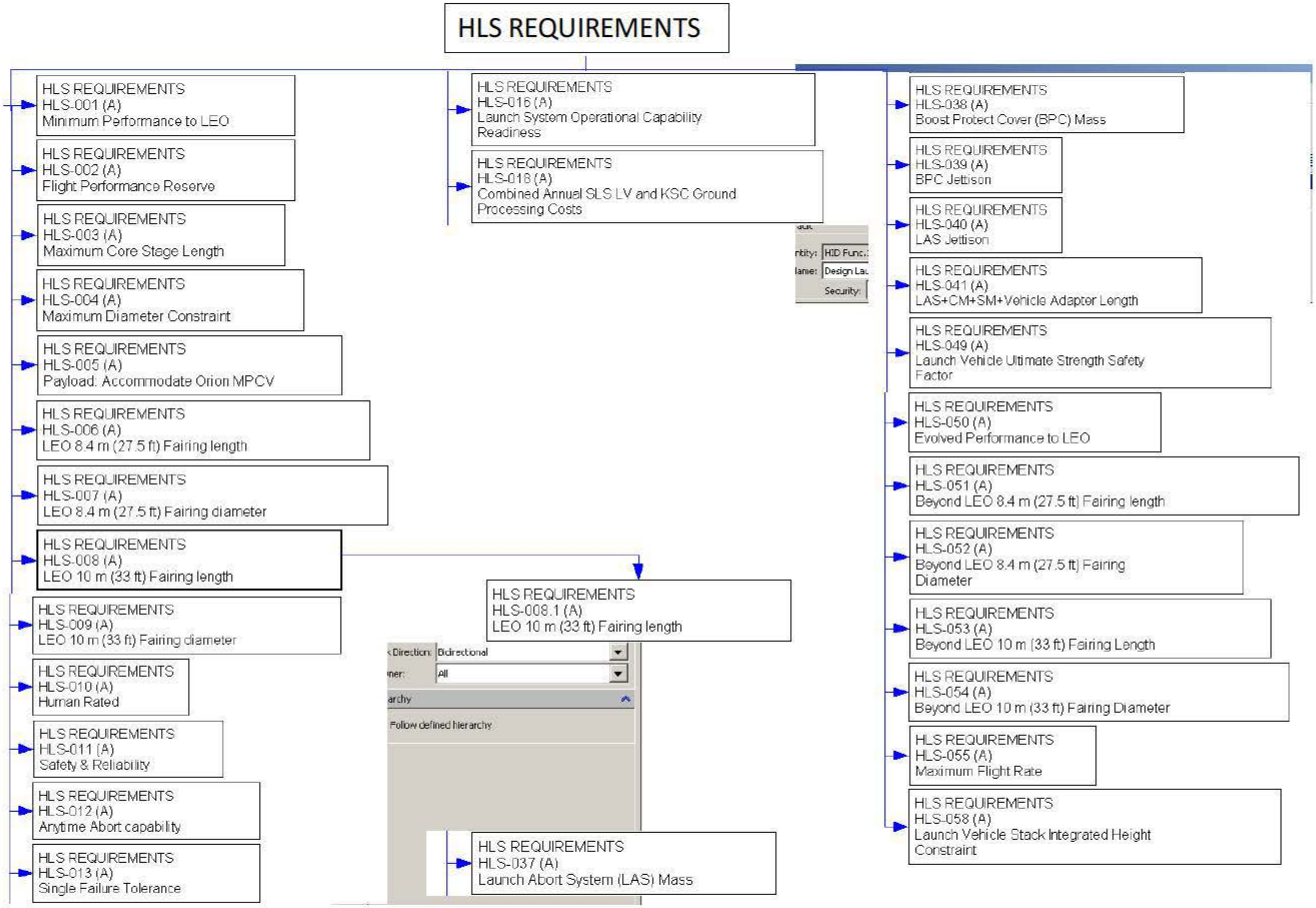
HLS REQUIREMENTS
HLS REQUIREMENTS HLS-001 (A) Minimum Performance to LEO
HLS REQUIREMENTS HLS-002 (A) Flight Performance Reserve
HLS REQUIREMENTS HLS-003 (A) Maximum Core Stage Length
HLS REQUIREMENTS HLS-004 (A) Maximum Diameter Constraint
HLS REQUIREMENTS HLS-005 (A) Payload: Accommodate Orion MPCV
HLS REQUIREMENTS HLS-006 (A) LEO 8.4 m (27.5 ft) Fairing length
HLS REQUIREMENTS HLS-007 (A) LEO 8.4 m (27.5 ft) Fairing diameter
HLS REQUIREMENTS HLS-008 (A) LEO 10 m (33 ft) Fairing length
HLS REQUIREMENTS HLS-008.1 (A) LEO 10 m (33 ft) Fairing length
HLS REQUIREMENTS HLS-009 (A) LEO 10 m (33 ft) Fairing diameter
HLS REQUIREMENTS HLS-010 (A) Human Rated
HLS REQUIREMENTS HLS-011 (A) Safety & Reliability
HLS REQUIREMENTS HLS-012 (A) Anytime Abort capability
HLS REQUIREMENTS HLS-013 (A) Single Failure Tolerance
HLS REQUIREMENTS HLS-016 (A) Launch System Operational Capability Readiness
HLS REQUIREMENTS HLS-018 (A) Combined Annual SLS LV and KSC Ground Processing Costs
HLS REQUIREMENTS HLS-037 (A) Launch Abort System (LAS) Mass
HLS REQUIREMENTS HLS-038 (A) Boost Protect Cover (BPC) Mass
HLS REQUIREMENTS HLS-039 (A) BPC Jettison
HLS REQUIREMENTS HLS-040 (A) LAS Jettison
HLS REQUIREMENTS HLS-041 (A) LAS+CM+SM+Vehicle Adapter Length
HLS REQUIREMENTS HLS-049 (A) Launch Vehicle Ultimate Strength Safety Factor
HLS REQUIREMENTS HLS-050 (A) Evolved Performance to LEO
HLS REQUIREMENTS HLS-051 (A) Beyond LEO 8.4 m (27.5 ft) Fairing length
HLS REQUIREMENTS HLS-052 (A) Beyond LEO 8.4 m (27.5 ft) Fairing Diameter
HLS REQUIREMENTS HLS-053 (A) Beyond LEO 10 m (33 ft) Fairing Length
HLS REQUIREMENTS HLS-054 (A) Beyond LEO 10 m (33 ft) Fairing Diameter
HLS REQUIREMENTS HLS-055 (A) Maximum Flight Rate
HLS REQUIREMENTS HLS-058 (A) Launch Vehicle Stack Integrated Height Constraint
Bidirectional
All
Follow defined hierarchy
Security:

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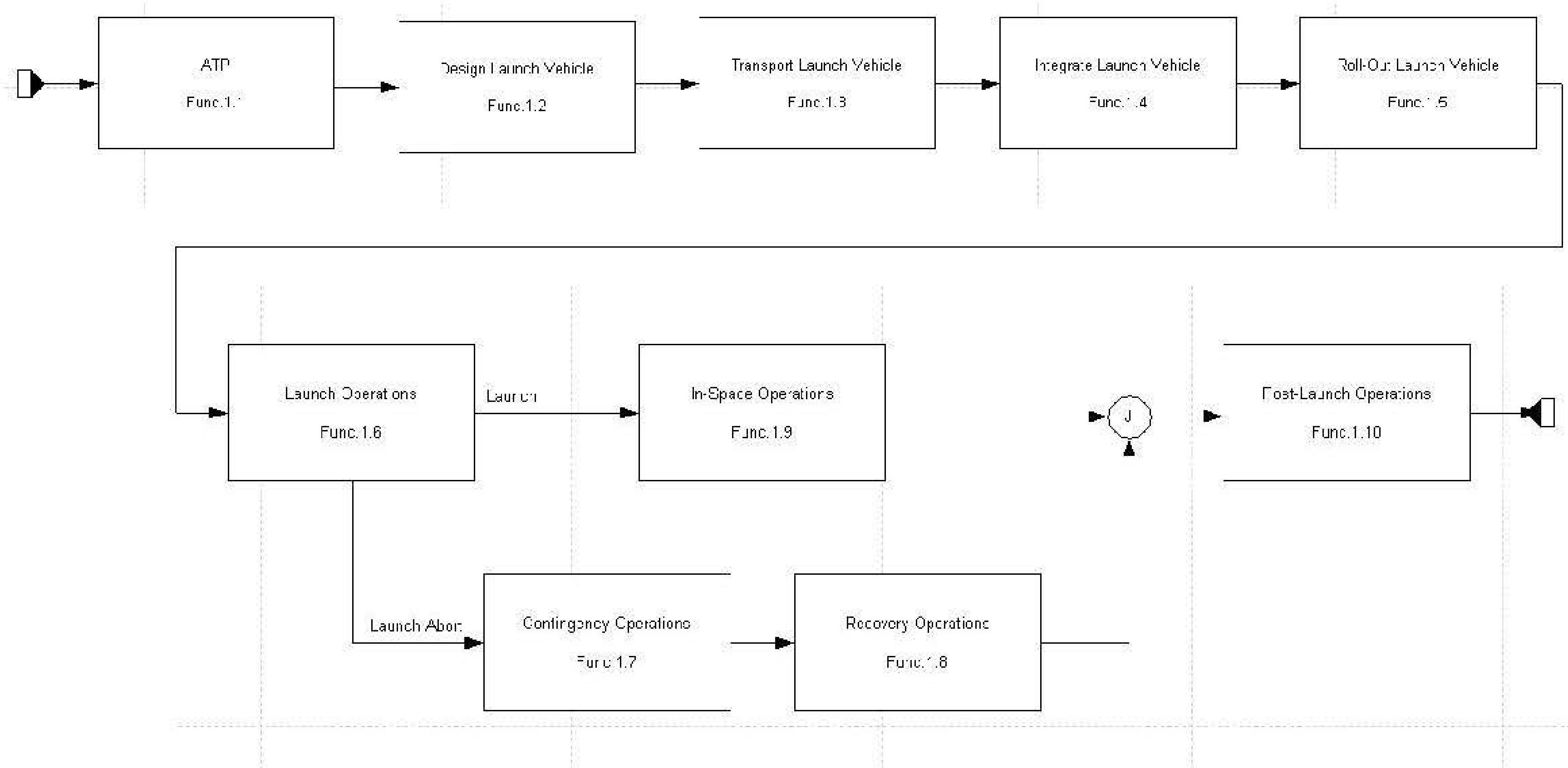

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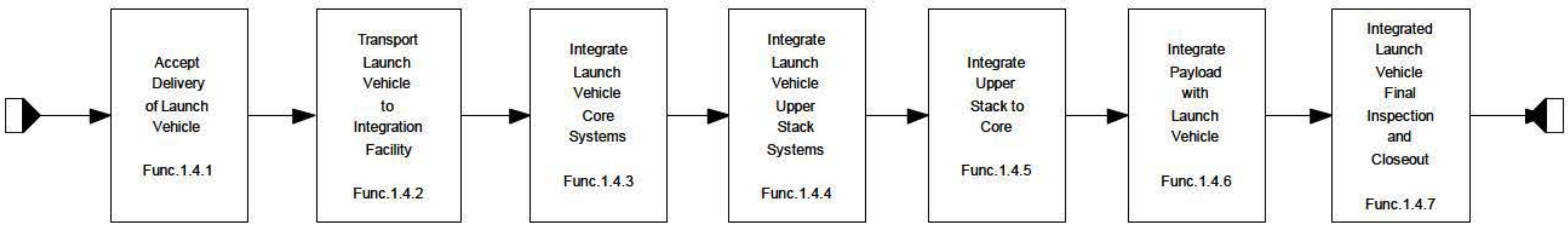

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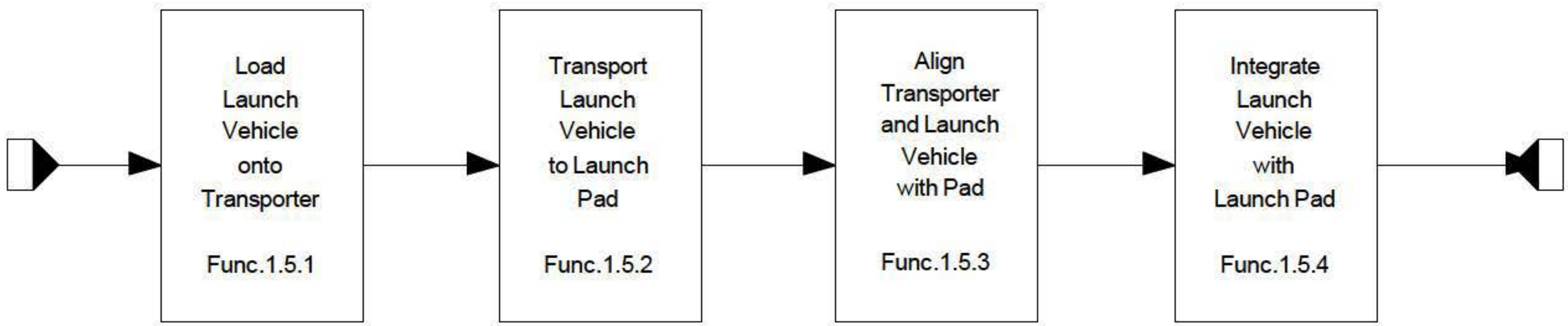

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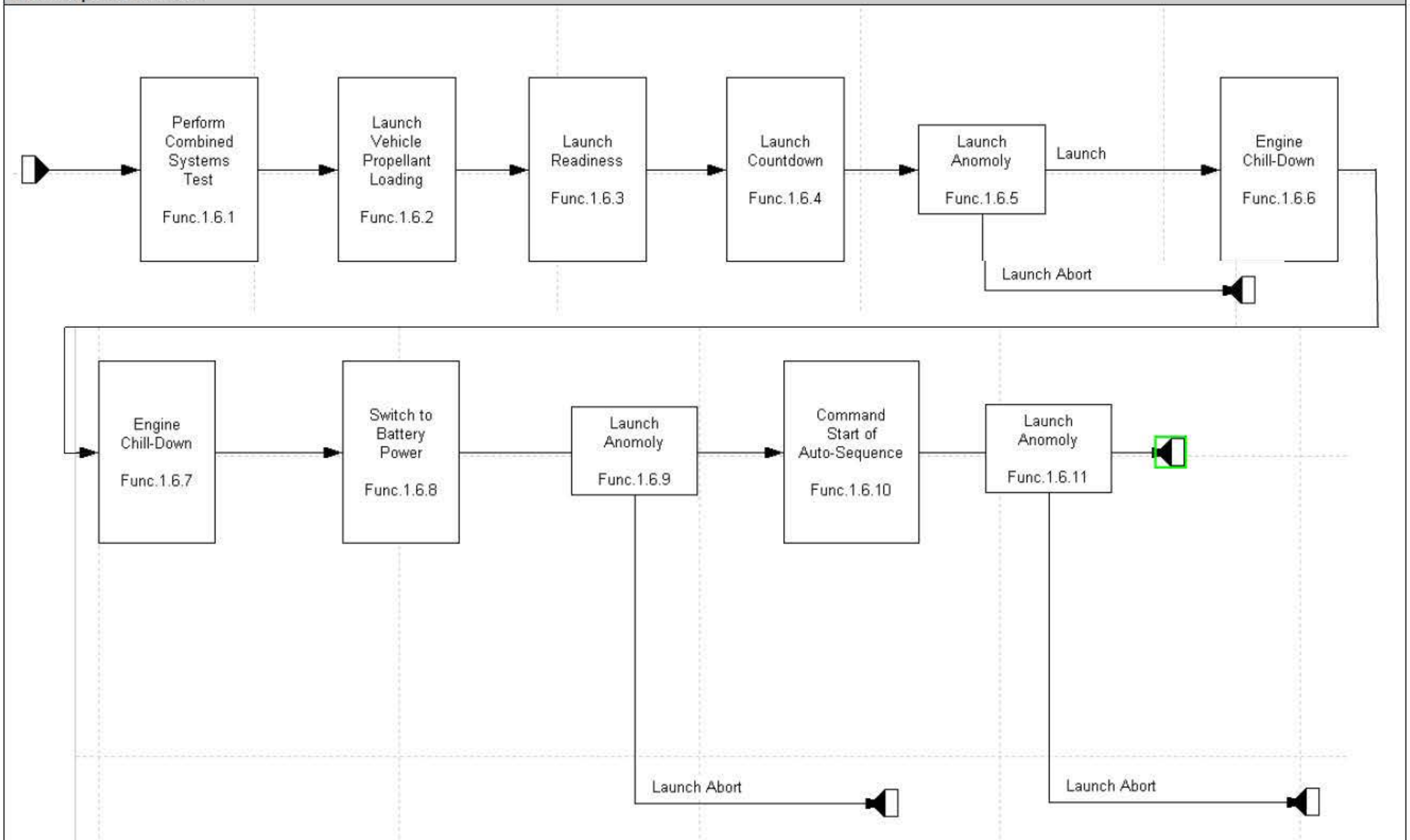

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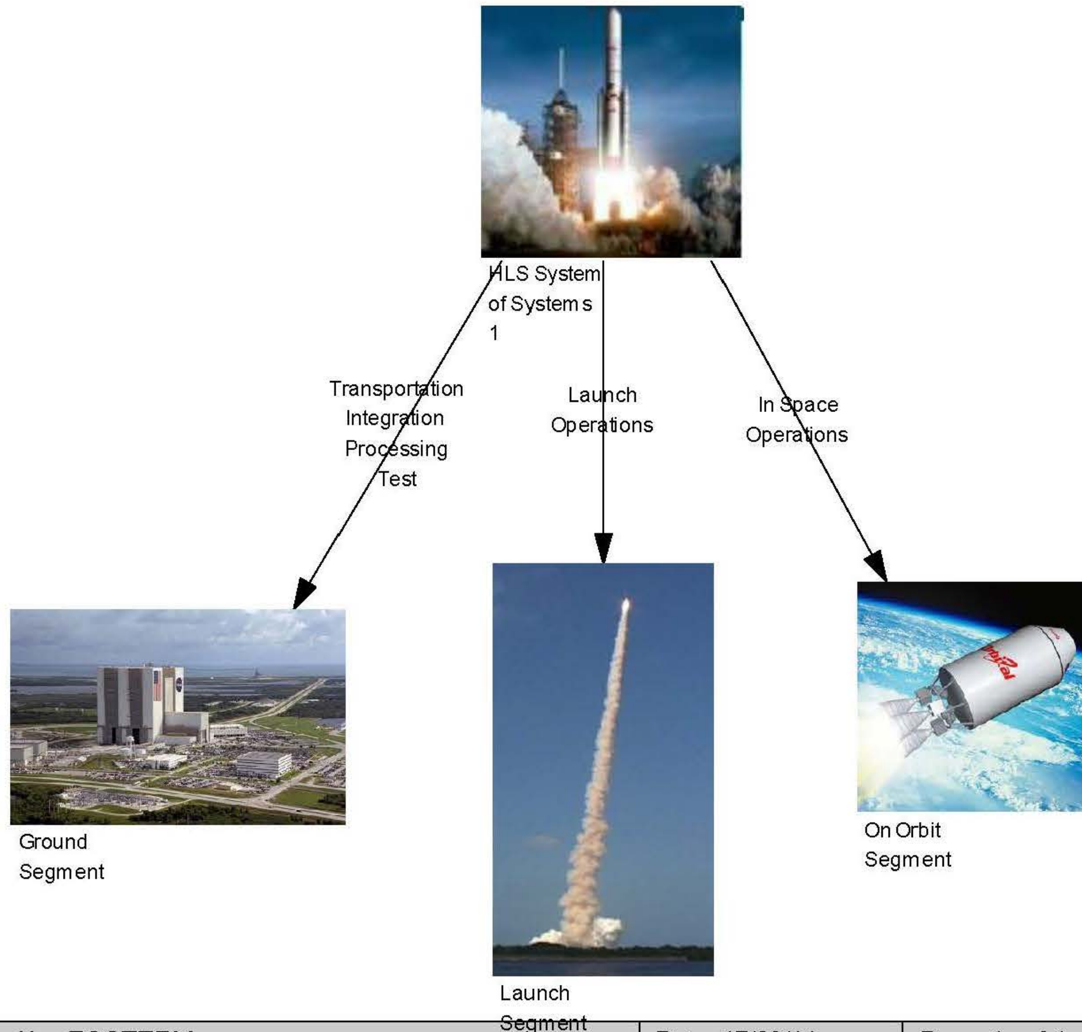

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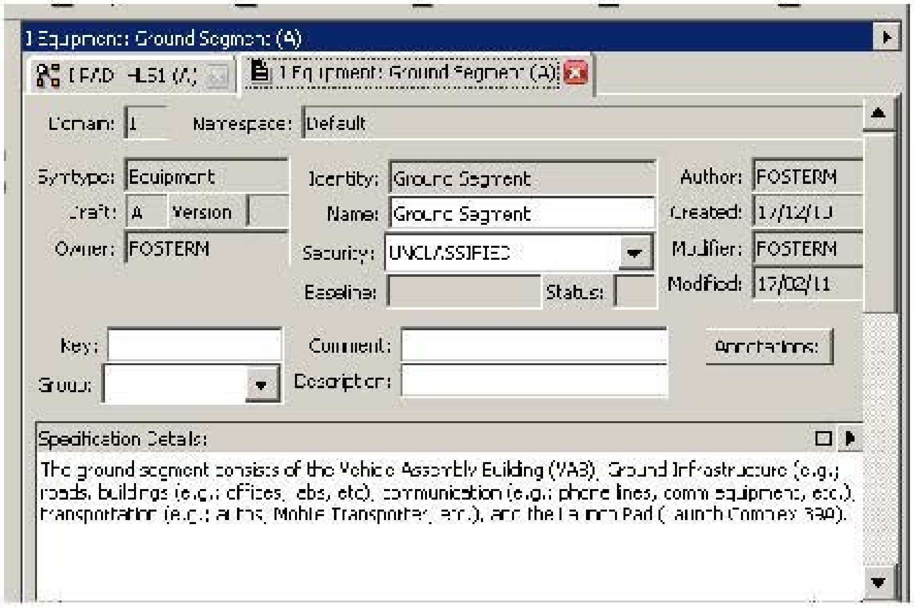
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Key:
Comment:
Annotations:
Description:
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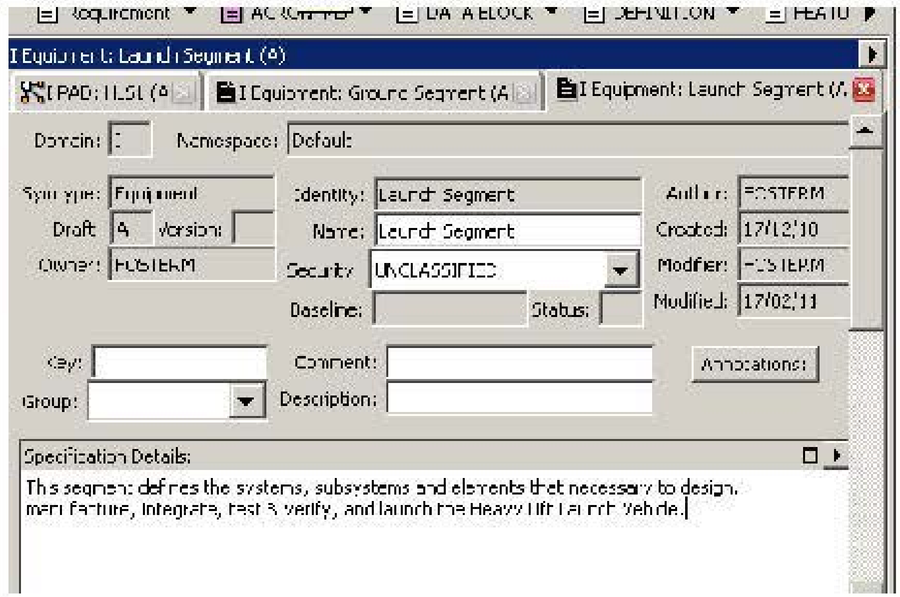
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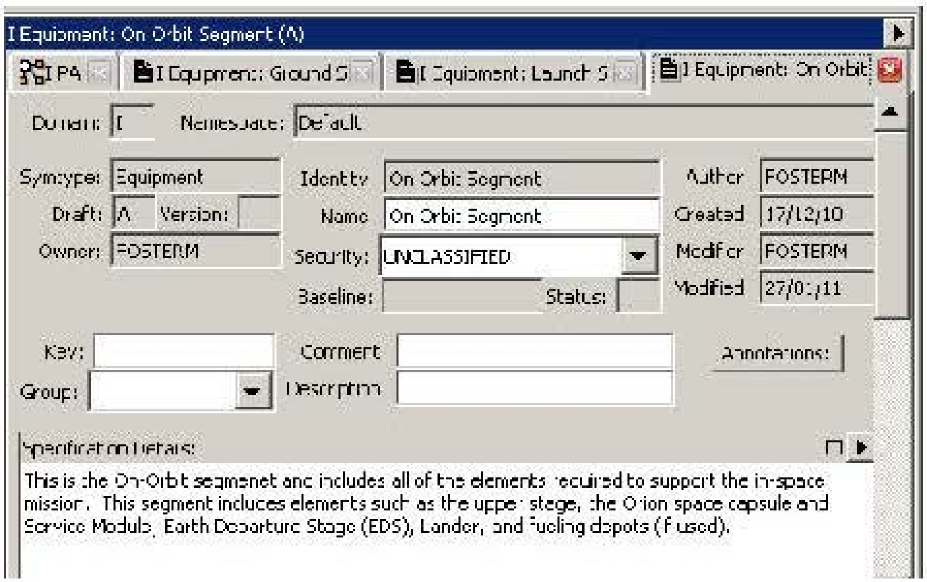
Modified: 17/02/11
Comment:
Annotations:
Group:
Description:
Specification Details:
This segment defines the systems, subsystems and elements that necessary to design, manufacture, integrate, test & verify, and launch the Heavy Lift Launch Vehicle.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Key:
Comment:
Annotations:
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This is the On-Orbit segmenet and includes all of the elements required to support the in-space mission. This segment includes elements such as the upper stage, the Orion space capsule and Service Module, Earth Departure Stage (EDS), Lander, and Fueling depots (if used).

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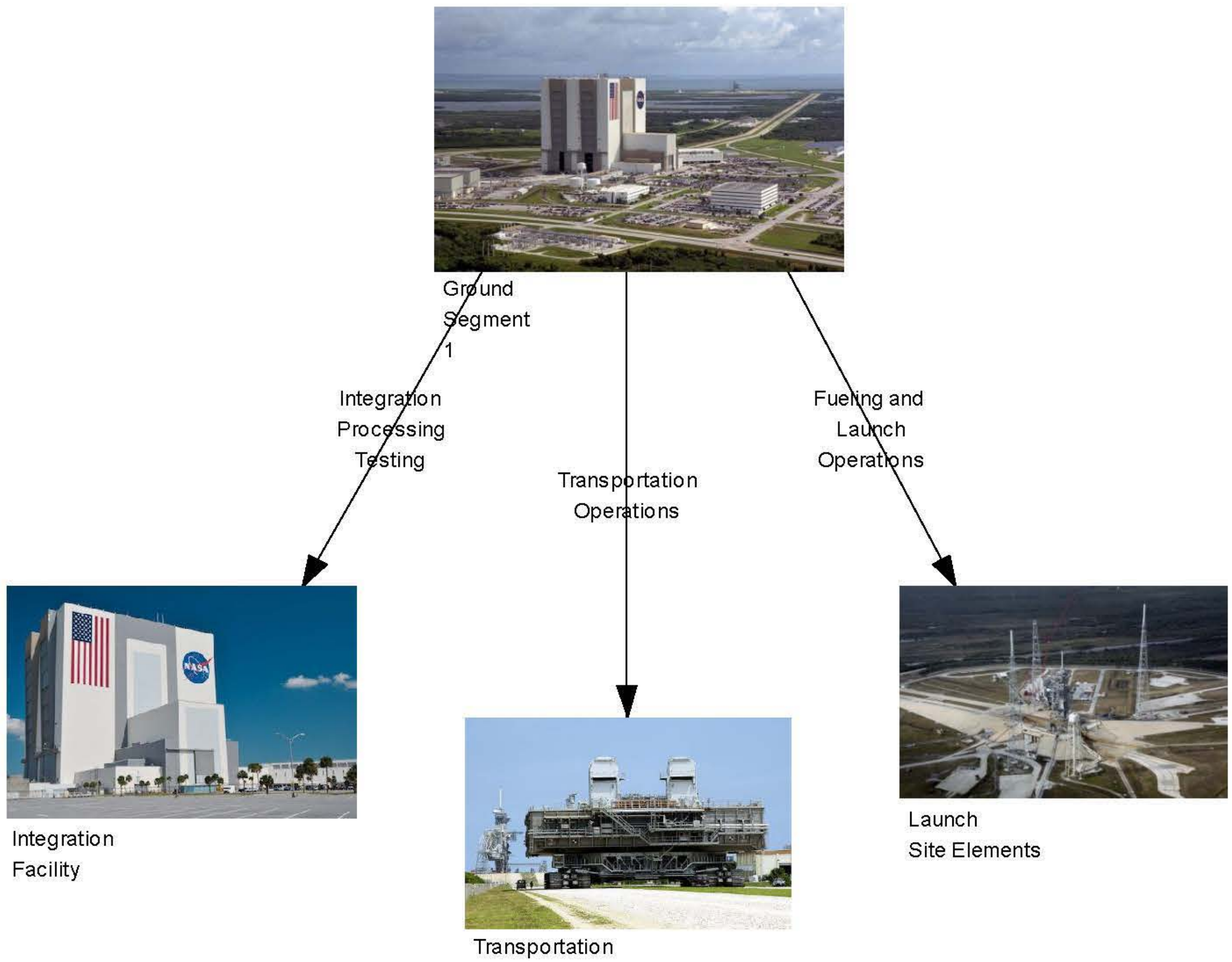
Namespace: Default					



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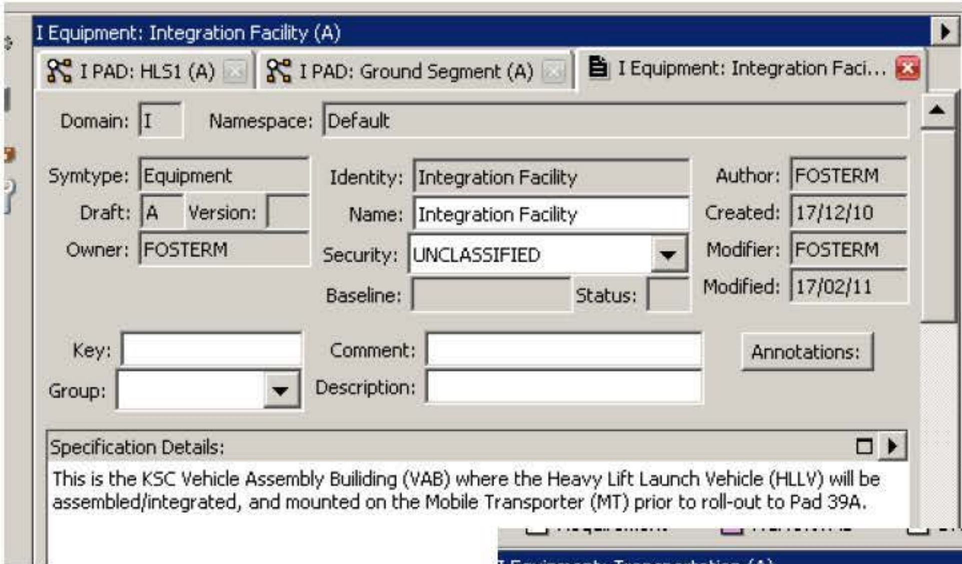
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Annotations:
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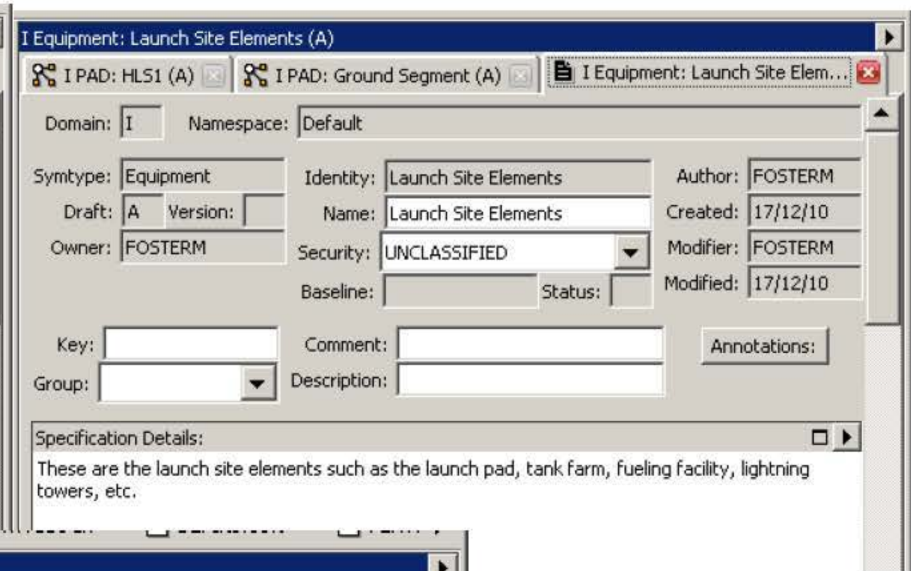
This is the KSC Vehicle Assembly Builiding (VAB) where the Heavy Lift Launch Vehicle (HLLV) will be assembled/integrated, and mounted on the Mobile Transporter (MT) prior to roll-out to Pad 39A.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Modifier: FOSTERM
Modified: 17/12/10
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Group:
Comment:
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Annotations:
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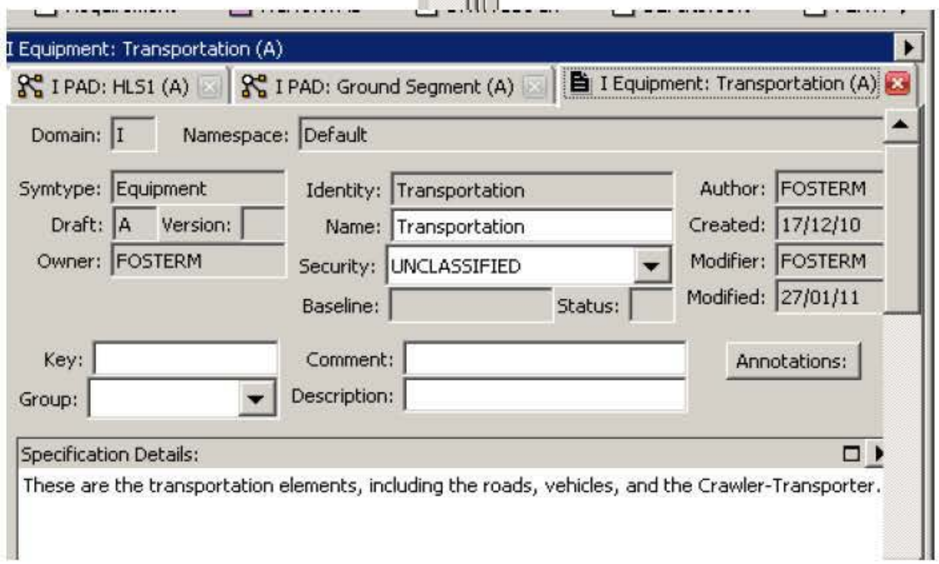
These are the launch site elements such as the launch pad, tank farm, fueling facility, lightning towers, etc.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Modifier: FOSTERM
Modified: 27/01/11
Key:
Group:
Comment:
Description:
Annotations:
Specification Details:
These are the transportation elements, including the roads, vehicles, and the Crawler-Transporter.

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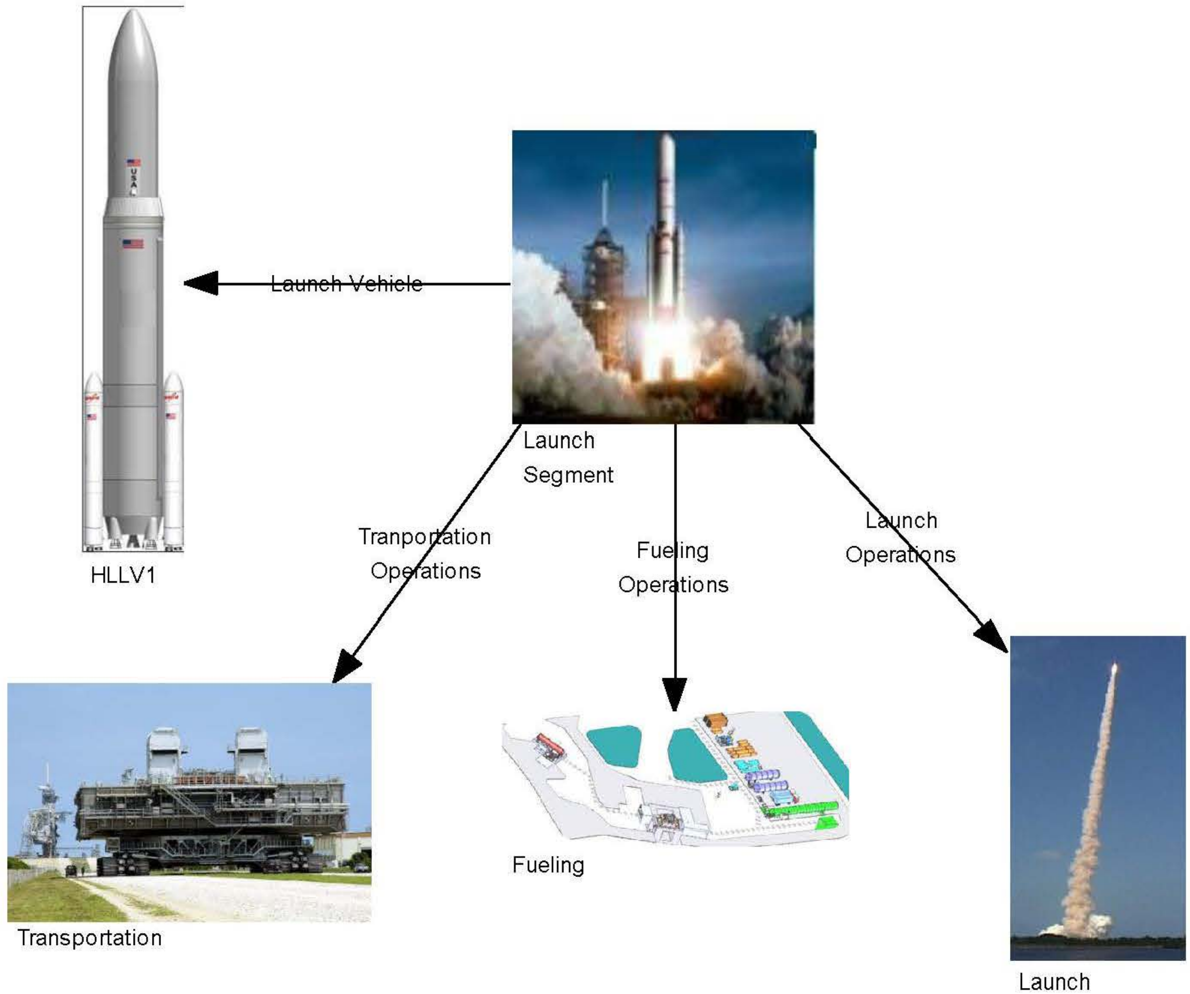

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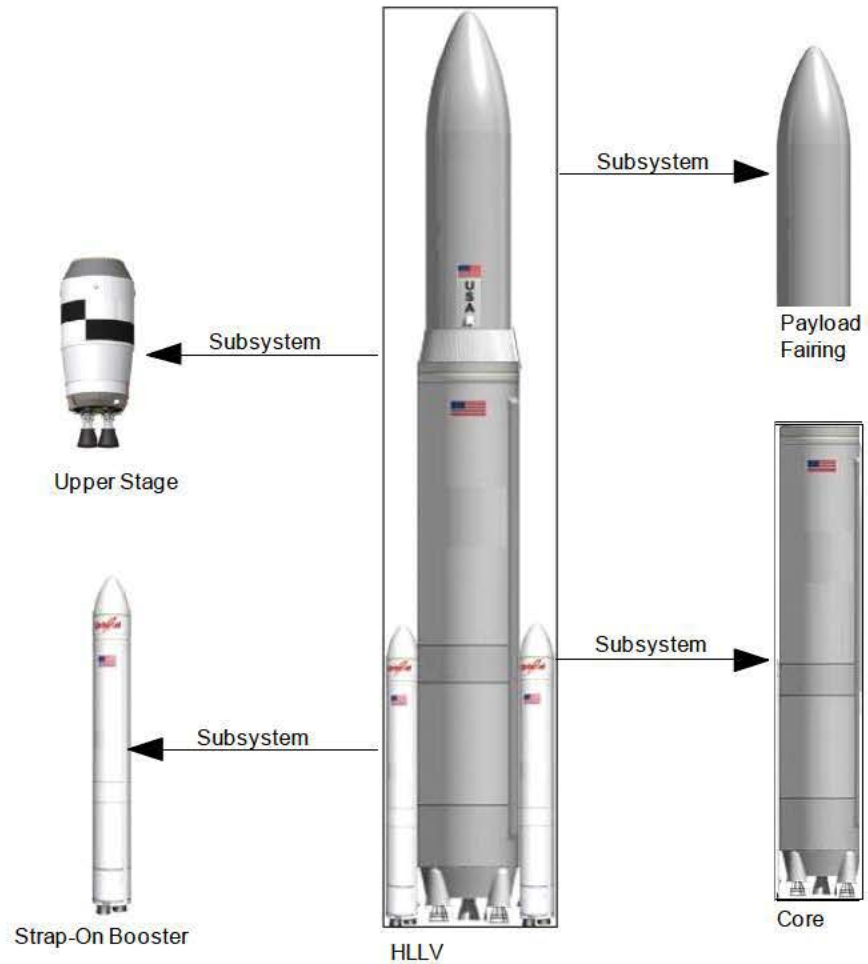

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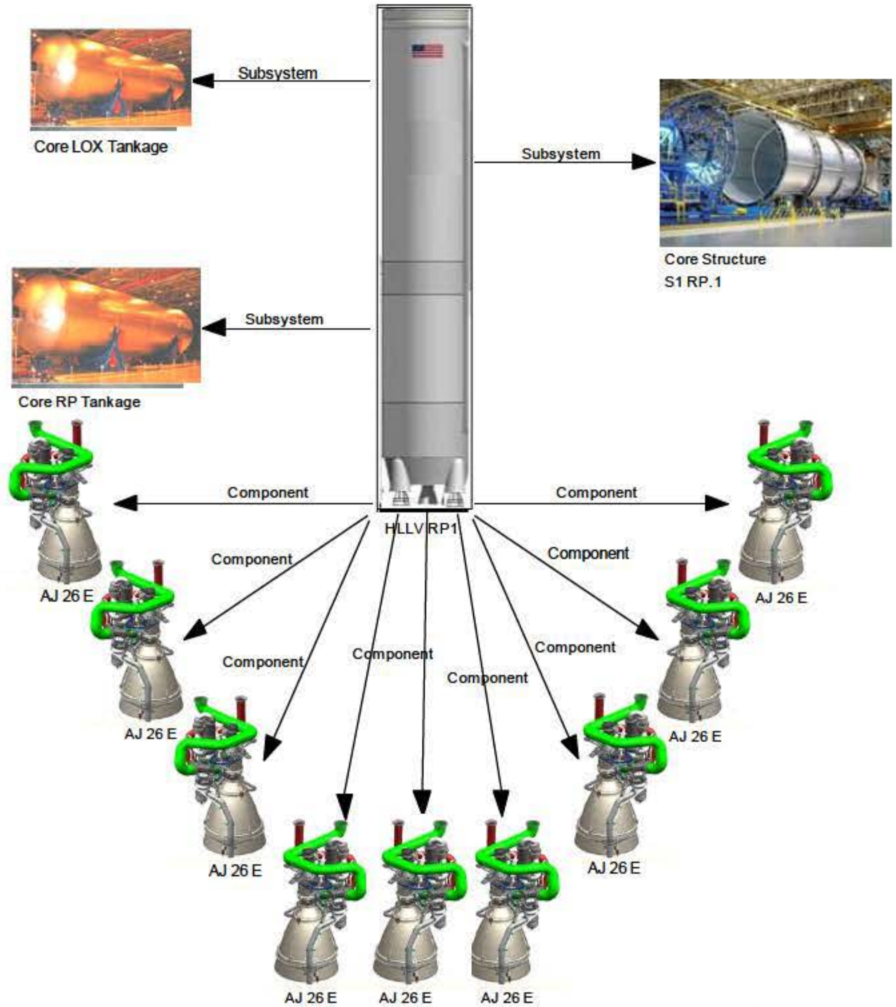

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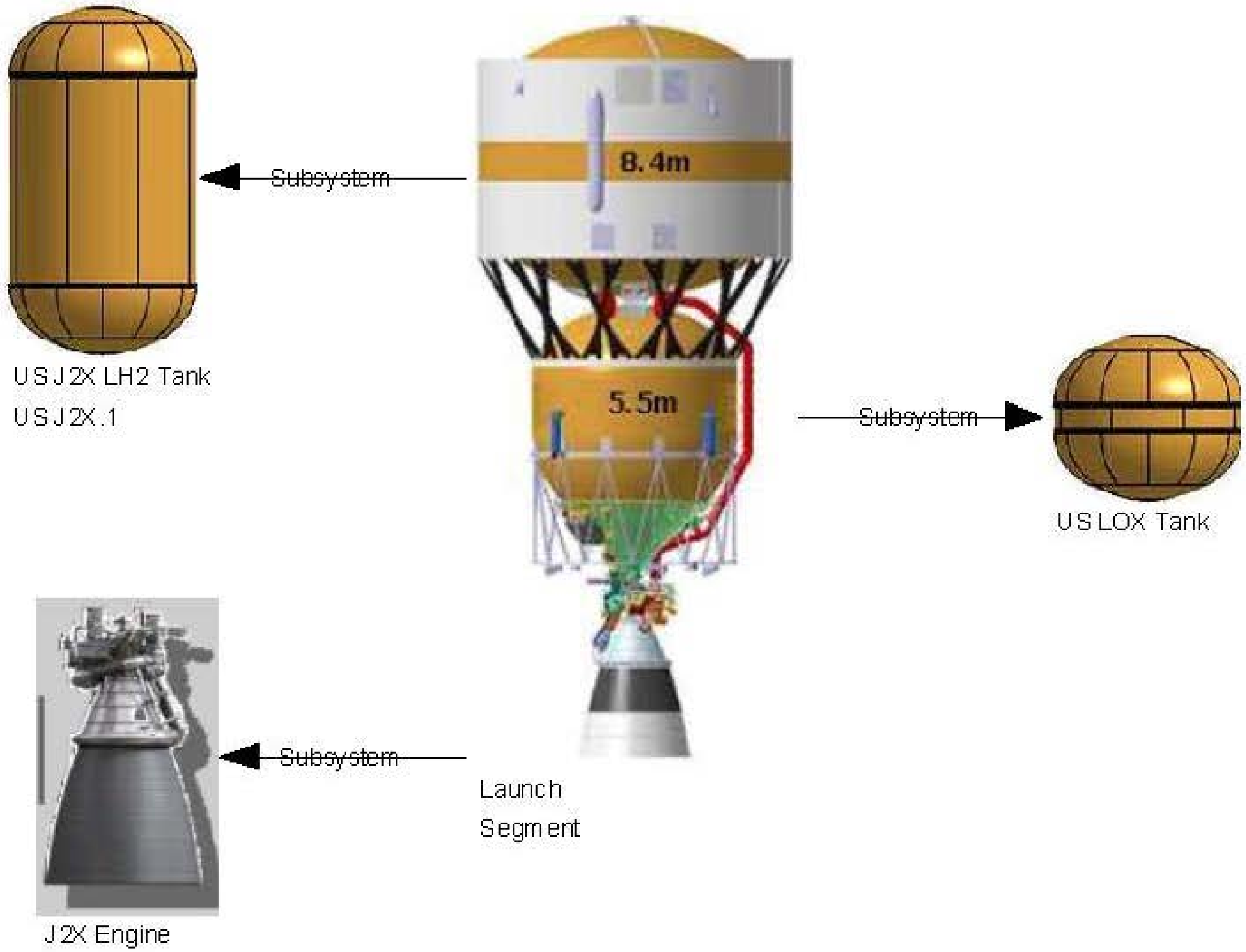

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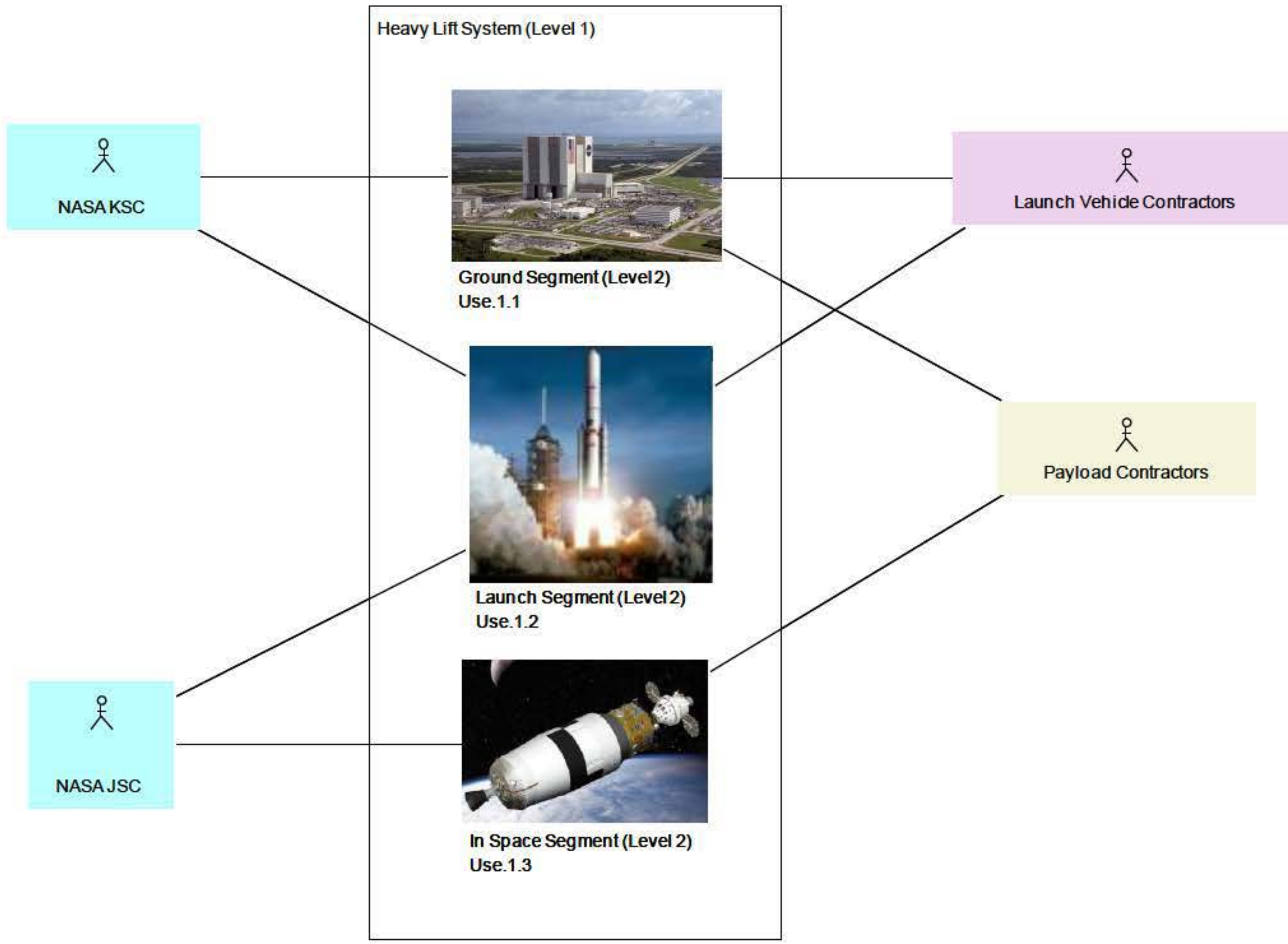

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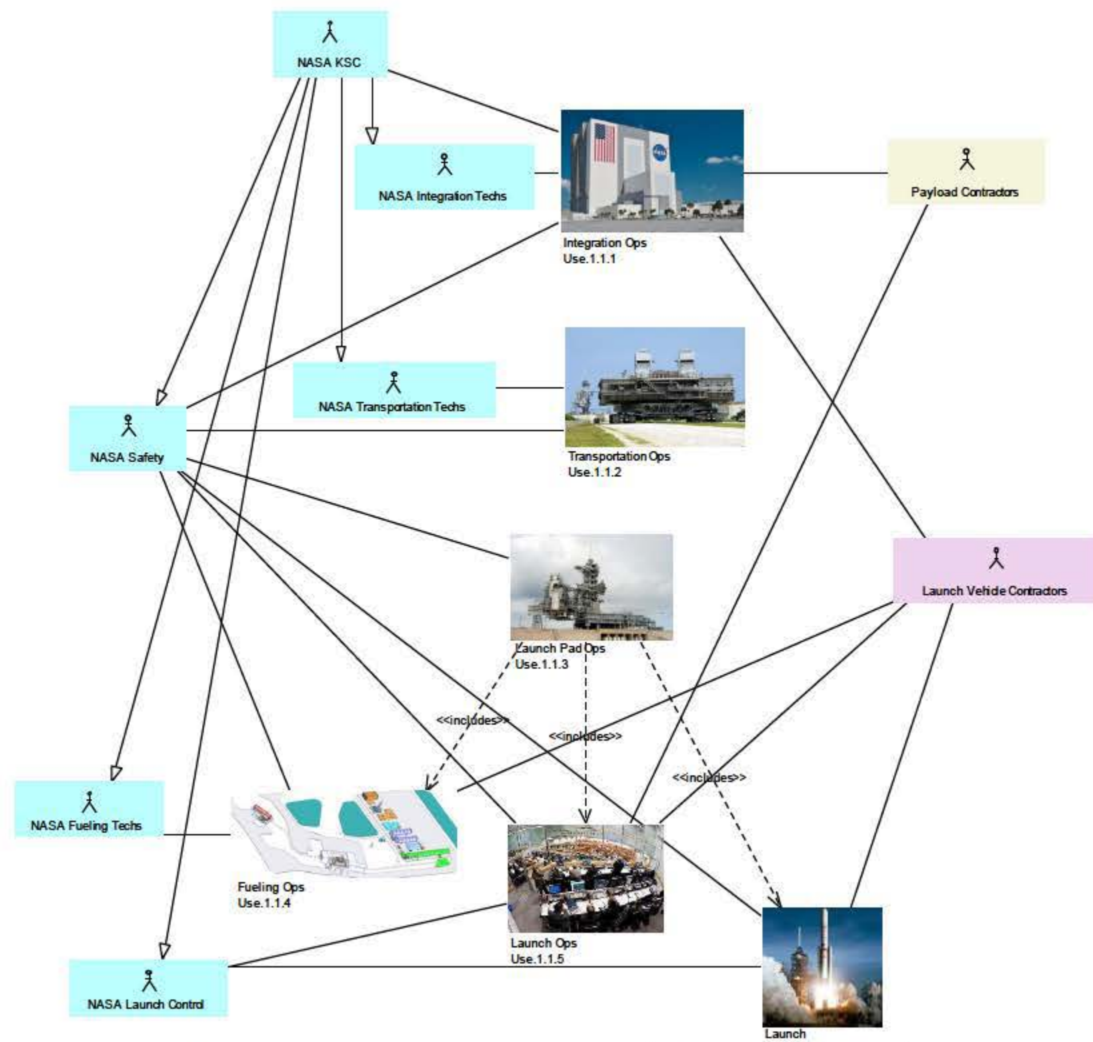

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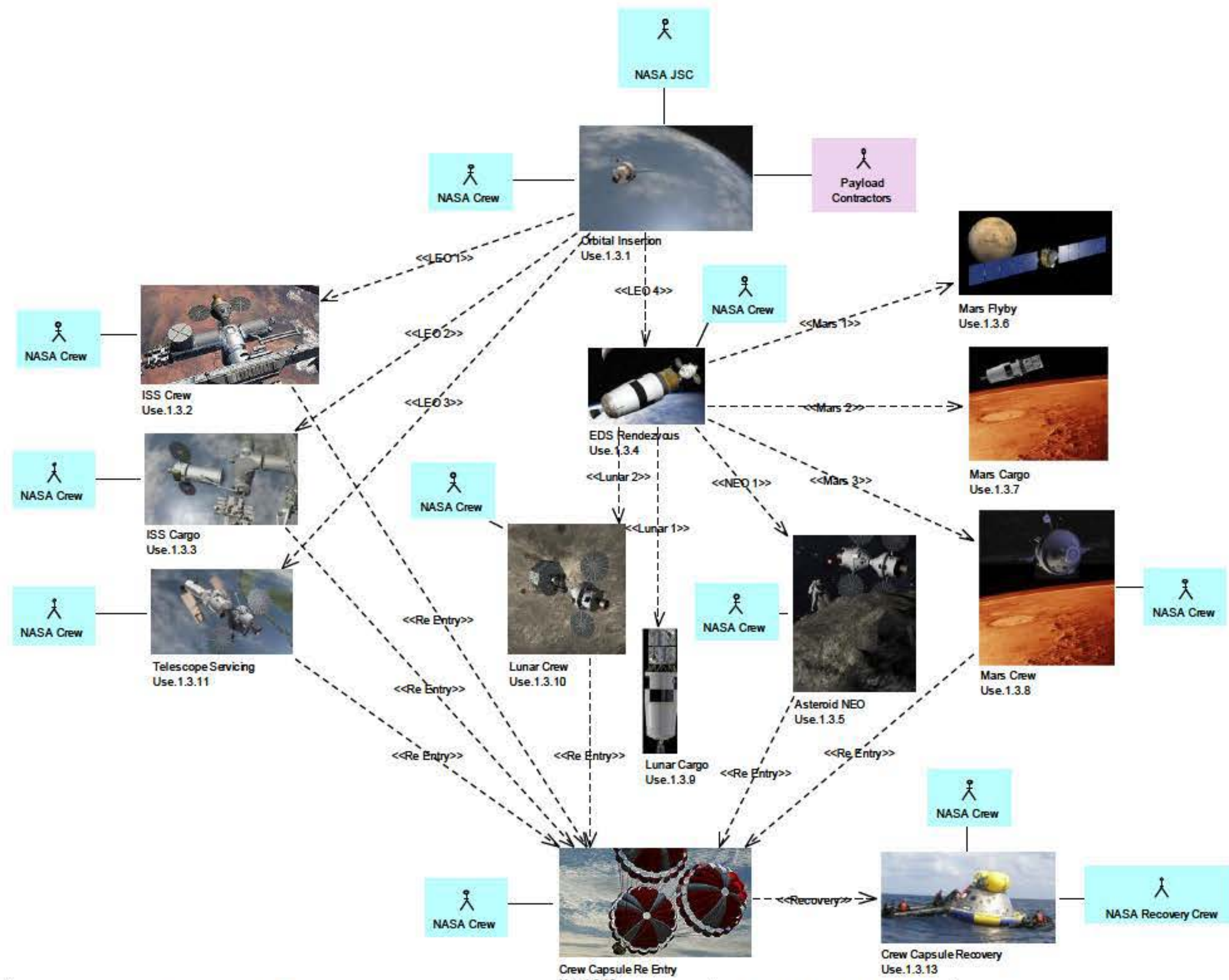

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Re-Usable Component Library

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(b) (4)

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Pratt & Whitney Rocketdyne, Inc.
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P.O. Box 7922
Canoga Park, California 91309-7922

Heavy Lift & Propulsion Technology (HLPT) Systems Analysis and Trade Study

NASA Contract: NNM11AA14C

Final Study Report

June 3, 2011

Prepared for:
NASA Marshall Space Flight Center

Acknowledgments

The work presented herein was conducted from November 22, 2010 to June 3, 2011 by personnel from the Engineering and Future Programs organizations of Pratt & Whitney Rocketdyne. The effort was completed under the NASA Heavy Lift & Propulsion Technology (HLPT) Systems Analysis and Trade Study contract (NNM11AA14C). Overall leadership for this HLPT study was provided by the program management team of Mr. Timothy Flores and Ms. Karen Gelmis of the NASA Marshall Space Flight Center. Programmatic and technical leadership was provided by Mr. Frederick Widman and Dr. Timothy Kokan of Pratt & Whitney Rocketdyne. Special thanks are extended to Mr. Rick Bachtel, Dr. Matthew Long, Mr. Peter White, Dr. Christopher Garrett, Mr. Jonathan Laurie, Mr. Russ Joyner, Mr. Daniel Levack, and Mr. Gary Gilmartin of Pratt & Whitney Rocketdyne for their support on this contract. Special thanks are also extended to Ms. Elizabeth Buchen, Mr. Andrew Willis, and Dr. John Bradford of SpaceWorks Engineering, Inc. for their support on this contract.

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Acronym	Definition	Acronym	Definition
Al-Li	Aluminum-Lithium	LOC	Loss of Crew
APAS	Aerodynamic Preliminary Analysis System	LOX	Liquid Oxygen
ART	Advisory Review Team	LRB	Liquid Rocket Booster
ATP	Authority to Proceed	MCC	Main Combustion Chamber
BAA	Broad Agency Announcement	MER	Mass Estimating Relationship
CBC	Common Booster Core	MMBF	Mean Missions Between Failure
CCAFS	Cape Canaveral Air Force Station	MPCV	Multi-Purpose Crew Vehicle
CECE	Common Extensible Cryogenic Engine	MPS	Main Propulsion System
CH_4	Methane	MSFC	Marshall Space Flight Center
COMET	Conceptual Operations Manpower Estimating Tool	NAFCOM	NASA / Air Force Cost Model
DATCOM	Missile Data Compendium	OCM	Operations Cost Model
DDT&E	Design, Development, Test, and Evaluation	OCR	Operational Capability Readiness
DoD	Department of Defense	ORSC	Oxygen-Rich Staged-Combustion
ΔV	Delta Velocity	Pc	Chamber Pressure
EELV	Evolved Expendable Launch Vehicle	POST	Program to Optimize Simulated Trajectories
ESMD	Exploration Systems Mission Directorate	PWR	Pratt & Whitney Rocketdyne
Ex-HEX	Expander-Heat Exchanger	RECM	Rocket Engine Cost Model
FMM	Flexible Mixed Model	ROM	Rough Order of Magnitude
FPR	Fuel Propellant Reserve	RP	Rocket Propellant-1
GG	Gas-Generator	RSE	Response Surface Equation
HIP	Hot-Isostatic Pressure	SE&I	Systems Engineering and Integration
HLLV	Heavy Lift Launch Vehicle	SEI	SpaceWorks Engineering, Inc.
HLPT	Heavy Lift and Propulsion Technology	SLS	Space Launch System
IMLEO	Initial Mass in Low Earth Orbit	SOW	Statement of Work
IMS	Integrated Master Schedule	SRB	Solid Rocket Booster
INTROS	INTegrated Rocket Sizing Program	SSC	Stennis Space Center
IOC	Initial Operating Capability	TRL	Technology Readiness Level
KDA	Key Decision Attribute	ULA	United Launch Alliance
KSC	Kennedy Space Center	USAF	United States Air Force
LAS	Launch Abort System	VAFB	Vandenberg Air Force Base
LCC	Life Cycle Cost	WBS	Work Breakdown Structure
LEO	Low Earth Orbit	WPB	West Palm Beach
LH_2	Liquid Hydrogen		

1 Summary

Pratt & Whitney Rocketdyne (PWR) conducted a Heavy Lift & Propulsion Technology (HLPT) Systems Analysis & Trade Study under contract NNM11AA14C for NASA Marshall Space Flight Center (MSFC). The firm fixed price study contract was awarded under Broad Agency Announcement (BAA) NNM10ZDA001K[1] in support of NASA's Space Launch System (SLS) planning activities. The systems analysis and trade study approach involved (1) assessment of SLS requirements and key decision attributes; (2) analyses of SLS vehicle concepts; and (3) assessment of capability gaps and technology, affordability, and innovative ideas with a focus on SLS affordability. The study included systems engineering utility analysis to capture the sensitivity of customer expectations as Key Decision Attributes (KDAs) and systematically balanced all objectives in proportion to their assessed importance. The KDAs and utility analysis were used to assess SLS vehicle configurations and provided a methodology to downselect a set of eight robust vehicle configurations covering a wide range of options for more detailed assessment. The study identified and assessed the technology, affordability, and innovative ideas for each of the downselected robust vehicle configurations and determined the improvements in development and recurring cost at the SLS vehicle system level.

The following key conclusions and observations were identified in the study:

1. Viable Shuttle-Derived vehicle configurations in the 130 mT class using a LOX/LH_2 first stage, a LOX/LH_2 second stage, and either solid or liquid strap-on boosters can meet the NASA budget profile and provide a 130 mT full-up payload capability at first operational flight.

2. Utilization of existing and derivative engines provides a benefit of low Design, Development, Test and Evaluation (DDT&E) costs, less risk, and high demonstrated reliability.
   - Effective use of existing RS-25D assets during development (use on 2 test flights) with early start and concurrent development of the improved-affordability RS-25E provides significant DDT&E and recurring cost savings.
   - The F-1A gas-generator cycle engine provides the best option for a LOX/RP large booster propulsion application by leveraging PWR/NASA program heritage.
   - The J-2X engine provides flexibility and extensibility as propulsion for an upper stage and in-space/Beyond-Earth-Orbit system.

3. Several in-space propulsion system technology development opportunities were identified and preliminary technology development roadmaps were defined for Nuclear Thermal Propulsion and Power, Nuclear/Solar Electric Propulsion and Power, and LOX/Methane Chemical Propulsion.

4. Significant PWR propulsion system cost reductions are enabled with incorporation of programmatic, technology, and engine-specific affordability innovations.
   - PWR Consolidation Initiatives will reduce infrastructure cost by more than 40% and enable better fixed costs sharing through Flexible Mixed Model (FMM) operations.
   - PWR has adequate production capacity for foreseeable NASA and Department of Defense (DoD) needs with some nominal capital required for higher rates.
   - PWR's Flexible Mixed Model brings the benefit of PWR overall production rate across all customers to each program with up to 27% engine cost reduction.
   - PWR is actively working specific affordability improvements for the RS-25E, J-2X, RS-68B, and F-1A engines that are applicable to SLS.
   - Opportunity exists to leverage PWR-NASA experience to reduce traditional NASA oversight costs.

- PWR advanced materials and processes are showing strong near-term potential for reducing component-specific production costs.

2 Introduction

PWR conducted a Heavy Lift & Propulsion Technology (HLPT) Systems Analysis & Trade Study under contract NNM11AA14C for NASA Marshall Space Flight Center (MSFC) with a period of performance from November 22, 2010 through June 3, 2011. This report documents the study methodology, systems analysis results, downselected set of robust SLS vehicle configurations, and prioritized set of quantified technology, innovations, and affordability improvements.

2.1 Background

NASA is seeking an innovative path for human space exploration which strengthens the capability to extend human and robotic presence throughout the solar system. NASA is laying the ground work to enable humans to safely reach multiple potential destinations, including the Moon, asteroids, Lagrange points, and Mars and its environs. The Exploration Systems Mission Directorate (ESMD) is leading the Nation on a course of discovery and innovation that will provide the technologies, capabilities and infrastructure required for sustainable, affordable human presence in space.

NASA is examining the trade space of potential heavy lift launch and space transfer vehicle concepts under the Space Launch System (SLS) planning activities. The focus is on affordability, operability, reliability, and commonality with multiple end users (NASA, Department of Defense (DoD), commercial, international partners, etc.) at the system and subsystem levels. For the purposes of this BAA, affordability is defined as lifecycle cost which consists of DDT&E, production and operations (fixed and variable). A major thrust of this activity is space launch propulsion technologies that will enable a more robust exploration program, support commercial ventures, and related national security needs.

NASA BAA NNM10ZDA001K solicited proposals for Heavy Lift and Propulsion Technology Systems Analysis and Trade study to seek industry input on technical solutions in support of heavy lift system concepts studies. The objectives of the studies were to capture potential system architectures and identify propulsion technology gaps. The study efforts were to include architecture assessments of a variety of heavy lift launch vehicle and in-space vehicle architectures employing various propulsion combinations and how they can be employed to meet multiple mission objectives. The focus of the studies were to be on developing system concepts that can be used by multiple end users with a strong emphasis on affordability, based on the contractor's business assumption.

PWR developed a detailed study plan that addressed each of the Technical Objectives called out in the BAA. Figure 1 summarizes the BAA Technical Objectives and indicates where each of the objectives were addressed in the study plan and statement of work as well as where they are discussed in this Report.

2.2 Program Team

Performing the HLPT study required forming a team with expertise in a wide range of disciplines encompassing vehicle, mission, con-ops, cost, reliability, and utility analysis; main propulsion system and engine subject matter experts. An experienced and committed team was assembled to conduct the HLPT study. As detailed in the program methodology, the study logic was established at the inception of the contract so that the appropriate team members could be identified and the study team could be formed. The PWR study team organization is shown in Figure 2.

PWR formed a study team with proven experience and capabilities in the design and development of space flight-qualified systems for both NASA and the United States Air Force (USAF) and NASA MSFC propulsion system and technology development. The team used data based on space-qualified flight hardware programs, as well as data and knowledge gained on NASA MSFC propulsion system programs in performance of the HLPT program. Detailed data on propulsion system life cycle cost (LCC) [DDT&E, production and operations (fixed and variable)], safety and reliability, development schedules, and technology roadmaps generated on prior and current contracts and internally-funded projects, was leveraged in support of this study.

#	BAA Technical Objective Statements	✓	PWR SOW	PWR Final Study Report Section(s)
i	Determine the technology, and research and development, required for a Heavy Lift System (HLS). Identify and analyze multiple alternative architectures (expendable, reusable, or some combination).	✓	20000 30000 40000	3, 4, 5
ii	Identify how alternative HLS solutions address key decision attributes	✓	31000	4.1, 4.2, 4.3
1	Provide a recommended list of key decision attributes and rationale	✓	22000	3.2, 3.3
2	Provide a recommended weighting of the key decision attributes	✓	22000	3.4
3	Identify impact of changes in weighting of key decision attributes on the architectures	✓	31000	3.5, 4.3
4	Identify how alternative ground rules and assumptions impact the identified alternative system solutions	✓	31000	4.1, 4.3
5	Identify how innovative or non-traditional processes or technologies can be applied to the HLS to improve affordability and sustainability	✓	42000	5.1, 5.2, 5.3, 5.4
6	Identify how aspects of a HLS (including stages, subsystems, and major components) could have commonality with other user applications, including NASA, DoD, commercial, and international partners	✓	31000	4.3
7	Identify how incremental development testing, including ground and flight testing, of HLS elements can enhance the heavy lift system development	✓	42000	5.3
8	Identify capability gaps associated with the HLS using established metrics, i.e. NASA Technology Readiness Level (TRL), etc.	✓	41000	5.3
9	Identify capability gaps associated with the first-stage main engine functional performance and programmatic characteristics. Identify any impacts to overall life cycle costs of the HLS based on the engine studied.	✓	31000 32000 41000	4.2, 4.3, 5.2
10	Identify capability gaps associated with the upper-stage main engine functional performance and programmatic characteristics. Identify any impacts to overall life cycle costs of the HLS based on the engine studied.	✓	31000 32000 41000	4.2, 4.3, 5.2
11	Identify capability gaps associated with all other technical aspects of the HLS. Identify test and integrated demonstrations to mitigate risk associated with the gaps.	✓	41000	5.2
12	Identify capability gaps associated with the in-space propulsion elements functional performance and programmatic characteristics. Identify any impacts to overall life cycle costs of the HLS based on the engine studied.	✓	31000 41000	5.3
13	Identify capability gaps associated with all other technical elements of the in-space space propulsion element. Identify test and integrated demonstrations to mitigate risk associated with the gaps.	✓	31000 41000	5.3
14	Identify what in-space space propulsion elements should be demonstrated via space flight experiments	✓	42000	5.3

Figure 1. PWR study addressed all technical objectives in the BAA.

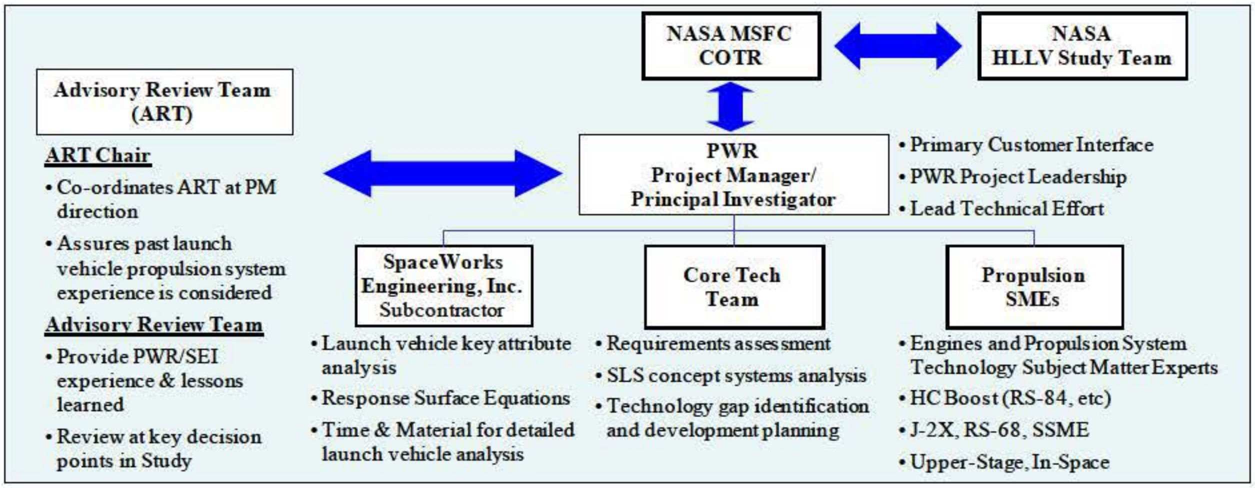

Figure 2. Team orginazation was structured to responsively meet NASA needs.

PWR's key subcontractor, SpaceWorks Engineering, Inc. (SEI), has over 10 years experience conducting similar vehicle and technology analysis studies for NASA, USAF, and commercial customers. Since 2007, SEI has provided cost estimating and affordability integration support to the NASA Constellation Program Systems Engineering and Integration (SE&I) Strategic Analysis Cost Team, including model sensitivity analysis and integrated affordability assessments. SEI's work with NASA provided anchored LCC analysis for exploration missions and launch systems in the HLPT study.

2.3 Program Approach

Details of the study approach and results are presented in the following sections of the report. Figure 3 summarizes the key study activities. The requirements assessment and key decision attribute discussion is detailed in Section 3. The SLS Vehicle Concept Analysis study activities and results are detailed in Section 4. The Affordability, Innovations, and Technology Assessment work is detailed in Section 5.

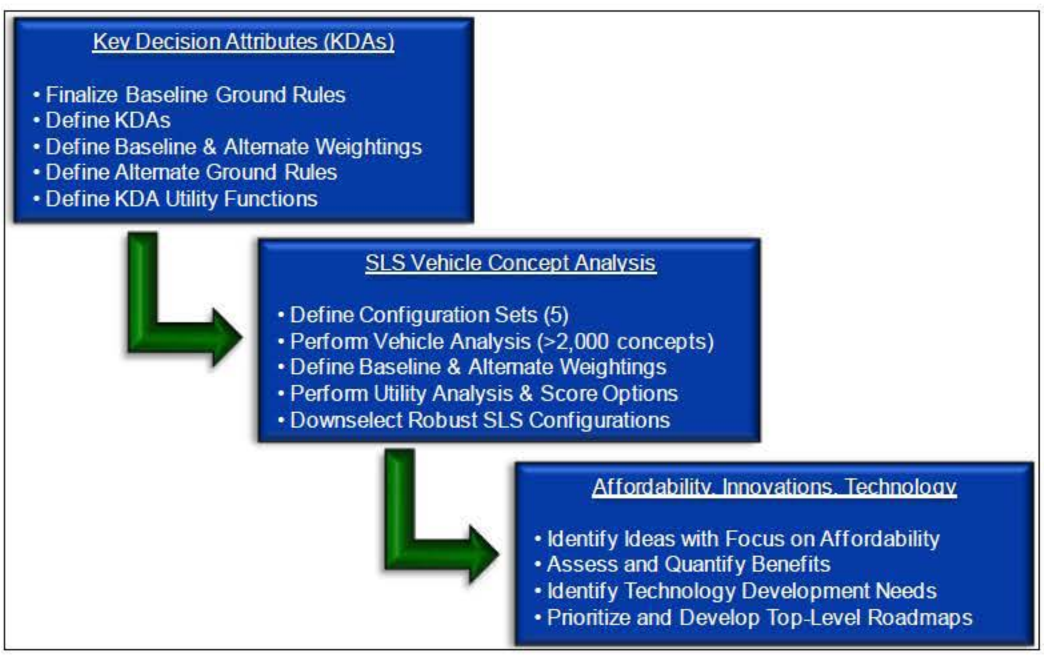

Figure 3. Key activities of study approach.

PWR used a detailed Study Plan that focused the technical key contributors to meet our commitments on cost, schedule, and quality of deliverables. PWR continuously reviewed the study plan with NASA to ensure the study had correct and up-to-date focus. The study plan/schedule, provided in Figure 4, includes all summary project activities and obligations.

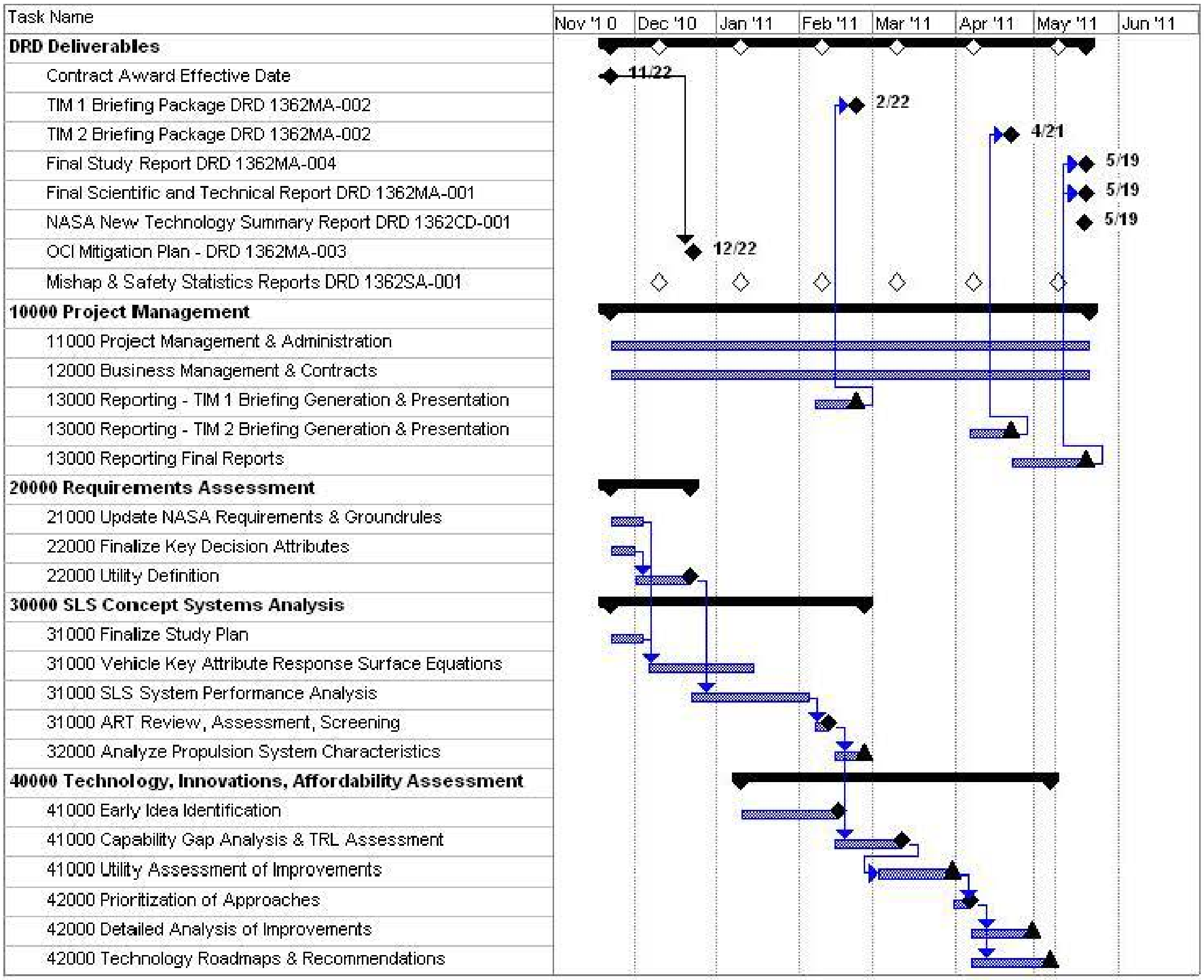

Figure 4. Integrated Master Schedule was aligned to meet customer needs and milestones.

Our Technical Approach followed a clearly defined integrated Work Breakdown Structure (WBS), Statement of Work (SOW) and Integrated Master Schedule (IMS). The study involved Requirements Assessment (WBS 20000), SLS Concept Systems Analysis (WBS 30000), Technology, Innovations, Affordability Assessment (WBS 40000), and Project Management (WBS 10000). The study used a standard process that features identification of ground-rules and assumptions, utility based selection criteria, alternate candidate development and maturation, independent key decision attribute assessments, capability gap identification, technology/innovations/affordability idea prioritization, and roadmap/risk reduction planning. Our Advisory Review Team (ART), with broad experience in space-qualified systems and studies, provided critical review and guidance at key decision points in the study, infused our industry domain knowledge and lessons learned to the core technical team, and provided a consistent view of all analytical results.

Figure 5 provides a more detailed depiction of the study approach including screening the SLS vehicle configuration point designs to enable downselecting a set of robust concepts and screening the technology, innovations, and affordability improvement ideas to provide recommendations and quantification of the potential benefits. Task boxes shaded in green were completed prior to the study contract work.

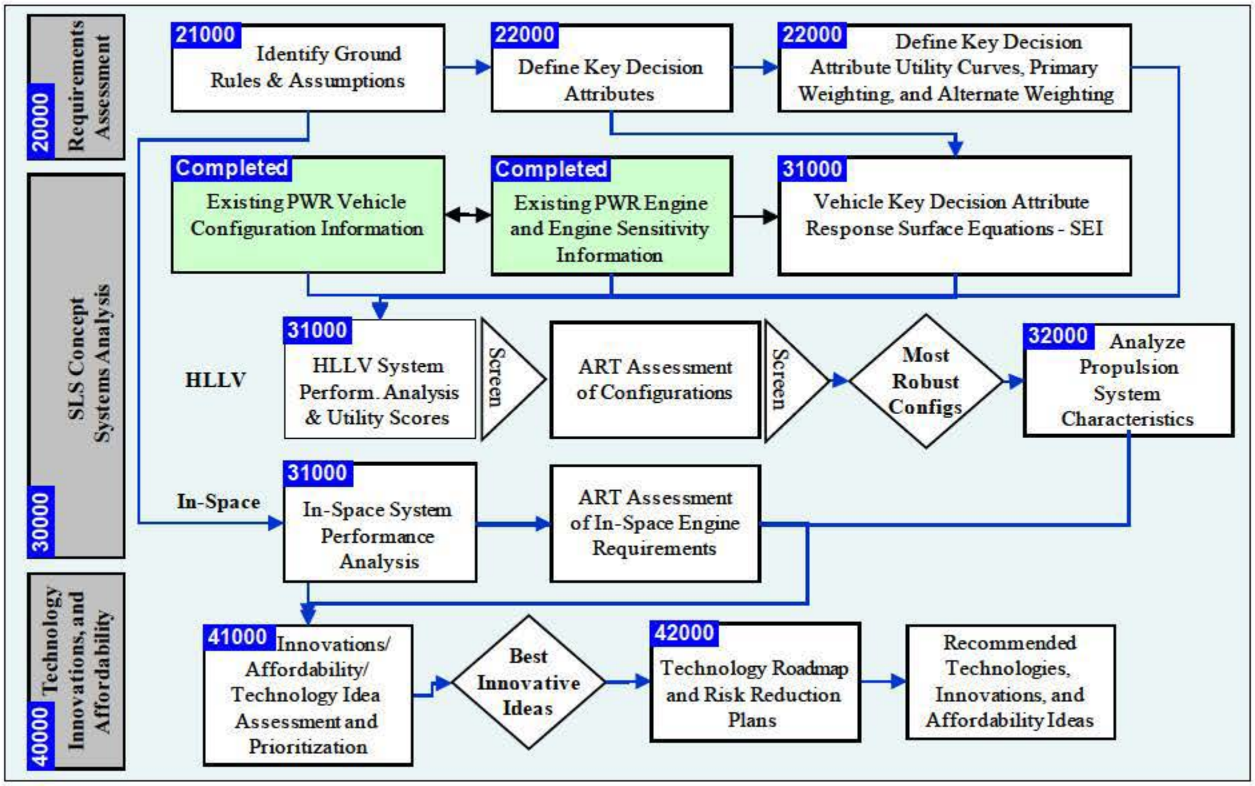

Figure 5. Our Team performed utility trades and analysis using a well defined process that leveraged the team capabilities and prior internally-funded work to assess over 2,200 SLS vehicle configuration point designs.

3 Requirements and Key Decision Attributes Assessment

The PWR study team reviewed all documents associated with SLS requirements and ground rules and assumptions that were provided as reference material under the HLPT contract.[2, 3, 4, 5, 6, 7, 8] After reviewing the reference material and incorporating lessons learned from prior PWR vehicle system study experience, a set of baseline ground rules and assumptions and conceptual definition requirements were developed for use in the HLPT study. The major study ground rules and assumptions are described in Section 3.3 under the related key decision attributes.

3.1 Utility Analysis

Utility analysis methodology was used in the study to identify and assess the key decision attributes relevant to the SLS. The utility analysis was led by a PWR subject matter expert with experience performing propulsion system level utility analysis on the J-2X engine program with NASA MSFC. The methodology enables assessment of competing design influences by balancing the identified objectives in proportion to their system-level importance. The utility analysis enabled quantitative comparisons among dissimilar attributes by developing mathematical models to map customer, end-user, and decision-maker preferences over the SLS trade space. A multi-functional team was used to assess the SLS objectives with the intent to capture the sensitivity of end customer expectations as defined key decision attributes.

Utility analysis was used to score each SLS configuration using a common methodology. The overall utility is a function of the value of each attribute and represents overall strength of customer preference and is defined by the following equation:

$$U = k_1u_1(x_1) + k_2u_2(x_2) + k_3u_3(x_3) + k_4u_4(x_4) + k_5u_5(x_5) + k_6u_6(x_6)$$

where:
k_i = weighting of attribute i;
u_i = utility function of attribute i;
x_i = value of attribute i

SLS key decision attributes and lower and upper bound values were defined as discussed in Section 3.2. The utility function for each of the defined key decision attributes was developed as described in Section 3.3. The baseline weighting (or scale factor, k) for each KDA was systematically determined as detailed in Section 3.4

3.2 Key Decision Attribute Definitions and Ranges

The PWR multi-functional team including engineering, program management, manufacturing, and NASA customer relations representatives reviewed the NASA SLS and Heavy Lift Launch Vehicle (HLLV) objectives and pertinent reference material and defined a set of six SLS key decision attributes. The focus of the KDA definition activity was to capture quantifiable attributes that were expected to strongly influence the decision making process. The identified KDAs cover the life cycle of the SLS and therefore include the design, development, and test period as well as recurring operations for SLS launches and missions.

Figure 6 provides a summary of the six SLS key decision attributes identified and assessed in the HLPT study.

KDA #	Key Decision Attribute	Definition	Units	Lower Bound	Upper Bound
1	**SLS Payload to LEO**	Gross payload delivered to LEO (30x130 nmi @ 28.5°) from KSC	mT	70	160
2	**SLS Development Schedule**	Months from Authority to Proceed (ATP) to Operational Capability Readiness (OCR) for SLS configuration	Months from ATP to OCR	72	96
3	**SLS Safety**	Probability of loss of crew	MMBF	500	2000
4	**SLS Development Cost through 1st Operational Flight**	DDT&E cost (including two test flights) + production and operations for the first operational flight	FY10$B	5	15
5	**SLS Recurring Cost per Year**	Production (fixed and variable) and operations costs per year	FY10$M	400	1800
6	**Over/Under NASA Yearly Budget Profile**	Highest yearly SLS cost relative to NASA budget profile. A negative value means that the predicted yearly SLS cost profile is under the NASA budget profile throughout the SLS development and production.	FY10$M	-1000	+1000

Figure 6. SLS Key Decision Attributes were defined and ranges were set to facilitate utility analysis.

The six KDAs are defined as follows:

1. SLS Payload to LEO: The gross payload delivered by the SLS vehicle to a 30x130 nmi low Earth orbit (LEO) at 28.5° inclination from KSC. Payload is defined as the total injected mass at the destination orbit minus the burnout mass of the final stage. Quoted payload capabilities are gross mass delivered to final destination which includes any payload margin. The unit of measurement for this KDA is metric tons (mT).

2. SLS Development Schedule: The period from Authority to Proceed (ATP) to Operational Capability Readiness (OCR) for the SLS configuration. Operational Capability Readiness is defined as being the date which flight tests have been successfully conducted and the Design Certification Review has been completed. The unit of measurement for this KDA is months.

3. SLS Safety: The probability of loss of crew for an SLS ascent to orbit. The unit of measurement for this KDA is Mean Missions Between Failures (MMBF).

4. SLS Development Cost through 1st Operational Flight: The Design, Development, Test, and Evaluation (DDT&E) cost including two test flights and the production and operations cost of the first operational flight. This KDA captures all SLS cost from ATP through and including the first operational flight. The unit of measurement for this KDA is Fiscal Year 2010 billion dollars (FY10$B).

5. SLS Recurring Cost per Year: The production (including all fixed and variable) cost and operations cost per year. The unit of measurement for this KDA is Fiscal Year 2010 million dollars (FY10$M).

6. Over/Under NASA Yearly Budget Profile: The highest yearly SLS cost relative to an assumed NASA budget profile. A negative value means that the predicted yearly SLS cost profile is under the NASA budget profile throughout the SLS development and production. The unit of measurement for this KDA is Fiscal Year 2010 million dollars (FY10$M).

As part of the utility analysis methodology, lower and upper bound values for each KDA were defined. The KDA value bounds were defined to be wide enough to be able to capture and assess all reasonable SLS configuration options while not being so spanning that the impact of differences in KDA values would be diluted when comparing SLS configurations. Figure 6 provides the lower and upper bound values for each of the six KDAs.

The lower and upper values for the six KDAs are as follows:

1. SLS Payload to LEO: 70 mT to 160 mT

2. SLS Development Schedule: 72 months to 96 months

3. SLS Safety: 500 MMBF to 2000 MMBF

4. SLS Development Cost through 1st Operational Flight: $5B (FY10$B) to $15B (FY10$B)

5. SLS Recurring Cost per Year: $400M (FY10$M) to $1800M (FY10$M)

6. Over/Under NASA Yearly Budget Profile: -$1000M (FY10$M) to +$1000M (FY10$M)

3.3 Key Decision Attribute Utility Functions and Ground Rules and Assumptions

A utility function, $u_i(x_i)$, was constructed for each key decision attribute to model the strength of preference over the feasible range as defined in the prior section. The utility functions were developed through set of detailed exercises with the multi-functional PWR team evaluating hypothetical trades and degrees of risk aversion and facilitated by our subject matter expert. With appropriate ranges, the utility functions result in an "S"-shaped curve representing three distinct regions. The slope of each curve indicates the relative worth of marginal improvement.

3.3.1 *SLS Payload to LEO*

The utility function for this KDA is depicted in Figure 7. The shape of the utility function indicates that, above approximately 140 mT, there is a diminishing return on additional payload capability. Incremental improvements are most valuable in the mid-range of about 100 mT to 130 mT.

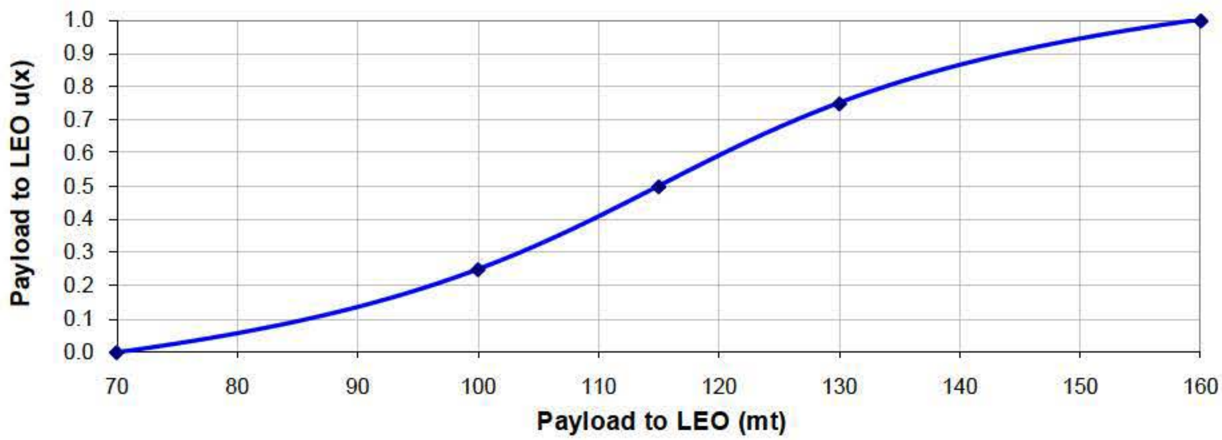

Figure 7. Payload to LEO Utility Function.

The study team assessed the SLS reference material and established the following ground rules and assumptions relevant to the SLS Payload to LEO KDA: (1) the payload range of 70 mT to 160 mT was intended to capture a wide range of potential SLS missions; (2) the vehicle and mission analysis methodology utilized the NASA HLLV trajectory / ascent flight profile ground rules and assumptions [2]; (3) the destination orbit was set at 30x130 nmi at 28.5° orbit from KSC; (4) all quoted payloads are based on a cargo configuration SLS; (5) vehicle weights and sizing includes 15% mass growth allowance on all dry masses excluding the engine; (6) vehicle weights and sizing includes fuel propellant reserve (FPR) of 1% of total ideal ΔV to LEO with the final stage carrying the entire FPR; (7) vehicle weights and sizing assumes Aluminum-Lithium (Al-Li) tanks, dry structures, and shroud; and (8) utilized NASA HLLV payload shroud volume assumptions for in-line configurations to LEO. Additional details of the SLS vehicle analysis are included in Section 4.2.

3.3.2 *SLS Development Schedule*

The utility function for this KDA is shown in Figure 8. As indicated, the utility function for development schedule was determined to be highly linear over the defined range of 72 months to 96 months.

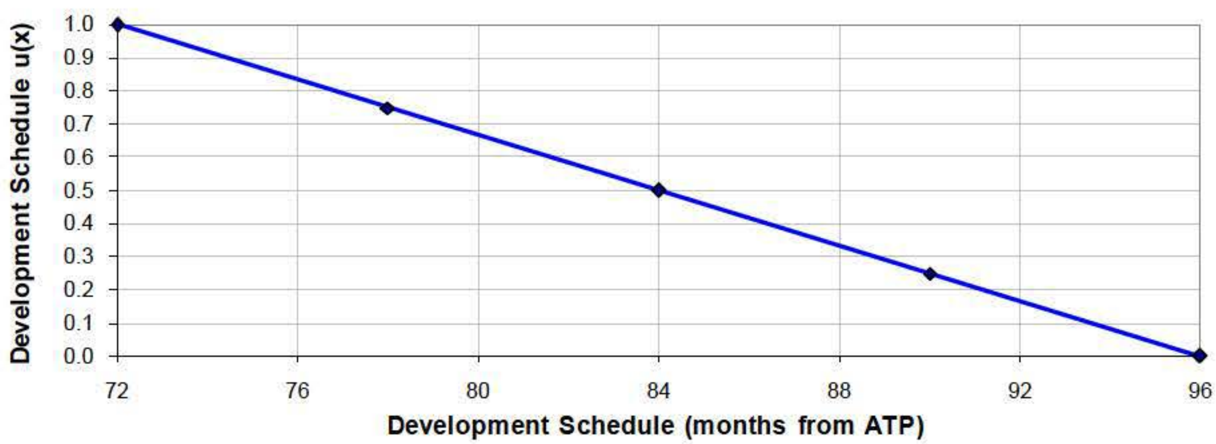

Figure 8. Development Schedule Utility Function.

The major ground rules and assumptions associated with the SLS development schedule are: (1) the period covers all work from ATP to OCR; (2) OCR occurs 6-12 months after the second test flight; (3) stage development schedules are based on the output from running the NASA/Air Force Cost Model (NAFCOM); (4) engine development schedules are based on internal-PWR bottoms-up analyses; and (5) two test flights are included as part of the vehicle development schedule. Additional details of the development schedule determination for SLS configuration options are included in Section 4.2.

3.3.3 *SLS Safety*

The utility function for this KDA is shown in Figure 9. The key ground rules and assumptions for SLS safety are: (1) all engines and stages were assumed to be human-rated and have reached operational maturity; (2) engine out capability was not included in the analyses; (3) each vehicle configuration assumed a launch abort system (LAS) with a common 95% LAS success; (4) event tree analysis-based vehicle reliability prediction with historically-based subsystem and event reliabilities was used for all SLS configurations.

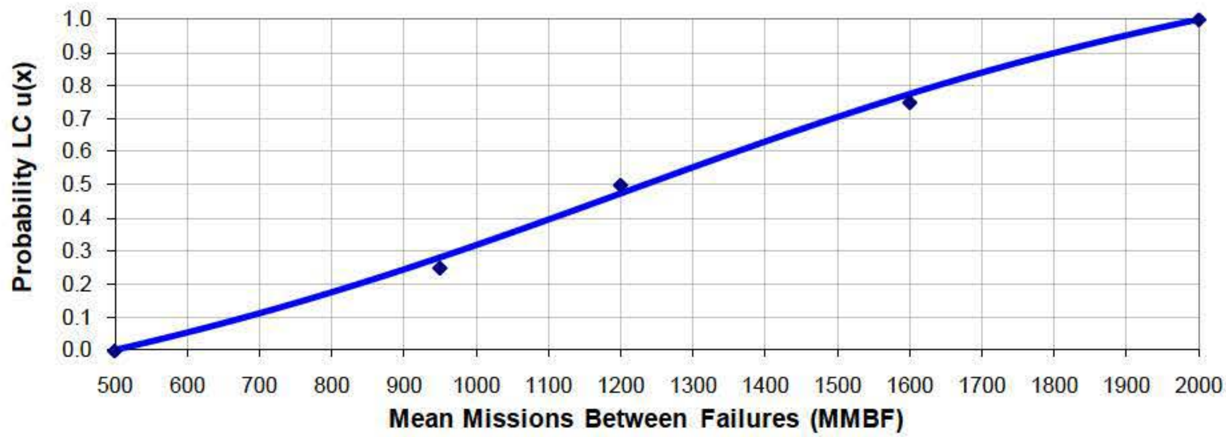

Figure 9. Probability Loss of Crew Utility Function.

3.3.4 *SLS Development Cost through 1st Operational Flight*

The utility function for this KDA is shown in Figure 10. The shape of the utility function indicates that, below approximately \$8B, there is a diminishing return on additional reduction in development cost. Incremental improvements are most valuable in the mid-range of about \$8B to \$12B.

The major ground rules and assumptions associated with the development cost are: (1) NAFCOM was used for all stage and vehicle-level DDT&E cost estimates; (2) detailed PWR estimates were used for all engine DDT&E costs; (3) estimated costs include facility modifications covering modification to mobile launch platform, crawler, pad (one of each at a flight rate of 1/year), and pad propellant tankage and includes development of vertical payload transporter for Multi-Purpose Crew Vehicle (MPCV)/LAS integration with the SLS; (4) includes the cost of two test flights during DDT&E and first operational flight; (5) includes any configuration-dependent fixed costs; (6) all estimates are prices in fixed FY2010\$; and (7) all values include a standard 20% contingency/rough order of magnitude (ROM) factor. Additional details of the development cost estimates for SLS configuration options are included in Section 4.2.

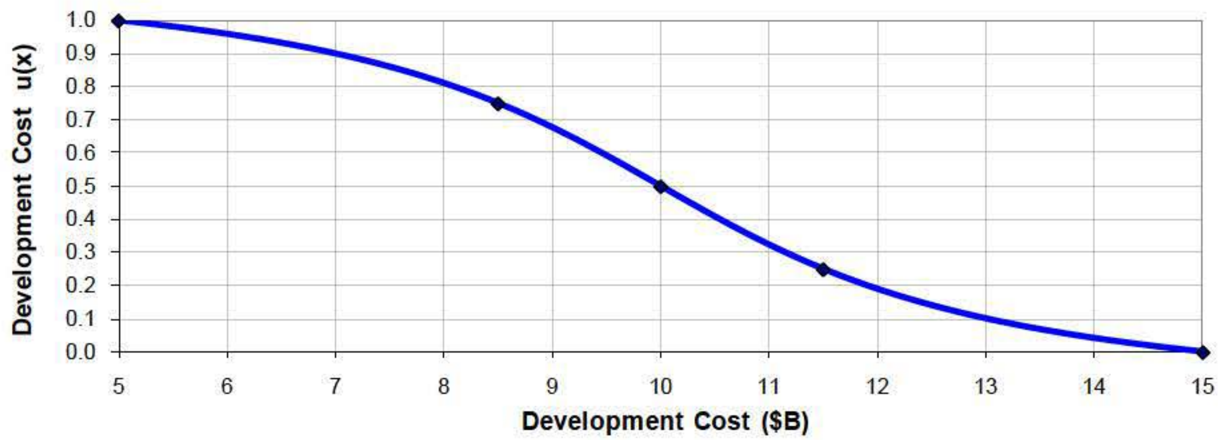

Figure 10. Development Cost Utility Function.

3.3.5 *SLS Recurring Cost per Year*

The utility function for this KDA is shown in Figure 11. The shape of the utility function indicates that, below approximately $800M, there is a diminishing return on additional reduction in recurring cost. Incremental improvements are most valuable in the mid-range of about $800M to $1300M. The utility function range was set at $400M to $1800M to capture all configurations assessed and also to address all affordability improvements

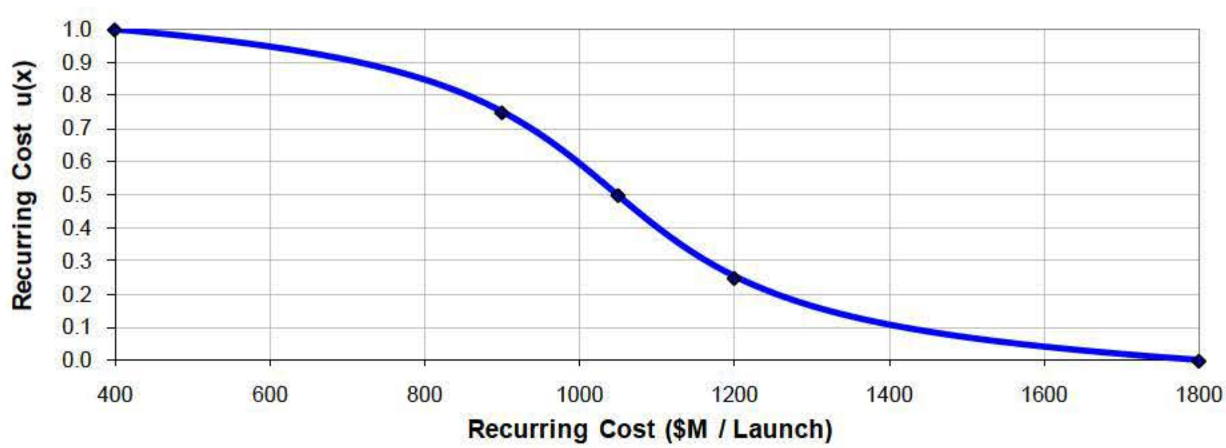

Figure 11. Recurring Cost Utility Function.

The major ground rules and assumptions associated with the recurring cost are: (1) NAFCOM was used for all stage and vehicle-level production cost estimates; (2) detailed PWR estimates were used for all engine production costs; (3) Conceptual Operations Manpower Estimating Tool/Operations Cost Model (COMET/OCM) was used for configuration-based vehicle ground operations cost estimates; (4) includes any configuration-dependent fixed costs; (5) assumes one flight per year for initial trades; (6) all estimates are prices in fixed FY2010$; and (7) all values include a standard 20% contingency/ROM factor. Additional details of the recurring cost estimates for SLS configuration options are included in Section 4.2.

3.3.6 *Over/Under NASA Yearly Budget Profile*

The utility function for this KDA is shown in Figure 12. The shape of the utility function indicates that incremental improvements are most valuable in the mid-range of about -$250M to +$250M.

The key ground rules and assumptions associated with the over/under cost are: (1) a fixed NASA yearly budget of $1.8B for SLS in fixed FY2010$; (2) captures the ability of an SLS configuration's development and production cost profile to fit a flat NASA SLS budget profile; (3) assumes the ability to stretch and/or phase the stage, engine, and facility development in order to fit to the NASA budget profile where feasible. Additional details of the over/under estimates for SLS configuration options are included in Section 4.2.

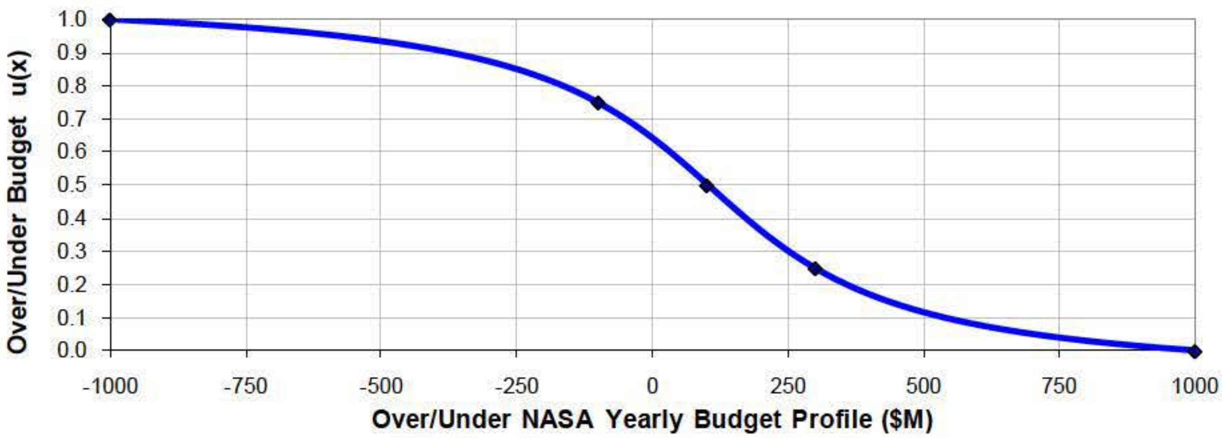

Figure 12. Over/Under NASA Yearly Budget Utility Function.

3.4 Key Decision Attribute Baseline Weightings

Baseline KDA weighting (scale factors) were developed by the team using a systematic process led by the PWR utility analysis subject matter expert. A set of detailed exercises were conducted to consider pairs of KDAs together to establish the relative scales of the individual KDA utility curves and to determine the trade factors between attributes. The results of the baseline KDA weighting assessment are shown in Figure 13. As indicated, the KDAs associated with affordability (Development Cost, Recurring Cost, Over/Under NASA Yearly Budget) account for 58% of the total weighting. It should be noted that SLS Safety was weighted at 8% as a result of the range chosen for that attribute (500-2000 mean missions between failures) with all configurations having improved safety with a higher MMBF than Shuttle ascent. The KDA weighting is indicative of the importance of the six KDAs in differentiating between SLS configuration for decision-making purposes and does not reflect the inherent importance of an essential attribute such as crew safety.

KDA #	Key Decision Attribute	Definition	Units	Weighting / Scale Factor
1	**SLS Payload to LEO**	Gross payload delivered to LEO (30x130 nmi @ 28.5°) from KSC	mT	15%
2	**SLS Development Schedule**	Months from Authority to Proceed (ATP) to Operational Capability Readiness (OCR) for SLS configuration	Months from ATP to OCR	18%
3	**SLS Safety**	Probability of loss of crew	MMBF	8%
4	**SLS Development Cost through 1st Operational Flight**	DDT&E cost (including two test flights) + production and operations for the first operational flight	FY10$B	31%
5	**SLS Recurring Cost per Year**	Production (fixed and variable) and operations costs per year	FY10$M	6%
6	**Over/Under NASA Yearly Budget Profile**	Highest yearly HLLV cost relative to NASA budget profile. A negative value means that the predicted yearly HLLV cost profile is under the NASA budget profile throughout the HLLV development and production.	FY10$M	21%

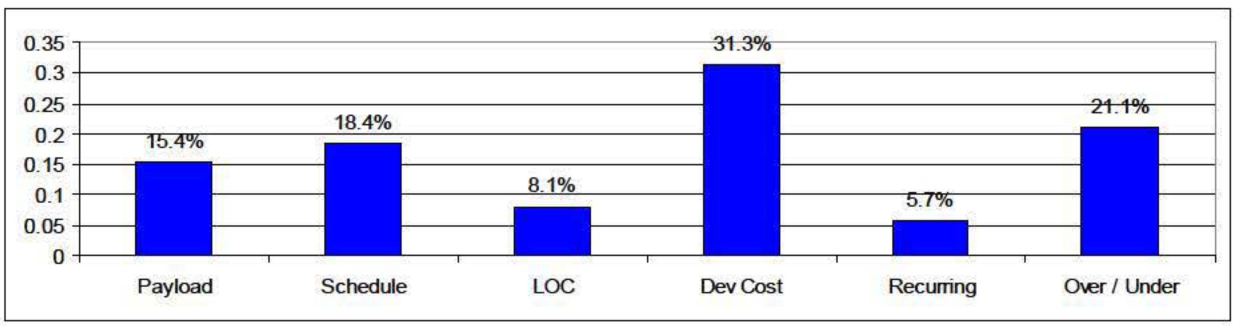

Figure 13. Baseline KDA weighting was developed using systematic utility analysis process.

3.5 Key Decision Attribute Alternate Weightings

The baseline KDA weightings were developed to describe the end-user preferences to the best ability of the study team. In addition to the baseline KDA weightings, the study team assessed and developed three sets of alternate weighting scenarios to capture the sensitivity to alternative customer viewpoints. Figure 14 provides the baseline and alternate weightings for the six KDAs. Alternate Weighting Scenario 1 assumed the viewpoint where the 130 mT LEO Payload value (stated in congressional language) was of higher importance. Alternate Weighting Scenario 2 expressed a viewpoint where recurring cost and life cycle cost was of higher importance during the operational phase. Alternate Weighting Scenario 3 placed a higher importance on the Initial Operational Capability (IOC) date. In each of the alternate weighting scenarios, a weighting of 30% was assigned to the highest importance KDA and all other KDA weightings were proportionally adjusted downward.

KDA #	Key Decision Attribute	Baseline	Alt-1 130 mT Payload Higher Importance	Alt-2 Recurring Cost & LCC Higher Importance	Alt-3 IOC Date Higher Importance
1	SLS Payload to LEO	15%	**30%**	11%	13%
2	SLS Development Schedule	18%	15%	14%	**30%**
3	SLS Safety	8%	7%	6%	7%
4	SLS Development Cost through 1st Operational Flight	31%	26%	23%	27%
5	SLS Recurring Cost per Year	6%	5%	**30%**	5%
6	Over/Under NASA Yearly Budget Profile	21%	18%	16%	18%

Figure 14. Study assessed three alternate weighting scenarios to capture sensitivity to alternative customer viewpoints.

As stated previously, utility analysis was used to score each SLS configuration using a common methodology. The overall utility is a function of the value of each attribute and represents overall strength of customer preference and is defined by the following equation:

$$U = k_1u_1(x_1) + k_2u_2(x_2) + k_3u_3(x_3) + k_4u_4(x_4) + k_5u_5(x_5) + k_6u_6(x_6)$$

where:
k_i = weighting of attribute i;
u_i = utility function of attribute i;
x_i = value of attribute i

The mathematical relationships developed for each of the six key decision attribute utility functions were incorporated into utility score calculations. Utility scores were calculated for each of the over-2,200 SLS vehicle configuration point designs using the Baseline Weighting Scenario as well as each of the three Alternate Weighting Scenarios. The resulting scores for the baseline and alternate weightings were used in downselecting the set of eight robust SLS vehicle configurations as discussed in Section 4.3.

4 Space Launch System Vehicle Concept Analysis

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5 PWR Innovation, Technology, and Affordability Improvements

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6 Results and Conclusions

The objective of the PWR HLPT study was to use systematic analytical methodology to determine a set of robust SLS configurations that scored consistently high across a range of baseline and alternate key decision attribute weightings as opposed to promoting a single specific vehicle design. Based on the downselected set of eight robust SLS configurations, main propulsion system innovations, technology & affordability improvements were assessed and quantified. The results of the study as well as several key observations and conclusions are described below.

6.1 Results - Key Decision Attributes

As detailed in Section 3, the PWR multi-functional team reviewed the NASA SLS and HLLV objectives and pertinent reference material and defined a set of six SLS key decision attributes with a focus on capturing quantifiable attributes that were expected to strongly influence the SLS decision making process. The identified KDAs cover the life cycle of the SLS and therefore include the design, development, and test period as well as recurring operations for SLS launches and missions. Utility analysis was performed to define a set of equations to score and compare SLS configurations. A baseline set of weightings and three sets of alternate weightings were developed to assess the SLS configurations with the objective of downselecting a set of robust vehicle configurations.

The six KDAs identified and used in the HLPT study are defined as follows:

1. SLS Payload to LEO (mT): The gross payload delivered by the SLS vehicle to a 30X130 nmi orbit at 28.5° inclination from KSC.
2. SLS Development Schedule (months): The period from Authority to Proceed (ATP) to Operational Capability Readiness (OCR) for the SLS configuration.
3. SLS Safety (MMBF): The probability of loss of crew for an SLS ascent to orbit.
4. SLS Development Cost through 1st Operational Flight (FY10$B): The Design, Development, Test, and Evaluation (DDT&E) cost including two test flights and the production and operations cost of the first operational flight.
5. SLS Recurring Cost per Year (FY10$M): The production (including all fixed and variable) cost and operations cost per year.
6. Over/Under NASA Yearly Budget Profile (FY10$M): The highest yearly SLS cost relative to an assumed NASA budget profile.

6.2 Results - SLS Vehicle Concept Analysis & Robust Configuration Downselect

As described in Section 4, over 2,200 SLS vehicle configuration point designs were analyzed spanning five configuration sets with numerous engine options: (1) LOX/LH_2 Core, LOX/LH_2 Upper Stage, Solid Boosters; (2) LOX/RP First Stage, LOX/LH_2 Upper Stage, (3) LOX/LH_2 Core, LOX/LH_2 Upper Stage, Liquid Boosters; (4) LOX/RP Core and Boosters, LOX/LH_2 Upper Stage; and (5) LOX/LH_2 Core and Boosters, LOX/LH_2 Upper Stage.

A robust conceptual vehicle analysis was performed on each of the 2,200+ SLS vehicle configuration point designs to determine estimates for each of the SLS KDAs. Conceptual vehicle design disciplines analyzed include aerodynamics, propulsion, weights and sizing, trajectory, ground operations costs, DDT&E and production costs, safety and reliability, and DDT&E and production schedules.

The results of each of these discipline analyses were used to calculate a utility score for each of the SLS vehicle configuration point designs using both a baseline set and three alternate sets of utility score weightings. These utility score results, along with calculations for overall configuration LCC and information about configuration applicability to non-NASA customers were used to downselect to a set of eight robust SLS configurations.

Figure 22 summarizes the eight downselected configurations. Three of the configurations are in the class of the NASA SLS reference using a LOX/LH_2 core (RS-25E or RS-68B engines), a LOX/LH_2 upper stage (J-2X or RS-25E engines), and two 5-segment PBAN SRBs. Two of the configurations also use a LOX/LH_2 core and LOX/LH_2 upper stage (J-2X) with four liquid boosters derived from the Atlas V LOX/RP core stage (RD-180 or F-1A engines). One of the configurations is an in-line two-stage vehicle with a LOX/RP first stage (F-1A engines) and a LOX/LH_2 second stage (J-2X engine). An additional configuration uses three LOX/RP common booster cores (F-1A engines) and a LOX/LH_2 upper stage (J-2X engines). The final configuration uses three LOX/LH_2 common booster cores (RS-68B engines) and a LOX/LH_2 upper stage (J-2X engines).

These eight downselected configurations were each examined in detail with a range of potential PWR innovation, technology, and affordability improvements incorporated.

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6.4 Key Conclusions & Recommendations

PWR implemented a systematic analytical methodology which resulted in the downselection of eight robust SLS vehicle configurations. Innovation, technology, and affordability improvements were applied to these downselected SLS vehicle configurations with a focus on reducing SLS vehicle DDT&E and production costs.

The following key conclusions and observations were identified as a result of this study:

1. Viable Shuttle-Derived vehicle configurations in the 130 mT class using a LOX/LH_2 first stage, a LOX/LH_2 second stage, and with SRBs or LOX/RP LRBs can meet the NASA budget profile and provide 130 mT full-up payload capability at first operational flight.

2. Utilization of existing and derivative engines provides a benefit of low Design, Development, Test and Evaluation (DDT&E) costs, less risk, and high demonstrated reliability.

   a. Effective use of existing RS-25D engine assets during development (use on 2 test flights) with early start and concurrent development of the improved-affordability RS-25E engine provides significant DDT&E and recurring cost savings.

   b. The F-1A gas-generator cycle engine provides the best option for a LOX/RP large booster propulsion application by leveraging PWR/NASA program heritage for a low risk and low cost propulsion development program.

   c. The J-2X engine provides flexibility and extensibility as propulsion for an upper stage and in-space/Beyond-Earth-Orbit system in the SLS architecture for human space exploration.

3. Several in-space propulsion system technology development opportunities were identified and preliminary technology development roadmaps were defined for Nuclear Thermal Propulsion and Power, Nuclear/Solar Electric Propulsion and Power, and LOX/Methane Chemical Propulsion.

4. Significant PWR propulsion system cost reductions are enabled with incorporation of programmatic, technology, and engine-specific affordability innovations.

a. PWR Consolidation Initiatives will reduce infrastructure cost by more than 40% and enable better fixed cost sharing through Flexible Mixed Model operations.

b. PWR has adequate production capacity for foreseeable NASA and DoD needs with some nominal capital required for high production rates.

c. PWR's Flexible Mixed Model brings the benefit of PWR overall production rate across all customers to each program with up to 27% engine cost reduction.

d. PWR is actively working specific affordability improvements for the RS-25E, J-2X, RS-68B, and F-1A engines that are applicable to SLS.

e. Opportunity exists to leverage PWR-NASA experience to reduce traditional NASA oversight costs by formulating a shared/blended workforce with NASA and contractor participation to reduce the cost of customer oversight through innovative contracting.

f. PWR advanced and low-cost materials and processes are showing strong near-term potential for reducing propulsion system component-specific production costs.

Taken together, these affordability innovations were calculated to provide significant cost savings for the downselected SLS vehicle configurations. These cost savings range from $1.0B to $1.7B for the combined costs of DDT&E, two test flights, and first operational flight. The corresponding production cost savings ranged from $200M to $400M per flight. PWR is actively working many of these affordability improvements and stands ready to work together with NASA in implementing these and other innovations in support of an innovative, affordable, and sustainable path for human space exploration.

References

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Space Launch Systems

Systems Analysis and Trade Studies

Final Report

(b) (4) [redacted]

Northrop Grumman Systems Corporation, Aerospace Systems, El Segundo, California

Dated: June 3, 2011

Updated: June 26, 2011 to incorporate NASA comments

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Prepared for Marshall Space Flight Center
Under Contract NNM11AA11C

Foreword

This report documents work completed on the Space Launch Systems Systems Analysis and Trade Studies (abbreviated SLS Studies), a Broad Agency Announcement (BAA) contract NNM11AA11C. The work was performed by Northrop Grumman's Aerospace Systems, Huntsville, Alabama and Redondo Beach, California. Mr. Timothy Flores, NASA Marshall Space Flight Center, was the NASA contracting officer's technical representative. [redacted] was program manager for Northrop Grumman Corporation (NGC). [redacted] led NGC's system analysis; [redacted] was the NGC chief engineer and led the cost analysis and [redacted] was the configuration design and manufacturing engineer. Support contractors KT Engineering of Madison, Alabama, Nelson Engineering of Merritt Island, Florida and the University of Alabama Huntsville Propulsion Research Center provided technical guidance and assistance to the project. A complete list of team members is given below.

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Abstract

This is the Northrop Grumman final report for the Space Launch Systems (SLS) Systems Analysis and Trade Study performed for the National Aeronautics and Space Administration (NASA) Marshall Space Flight Center. The SLS configuration design space has been thoroughly explored with a compelling focus on sustainability. Key sustainability drivers are; incremental development within available funding profile, low production and operations fixed costs and SLS flexibility. Hundred of SLS configurations were assessed using the Northrop Grumman quick-sizer launch vehicle conceptual design tool. Promising concepts representing the entire design space were then sized using a full trajectory multi-disciplinary optimization simulation. Computer aided design solid models were built to verify configuration geometry. Payload fractions, mission reliability and lifecycle costs were analyzed for a variety for human space exploration missions from low Earth orbit to Mars surface. Configurations were ranked using a Pugh relative rating method with a variety of weighting factors to test sensitivities for near and long term emphasis. Many good SLS configuration options were identified. Northrop Grumman recommends a three body liquid oxygen and rocket propellant (LOX/RP) first stage with a liquid oxygen and liquid hydrogen (LOX/LH2) multi-purpose upper stage. This configuration ranks well for all criteria and sensitivities. The major discriminator for the recommended configuration is the robust family of SLS vehicles that can be incrementally developed, using a single production system, with shared fixed costs, for a wide variety of exploration, science and national security missions. This SLS family will be sustainable for many decades of space exploration.

Key Words

Space Launch System (SLS)

National Aeronautics and Space Administration (NASA)

Marshall Space Flight Center

Sustainable

Exploration beyond Earth orbit

Northrop Grumman

1. Introduction and Overview

Our Space Launch Systems (SLS) Systems Analysis and Trade Study has explored an unencumbered trade space with a wide variety of exploration architecture and SLS configuration options. The results are presented in this non-proprietary final report. This study evaluated a thorough set of SLS Vehicle configuration options using a set of decision attributes. The products are used to select a recommended SLS configuration that is sustainable for human exploration beyond Earth orbit including Mars surface missions. The study also identifies any major capability and technology gaps and opportunities to assist NASA in planning future SLS technology development activities.

Figure 1-1 summarizes our key sustainability drivers for the space launch system based on 83 years of corporate experience in aerospace systems development, production and operations support. Our single greatest concern is that the SLS is sustainable for many decades of space exploration. We strongly believe in these drivers and recommend that the reader to refer to them often while evaluating this report.

1. **Incremental development within HSF funding profile** ***human spaceflight (HSF) budget profiles are flat with very limited ability to absorb non-recurring spikes. Stretched out design development test and evaluation (DDT&E) to meet funding profile adds costs and inhibits progress***
2. **Low Production and Operations Fixed Costs** ***Low mission rate, flexible missions and long system life drive a compelling need for low fixed costs***
3. **SLS Flexibility** ***Exploration architecture and launch needs will evolve. Need to react to currently unknown missions and architectures. Needs to be available for unplanned and other user missions***

Figure 1-1 SLS Sustainability Drivers

For our study approach we addressed the need to be flexible for 30+ years of exploration beyond LEO. This included selection of three reference missions to span near and long term. These included missions to

- Low Energy Near Earth Objects (including Lunar surface)
- High Energy Near Earth Objects and Mars orbit
- Mars Surface

We recognize that allowances and robustness in the launch system architecture must accommodate the large uncertainties in inherent in this 30+ year span, including questions about long duration crew requirements for habitability, consumables, micro gravity tolerance and radiation tolerance, means for capture, entry descent and landing at Mars and maturation of capability to remotely produce propellants, such as using Mars in-situ resource utilization (ISRU.) Other uncertainties include in space propulsion capabilities and roles (cryo-storage, electric propulsion, nuclear propulsion) and the potential for application of pre-deployed propellant and equipment on surface and in orbit.

In addition to payload to orbit and escape, this study considered the payload volume and accommodations for a full range of missions, including large scale scientific missions, human in-space servicing of high value science and national security assets, and a the deployment of future national security assets in higher orbits.

Our study approach is established on evaluation of key affordability drivers and figures of merit (FOMs). In particular these evaluations were prioritized to the SLS key sustainability drivers:

Our study of potential SLS vehicle configurations considered a wide ranging trade space. This included exploration of improvements to current systems, such as EELVs, improvements to Ares V and application of Shuttle derived elements, such as the reusable solid rocket motors (RSRMs) and Space Shuttle main engines (SSME's). The evaluations applied two performance modeling tools that operated with successive levels of turnaround time versus model complexity and fidelity: quick-sizer produces truly instant configuration assessments and FASTPASS, provides high fidelity, industry benchmark class outputs. The tools were calibrated to Saturn V, Ares V and Delta IV Heavy performance and mass properties. We rough sized hundreds of initial concepts representing eight families and four paths to escape. From this we prescreened to 14 concepts using a relative rating, Pugh matrix approach.

We conducted trajectory and vehicle sizing to optimize payload to escape velocity for direct and in-space LOX refueling paths and prepared CAD models to validate vehicle general arrangement and iterated sizing as required. This information is captured in sets of concept description packages. We prepared relative ratings to Ares V as a standard reference. In concert with these evaluations the team conducted capability and technology push sessions with suppliers in order to identify emerging launch vehicle capabilities.

We then down selected ten concepts representing five families which were presented at the study technical interchange meeting-1 (TIM-1). Assessments included reliability analysis for three sample missions and two paths and a lifecycle cost analysis for DDT&E and fixed and variable production and operations costs for 40 missions through 2036. Based on information gathered at TIM1 and further study of in-space requirements, cross feed, engine layout, flexibility and growth paths we refined a subset of our configuration options and closed on a recommended SLS configuration. We recommend that the SLS be a three body liquid oxygen and rocket propellant (LOX/RP) first stage with a liquid oxygen and liquid hydrogen (LOX/LH2) multi-purpose upper stage.

Our recommended configuration and path forward were presented at the TIM-2 to complete the study.

2. Exploration Architecture Options

2.1. GR&As, Alternatives *(SOW 2.1)*

In this section we describe the overall study approach and provide example results presenting findings as well as illustrate the study methods. The study resulted in two basic conceptual trade study and analysis phases. The initial phase which was reported at the SLS Study TIM 1 stepped through the process for a wide range of SLS vehicle configuration options. It provided ground rules and assumptions (GR&As) assessment, exploration architecture definitions and exercised a FOM assessment and weighting approach to provide an evaluation of these configurations and an initial down select of the most effective concepts. This section presents the GR&A, FOM assessments and weighting approach for this initial cycle. Section 3 summarizes the conclusion on the preferred launch vehicle concepts and then presents the results of the second iteration of the cycle, which was provided in TIM 2 and presented the NGC preferred concept resulting from this FOM evaluation approach.

At the outset of the study we received the NASA GR&As supplied for consideration during the SLS Study. The GR&A's were assessed and categorized as to their applicability as shown in Figure 2-1. The set adopted as "OK as is" are found in Figures 2-2, 2-3 and 2-4. These GR&A's are useful for narrowing the option space and providing a point of departure set of analysis assumptions and performance simulation modeling input. In a few instances we have departed or updated these values, such as: use of 400 km altitude LEO basing orbit which is more appropriate for LEO aggregation, refueling and beyond LEO departure. Alternate performance calculation assumptions are used in place of Figure 2-2 19-32. See descriptions of the quick-sizer and FASTPASS performance modeling tools in Sections 2.3.

Heavy Lift and Propulsion Technology Study					
Ground Rules and Assumptions Review					
NGC Assessment	Tech	Cost	SM&A	Total	%
Ok as is	32	12	6	50	37%
Concern for restrictions	11	2	4	17	13%
Concern for fair comparison	9	7	1	17	13%
Concern for study resources	2	3	0	5	4%
Details	32	3	4	39	29%
Ask for info reference	3	3	0	6	4%
Total	89	30	15	134	100%

Figure 2-1 GR&A Review

Technical GR&As	
1.	VAB Launch Vehicle Stack Integrated Height Constraint = 390 ft (potential trade CEV LAS integration at pad)
2.	Maximum Core Stage Length = 234 ft (VAB diaphragm limit)
3.	All Vehicle Stages: Diameter Constraint = up to 33 ft. Diameters reported as outside diameters.
4.	For Lunar Missions… Launch vehicle payload includes the CEV (CM/SM), and/or LSAM, payload to the lunar surface, LSAM adapter, and airborne support equipment (ASE).
5.	Quoted Launch vehicle payload capabilities are 'gross mass' delivered to final destination which includes any payload margin.
6.	POST 3D is a 3-DoF point mass trajectory simulation
7.	Launch from Pad 39A: gdlat = 28.608422 deg, long = 279.395910 deg, gdalt = 0 ft
8.	Grace GM02C gravity model
9.	Gram2007-Mean_Annual_Atmosphere-October2008 (EV44)
10.	Gram2007-Mean_Annual_Winds-October2008 (EV44)
11.	Start simulation at lift-off (all liquid) or SRB ignition (if using solids)
12.	Avoid instantaneous changes in vehicle attitude
13.	SRB apogee is unconstrained (product of analysis)
14.	Perigee and apogee are relative to a spherical earth whose radius equals earth's mean equatorial radius.
15.	MECO altitude is optimized for elliptical orbits, but must be ≥ 75 nmi (driven by heating rate constraint)
16.	For LEO Mission (No Upper Stage) –Gross payload capability will be analyzed for 30 nmi x 130 nmi @ 29°… orbital circularization assumed to be provided by the payload.
17.	Perigee and apogee are relative to a spherical earth whose radius equals earth's mean equatorial radius.
18.	Fairing jettison weight includes: structures, TPS and acoustic/thermal blankets
19.	Unusable Tank Volume (Ullage Gas/Manufacturing Tolerance/Loading Accuracy/Internal Equipment & Structures/Cryo Tank Shrinkage): For all stage concepts: 0.04 (3% for gas volume and 1% for cryo shrinkage & internal tank equipment)
20.	Miscellaneous Secondary Structures calculated as 5% of LVA Primary Structures for InLine Configurations
21.	Vehicle sizing is considered closed when the payload capability is between the target payload and the target payload plus 0.1%.
22.	FPR (SideMount and InLine): FPR is 1% of the total ideal dV for the mission; Final stage carries the entire FPR; Any excess FPR is not calculated as payload
23.	Fuel bias (SideMount): use SSP/ET values
24.	Fuel bias (InLine): Fuel bias mass (lbm) = 0.0013 * mixture ratio / 5.29 * usable propellant (based on INTROS mass estimating relationship); Applies to fuel tanks (core and upper stages)
25.	Residuals (SideMount): use SSP/ET values
26.	Residuals (InLine): Stage residuals mass (lbm) = 0.0631 * (usable propellant)^0.8469 (based on INTROS mass estimating relationship)
27.	Start propellant (SideMount and InLine): Core Stage calculated based on engine startup transients; Air Start Stages: zero start propellant allocated
28.	Other propellant note: Ascent Propellant includes all LOX in vertical portion of feedline
29.	Other INTROS configuration note concerning Aeroshell for Upper Stage: Aeroshell applied to inline configuration upper stage to protect MLI during ascent from aero heating environment. Aeroshell jettisoned prior to earth departure burn; No Aeroshell applied to sidemount configuration upper stages
30.	No proof analysis on tanks.
31.	Used combined worst-case loads analysis in LVA (i.e. all worst case loads happen simultaneously) with a 1.3 load uncertainty factor applied. (This matches very closely individually run load cases with a dispersed max q and a 1.5 load uncertainty factor).
32.	Upper Stages use Cryogenic material properties (if available)

Figure 2-2 Adopted Technical GR&As

Cost GR&As	
1.	Use a common WBS to ensure a consistent and complete capture of the vehicle life-cycle costs
2.	Reserves will be applied appropriate to the level of maturity of the cost data or estimate
3.	Actual Cost History - use "as is" if no changes
4.	Vehicle development (Phase A) start in 2011
5.	HLLV Initial Operational Capability (IOC) determined by schedule estimates provided by data product teams: IOC as soon as possible; (if time allows) Assessment of IOC date of 2020
6.	All cost estimates provided in FY 2010 dollars
7.	All time phased/funding costs are in real year dollars
8.	Inflation factors from the 2008 NASA New Start Inflation Index for use in 2009: KSC cost estimates are phase by FY and reported in real year dollars
9.	Costs include the HLLV, Vehicle Integration, Ground Operations, and associated Full Cost Accounting elements
10.	Phasing of cost will be based on an HLLV development start in FY2011 with IOC at the earliest possible date based on schedule assessment
11.	Carrying cost defined as minimum level of resources needed to maintain production capability, until combination of DDT&E and Production is sufficient to sustain
12.	Assume Shuttle is retired by the end of FY 11

Figure 2-3 Adopted Cost GR&As

S&MA GR&As	
1.	All engines/vehicles have reached operational maturity. Therefore, data reflects inherent reliability (mature) for the design, not first flight risk
2.	Engineering specs/standards apply or waivers required
3.	Government oversight/insight required
4.	Constellation abort capability assumed
5.	LAS jettison assumed to occur 30 sec after final ascent stage engine start. Post jettison aborts are accomplished with service module for non-catastrophic failures.
6.	Abort Effectiveness is for the ascent phase only

Figure 2-4 Adopted S&MA GR&As

2.2. Exploration Architecture Options *(SOW 3.1)*

For the study we considered a range of alternative exploration architecture approaches for establishing an operational and performance context for the SLS. We incorporated the flexible path and capability based exploration approach as outlined in the Human Spaceflight Plans Committee findings and the findings of the Human Exploration Framework Team (HEFT). The range of missions considered include near term human spaceflight objectives this decade, near earth destinations including Lagrange points and lunar orbit, lunar surface, extending to the ultimate destinations, including human Mars surface exploration in the 2030's. This architecture allows for different and evolving operational approaches for aggregating assets from multiple launches of the SLS to achieve the more demanding out year mission objectives. These include aggregation of exploration vehicles and space transportation in LEO and other staging points, such as the Earth-Moon Lagrange point 1 (EML1) and various refueling approaches, including use of propellant depots at the different staging locations.

The adopted evolutionary exploration architecture shown in Figure 2-5 integrates the range of mission performance levels

- Low Earth Orbit (LEO), such as for support of ISS
- High Earth Orbit destinations such as GEO and Lagrange points
- Early, Low Energy NEO (LENEO) and Lunar
- Mid era, High Energy NEO (HENEO) and Mars orbit
- Late era Mars Surface human exploration.

This architecture includes a time ordered sequence of staging approaches, including direct launch, LEO Aggregation, combined LEO aggregation and propellant depots, and HEO. This architecture can include a full range of mission objectives, including beyond LEO exploration (Moon, Near Earth Objects, and Mars), servicing of large space assets in LEO, GEO, Lagrange points, deployment of large aperture science payloads and deployment of large national security payloads. Instead of evaluating the SLS performance against all the shown performance levels (up and down in Figure 2-5) and architectures modes (across left and right in the figure) the study focused on six reference missions, denoted 1a and 1b for low energy NEO/lunar, 2a and 2b for high energy NEO/Mars orbit, and 3a and 3b for Mars cargo and surface missions. This is an appropriate simplification to evaluate SLS performance. The other missions, in the direct category and deployment and servicing of large telescopes are still accommodated by the SLS, but these missions lie within the performance of the six reference missions.

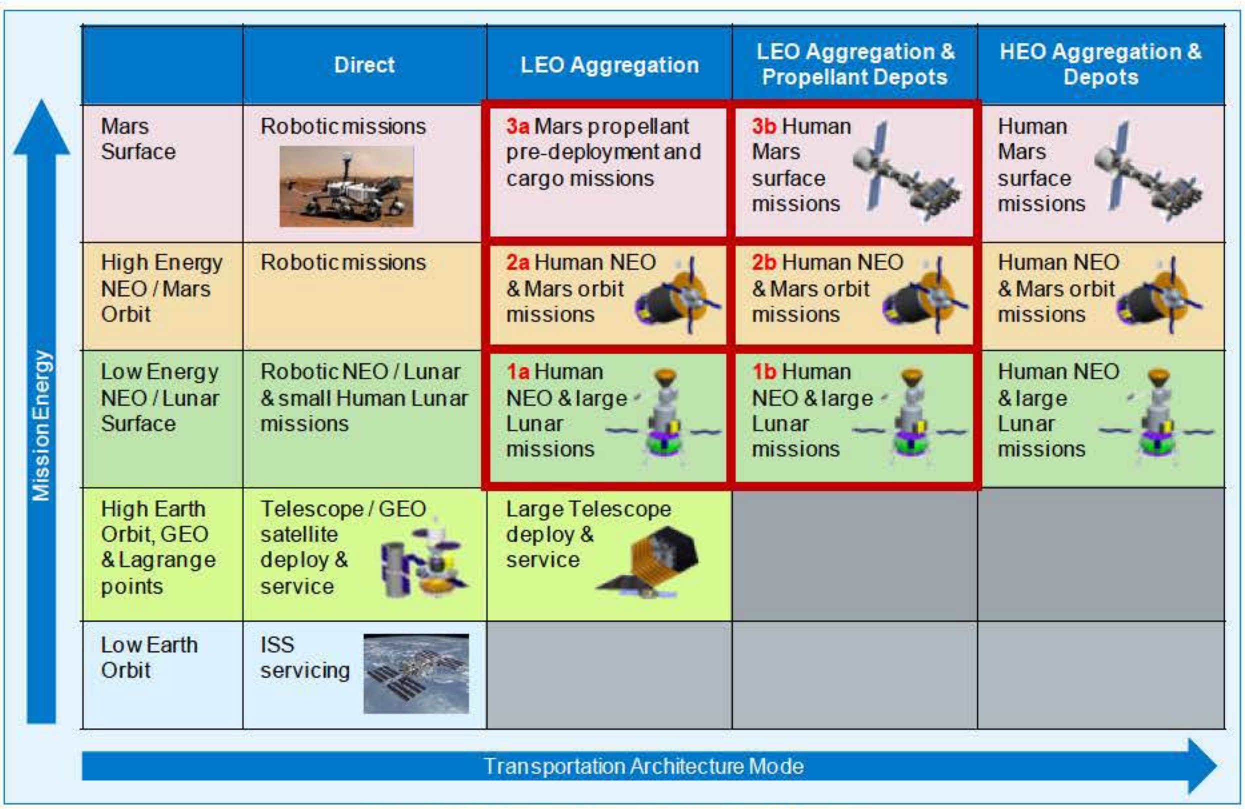
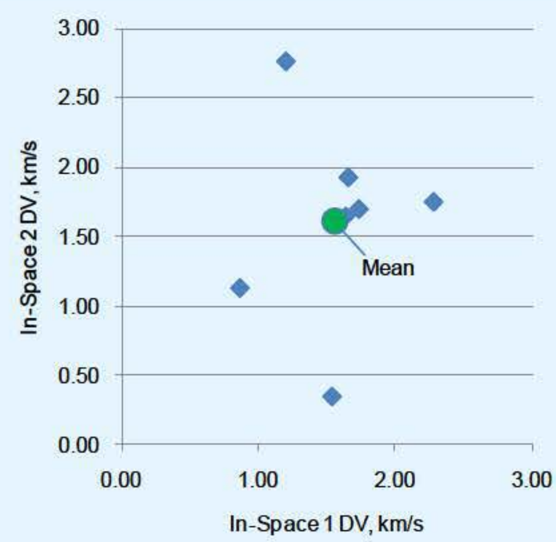


Figure 2-5 Space Transportation Architecture

Launch vehicle performance levels are referenced to delivery to velocity at earth escape (C3=0). Performance requirements at greater departure velocities (C3>0) are scaled up and for lower than escape velocities (C3<0) are scaled down. This sizes the SLS upper stage for efficiency for beyond earth orbit exploration missions. Performance required may alternatively be achieved by aggregation (multiple launches) or use of a propellant depot. Propellant depot concepts can be applied in different ways, including at LEO, MEO and Lagrange staging points and can be implemented for either LOX and LH2 or LOX, only. The latter is nearly as effective as LOX and LH2 in terms of additional performance, because of the approximately 6:1 mass ratio of LOX to LH2 for typical cryogenic engine mixture rations. The LOX only depot requires the needed LH2 to be launched with the refueled vehicles, but the added mass is relatively small. LOX only depots are much less demanding for cryo-fluid management and near zero boil off storage. This is particularly true for the thermally demanding LEO orbit staging location. It is anticipated that the propellant depot capability will evolve to both LOX and LH2 with the maturing of cryo-fluid management technology. In addition, evolving depots can be expect to be located at high energy orbit locations, such as the EML1 which would further simplify LH2 cryo-fluid management and would support increasing traffic for human exploration.

The LENO/Lunar case is a lunar surface exploration mission, from the Constellation program. For the LENO variant we used a 52 metric tonne (t) exploration vehicle and in-space delta velocity (ΔV) obtained from the MSFC-supplied NEO missions. The LENO mission in-space ΔV is 0.338 km/s for arrival at the target and 0.540 km/s for a return to earth departure for a total, 0.878 km/s. Using rocket equation sizing and the exploration vehicle mass this determines the SLS payload required at departure. The payload at escape is bounded by 66t (including crew vehicle and in-space propulsion) for both Lunar and LENO missions.

Although the payload mass was calculated separately for the HENO mission, Mars Orbit and Mars Surface missions the performance at escape velocity (Ve) are all approximately 165t and this value was used in the mission evaluation for all as shown in Figure 2-6. Considering reference NEO missions, the HENEO mission ΔVs were reduced down to the mean of the reference list providing a more reasonable requirement for this class of missions versus using the maximum energy version of the HENEO missions. Using the 50th percentile ΔV the payload at escape is also 165t. Using the maximum HENEO case results in a payload of 282 t. Figure 2-7 illustrates these performance levels at the earth escape reference point. Mars human surface exploration mission performance requirement is based on the NASA Design Reference Architecture five (DRA5), with in space ΔV of 1.7 km/s at destination arrival and 1.5 km/s for Earth return. Earth departure C3 = 13.9 km^2/s^2 which equates to 3.8 km/s departure from LEO velocity. Unlike the HENEO mission which has a single flight to the NEO,



Reference Missions	Mass at Ve, t
LENO/Lunar	66
HENO/Mars	165

Figure 2-6 In Space Delta V

DRA5 requires three separate flights. There are two cargo flights in the first Mars opportunity followed by the crew at the next opportunity. Each flight requires 165 t payloads to earth escape.

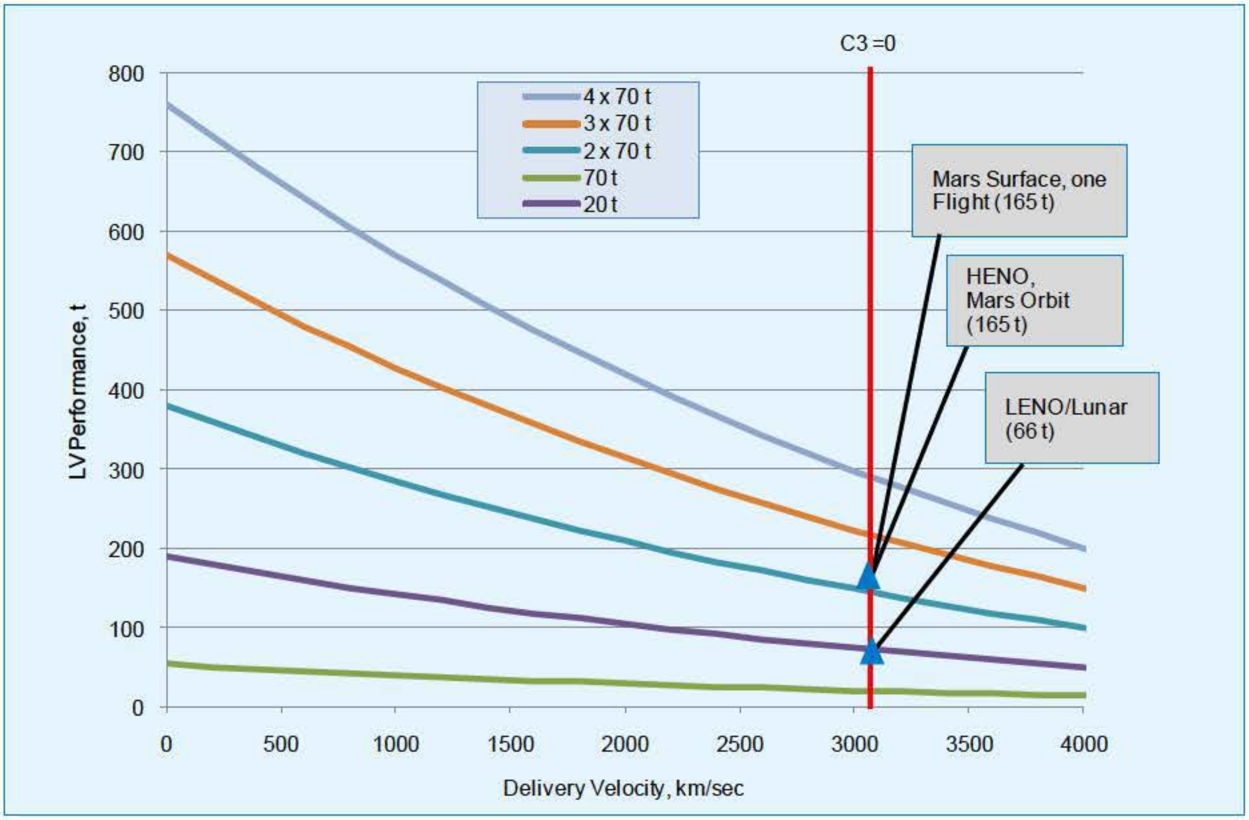


Figure 2-7 Mission Performance

We also considered the volume and accommodations for the SLS payload fairing for a range of applications, including large scale scientific missions, human in-space servicing of high value assets and deployment of future national security assets as shown in Figure 2-8.

Servicing Location	Candidate Servicing Missions
LEO 200 km- 1500km; 360 deg long. Ascending node	Multiple civil, military and international satellites. Refuelling, equipment and payload exchange, deorbit and debris removal.
LEO ISS Orbit, 52 deg, 500 km	Focused on ISS utilization. Servicing may occur at ISS, or if traffic warrants, remote servicer could be deployed.
LEO polar orbit, sun synchronous orbits	A cycling servicing satellite could be used, applying differential nodal regression.
GEO	Dominated by commercial communication satellites. High potential for refuelling followed by ORU change outs
Earth-Sun Lagrange points 1 and 2	Beneficial location for large space observatories. Potentially very high value for servicing.

Hubble Servicing Mission 4
J. Grunsfeld

Servicing and Lagrange Point Ops for Astronomy, D. Lester

Orbital Express, F. Kennedy

Figure 2-8 SLS Servicing Missions

2.3. Study Approach Figures of Merit (FOMs), Weighting Factors and Sensitivity *(SOW 2.2, 2.3, 4.1)*

Our study of potential SLS vehicle configurations considers a wide ranging trade space. This includes exploration of improvements to current systems, such as EELVs, improvements to Ares V and application of Shuttle derived elements, such as the RSRMs and SSME's. We rough sized hundreds of initial concepts representing eight families for four paths. From this we prescreened to fourteen concepts using a FOM based relative rating Pugh matrix approach which is described in the following.

We have considered FOMs from previous relevant programs and establish a set for this study with a strong focus on sustainability. FOMs are checked to assure they are quantifiable, objective and truly drive a sustainable SLS system. In particular these evaluations are prioritized to the previously introduced SLS key sustainability drivers discussed earlier in Figure 1-1.

SLS payload performance is an essential performance indicator. See Section 3.1 showing the initial set of SLS configurations and families with corresponding payload performance which is referenced as we have previously indicated, at earth escape velocity (C3=0). This is determined by the essential role of the SLS to deliver large scale exploration payload to destinations beyond LEO. Similar to performance at LEO, the earth escape velocity point provides a common reference against which different SLS configurations can be compared.

The selected payload performance FOM is the payload fraction defined as the payload divided by total vehicle dry mass for payload to LEO and to Earth escape. This FOM is a strong indicator of the configuration payload lifting efficiency, since the dry mass drives development, production and even operations cost. The next principal FOM is vehicle reliability for delivery of payloads to escape. Mission reliability to escape velocity (complement of probability of loss of mission, PLOM) is used as the FOM, since vehicle reliability for the large very high value human exploration payloads becomes a strong discriminator and is sensitive to vehicle complexities and number of failure critical components such as engines and separation systems. The last selected key FOM is life cycle cost (LCC). This is our primary metric and complements the payload FOM. Figure 2-9 summarizes these FOMs and shows how they are composed by FOM sub-elements and how the evaluation weighting applies to each of the FOMs and sub-elements. Three different weightings are employed, W1, W2 and W3 which are used to test sensitivities for equally weighed, near term and far term emphasis.

FOM Categories			Weighting Approach		
FOM Area	FOM Element	FOM Sub element	W1 equal wt.	W2 near term	W3 far term
Payload	Payload Fraction 25%	LEO	0.08	0.19	0.01
		Direct to Ve	0.08	0.05	0.05
		LOX only Depot to Ve	0.08	0.01	0.19
Reliability	Reliability to Ve 25%	Direct	0.13	0.19	0.06
		LOX only Depot	0.13	0.06	0.19
Cost	Life Cycle Cost 50%	DDT&E (2011-2016)	0.25	0.38	0.03
		Fixed (2016-2036)	0.13	0.10	0.10
		Variable (2016-2036)	0.13	0.03	0.38
		Check Total	1.00	1.00	1.00

Figure 2-9 FOM Evaluation and Weighting Approach

Figure 2-10 shows our configuration naming convention used for the study. For the first technical interchange meeting (TIM1) we down selected 10 concepts in five families. Assessments included payload performance to escape velocity and reliability analysis for three sample missions with two paths. Life cycle costs were estimated for DDT&E and production and operations fixed and variable costs for a 40 missions manifest through 2036. The configuration options were refined based on TIM1 information and additional studies and resulted in a specific SLS configuration recommendation and growth path for TIM2.

Sym	Meaning		Families
N	NGC	C	Constellation derived
A	Family	A	Atlas V derived
6	# Core Engines	D	Delta IV derived
01	Sequence 1	H	New in-line LH2 core
		K	New in-line RP core
		S	New side by side LH2
		T	New side by side RP
		U	Solid CS
		R	Reusable Strap-Ons
		Z	In-Line 3 Stage

Figure 2-10 NGC Configuration Naming Convention

Figure 2-11 shows the payload fraction to escape velocity for each TIM1 configuration. The blue diamonds are the payloads that can be delivered to escape velocity directly. The red squares are the payload to escape if the upper stage LOX tank is refueled in LEO. The direct payload fraction trends upward from 0.12 to 0.29 indicating a significant improvement versus the NC601 reference. Note in particular the increased efficiency of the RP liquid booster configurations, NK403 and NZ401. The increases going from the direct mode to the LOX refueling is dramatic and highlights how dominant LOX is to total mission mass.

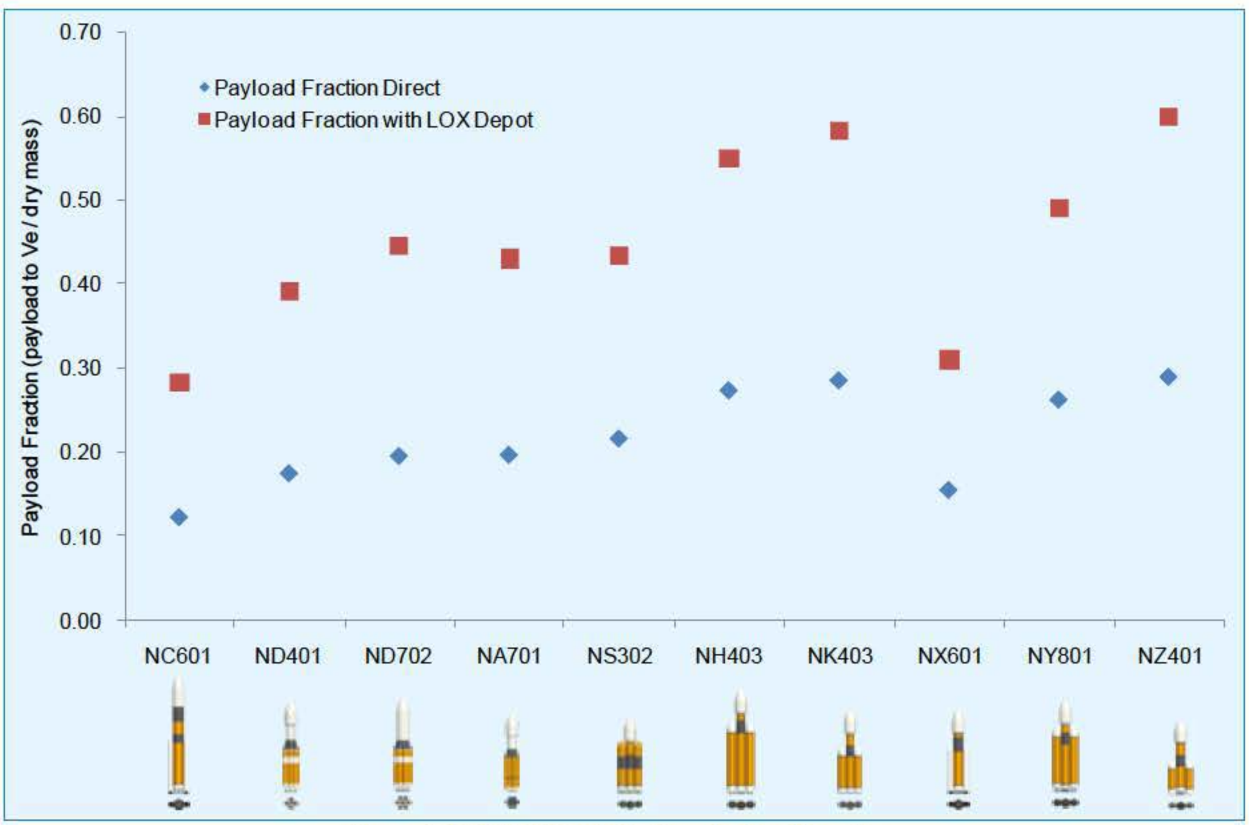


Figure 2-11 Initial Trade Options Payload Fraction

We used a Pugh relative rating matrix where each FOM sub-element is scaled to a range between -2 and 2 where 0 is the NC601 comparison standard. Weighting factors are then applied and summed across all sub elements for each configuration. Figure 2-12 shows the ratings for payload fraction. Configurations NH403, NK403 and NZ401 have superior performance for all weighting sensitivities.

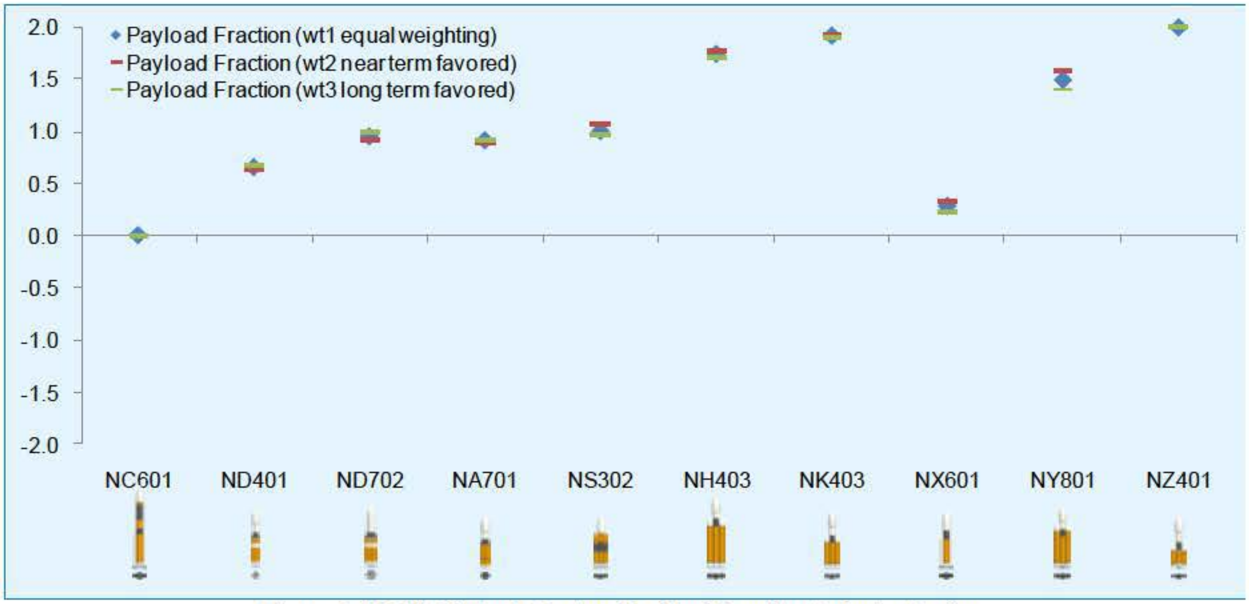


Figure 2-12 Initial Trade Options Payload Fraction Relative Rating

Mission reliability weighting in Figure 2-13 indicates reasonable values for NH403 and NK403, but NZ401 is degraded because of reduced reliability due to increase number launches to carry out the same number of missions.

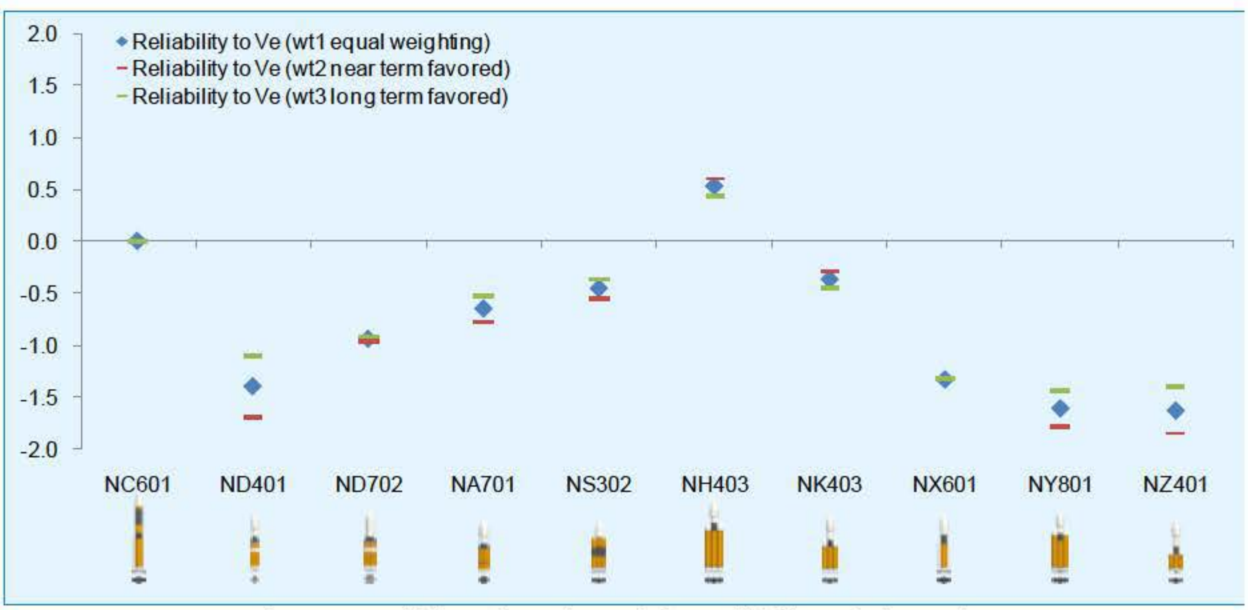


Figure 2-13 Initial Trade Options Mission Reliability Relative Rating

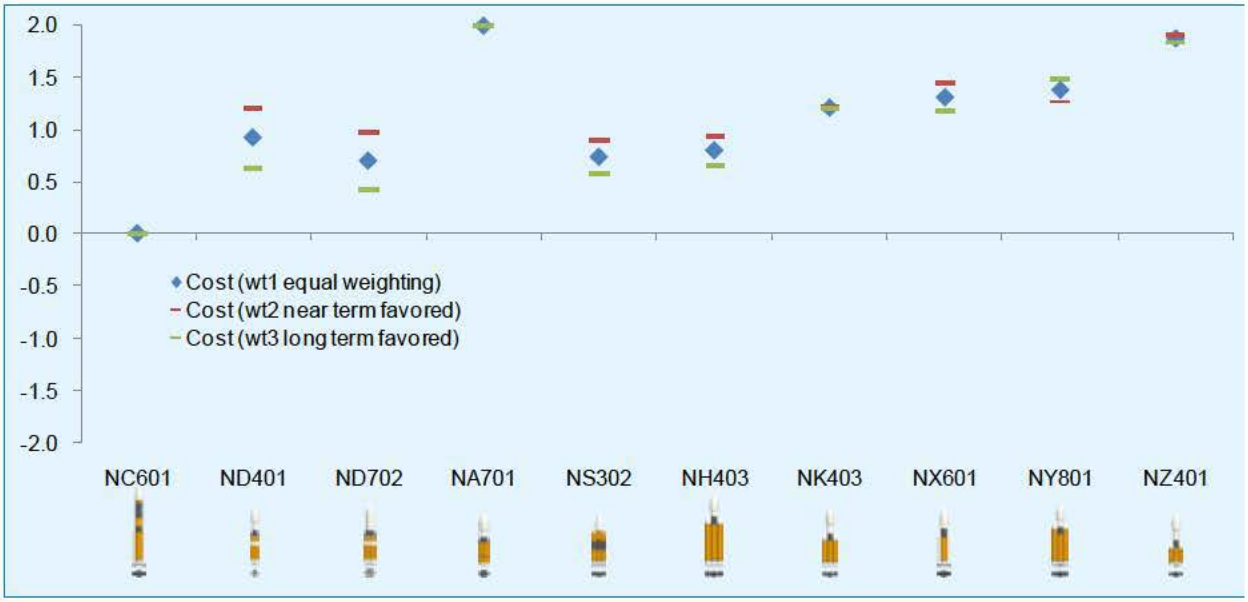


Figure 2-14 Initial Trade Options Cost Relative Rating

The cost ratings (Figure 2-14) show trends similar to the payload fraction and this time NA701 and NZ401 shows superior performance.

Figure 2-15 shows weighted total ratings in the form of high-low bars and indicating the ranking across all candidate configurations. Generally the presented configurations all rank better than the reference NC601 and collectively point to improvements in alternative approaches for achieving a sustainable SLS. NA702, NH403, NK403 and NZ401 all rank highly. NZ401 is high, but shows more sensitivity to the weighting scheme.

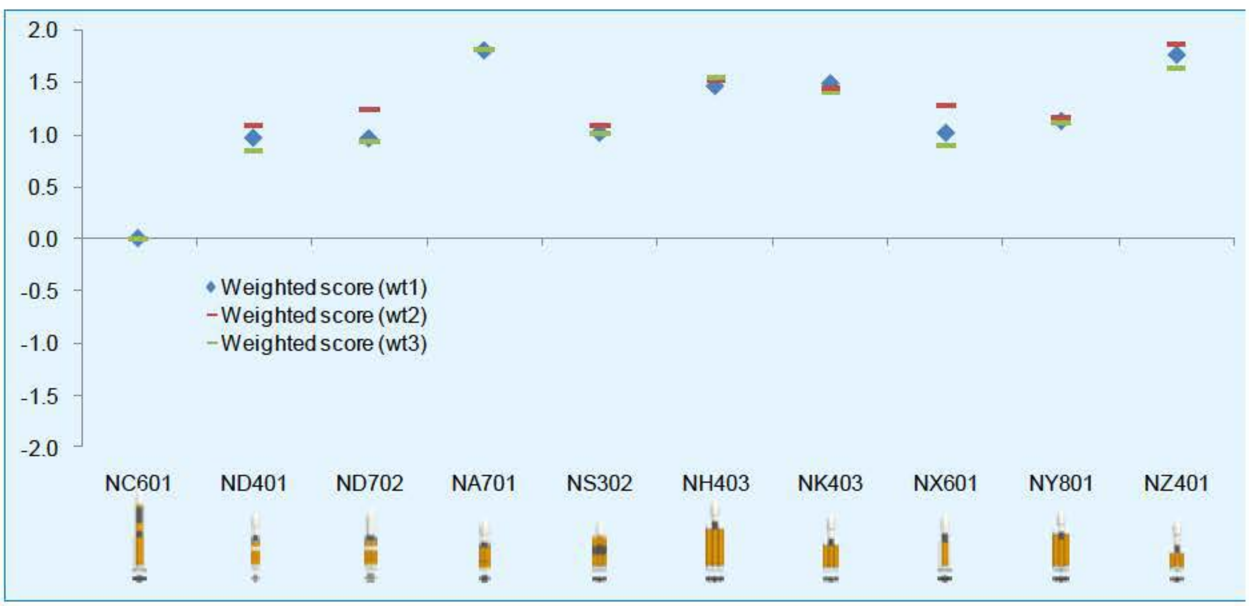


Figure 2-15 Initial Trade Options Weighted Total Rating

The reliability and cost model FOM parameters are built up based on a mission and manifest established from the exploration architecture described in Section 2.2. This modeling is described in some detail below. An intermediate input to the reliability models and LCC are the flight rates for both critical (crew exploration vehicle and key cargo elements) and non critical (propellant or propulsion elements or stages that can be replaced by spares) missions shown in Figures 2-16 and 2-17.

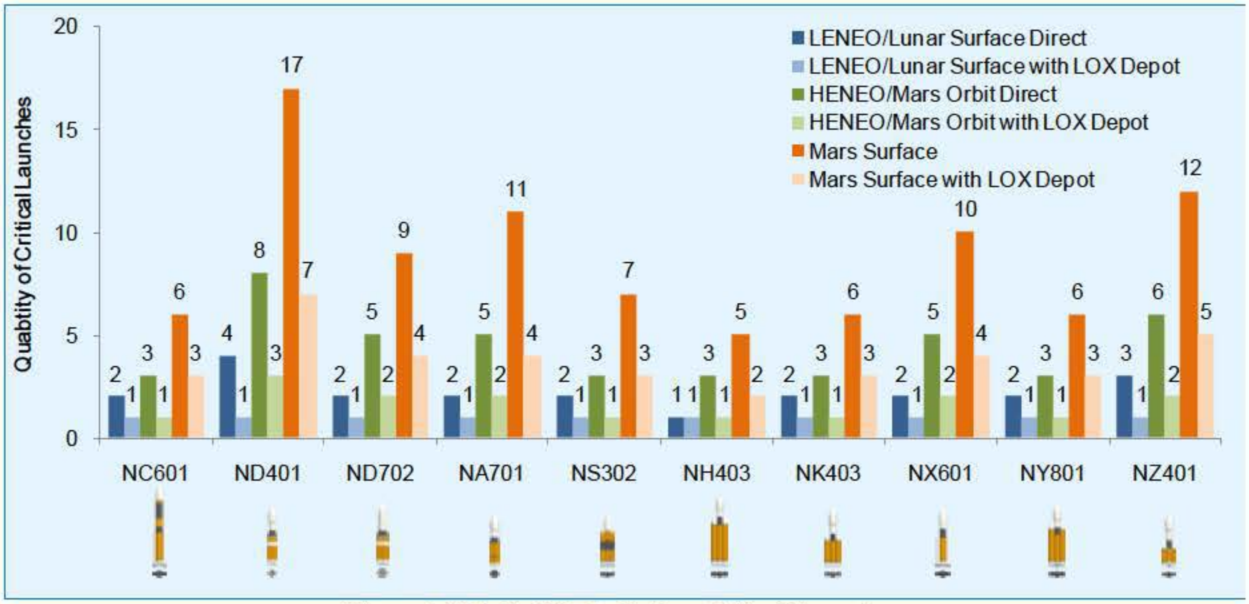


Figure 2-16 Initial Trade Options Critical Launches

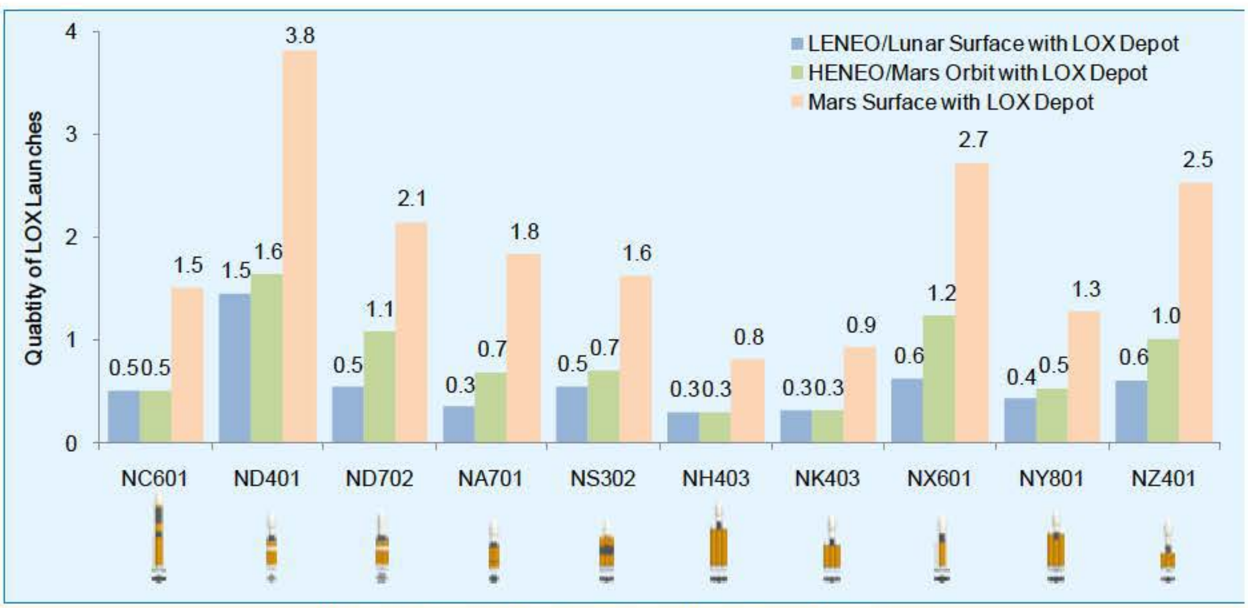


Figure 2-17 Initial Trade Options Non-Critical Launches

These are input to determine the reliability evaluations and LCC across the different configurations and mission modes such as described in Section 2.2, e.g., direct, LEO aggregation, and LOX-only depot mission modes.

The beneficial effect on mission reliability when using the LOX only depot mission mode can be seen in Figure 2-18. The values are normalized to the reliability of the Ares V, NC601 reference configuration and the LENO/Lunar Surface Direct mission mode (solid blue line.) The solid lines show for the three missions (LENO/Lunar Surface, HENO/Mars Orbit and Mars Surface) reliability using the direct mode. For these the payload is either directly launched to its velocity at departure, or in the case where there are multiple critical payloads (Figure 2-16.), they are aggregated with associated departure stages for the earth departure launch. The primary sensitivity in reliability is shown to be due to moving from the direct mode to the LOX depot mode, i.e., going from the solid lines to the dashed lines in the figure. As shown in the reliability and safety modeling approach section, below, this reliability improvement is due to the economic feasibility of providing launch spares for the "non critical" propellant launches, see Figure 2-26 showing this effect for the study downselect families mission reliability.

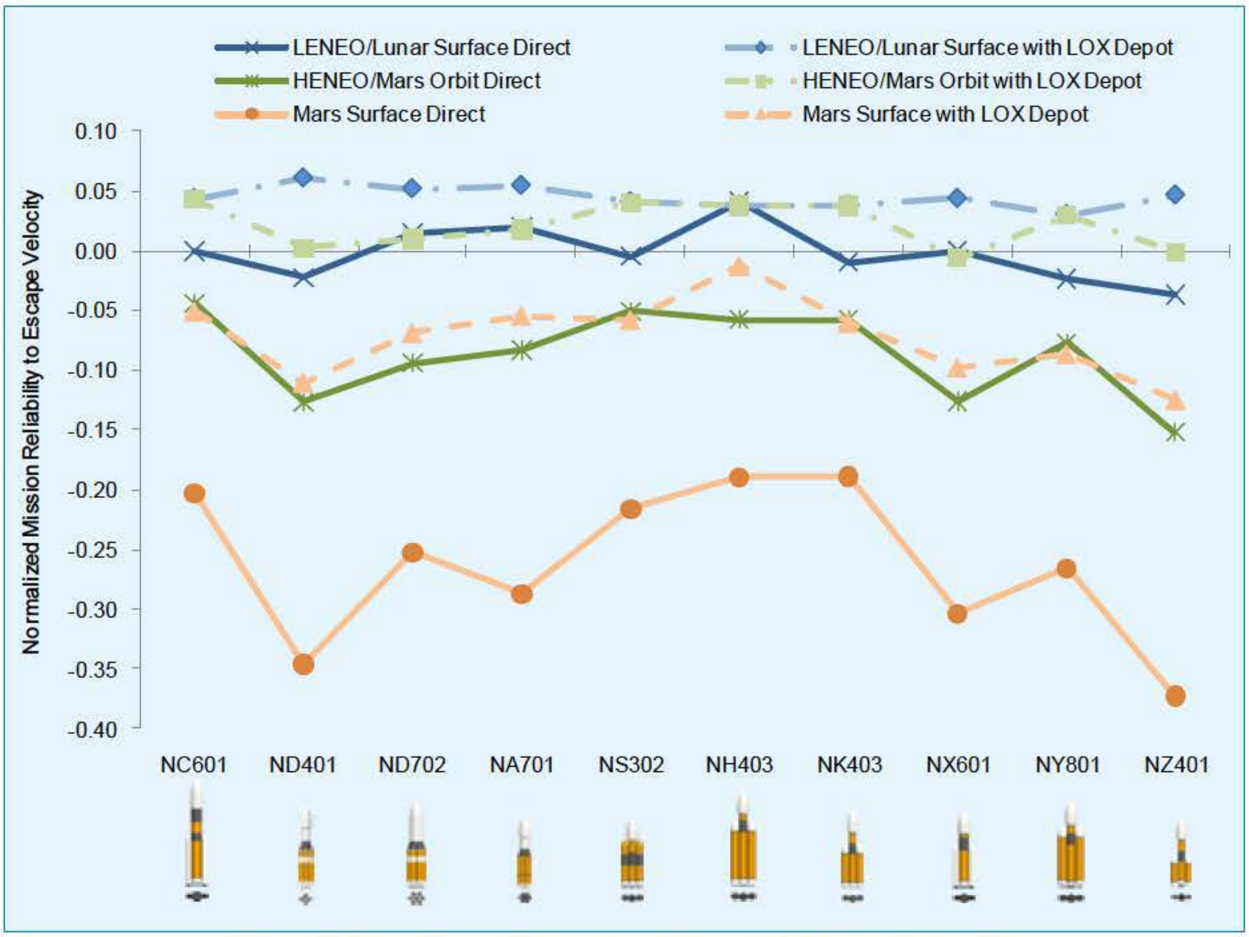


Figure 2-18 Initial Trade Options Normalized Mission Reliability to Escape Velocity

The initial trade options life cycle costs in Figure 2-19 also use the critical and non critical launches versus configuration to determine the overall flight rate which drives the LCC calculations. Our configuration development emphasis on affordability and sustainability results in significantly lower fixed costs for production and operations over the period 2016-2031 vs. the NC601 reference. The lowest LCC example is the NA701, cluster configuration vehicle which leverages the EELV vehicle production infrastructure.

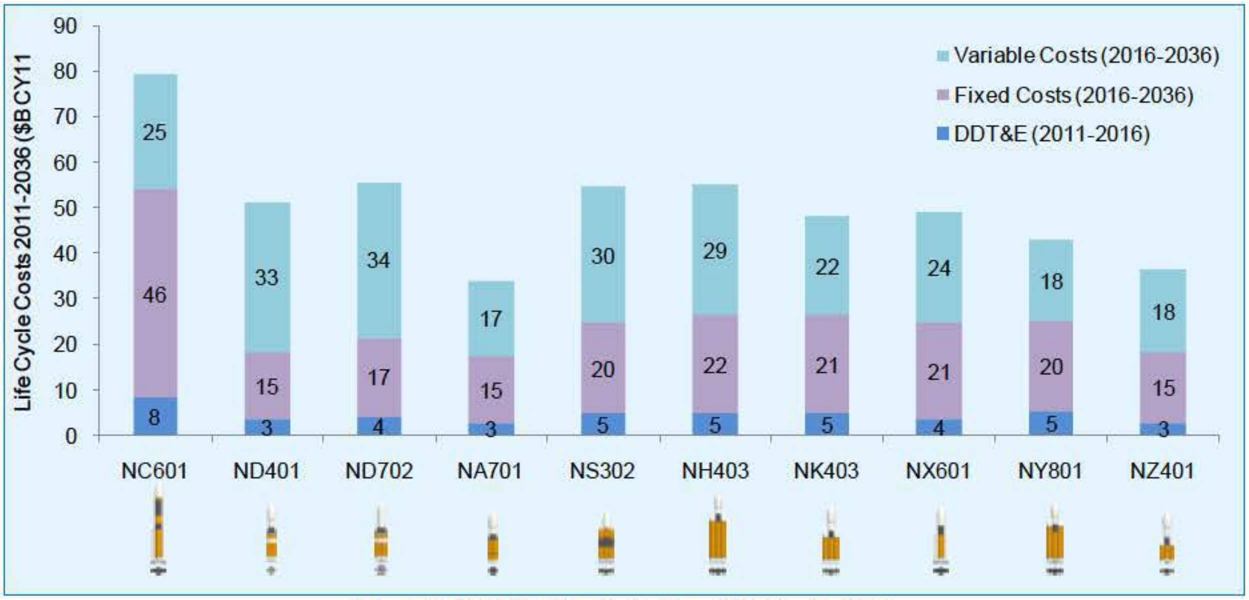


Figure 2-19 Initial Trade Options Life Cycle Cost

Trajectory Modeling and Optimizing Tools

The evaluations applied two performance model tools that operated successive levels of turnaround time vs model complexity and fidelity: quick-sizer yielding truly instant configuration assessments and FASTPASS, providing the high fidelity, industry benchmark class outputs. There tools were calibrated to Saturn V, Ares V and Delta IV Heavy characteristics. This optimization process operates at the vehicle, subsystem, and component levels. At each level, a suite of analytical tools were used to model various aspects of the problem. User requirements for payload size, desired orbit, environmental limitations, etc. were collected and used as design constraints or objective functions in vehicle synthesis.

Quick-Sizer: To facilitate a thorough exploration of the SLS configuration trade space we developed a launch vehicle quick-sizer before study go ahead and updated it during study execution using company discretionary funds. The quick-sizer uses the ideal rocket equation, catalogue solid rocket booster (SRB) and liquid rocket engine performance parameters, legacy stage mass fractions and deliberately conservative thrust to weight constraints. It solves for payload to a selected mission delta velocity (ΔV) and sizes propellant volumes, stage mass, residuals mass and dry mass for each stage. Our sizer is built in MS Excel and uses a series of look up tables to very quickly populate an SLS configuration option with catalogue SRB parameters, liquid rocket engine parameters and legacy stage mass fractions. The investigator defines the number a type of stages, quantity and type of engines, most relevant legacy stage, payload fairing shape and size, any throttle back settings and whether cross feed is used or not. In addition the investigator can select whether liquid rocket boosters have common tanks with the core or are unique. The investigator closes the configuration using a variety of built-in sizing and convergence macros. There are macros to solve for a given payload, to adjust payload and staging point for thrust to weight constraints, to update tank volumes and to solve payload for a given set of tank volumes and thrust to weight constraints. For burns with multiple propulsion types such as a first stage burn with SRBs assisting a liquid rocket core, the sizer calculates a composite specific impulse (ISP) from the mass flow, thrust and quantity of engines. Gravity loss, nozzle pressure loss and aerodynamic drag are approximated by adding an increment to the mission ΔV. Legacy stage mass fractions without engines are used to calculate dry mass without engines and engine dry mass is discretely added using the number of engines and catalogue engine dry mass data. The payload fairing and interstage mass are estimated discretely using calculated or CAD modeled surface areas, payload fairing frontal area, maximum acceleration and legacy areal densities. Using the quick-sizer an investigator can setup and solve an SLS configuration in a few minutes. This has allowed us to assess hundreds of options with consistent and conservative methods. Often we performed these assessments during team web meetings and were able to quickly model anyone's proposed SLS configuration option in real time. We compared quick-sizer results for known launch vehicles including the Saturn V and Delta IV Heavy and the predicted performance for the Lunar Capability Concept Review (LCCR) version of Ares V. Qick-sizer results are typically 5-10% less payload using our conservative thrust to weight constraints and mass fractions. Promising SLS options are much more rigorously analyzed using the FASTPASS full simulation described below. In all cases FASTPASS predicts more payload than our quick-sizer usually within 5-10%. Our sizer has allowed us to explore many more SLS configuration options than we could possibly have accomplished without it and it enabled all study members to participate and have their ideas evaluated. In retrospect we probably could have included a step wise calculation of gravity loss, nozzle pressure loss and aerodynamic drag using numeric methods within MS Excel and could have eliminated most of the conservatism and had better resolution for detail features such as cross feed. For this conceptual level study though our quick-sizer worked very well.

FASTPASS: The Flexible Analysis System for Trajectory Performance of Advanced Space Systems (FASTPASS) computer program was used to model and optimize each promising launch vehicle configuration. FASTPASS is a flexible, multidisciplinary design optimization framework with embedded trajectory optimization similar to the industry standard Program to Optimize Simulated Trajectories (POST). FASTPASS has a simpler and more powerful user interface and incorporates a comprehensive sizing model to enable rapid simultaneous optimization of the vehicle and trajectory. FASTPASS has been used in support of many NASA programs and studies, including but not limited to the Space Launch Initiative, the Space Exploration Initiative, Rocket Based Combined Cycle Single and Two Stage to Orbit studies, National Launch System, Advanced Launch System, Bantam Low Cost Booster, Orbital Space Plane, Shuttle-C, Liquid Rocket Booster, Liquid Fly-Back Booster, X33/Venturestar and Ares V. FASTPASS results have been compared with POST on a number of occasions and found to consistently match within 0.5%

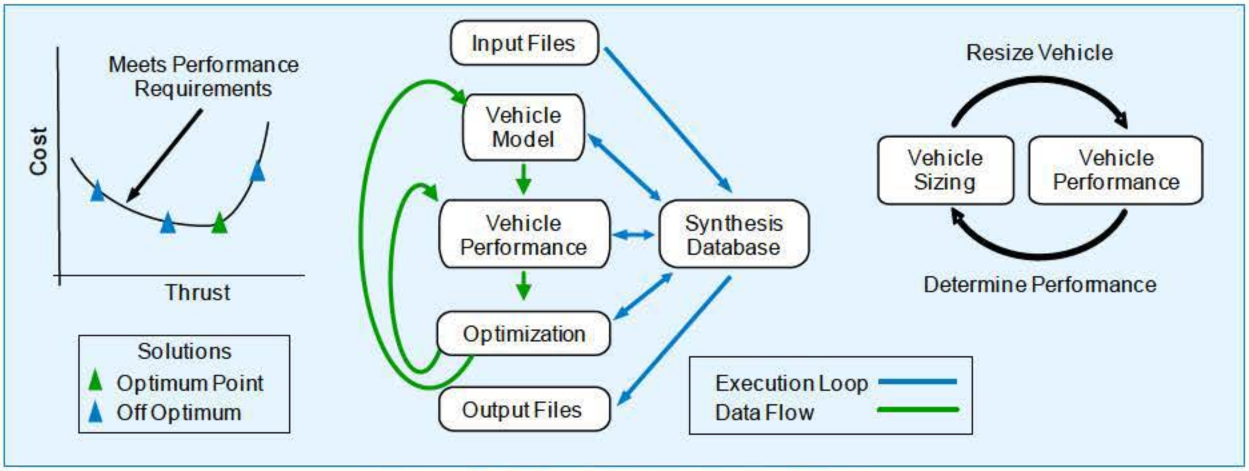


Figure 2-20 FASTPASS Optimization Process

FASTPASS uses a central synthesis database to store information throughout the optimization loop. The vehicle, performance, and optimization modules all communicate through this database. Descriptions of a few of the critical modules and groundrules used for SLS simulations are summarized in the following paragraphs. The FASTPASS program structure is illustrated in Figure 2-20. FASTPASS has an extensive library of liquid and solid propellant propulsions systems, weight estimating relationships, and aerodynamic models that were used to rapidly synthesize new launch vehicle configurations.

Reliability and Safety Modeling Approach

We accumulate the reliability of the launch vehicle using bulk reliability for each subsystem functional block for each of the launch vehicle elements, such as stages (Figure 2-21). The plot shows in blue the estimated failure rates in ppm for each of these subsystems. The reliability product across all these elements results in the estimated vehicle reliability, alternately the PLOM.

Subsystem failure rates are estimated using analysis of prior US launch vehicle experience. We factor these to represent a human rated launch vehicle to achieve nominal launch reliability of 99%. This is to allow comparison of various mission approaches and architectures since we were mainly interested in the differences; however we also wanted to start with something reasonable. The study also evaluates the SLS for PLOC considering its possible role of delivering crew to orbit. We can achieve good crew safety given a 95% abort capability. We incorporated engine catastrophic failure fractions and estimated a much reduced 20% abort reliability in those cases.

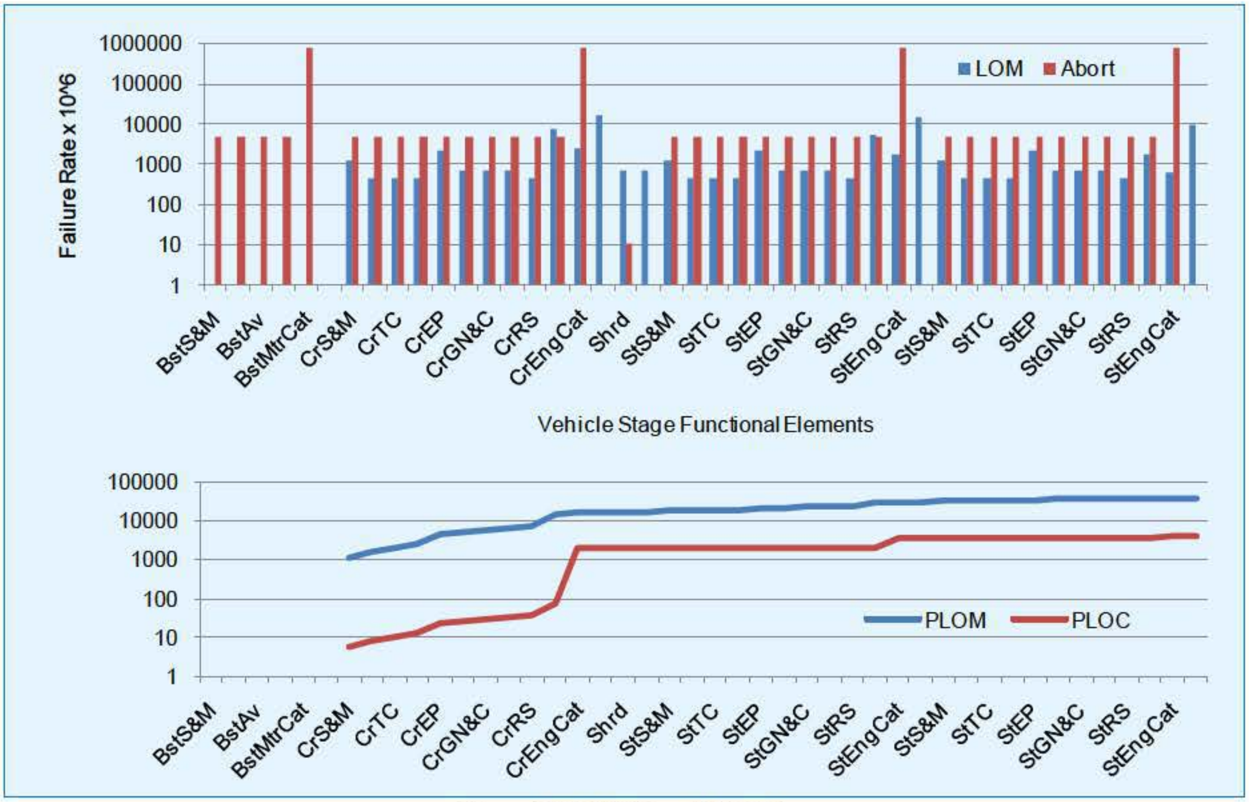


Figure 2-21 PLOC and PLOM

The main thrust of our reliability analysis is to look at the objective of aggregating multiple-vehicle launches of the SLS and loitering on orbit prior to departure. The aggregation can be in the form of component or stage aggregation, or by using depots. The reliability problems for both are similar, but to keep it manageable, this discussion concentrates on depots.

For a depot to be effective in supporting a large vehicle departure to deep space, it must successfully operate in low earth orbit while the propellant is brought up. This could be for a period of one to two years. We have applied understanding gained from Northrop Grumman unmanned satellites, such as Compton, Chandra, and Aqua/Aura that must operate reliability for 5, 10 and 15 years. The same models that are used to design these satellites can be used to show that high, 99% reliability over two years can be expected for an appropriately designed depot, or loitering stage as shown in Figure 2-22.

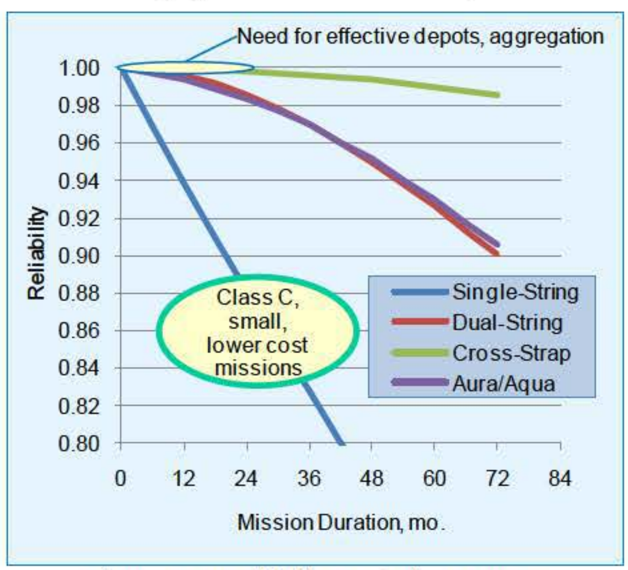


Figure 2-22 Reliability Analysis – Depots

Our reliability models for depots include the high reliability subsystems and include the reliability (probability of no penetration) of tank MMOD shields. These results are derived from detailed modeling that was done on our CEV Service Module design for safe operations at ISS over 60 months total proximity period. (Figure 2-23)

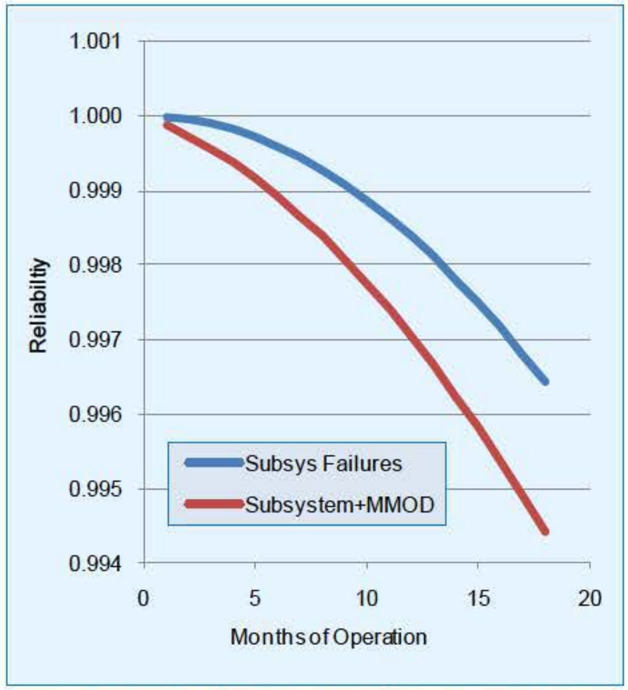


Figure 2-23 Reliability Analysis – Subsystems & MMOD

This analysis of reliability addresses the fundamental problem of how big the SLS vehicle should be. Ideally the launch vehicle would launch the missions to the Asteroids, the Moon or Mars using a single launch. That of course would be too large. However if the mission is reduced to many flights using smaller launches, the probability of successfully executing the launch campaign becomes prohibitively low. The problem is addressed by dividing the vehicles into two classes of payload, critical and non critical (Figures 2-24 and 2-25). Critical payloads need to be launched with high assurance, since their loss would cause a catastrophic loss and end to potentially a multi-billion dollar mission. Non critical payloads have much more replaceable items, such as propellant. Figure 2-26 shows how each of the candidate vehicle configurations perform for our reference missions. It is assumed that without a depot all payloads are critical. A depot, in this case for LOX-only allows the grouping of critical and non critical payloads. These are the corresponding

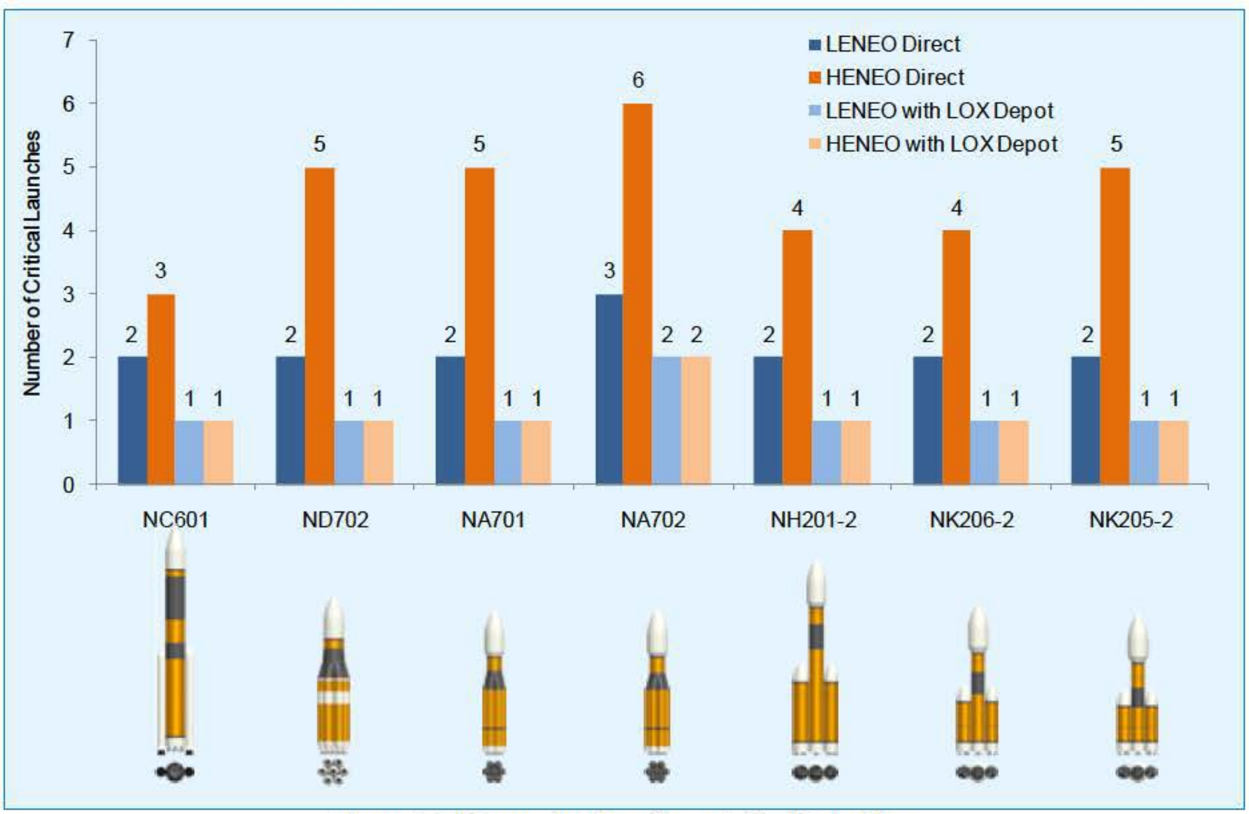


Figure 2-24 Downselect Families - Critical Launches

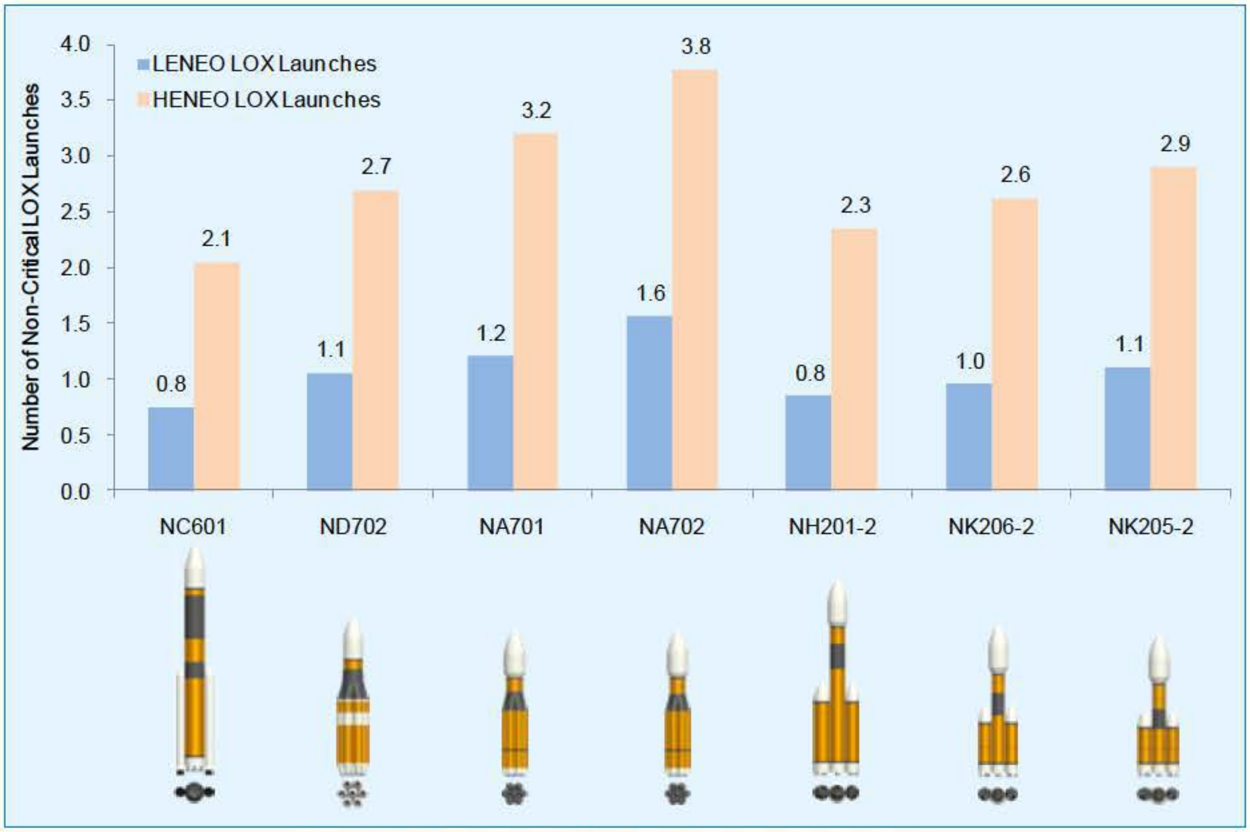


Figure 2-25 Downselect Families - Non-Critical Launches

non critical payloads for the reference missions. As can be seen, since one can arrange to replace the loss of the non critical payloads (a single spare is usually sufficient) there results a dramatic improvement in mission reliability.

Right after TIM1 NGC met with the TMT Safety and Mission Assurance group to review our approach. In summary the approaches we have used and the one used by MSFC are very similar and appear to produce comparable results.

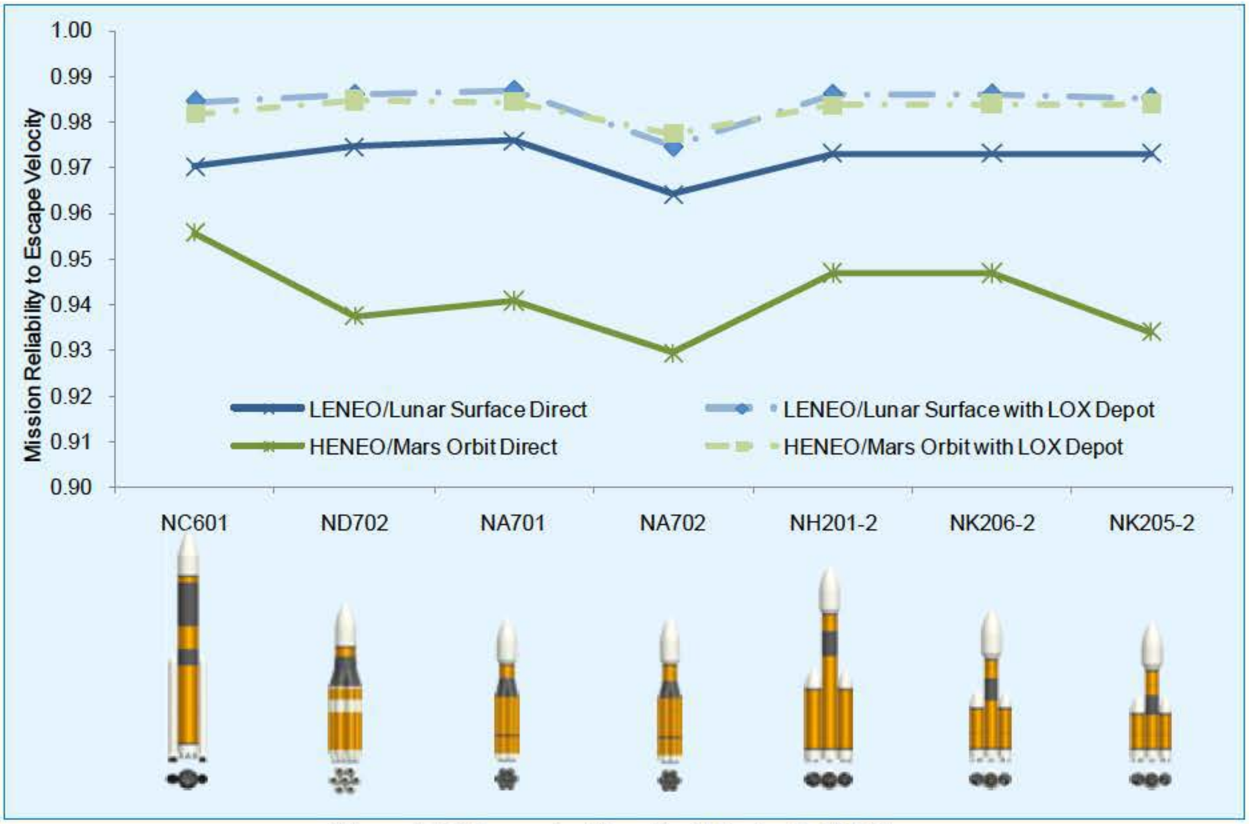


Figure 2-26 Downselect Families Mission Reliability

Life Cycle Cost Evaluation

The updated missions are fed back into the mission manifest (Figure 2-27) for the life cycle costing. The life cycle costs are computed for the candidate vehicles broken down into DDT&E, fixed and variable cost for the 26 years of development and operations and launching 40 SLS supported missions as shown in Figure 3-7. Generally multiple launches are performed for each mission as shown previously in Figure 2-24 and 2-25. The salient result is reduced fixed costs shown for the selected vehicle configurations in comparison with the NC601 with its large fixed production and operations infrastructure. Figure 2-28 presents the breakout according to the recommended configuration component elements.

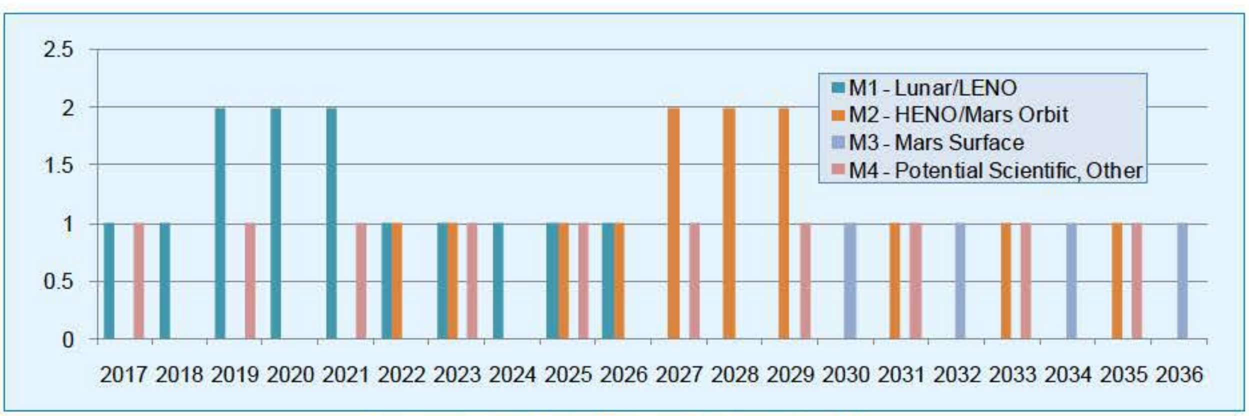


Figure 2-27 Life Cycle Cost Mission Manifest

Cost modeling is based on the following assumptions, input date and ground rules.

Assumptions:

- FY11 Constant Dollars
- Industry development with 20% NASA program support
- 6 year development from 2011 through 2016
- DDT&E program includes a single unmanned demo flight in 2016
- 25 years of operations from 2016 through 2036

Mission Manifest:

- 13 LENEO/Lunar Surface Missions
- 13 HENEO/Mars Orbit Missions
- 4 Mars Surface Missions between
- 10 Science or DoD Missions

Shared Infrastructure where applicable:

- EELV stages and derivatives
- EELV engines (incl. J-2X)
- Common stage designs
- Shuttle infrastructure used for RSRMs

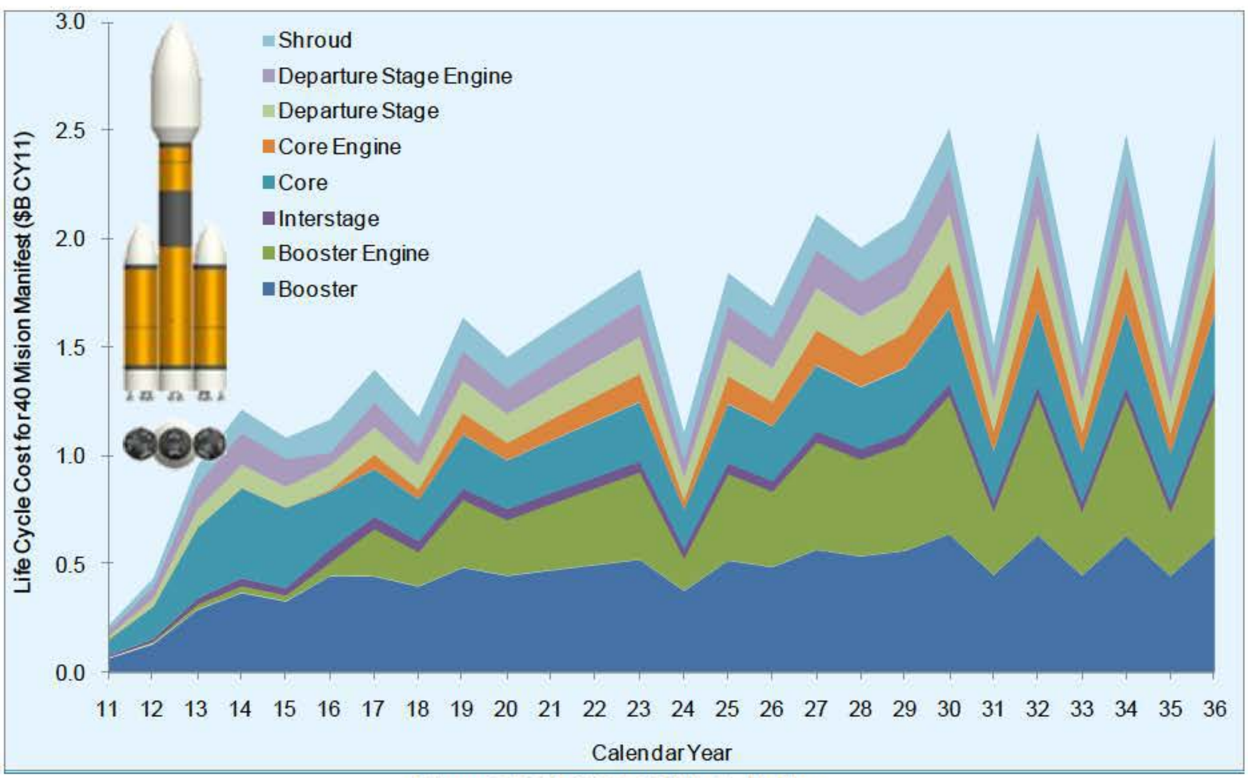


Figure 2-28 NK206-2 Life Cycle Cost

3. SLS Configuration Options for Alternative GR&As *(SOW 3.2, 4.2)*

A key aspect of our Northrop Grumman SLS configuration trade study approach was our ability to consider a wide variety of vehicles in a broad range of vehicle families. We took an unconstrained view, enabling us to remain creative in our concept development and ensure that the SLS is truly sustainable within the expected funding profile. Although a number of traded options and associated vehicles were eliminated almost immediately, we uncovered combinations of very promising features that prompted further analysis and then are used improve the performance of a variety of vehicle configurations. This section documents our trade space evolution from an initial trade set of five configuration families to our down selected set, consisting of three families supporting further, detailed analysis. The study culminates with the Northrop Grumman recommended SLS configuration, NK206-2 described in detail, below. We also summarize our analysis of critical features that apply to a range of SLS elements and warrant further consideration.

3.1. SLS Configuration – Initial Trade Options

Hundreds of configurations were analyzed using our in-house quick-sizer tool (see description of tool in Section 2.3). The ten vehicles shown in Figure 3-1 below represent the five SLS configuration families we selected for detailed analysis and presented at TIM1. Refer to Figure 2-10 for an explanation of the naming convention. NC601, our Ares V-like reference vehicle, is characteristic of Constellation-derived options with a constant 10m diameter. Tank layouts for NC601 and all other initial trade vehicles are illustrated in Figure 3-2.

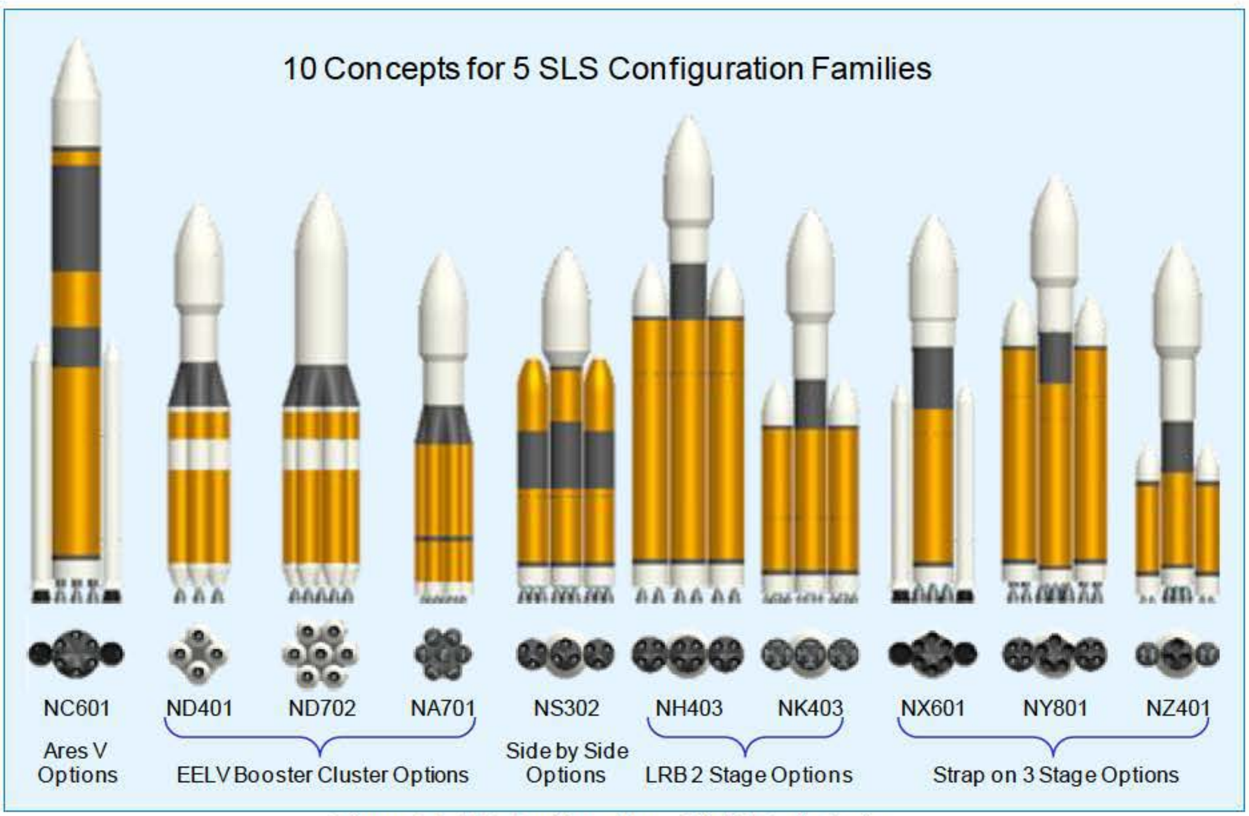


Figure 3-1 SLS Configuration – Initial Trade Options

Our novel composite clustered interstage concept described in Section 3.9 is a key feature of the vehicles shown in the EELV Booster Cluster family. These vehicles take advantage of existing EELV booster core designs and production infrastructure and tend to reduce vehicle mass and stack height. ND401 is a cluster of four Delta IV Common Booster Cores (CBCs) while ND702 incorporates six around a central CBC for a total of seven. The natural geometry of six equally-sized circles spaced evenly around a central seventh optimizes the launch pad footprint. NA701 is similar to ND702 but utilizes seven Atlas V Common Core Boosters (CCBs). The primary launch loads are uniformly distributed through the clustered interstage in all of the cluster configurations, so the booster forward mounts are unchanged from the current design. A key affordability feature of this family is that few modifications are required to existing EELV boosters. The clustered interstage is the enabling feature.

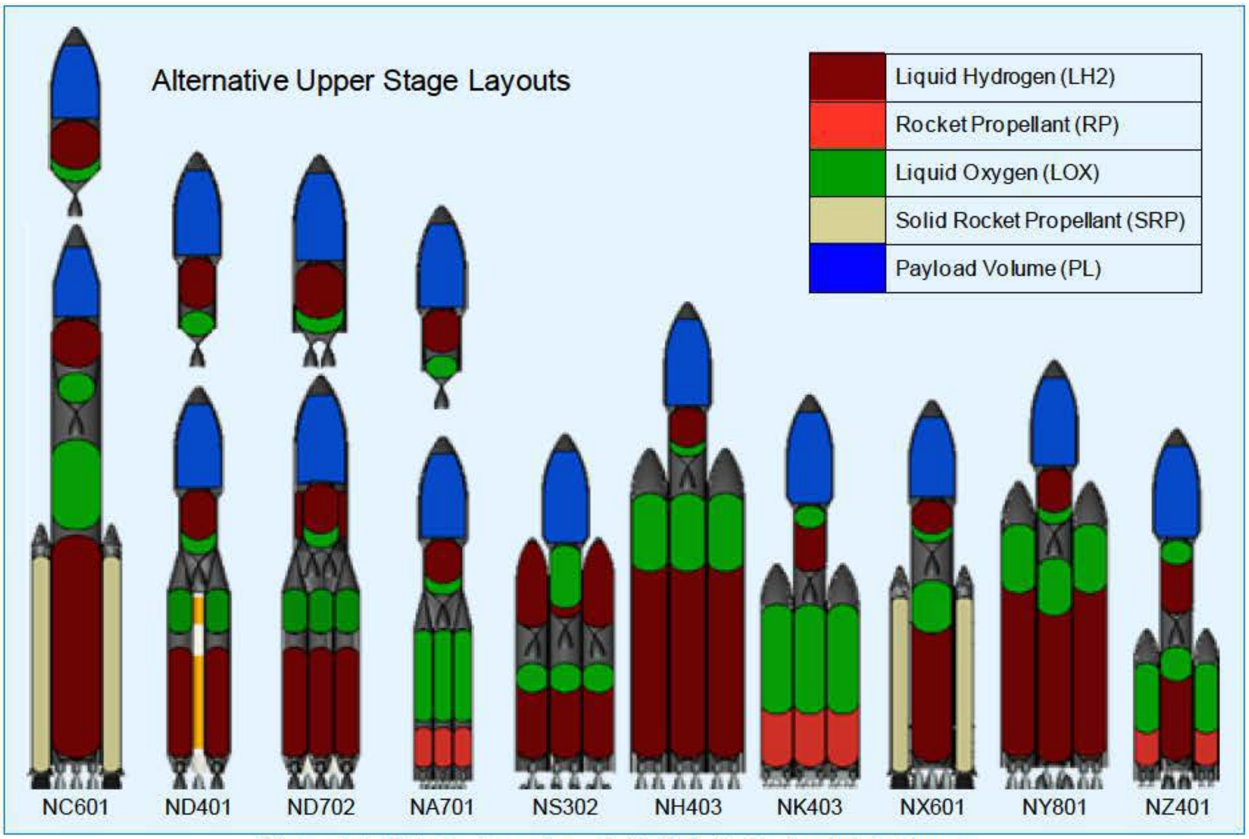


Figure 3-2 SLS Configuration – Initial Trade Options Tank Layouts

NS302 represents a family of vehicles that feature common diameter side-by-side core and upper stages with all LOX/LH2 propulsion. Three common core segments are staged simultaneously. The upper stage side-mounted drop units carry ascent LH2 and are jettisoned in LEO. The upper stage center unit carries all upper stage LOX and only the LH2 required for the escape burn. Affordability drivers include a common reduced diameter throughout the vehicle, common major unit production in a reduced footprint infrastructure and a steady state line of balance at low production rates.

The fourth family includes two-stage vehicles with liquid rocket boosters (LRBs). NH403 is an all LOX/LH2 vehicle with common 7.5m diameter LRBs, core and upper stage. NK403 features common 6.5m diameter elements, but utilizes LOX/RP LRB and core propulsion. In this family, the core stage and LRBs all fire on liftoff. Similar to the side-by-side concepts, this family benefits from common diameter elements throughout, common major unit production in a reduced footprint and a steady-state line of balance at low rates, yet designed into a more traditional LRB two-stage configuration.

The final trade options family consists of three strap-on three-stage vehicles with varying core stages and LOX/LH2 second and upper stages. Although they may appear similar to other vehicles in our trade space, the solid rocket booster, LOX/LH2 and LOX/RP core stages (NX601, NY801 and NZ401, respectively) contain the only engines that fire at liftoff. What looks like a central core stage is actually the second stage in the strap-on 3-stage configurations. All vehicles are shown in Figure 3-1 and Figure 3-2 with upper stages encapsulated by extended payload fairings. This is a critical feature for dedicated in-space vehicle architectures (ref Section 4.7) but not a requirement for a multi-purpose upper stage.

Performance for the ten initial trade vehicles is shown below in Figure 3-3 in the form of payload to escape velocity. The payload capability of our trade set ranged from 21 (ND401) to 74 (NH403) metric tons direct to escape velocity. As expected, all vehicles significantly benefit from the availability of a propellant depot (reference Section 2). With LOX only refueling, the smallest vehicle in our trade space, ND401 performance increases from 21 to 47 metric tons, while the largest, NH403 increases from 74 to 148 metric tons to escape velocity. All vehicles at least double their payload performance with the addition of a LOX only propellant depot, with the exception of NY801 at 186%. The Constellation-derived NC601 vehicle witnesses the greatest performance increase at 232%. Payload to escape velocity is one metric we used to compare our initial trade vehicles prior to our configuration downselect. For all others, reference Section 2.

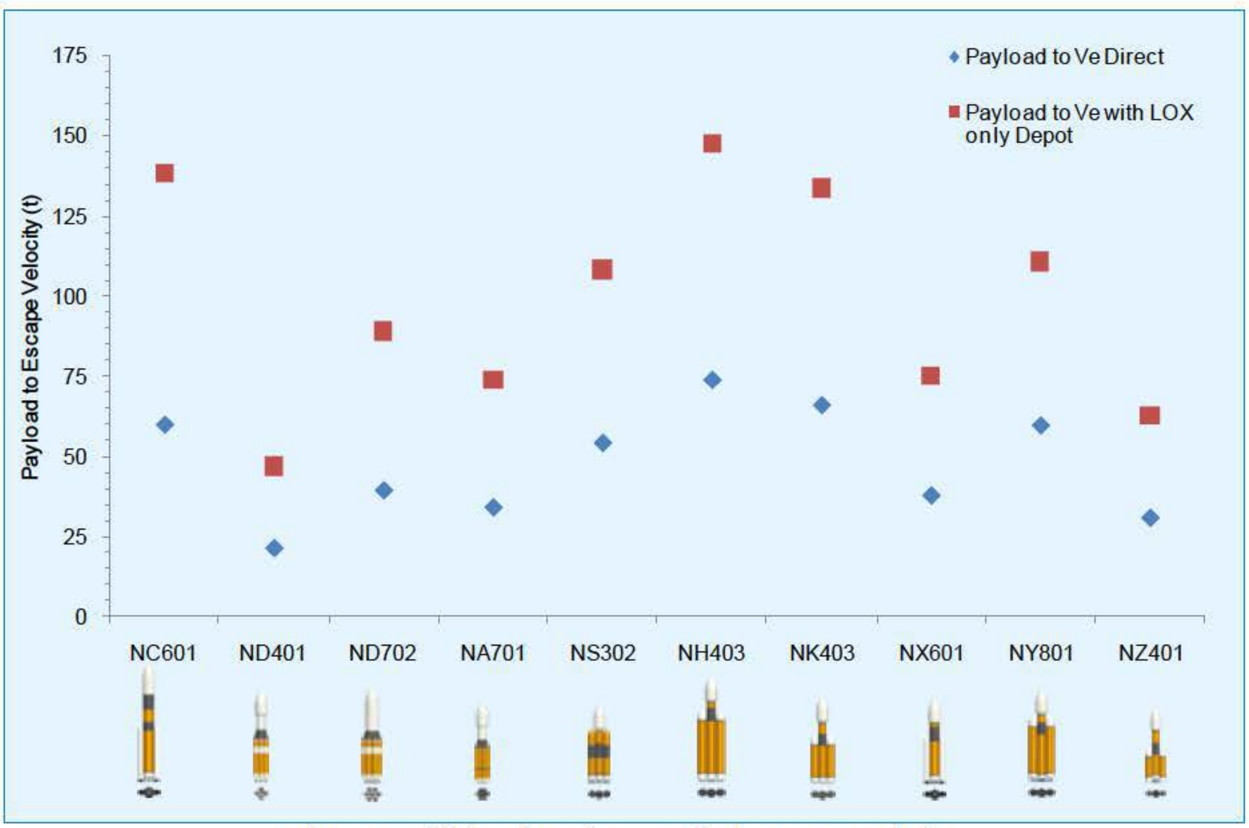


Figure 3-3 Initial Trade Options – Payload to Escape Velocity

3.2. SLS Configuration Downselect

After TIM1, we conducted a configuration downselect resulting in the set of vehicle families shown in Figure 3-4. Whereas the TIM1 initial configurations were representative of a wide trade space within each family, these particular vehicles are the result of our first level of detailed analysis. All TIM1 options outscored our Constellation-derived NC601 vehicle (reference Figure 2-15) but, as our baseline, we continued to trade our downselected vehicles against it. The highest scoring families contained the EELV Booster Cluster and the LRB 2-Stage vehicles. We eliminated the Side-by-Side family. The NS302 representative vehicle was too complex for its limited staging benefit. All three-Stage options were eliminated because we determined that cross feed can provide the same benefits in two-stage LRB configurations at lower cost (reference Section 3.8). We replaced ND401 with NA702 within the EELV Booster Cluster family and added NK205-2 based on promising results from our all-RP commonality study described in Section 3.7. ND401 is a nice compact vehicle but it was just too small with many critical launches required per mission. We also replaced NH403 and NK403 with NH201-2 and NK206-2 respectively to reduce the number of total engines and take advantage of the natural staging benefit of three-engine LRBs combined with a two-engine core.

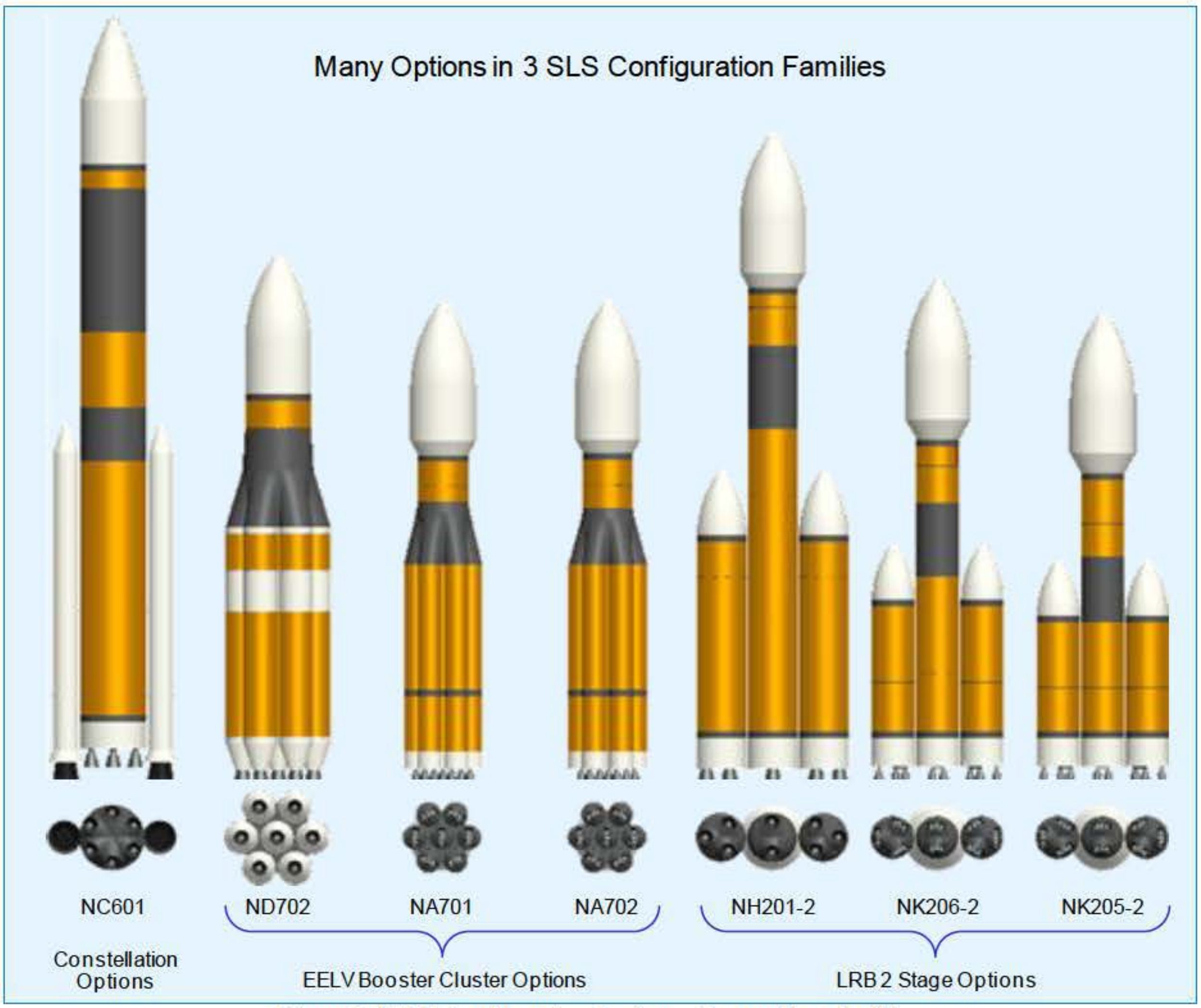


Figure 3-4 SLS Configuration Options – Downselect Families

Our downselected vehicle tank layouts are shown below in Figure 3-5. The added vehicles, NA702 and NK205-2 take advantage of common propulsion throughout. Similar to NA701, NA702 features seven Atlas V CCBs but has a single RD-180 upper stage. NK205-2 has two LRBs with three RD-180s each, a central core stage with two RD-180s and a single RD-180 upper stage. NH201-2 utilizes two LRBs, each with three RS-68C engines, a core stage with two RS-68Cs and a single J-2X upper stage. Its TIM1 predecessor, NH403, was a common core/LRB configuration with four RS-68B engines each. NK206-2 replaced NK403 and again reduced the number of engines moving from a common core/LRB configuration with a total of twelve RD-180s to its design, sized with two LRBs with three RD-180s each, a central core stage with two RD-180s and a single J-2X upper stage. Alternative upper stage layouts are also included to demonstrate the flexibility that still remains within the downselected configurations.

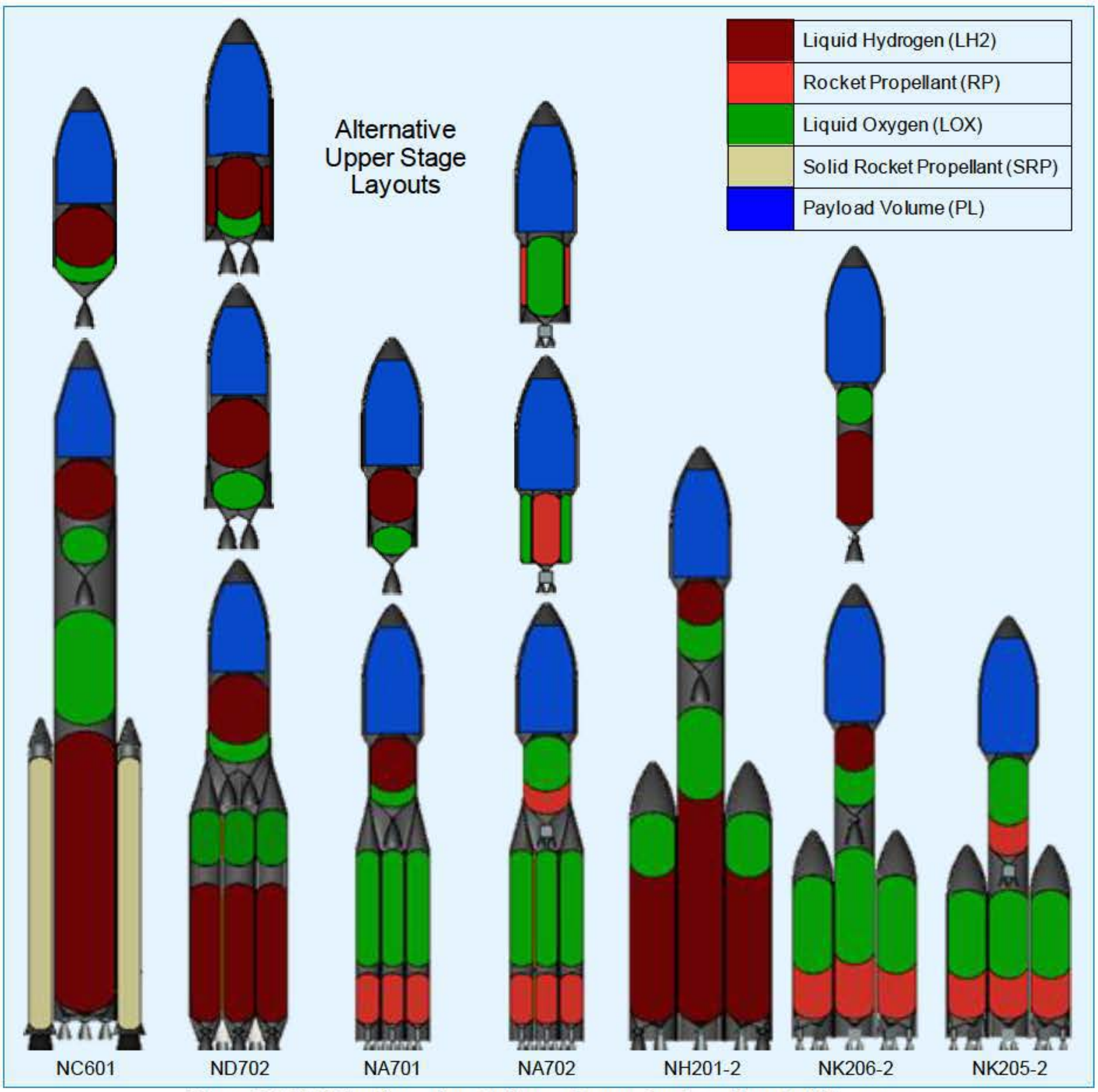


Figure 3-5 SLS Configuration Options – Downselect Families Tank Layouts

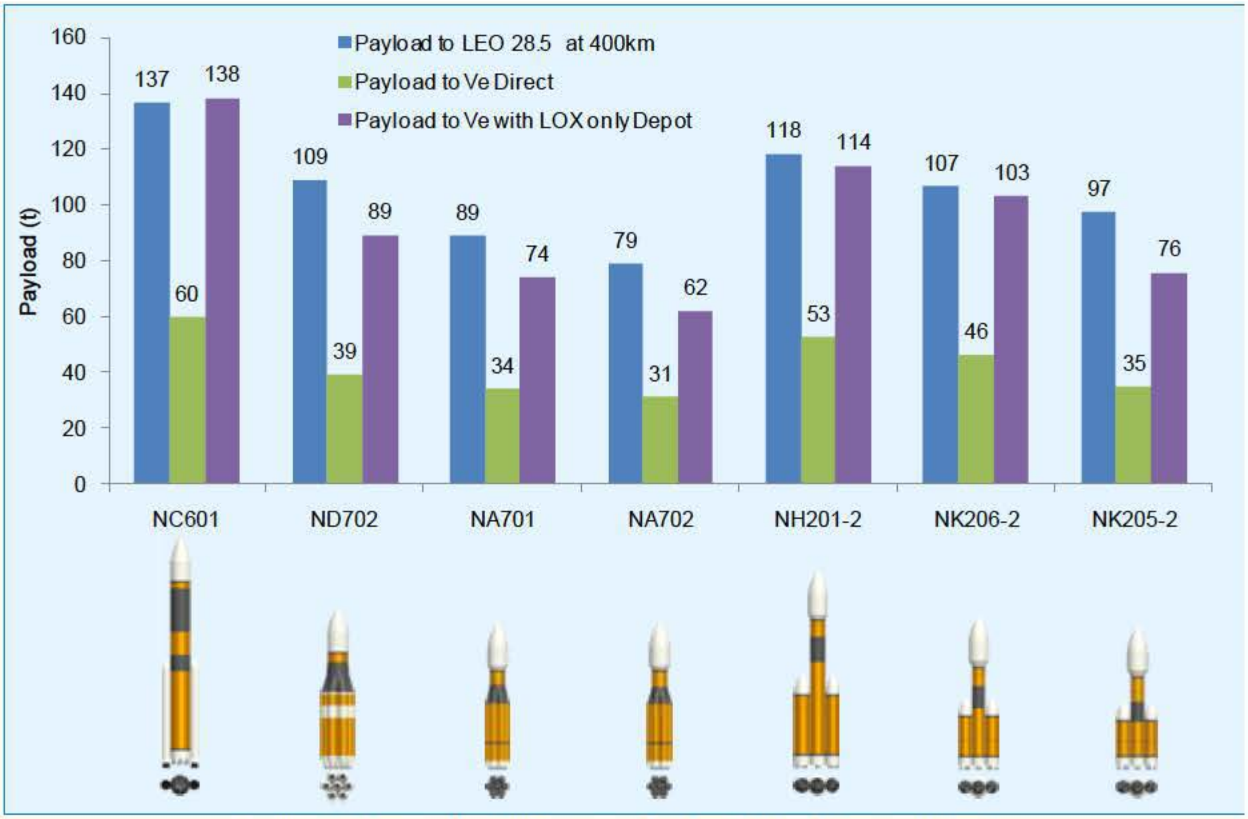


Figure 3-6 Downselect Families – Payload to LEO and Escape Velocity

We compared the performance of our downselected vehicles as shown in Figure 3-6, again in terms of payload to escape velocity. We also included the fallout LEO payload (28.5° at 400km) for reference. On average, the LRB 2-Stage vehicles outperformed the EELV Booster Cluster options but fell short of the NC601 baseline. In terms of affordability, the Life Cycle Cost (LCC) analysis in Figure 3-7 designated NK206-2 as the lowest cost option while still placing 107 metric tons to LEO. Much of the LCC cost benefit of the cluster and LRB 2-Stage vehicles is rooted in fixed cost stemming from the reduced infrastructure required to produce and operate each configuration, which we discuss further in Sections 3.5 and 3.6. Reduced infrastructures are largely the result of the application of smaller, multiple core and booster elements which reduces the factory and transportation "footprint" and multiplies the opportunism for commonality of structural elements and subsystems and introduce opportunities for more efficient, parallel line production systems. More commonality ripples also into reduced need for large support teams with large numbers of specialized groups to handle diverse vehicle components. The reference NC601 has the largest fixed costs resulting from its use of in-place, large infrastructures, such as for the RSRMs and the ET-derived vehicle core stage.

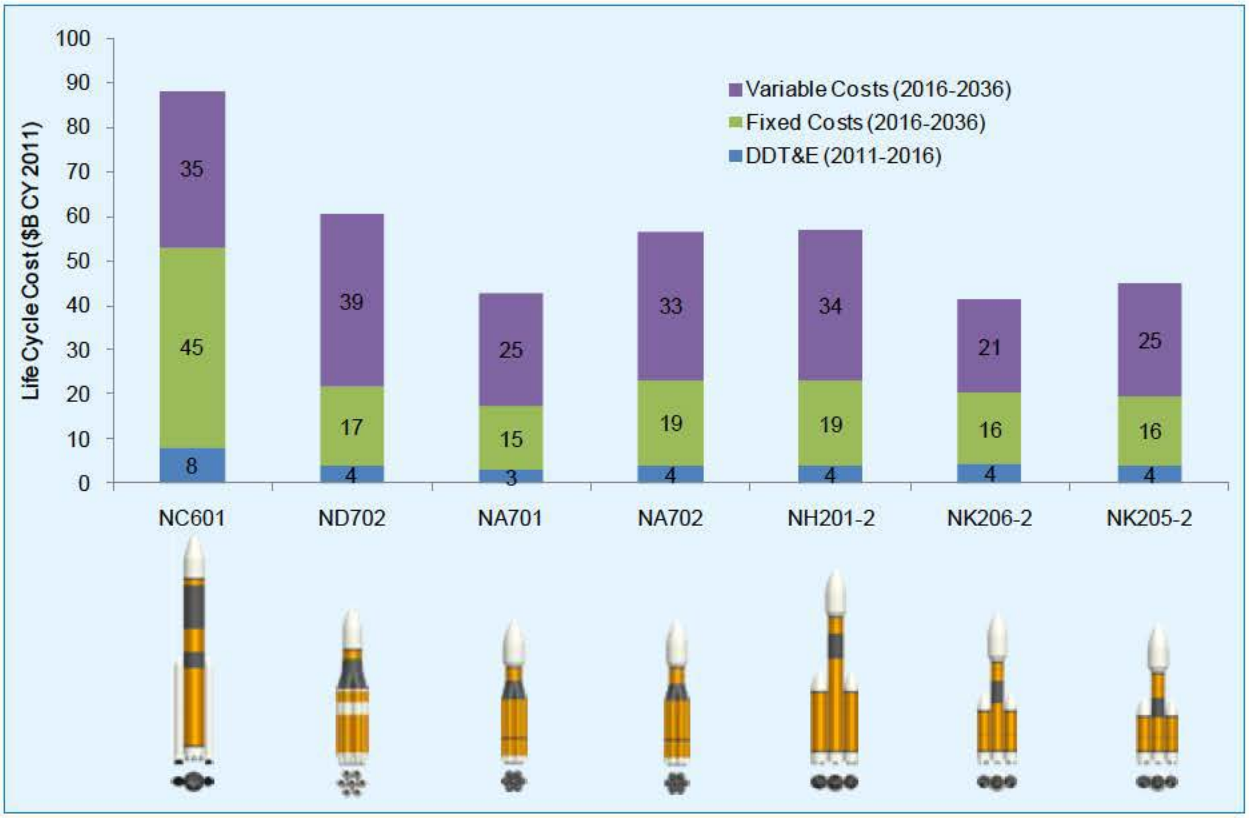


Figure 3-7 Downselect Families – Life Cycle Cost

3.3. Concept Description Packages

We created PowerPoint concept description packages (CDPs) for every interesting configuration identified and analyzed throughout our SLS trade study. These stand-alone packages summarize relevant descriptive data outlining each configuration and include a vehicle description pointing out affordability drivers, alternative configuration options, vehicle and tank layouts, concept of operations with FASTPASS mission timelines, and a look into the required production infrastructure in terms of major manufacturing units. Major manufacturing units are defined as shell elements and major engine procurements. These CDPs were key in transferring configuration data to the Technical Management Team immediately as they became available.

Figure 3-8 is a summary of the data contained in the ND702 CDP. This particular vehicle, ND702 features a cluster of seven Delta IV Common Booster Cores (CBCs) and distributes launch loads through our novel clustered interstage. The upper stage is sized for two J-2X engines and contains a tapered intertank between the LH2 tanks and a reduced major diameter LOX tank. Encapsulating the upper stage and/or reducing the mass/stack height by utilizing a common bulkhead approach are two noted configuration options. Affordability drivers include the utilization of existing CBC infrastructure, clustered interstage thrust shells that are manufactured in a reduced production footprint, and an overall reduced mass. ND702 requires seven off-the-shelf Delta IV CBCs and two J-2X engines. Six interstage thrust shells are built using our In-situ Manufacturing System (IsMS). The payload fairing barrel and ogive segments, aft skirt frustum and the tapered intertank are all built in panels to minimize the production footprint on the same IsMS using unique mandrel tooling. The upper stage thrust cone, the payload adapter fitting, two upper stage LH2 half tanks, two upper stage LOX half tanks and six interstage fillets are built vertically on unique tools. A snapshot of the ND702 FASTPASS mission timeline is also presented in Figure 3-8.

(b) (3)

[redacted]

Figure 3-8 ND702 Concept Description Package Summary

3.4. FASTPASS Trajectory Analysis

FASTPASS was utilized throughout this trade study to validate our quick-sizer results and provide optimized sizing information for promising vehicle configurations. One output from FASTPASS that contributed to our downselect decisions was a set of trajectory plots summarized below in Figure 3-9. The plots chart the following data, all as a function of the mission timeline:

- Thrust to Weight
- Dynamic Pressure, q (KPa) and Acceleration (g's)
- Relative and Inertial Velocity (m/sec)
- Altitude (km) and Acceleration (g's)
- Angle of Attack (deg) and q-alpha (KPa×deg)
- Heating Rate (KWm^2/sec) with Indicated Fairing Jettison

(b) (3)

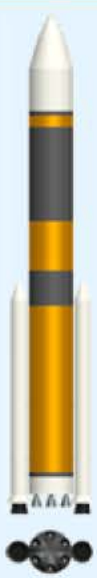
[redacted]

3.5. Production and Operations Plans

We developed production and operations plans for the five families presented at TIM1. In this section we present production and operations plans for a representative vehicle from each downselected family. These vehicles include NC601, our Constellation-derived baseline, NA702, a cluster of seven Atlas V CCBs, and NH201-2, the all LOX/LH2 vehicle from the LRB 2-Stage family. All composite tanks and dry structure are assumed in this analysis. Although these plans mention and picture specific vehicles, they apply to any configuration within that vehicle family. The example vehicles are compared in terms of major manufacturing units and their associated production footprint, and the required operations infrastructure. Note that the structural mass fractions used in the performance calculations assumed legacy metallic structure weights. Use of composites would reduce dry and tank structures by from 10% to 30%, indicating the conservative nature of our performance analysis. This approach leaves open the option to build low risk, high margin composite structures to further reduce costs.

Baseline Vehicle NC601 – Constellation Derived Family

The major manufacturing units and tooling required to produce NC601 are shown in Figure 3-10 below. Two fabrication stations and five unique tooling mandrels are required to build fifteen composite shells. The first station is a 10m-class IsMS with two unique tooling mandrels. Mandrel #1 is a cylindrical barrel section with a domed end used for fabricating half tanks and cylindrical dry structure. Since the infrastructure is already in place to produce 10m diameter structures, the payload fairing is built as one unitized structure on Mandrel #2 using tooling plugs along the future longitudinal splice joint locations. The fairing is trimmed into petals, the tooling plugs are removed and the separation system is installed. The second station is a Vertical Laminating Station (VLS). Shells with aspect ratios not conducive to our cantilevered IsMS approach are fabricated vertically using a variety of traditional laminating approaches like automated fiber placement or hand lay-up, depending on the design. Unique Mandrels #3, #4 and #5 are required for the core stage thrust shell, the upper stage thrust cone and the payload adapter fitting, respectively. The infrastructure required to produce the 5-segment solid rocket boosters



	NC601 MMUs	
5-Segment SRB	PF Biconic	5-Segment SRB
	PAF	
	US LH2 Shell A	
	US LH2 Shell B	
	US LOX Shell	
	US Thrust Cone	
	1X J-2X	
	Interstage	
	CS LOX Shell A	
	CS LOX Shell B	
	Intertank	
	CS LH2 Shell A	
	CS LH2 Shell B	
	CS LH2 Shell C	
	CS Thrust Cone	
	CS Boattail	
	6X RS-68	

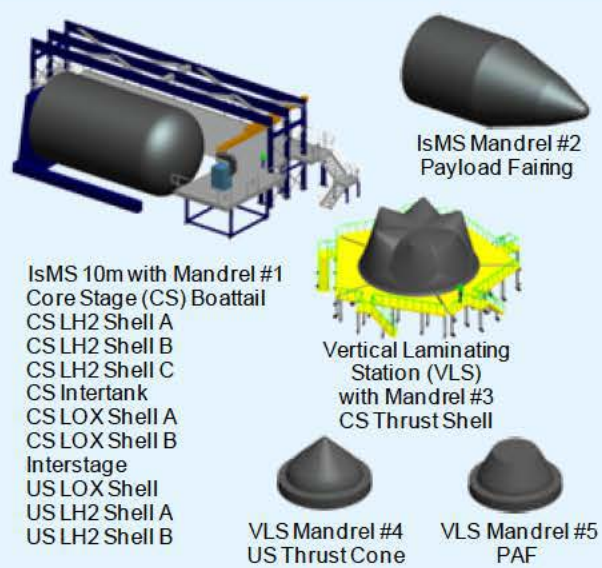


Figure 3-10 NC601 Major Manufacturing Units and Tooling

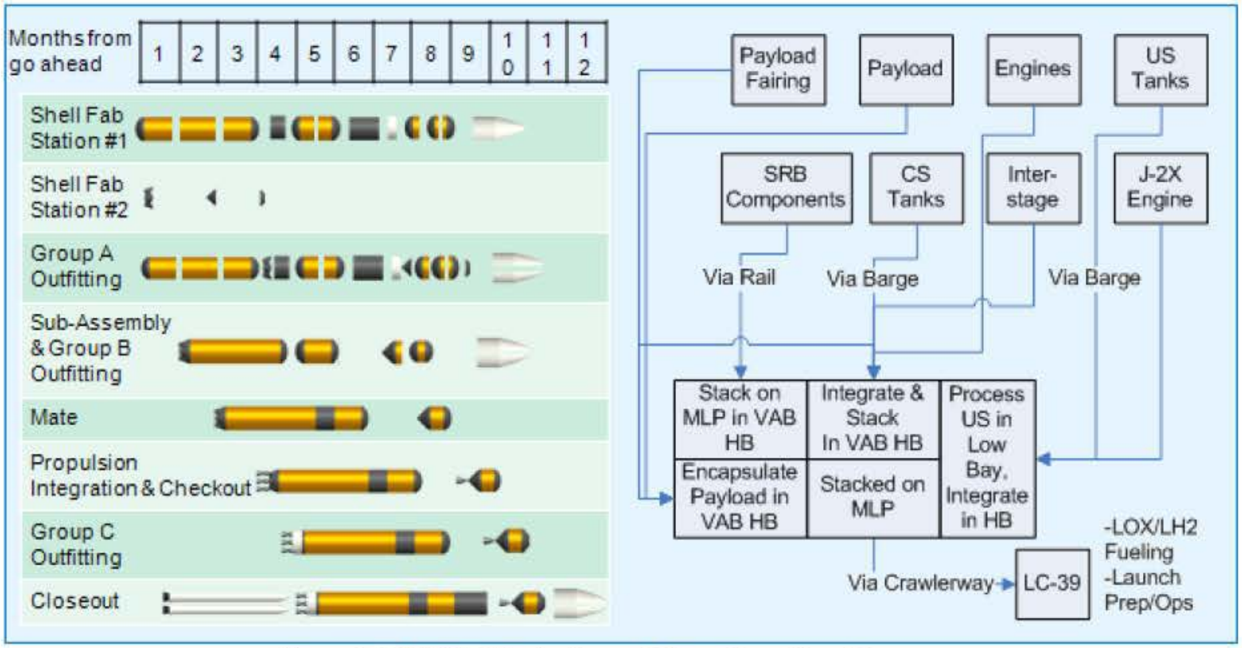


Figure 3-11 NC601 Production and Operations Flow Diagrams

and engines are not shown, but are considered in the summary analysis ahead in Figure 3-16.

With this approach we are able to fabricate the entire vehicle within a 12-month production interval as shown in Figure 3-11. A physics-based lamination model developed on an independent effort grounds our individual shell fabrication durations. We outfit as many subsystems and components while the tanks are in half segments in Group A. We join the tanks circumferentially prior to stage mate. Propulsion system integration and checkout follows and we bring in the boat tail and payload adapter fitting in Group C. The closeout row illustrates the major deliverables including the solid rocket boosters.

The NC601 operations flow diagram is also presented in Figure 3-11. Major elements are brought via rail or barge to a modified Vehicle Assembly Building (VAB) for vertical integration and processing. A fully integrated vehicle is transported to the launch pad using a new crawler and Mobile Launch Platform. This approach provides flexibility for an increased launch rate or adding vehicle elements to the stack due to the number of dedicated facilities and vertical transportation to the pad. Fixed costs are high with this approach, which was a major driver in our life cycle cost analysis.

Example Vehicle NA702 – EELV Booster Cluster Family

The major manufacturing units and tooling required to produce NA702 are shown in Figure 3-12. Three fabrication stations and eight unique tooling mandrels are required to build twenty-one composite shells. In comparison to the baseline NC601, the higher number of shells is due to the individual shell quantity required to build such a large payload fairing on reduced diameter infrastructure as described in Section 3.8. The individual thrust shells that eventually form the clustered interstage also add to the total. This does not account for the seven Atlas V CCBs as they are considered major procurements. This production infrastructure is much smaller than that of NC601. The first fabrication station is a 7.6m-class IsMS with three unique tooling mandrels. Mandrel #1 is a cylindrical barrel section with a domed end used for fabricating the upper stage half tanks. Mandrels #2 and #3 are used for payload fairing segments. Station #2 is a smaller

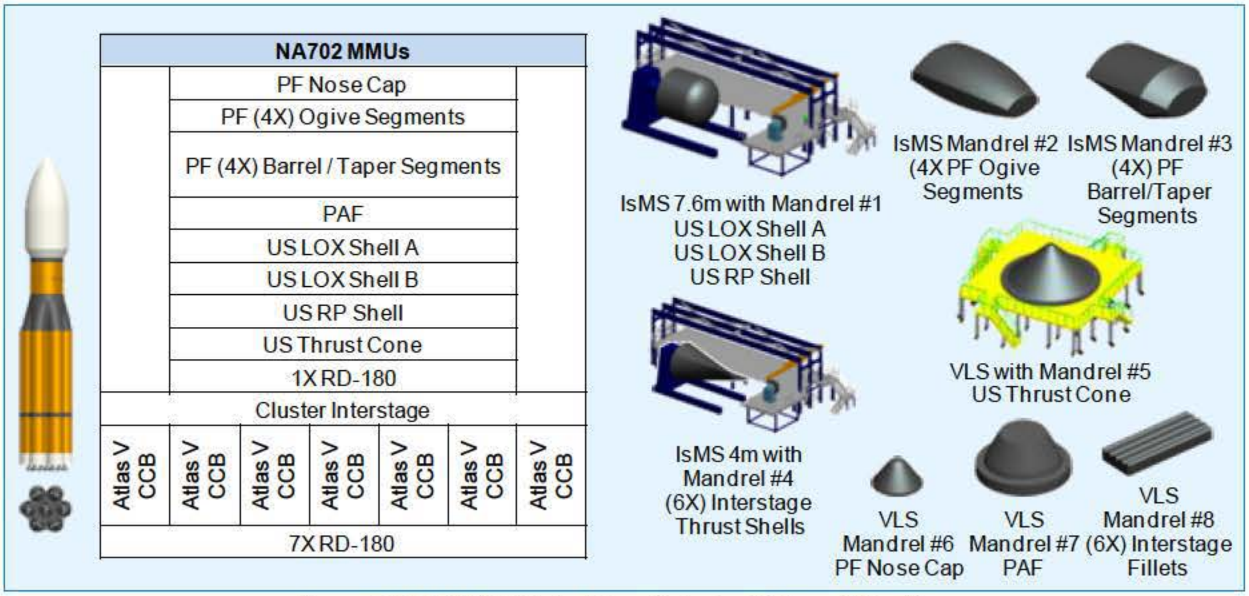


Figure 3-12 NA702 Major Manufacturing Units and Tooling

IsMS in the 4m range that is capable of building the six thrust shells. Smaller, but similar to NC601, a VLS is required with four unique mandrels to build the noted components.

Figure 3-13 shows the production plan and operations flow diagram for NA702. The number of shells per fabrication station for this particular vehicle is very low which leads to a production interval well within one year. Fundamentally, sub-assembly and group outfiting process are similar to NC601 although on a much smaller scale. The clustered interstage adds a significant sub-assembly and Group B outfitting task that can begin as soon as the first thrust shell leaves Fabrication Station #2.

CCBs and other components arrive by air or barge and are processed horizontally, eliminating

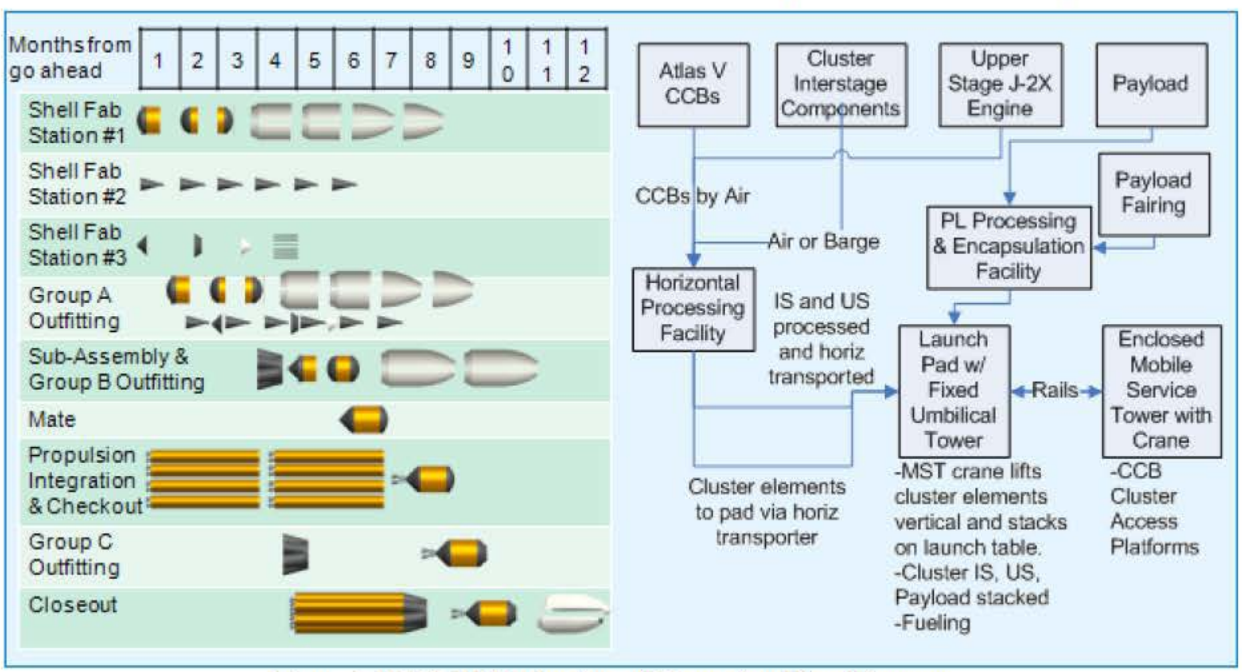


Figure 3-13 NA702 Production and Operations Flow Diagrams

any need for the VAB. Processed elements are moved to the launch pad using existing or modified CCB transportation equipment. On-pad integration is enabled by the relatively small individual elements and the overall reduced stack height. A mobile service tower with a reasonable crane is used to lift and stage the elements. This extra time on the launch pad is well-suited for the expected low flight rates and helps minimize the required infrastructure, thus reducing fixed cost. A rigid composite shell, the clustered interstage will even serve as the CCB forward alignment tooling fixture in this low cost operations approach.

Example Vehicle NH201-2 – LRB Two-Stage Family

The major manufacturing units and tooling required to produce NH201-2 are shown in Figure 3-14. Two fabrication stations and eight unique tooling mandrels are required to build thirty-five shell elements. The total number of NH201-2 shells is significantly higher than the previously discussed vehicle examples but they are all built in a common manufacturing footprint. Solid

NH201 MMUs		
	PF Nose Cap	
	PF (4X) Ogive Segments	
	PF (4X) Barrel / Taper Segments	
	PAF	
	US LH2 Shell A	
	US LH2 Shell B	
	US LOX Shell	
	US Thrust Cone	
	1X J-2X	
LRB Fairing #1	Interstage	LRB Fairing #2
LRB LOX Shell A#1	CS LOX Shell A	LRB LOX Shell A#2
LRB LOX Shell B#1	CS LOX Shell B	LRB LOX Shell B#2
LRB LH2 Shell A#1	CS LH2 Shell A	LRB LH2 Shell A#2
LRB LH2 Shell B#1	CS LH2 Shell B	LRB LH2 Shell B#2
LRB Thrust Cone #1	CS Thrust Cone	LRB Thrust Cone #2
LRB Boattail #1	CS Boattail	LRB Boattail #2
3X RS-68	2X RS-68	3X RS-68

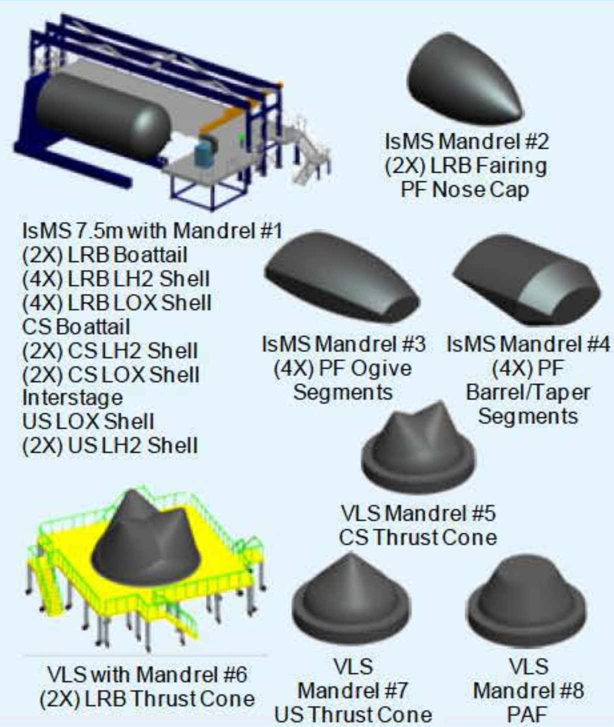


Figure 3-14 NH201-2 Major Manufacturing Units and Tooling

rocket or EELV booster infrastructure is not required. Fabrication Station #1 is similar to that of NA702 and falls into the 7.5m class. Its Mandrel #1 barrel section is extended to accommodate longer half tanks. IsMS Mandrel #2 is unique to this family of vehicles. As presented in Figure 3-12, NA702 requires a unique vertical tool for the nose cap. As discussed in Section 3.7, Mandrels #3 and #4 are required to build the 10m diameter payload fairing in this reduced diameter infrastructure. Finally, a VLS is required with four unique tooling mandrels.

NH201-2 will push Fabrication Station #1 to its maximum capacity as shown in Figure 3-15. It is possible to fabricate this relatively high number of shells and close out the major NH201-2 deliverables within a one year production interval, but only once the production line reaches steady state. Any production rate increase can be accommodated by adding a third IsMS 7.5m fabrication station while still maintaining a small footprint. This vehicle also requires increased

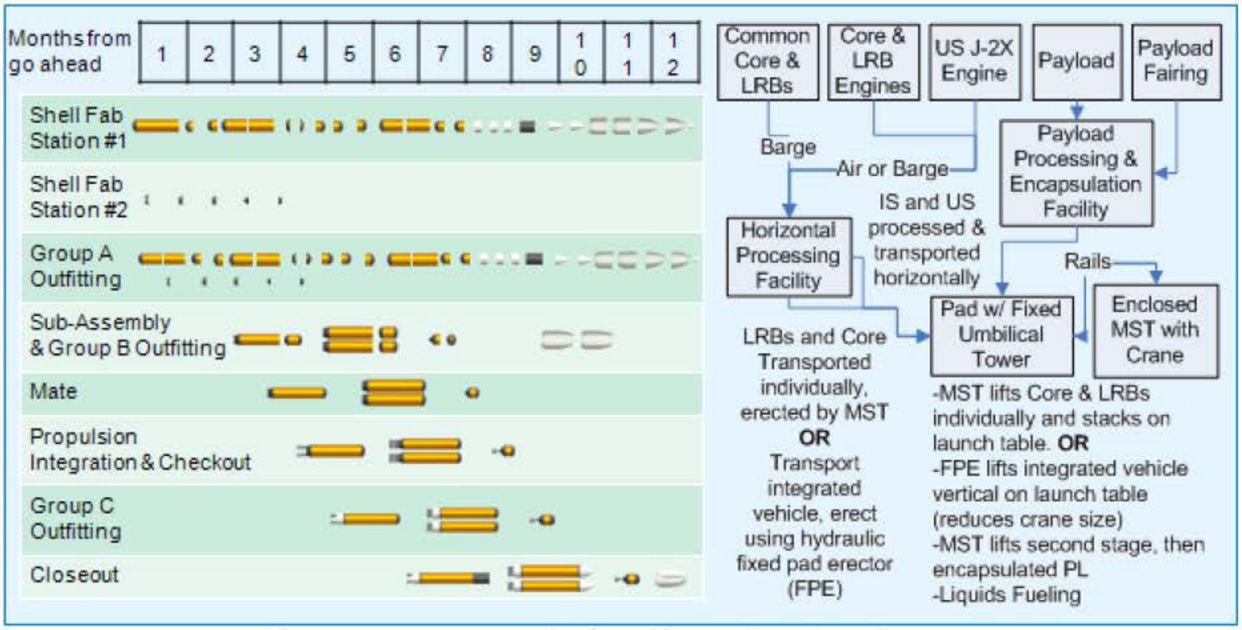


Figure 3-15 NH201-2 Production and Operations Flow Diagrams

outfitting and sub-assembly stations as compared to NC601 and NA702 due to the common core stage and booster production plan.

The Operations Flow Diagram is shown in Figure 3-15. The 7.5m core and LRBs can be barged to the processing facility similar to the Shuttle External Tank. Engines and payload fairing petals arrive to the appropriate facility via air or barge. NH201-2 is processed horizontally in either a dedicated or shared horizontal processing facility. This particular family offers flexibility when it comes to vehicle integration. One option is to transport the core and LRB segments individually and vertically integrate on the pad using a mobile service tower and crane. Alternatively, the core and LRBs are integrated horizontally, transported to the pad as a single unit and lifted to vertical using a fixed pad erector similar to Delta IV Heavy. Either way, the encapsulated payload is staged once the vehicle is vertical using the mobile service tower and crane. The operations costs are considered moderate for this family of vehicles. Although the VAB is again not required, there are increased on-pad facilities as compared to NA702.

Production and Operations Relative Rating

Relative qualitative analysis of NA702 and NH201-2 to the baseline NC601 vehicle is presented in Figure 3-16. Again, these comparisons are representative of the vehicle families and are not limited to the example vehicles. After summarizing critical features of the production and operations plans a comparison was made noting whether each example vehicle in the given category was clearly better (green), about the same (yellow) or clearly worse (red) than the Constellation-derived reference vehicle NC601.Both vehicle families scored better than the baseline vehicle in three of the four categories. Pad operations was the only category where the large Ares V-like vehicle competed with the smaller vehicles. In this case the dedicated processing and integration facilities decrease the total amount of activity performed at the pad, but the reduced stack heights and similar processes to current EELVs earned NA702 and NH201-2 a yellow rating. In the Fabrication, Assembly and Integration, and Transportation categories both the EELV Cluster family and the LRB two-stage family options scored clearly better. There were not any instances where the example vehicles scored clearly worse than the NC601 reference vehicle.

	Fabrication	Assembly & Integration	Transportation	Pad Operations
Constellation Options 	• 10m IsMS • Low # of Large MMUs • MMU Repair / Remake – High Schedule Impact • High Hook Height, Wide Aisle Facility • Maintain SRB Infrastructure	• Vertical Processing & Integration • Modifications to VAB • Existing RPSF for SRBs • US Processing Unique From CS • 7 Engines / 2 SRBs	• New 10m Fab to Integ Transport Required • Transport Integrated Vehicle Vertically to Pad is High Risk • New Crawler • New Mobile Launch Platform (MLP)	• Transfer – Launch (staging complete upon arrival) • Modifications to Support New Vehicle • LOX & LH2 Fluid Systems
EELV Cluster Options 	■ Clearly Better • 7.6m IsMS • 4m IsMS • Medium # of Small MMUs • MMU Repair / Remake – Low Schedule Impact • Utilize Shared CCB Infrastructure	■ Clearly Better • Horizontal Processing of All Elements in Shared Use or Modified Facility • No VAB or RPSF Requirement • US Processing Unique From CS • 8 Engines (7 CCBs)	■ Clearly Better • Existing Fab to Integ Transport Methods • Transport Individual Small Elements Horiz to Pad • Incremental Transport • Shared Use of Existing Transporters • New Dedicated Transporters • No Crawler or MLP	■ About the Same • On-Pad Integration / Staging • Mobile Service Tower with Crane • Cluster Interstage is FWD CCB Alignment Fixture • NA702 – LOX & RP Fluid Systems
LRB 2-Stage Options 	■ Clearly Better • 7.5m IsMS • High # of Medium MMUs • MMU Repair / Remake – Low Schedule Impact	■ Clearly Better • Horizontal Processing of All Elements • Horizontal Integration Option (CS/LRBs) • No VAB or RPSF Requirement • US Processing Common to CS • 9 Engines	■ Clearly Better • Existing Fab to Integ Transport Methods • Transport Horiz to Pad • Transport Integrated CS/LRBs -or- • Transport Individual CS & LRBs • Incremental Transport • New Transporter(s)	■ About the Same • On-Pad Integration Option • On-Pad Staging • Mobile Service Tower with Crane • Fixed Pad Erector (if CS/LRBs arrive integrated) • NH201-2 – LOX & LH2 Fluid Systems

Figure 3-16 Production & Operations Relative Rating

3.6. SLS Recommended Configuration

The Northrop Grumman team recommends configuration NK206-2 as shown in Figure 3-17. This two-stage vehicle features two liquid rocket boosters (LRBs), each powered by three RD-180 class LOX/RP engines. The core uses two RD-180 class engines and the upper stage a single J-2X. With cross feed NK206-2 can launch 122 t to LEO and 52t to escape. The three booster and two core engine layout provides staging that closely approximates the efficiency of a three stage vehicle without the attendant additional complexity and cost. Three stage performance and engine out capability can be achieved by incorporating cross feed as described in Section 3.8. Engine out capability may prove to be essential to successfully mount out year multiple launch missions to Mars and high energy NEO flights. The vehicle configuration is very compact. The 6m diameter core and liquid boosters are sized to accommodate the RD-180 engine envelope and can accept an upgrade to a domestic large RP engine when it becomes available. This compact configuration directly translates to a reduced production and operations footprint which are strong contributors to the lower LCC cost for this vehicle as shown previously in Figures 3-7 and 3-24.

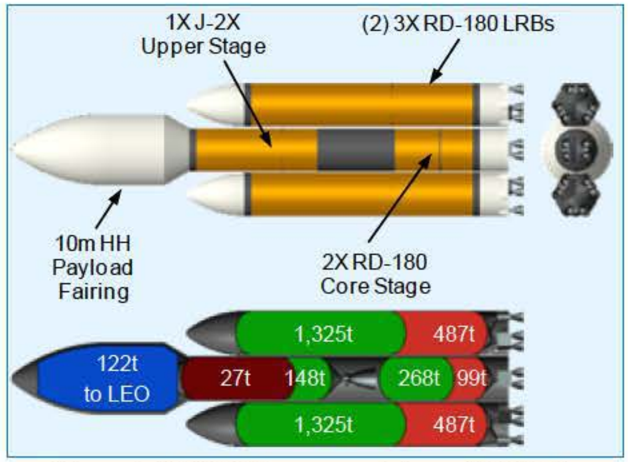


Figure 3-17 NGC Recommendation – NK206-2

NK206-2 scores well for all criteria and weightings and offers a family of options in addition to this very capable vehicle. NK 206-2 provides a high degree of adaptability and can evolve upward and downward in capacity to flexibly accommodate NASA exploration and other national user needs as shown in Figure 3-18 and discussed in Section 4. The NK206 family offers performance ranging between 30 and 196t to LEO. As already mentioned it is sized to optimally incorporate a domestic high thrust RP engine as discussed in Section 4.3. Variants include single stick vehicles that could be available if needed for very safe transport of crew to orbit and many other lower payload missions. The NK206 family allows development within NASA budget funding levels while making steady progress to a super heavy lift vehicle for Mars exploration.

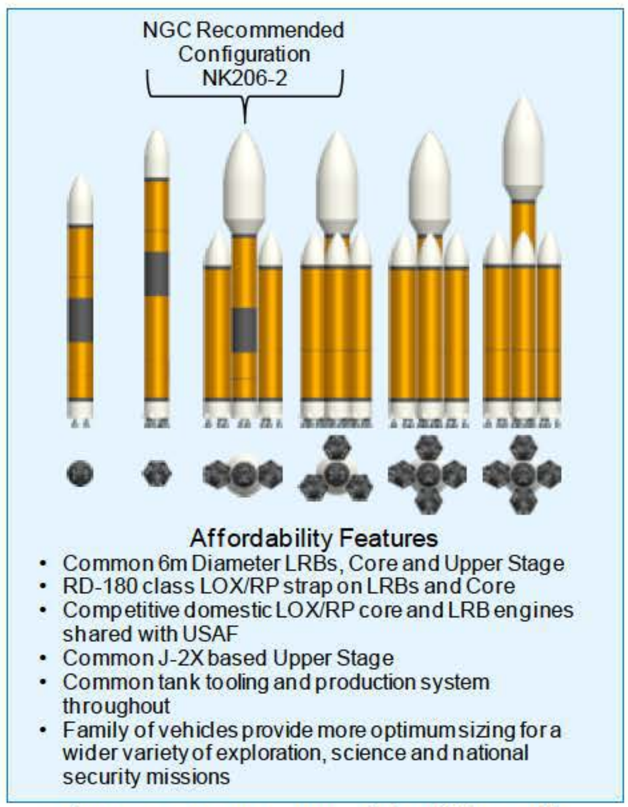


Figure 3-18 SLS Recommended Vehicle Family

Figure 3-19 shows the NK206-2 basic dimensions. The primary dimensions are in meters and the secondary dimensions in parentheses are in feet. Three RD-180 class engines nest nicely in the 6m base diameter with the addition of small nozzle fairings. A smaller base diameter may be possible with further refinement but will require a longer vehicle for the required propellant volume. The core with interstage is 27.5m long and the upper stage with payload adapter is 27.4m long. The payload fairing is a 1.375 aspect ratio tangent ogive with a 0.1m leading edge radius. The payload fairing length can be modified by varying the barrel length.

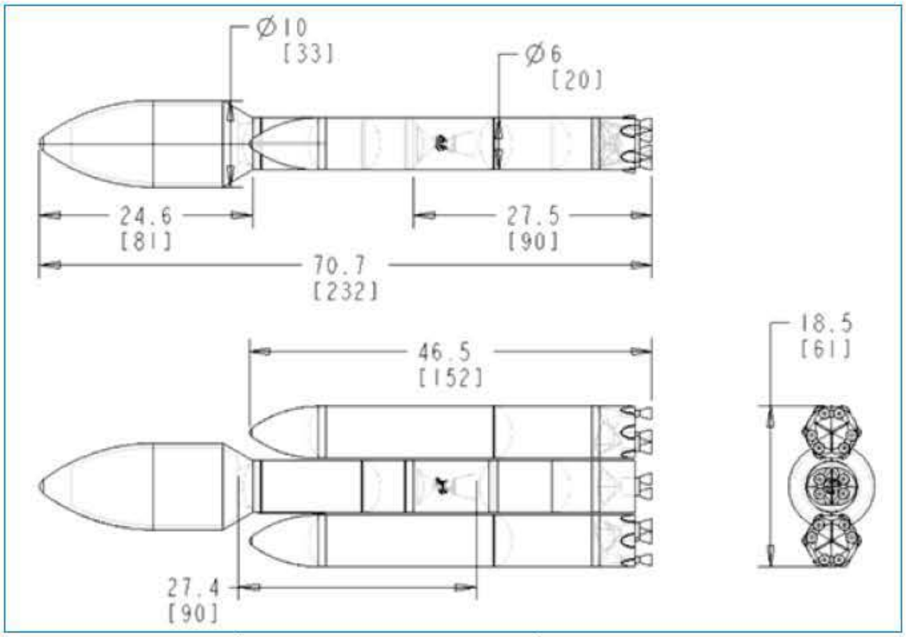


Figure 3-19 NK206-2 General Arrangement

Figure 3-20 shows the NK206-2 major manufacturing units. All tanks are built in unitized barrel and ellipsoid dome halves. The barrel length is stretched or shortened for the required propellant volume. The LRB nosecones, thrust structures, boat-tails, interstage and payload adapter are all built as unitized shells. The 10m diameter hammerhead payload fairing is broken up into a nose cap and panels to fit in the same production infrastructure as the rest of the 6m base diameter major units.

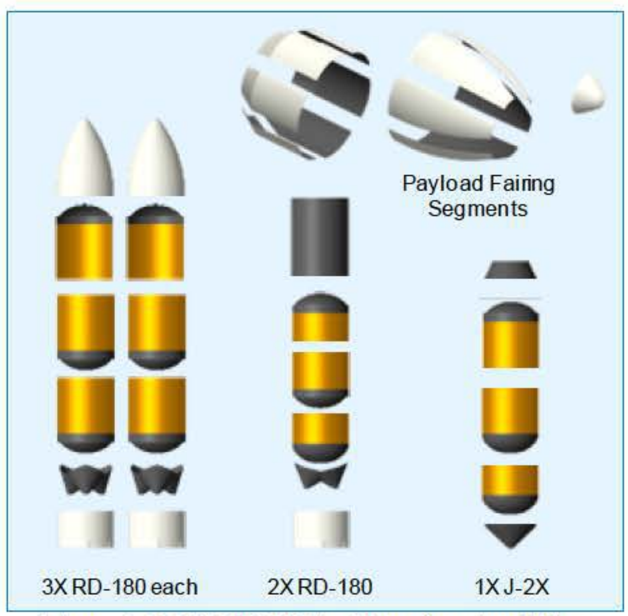


Figure 3-20 NK206-2 Major Manufacturing Units

Figure 3-21 is a summary NK206-2 production flow. After some initial learning the vehicle can be produced within a year. There are three shell fabrication stations that build all of the major units shown previously in Figure 3-20. Each major unit is stuffed with as much equipment and provisions as possible while in their most accessible state in the group A outfitting station. Thrust cones to tank halves and tank barrel subassemblies are next along with the installation of group B equipment. Major mate joins common bulkheads. At this point proof pressure acceptance testing can be performed. Next the engines and other high value propulsion equipment are installed and checked out. Next are the final group C installations and then the vehicle elements are closed out and shipped to the launch site ready for stacking.

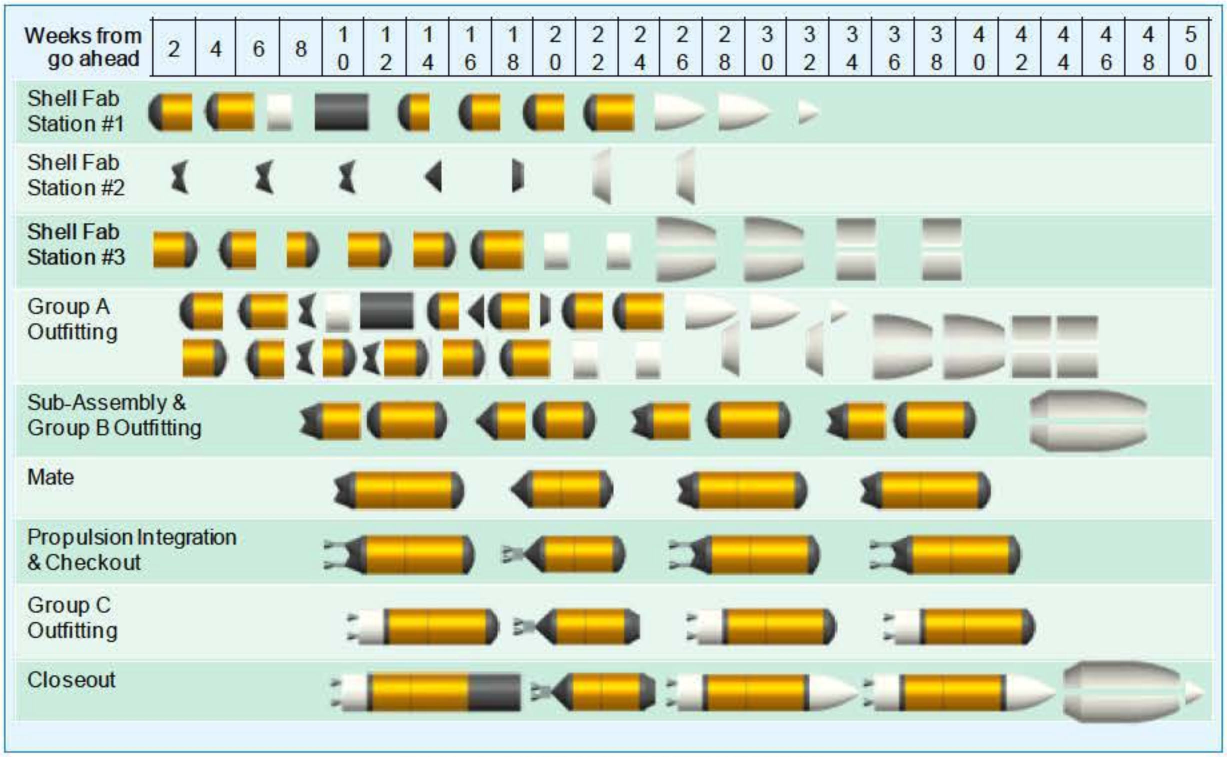


Figure 3-21 NK206-2 Production Flow

(b) (3)

[redacted]

ascent simulation data plots. Note that the minimum thrust to weight at lift off, LRB jettison and upper stage ignition are all healthy and the altitude profile is smooth. Maximum accelerations are reasonable along with all ascent characteristics. In this case the dynamic pressure is nil by the time the heating rate is low enough for payload fairing jettison.

(b) (3)

[redacted]

Figure 3-24 shows the NK206-2 life cycle cost for the 40 mission manifest shown previously in Section 2.3 Figure 2-27. The LCC is low for this vehicle due to its 6m base diameter, compact size and common production infrastructure. The small production and operations footprint keeps fixed costs low. The spikes and valley in this profile are driven by the multiple launches needed for higher energy missions can be smoothed by staggering the vehicle production.

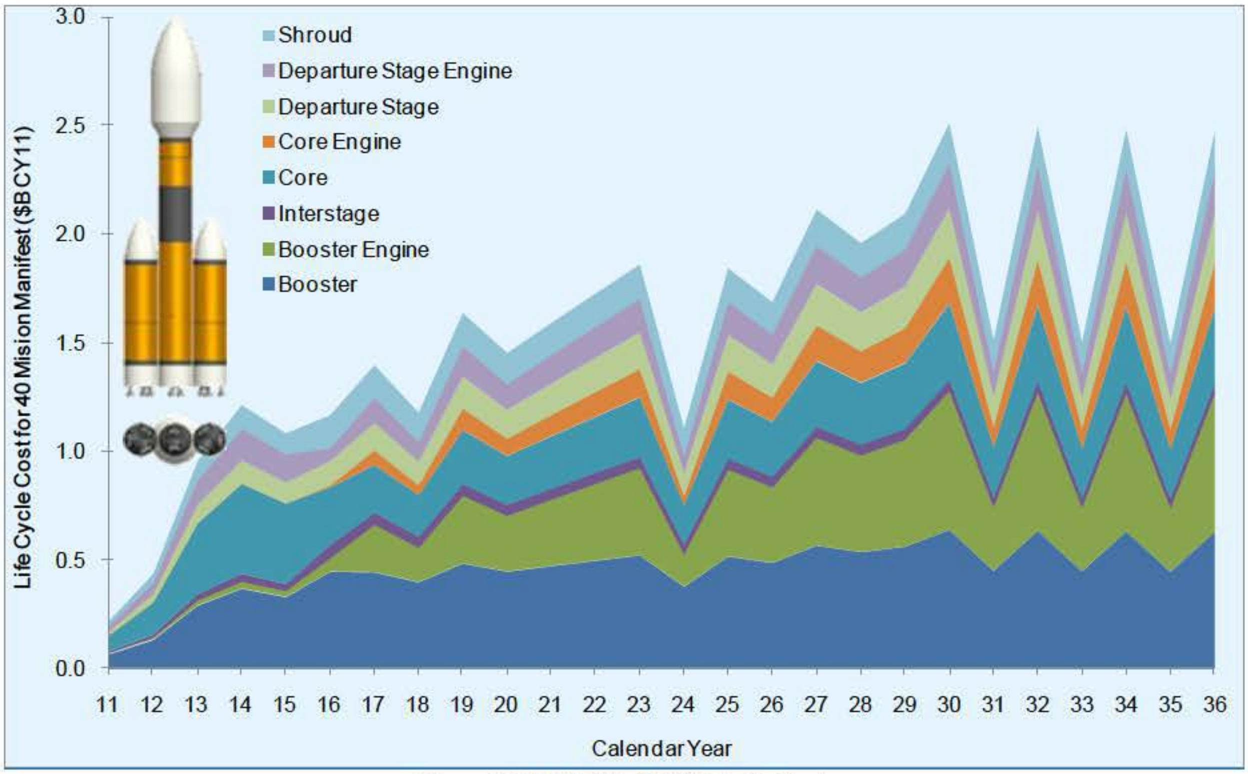


Figure 3-24 NK206-2 Life Cycle Cost

3.7. Commonality Opportunities *(SOW 3.4, 4.4)*

Main Propulsion Commonality – All LOX/RP Study

During our trade study we discovered that a dedicated in-space cryo-propulsion stage may be preferred over a multi-purpose upper stage (see Section 4.7). This opened up the design space for an all LOX/RP vehicle since the upper stage would no longer have to perform the escape burn. Due to common propulsion throughout the main engines, reduced tank sizes, simpler thermal management and the near term availability of a selection of capable engines, we hypothesized that an all LOX/RP vehicle would be more affordable. We optimized our highest scoring vehicles for RD-180s, RD-171s, F-1As and Merlin 2s using either demonstrated or predicted

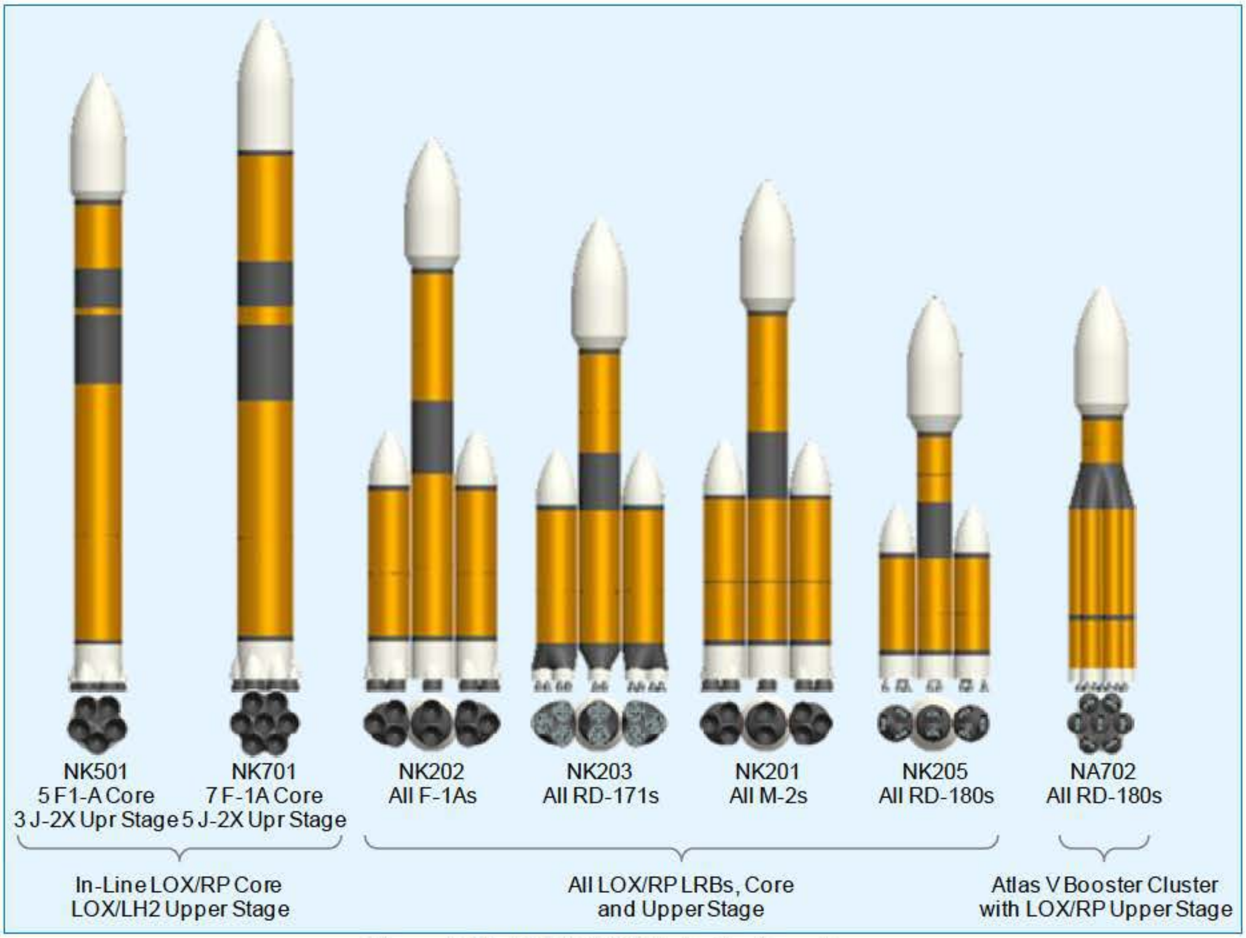


Figure 3-25 All LOX/RP Study Configurations

performance data. The resulting configurations are illustrated in Figure 3-25. Tank layouts for the study vehicles are presented in Figure 3-26. NK203, with its triple RD-171 LRBs, dual RD-171 core and single RD-171 upper stage is the highest payload performer out of the all LOX/RP vehicles as shown in Figure 3-27. Its large envelope RD-171 engines may call for some interesting thrust structures at such a small common core and LRB diameter.

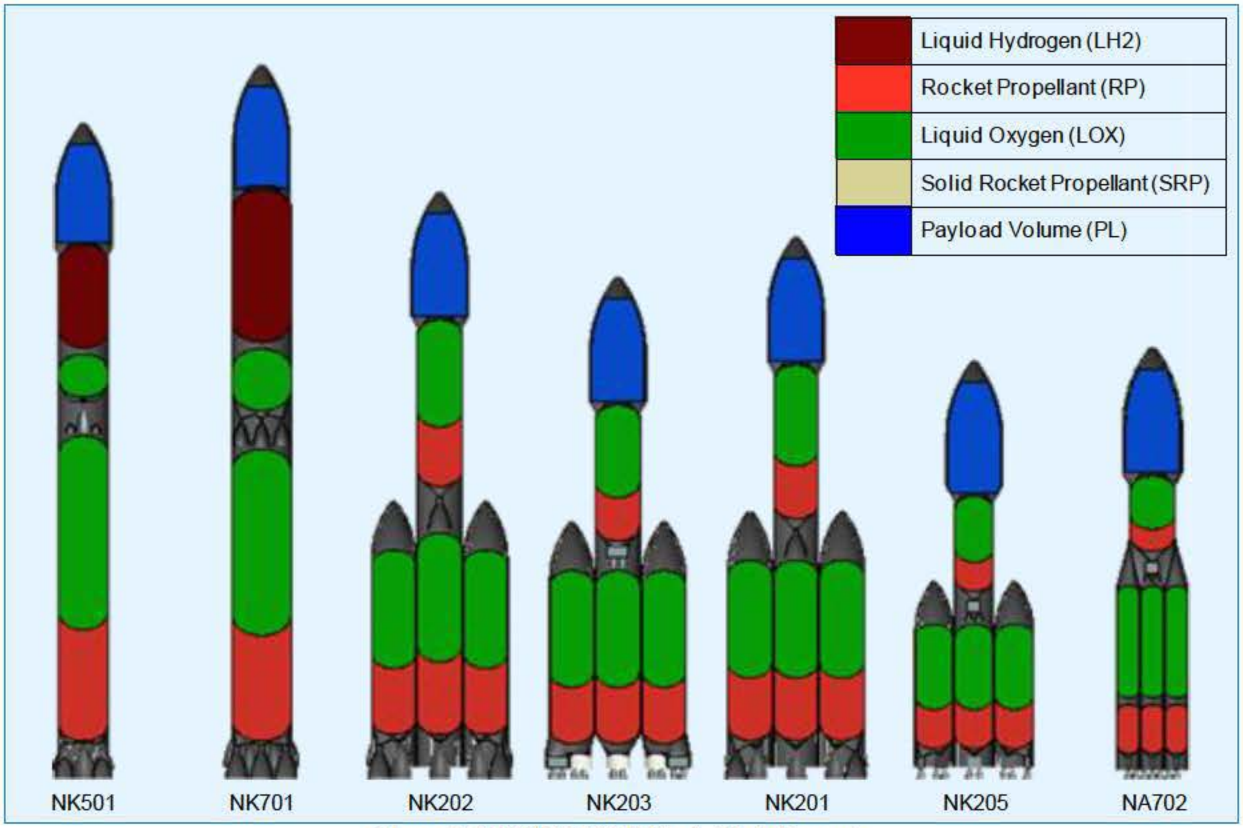


Figure 3-26 All LOX/RP Study Tank Layouts

Although the FASTPASS LEO payload prediction of nearly 100 metric tons for the smallest vehicle in this mix seems reasonable (Figure 3-27), our similarly sized SLS vehicle recommendation, NK206-2 with its J-2X upper stage, outperforms NK205 by 25 metric tons. Our study determined that, although an all-LOX/RP configuration looks very promising for a LEO only vehicle, we really need a multipurpose LOX/LH2 upper stage for early human exploration beyond earth orbit and long duration one-way science and exploration cargo missions. Therefore, we recommend developing a LOX/LH2 Upper Stage from the onset.

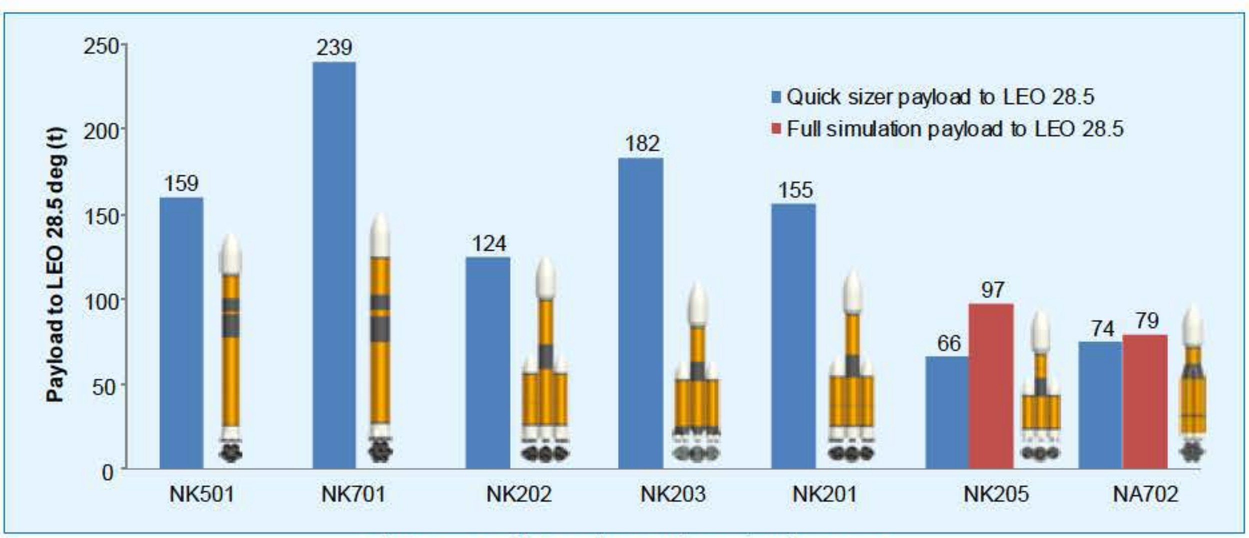


Figure 3-27 All LOX/RP Study Payload to LEO

Payload Fairing Commonality – 10m Hammerhead Fairing Study

Payload fairing size requirements have yet to be defined. It appears that volume driven payloads would benefit from a 10m diameter fairing, but the barrel length has not been determined. For our Constellation-derived baseline vehicle we have already discussed utilizing the available 10m In-situ Manufacturing System (IsMS) to build a unitized composite payload fairing with the expectation that it will be cut into petals for payload encapsulation and fairing jettison. Most of our highest scoring SLS configurations though have core stages in the 6m – 8m range. This study investigated production system commonality opportunities to build a 10m hammerhead payload fairing using existing reduced diameter infrastructure.

Our NGC recommended vehicle, NK206, was used as the example for this study, but this concept applies to virtually any reduced diameter vehicle with a hammerhead fairing requirement. This also enables flexibility with the fairing shape and barrel length and with today's analysis tools, mission-tailored fairings are possible with minimal fixed tooling cost deltas. Our goals in this study include the utilization of common infrastructure used throughout the vehicle, a minimized amount of unique tooling, and the ability to accommodate payload-specific barrel lengths. We assumed all composite construction, the capability to splice both longitudinally and circumferentially, common IML tooling and common manufacturing processes. This tooling approach enables the design flexibility to increase laminate gauges outward as required and common manufacturing processes throughout the vehicle allow for flexible production flow planning and a large fabrication learning curve.

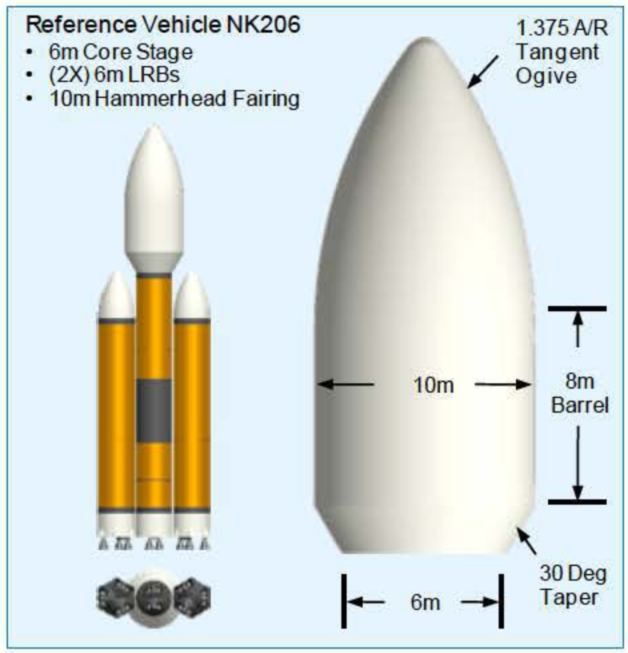


Figure 3-28 10m Hammerhead Fairing

NK206 is a 2-Stage vehicle with a 6m diameter core and two common-diameter liquid rocket boosters as shown in Figure 3-28. We have chosen a 1.375 aspect ratio tangent ogive fairing to reduce aerodynamic drag and provide a relatively benign and large payload environment as compared to a biconic design. The thirty degree taper is arbitrary and does not impact the feasibility of this study.

As shown in Figure 3-29 we split this fairing circumferentially in three locations creating a nose cap, ogive panels to include a portion of the cylindrical section for splicing, and barrel panels which include the hammerhead taper. The 2m nose cap as shown is also an arbitrary dimension and can be altered as required for optimized separation. For this 6m diameter production infrastructure we fabricate the panels in six 60 degree segments. For larger diameter core stage vehicles above 7.5m we are able to fabricate the panels in four 90 degree segments. In this case, the 6m IsMS is available for use from the basic vehicle production line. The only required major shell fabrication tooling additions consist of a two-sided barrel/taper mandrel, a similar ogive mandrel and a nose cap tool. The tool surfaces of the mandrels are nested to minimize the cross sectional area.

(b) (3)

[redacted]

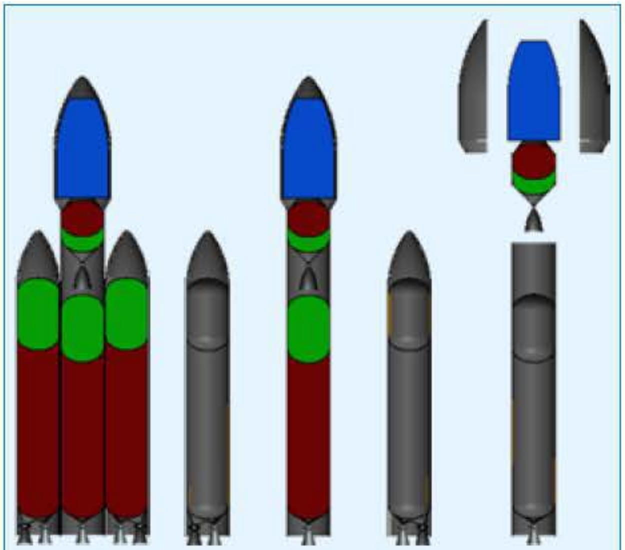
When the barrel and ogive segments are through fabrication we outfit any payload accommodations or required secondary features and assemble the segments into six 60 degree petals. Depending on the separation requirements we either splice the nose cap to one of the petals and deliver six 60 degree segments, or we longitudinally splice the petals together to form three 120 degree or two 180 degree fairing segments. To increase the barrel length we simply fabricate additional cylindrical sections and add circumferential splices as required.

The 6m diameter envelopes shown in Figure 3-29 illustrate that, with nested tooling surfaces and our vehicle-common manufacturing approach, 10m diameter hammerhead fairings can be built using the same reduced diameter production infrastructure as the rest of the vehicle.

3.8. Innovative or Nontraditional Processes or Technologies *(SOW 3.3, 4.3)*

Innovative Technology – Throttle Back and Cross Feed Study

Throttle back of the core engines after initial ascent is used successfully on the Delta IV Heavy to gain some staging benefit by depleting the LRB propellant before the core and jettisoning the LRBs. McDonnell Douglas, Boeing and now the United Launch Alliance have also studied cross feed for the Delta IV Heavy and SpaceX has stated they are considering cross feed for their Falcon 9 Heavy. We therefore decided to perform a throttle back and cross feed study for our SLS three body first stage configuration family. As shown in Figure 3-30 we chose our LOX/LH2 NH201-2 configuration as an example vehicle for this study. For throttle back and cross feed lift off has all first stage engines at 100% power to meet our lift off thrust to weight constraint of 1.2.



Example Configuration NH201-2

- (2) LRBs with (3) RS-68Cs each 7.5m diameter
- Core Stage with (2) RS-68Cs 7.5m diameter
- Upper Stage with (1) J-2X 7.5m diameter
- Hammerhead Payload Fairing to 10m diameter

Figure 3-30 Cross Feed Concept of Operations

For throttle back the core engines are set to their reduced power setting as specified by the engine supplier about 60 seconds into ascent. At that point in the ascent sufficient propellant has been burned to maintain adequate thrust to weight with the core engines at reduced power for acceptable gravity losses. The LRB engines maintain full power to LRB tank depletion and are then jettisoned. The core engines are then throttled up to full power for the remainder of the first stage burn. This provides some staging benefit and is a good solution for the Delta IV Heavy where the LRB and core tanks have common engines and propellant volume. Figure 3-31 shows there is a modest but easily achieved payload benefit due to throttle back for the NH201-2 configuration family.

For cross feed all first stage engines are fed from a common manifold and burn at full power until the LRB tanks until depleted. Then the LRBs are jettisoned. Since all engines are fed from a common manifold the core tanks are full at LRB staging. This provides staging efficiency nearly equal to a three stage vehicle since all the dry mass associated with the burn up to the LRB staging point is jettisoned with the LRBs and there is not any partially depleted core tank dry mass to carry further. [redacted]

[redacted]

[redacted] [redacted]

(b) (3) [redacted] (b) (3) [redacted]

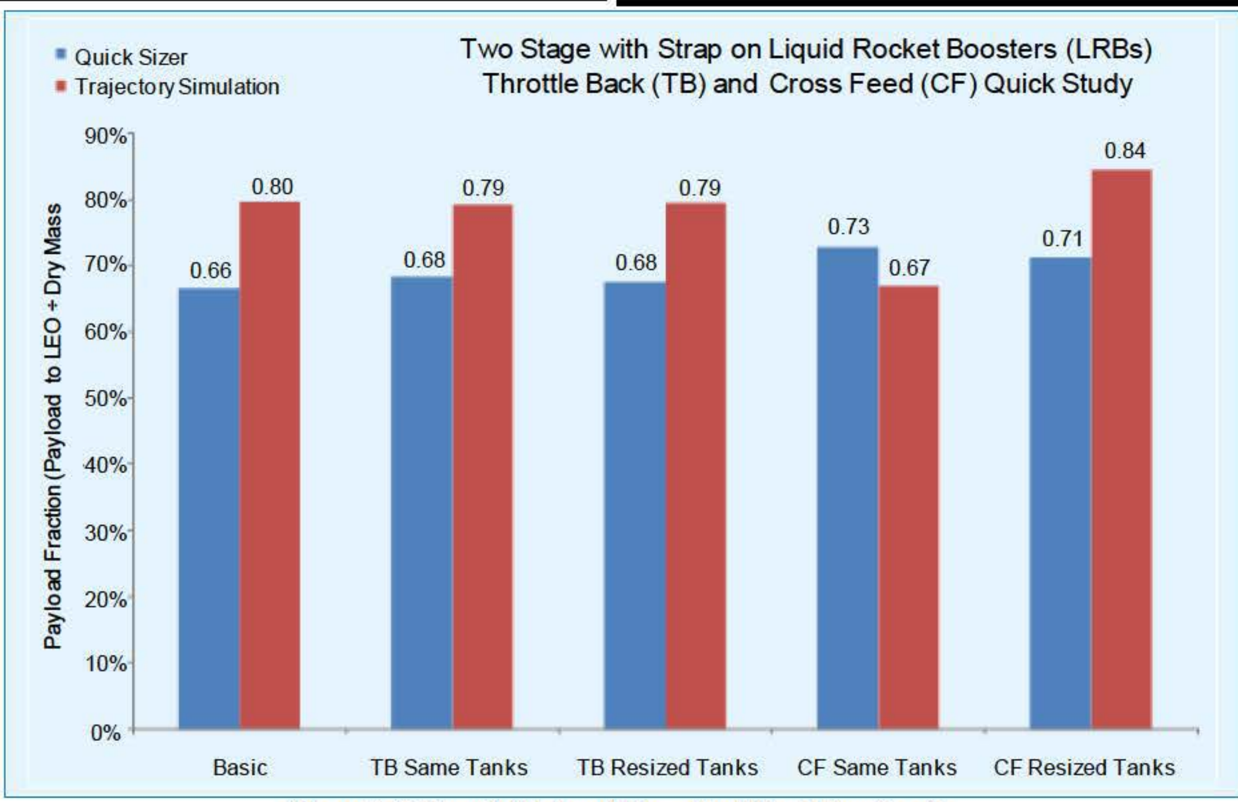


Figure 3-32 Throttle Back and Cross Feed Simulation Results

Cross feed also enables engine out options not otherwise available which can greatly improve launch reliability. Since all first stage tanks feed all first stage engines via a common engine manifold if any one of the first stage engines fails the stage can still feed all remaining engines through LRB tank depletion or longer off the core tank only. This provides a variety of engine out options depending on payload, which engine fails and at what time during ascent it fails.

(b) (3) [redacted]

(b) (3)

[redacted]

(b) (3)

[redacted]

(b) (3)

[redacted]

Innovative Technology – Clustered Interstage

Our clustered interstage study hypothesis is; if Atlas V Common Core Boosters (CCBs) or Delta IV Common Booster Cores (CBCs) can be aggregated with minimal changes, then SLS development costs would be greatly reduced and production fixed costs would be dramatically lower due to the smaller size of major elements and shared use with the USAF Evolved Expendable Launch Vehicle (EELV) program.

(b) (3)

[redacted]

(b) (3)

[redacted]

We developed a clustered interstage concept that enables the aggregation of off-the-shelf EELV boosters without forcing major changes to the booster design or production infrastructure. Our design features a set of composite thrust shells that smoothly distribute the individual booster loads to a uniform running load at the SLS upper stage interface. This concept can reasonably cluster between two and seven individual boosters. In the seven booster example the natural geometry of six equally spaced boosters provides a centerline zone for the seventh as described in the NA702 production plan in Section 3.5. (b) (3) [redacted]

(b) (3)

[redacted]

One of the integration benefits is that the blended thrust shells form a natural volume to nest the upper stage engine nozzle as shown in Figure 3-37. This open draft volume is more than sufficient to allow the upper stage to separate in a simple axial translation without any need to split or independently separate the interstage. The rigid interstage may also serve dual use as the forward alignment tool required for on-pad integration.

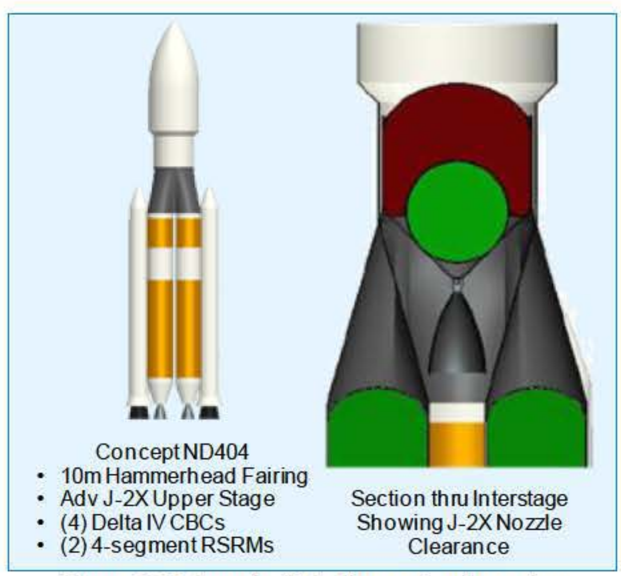


Figure 3-37 Sample SLS Cluster Configuration

4. Incremental Development Options *(SOW 3.5, 4.5)*

Incremental development is one of our key SLS sustainability drivers as discussed in Figure 1-1. Incremental development is also synergistic with our other sustainability drivers, low fixed costs and flexibility. Our recommended configuration NK206-2 with its three body LOX/RP core and J-2X LOX/LH2 upper stage is particularly strong in incremental development options. This section discusses major vehicle incremental development options and the resulting growth path for payload to low Earth orbit (LEO) and for payload beyond Earth orbit (BEO).

4.1. Configuration Family

One of the most attractive features of a three body first stage configuration is the family of vehicles that can be incrementally developed from the basic. Figure 4-1 shows the family of vehicles that can be developed from our recommended NK206-2 configuration. Our recommendation is to optimize the upper stage and liquid rocket boosters (LRBs) for the NK206-2 vehicle and then synthesize the rest of the family from the -2 configuration.

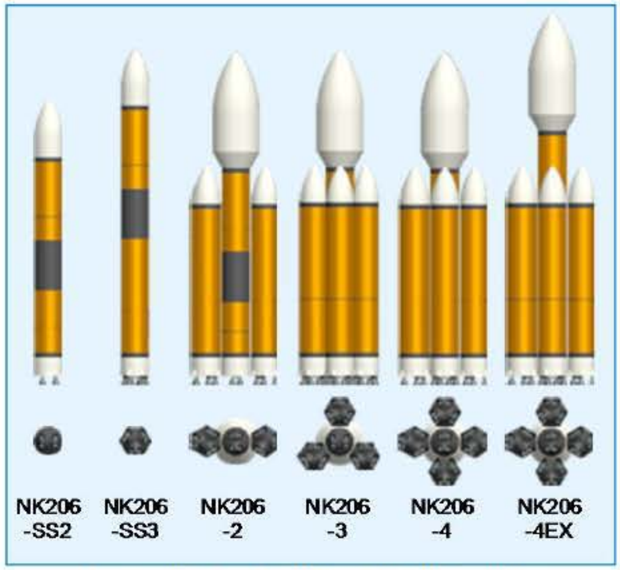


Figure 4-1 Configuration NK206 Family

A single stick two engine first stage vehicle (NK206-SS2) can be developed from the NK206-2 vehicle using the same upper stage and interstage and a modified two engine core. The –SS2 straight payload fairing can share the same production system and tooling as the basic LRB nosecones. As shown in Figure 4-2 the two engine core is stretched from the basic -2 core to provide a 30 metric tonne (t) payload to LEO. A trade should be conducted during preliminary design whether to update the structural sizing for the lower inertial loads or to use the -2 sized elements as is. If composite construction is selected the –SS2 shares the same production system and tooling as the basic -2 and can be resized for the lower loads by reducing the composite plies and or core thickness.

A three engine single stick vehicle (NK206-SS3) can also be developed from the basic -2 similar to the –SS2. For the –SS3 the three engine LRB is shortened to make the –SS3 core. Our tank production system fabricates half tank major units joined circumferentially at mid barrel. The barrel length can be stretched or shortened for optimum tank volume. As for the –SS2 the structural sizing can be updated for the lower inertial loads for a 48t payload or the basic -2 sizing can be retained with some payload penalty.

The single stick vehicles can be introduced first for an early or lower cost initial operating capability or added later after the basic -2 is deployed. Whether deployed first or later the single stick vehicles share the same production and operations system as the basic -2 and contribute to the SLS sustainability by sharing fixed costs and providing flexibility for a wide variety of exploration, science and national security missions.

Figure 4-2 NK206 Family Tank Layout

As our spacefaring capabilities grow over the decades a need for greater lift capability may arise. When greater lift is needed the basic -2 vehicle can be grown by adding LRBs to develop the NK206-3 and NK206-4 vehicles.

Since the LRBs are sized principally by tank pressure the basic -2 LRBs should be useable for the -3 and -4 without updates. The upper stage, interstage and core are the same volume as the basic -2 but will require structural sizing updates for the bigger payload inertial loads and LRB attach points. The launch pad will of course need to be updated for additional LRBs.

If our spacefaring requirements continue to grow the core stage can be stretched for an optimized 4 LRB configuration NK206-4EX. The LRBs are unchanged, the core is stretched, the LRB attach point is moved and the upper stage, interstage and core require updated structural sizing. The payload fairing can also be stretched if volume becomes a driver. The launch pad would be modified at that time for the taller stack.

A major strength of the NK206 configuration family is its incremental development robustness covering LEO payloads from 30 to 196t using a single production system.

4.2. Cross Feed Incremental Development

Section 3.8 discusses the significant payload and engine out benefits of cross fed LRB and core tanks. (b) (3) [redacted] (b) (3) [redacted]

(b) (3) [redacted]

4.3. Domestic LOX/RP Engine Incremental Development

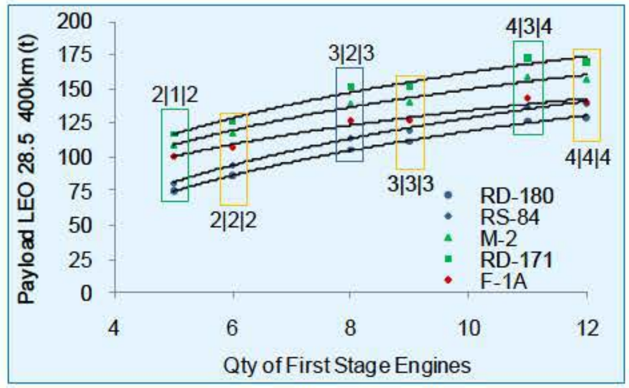


Figure 4-4 Three Body First Stage Engine Quantity

The quantity and type of LOX/RP engines is a critical consideration for three body first stage configurations. Figure 4-4 shows the LEO payload for combinations of LRB and core engines and log regressions through each engine type. The nomenclature is LRB1|Core|LRB2. Note that in all cases having more engines in the LRBs has better return than the same number of engines in each body. Note also that the point of diminishing returns is our recommended 3|2|3 engine layout. The regressions curve gently however which indicates that 2|1|2 and 4|3|4 layouts can be considered in early design trades.

Figure 4-5 shows the LEO payload versus total vehicle dry mass for the same combinations of engine quantity and type. This log regression is through all the options except for the F-1A options which are not competitive. The regression has a very good statistical correlation to the data. This plot again shows that options with more LRB engines than core engines are preferred. Our recommended 3|2|3 RD-180 class engine layout is in the desired zone of diminishing returns but the plot also shows that fewer larger engine 2|1|2 M-2 and RD-171 options are also interesting. We therefore recommend that the 3|2|3 RD-180 class option be the advanced design point of departure and that a higher thrust 2|1|2 option be thoroughly evaluated.

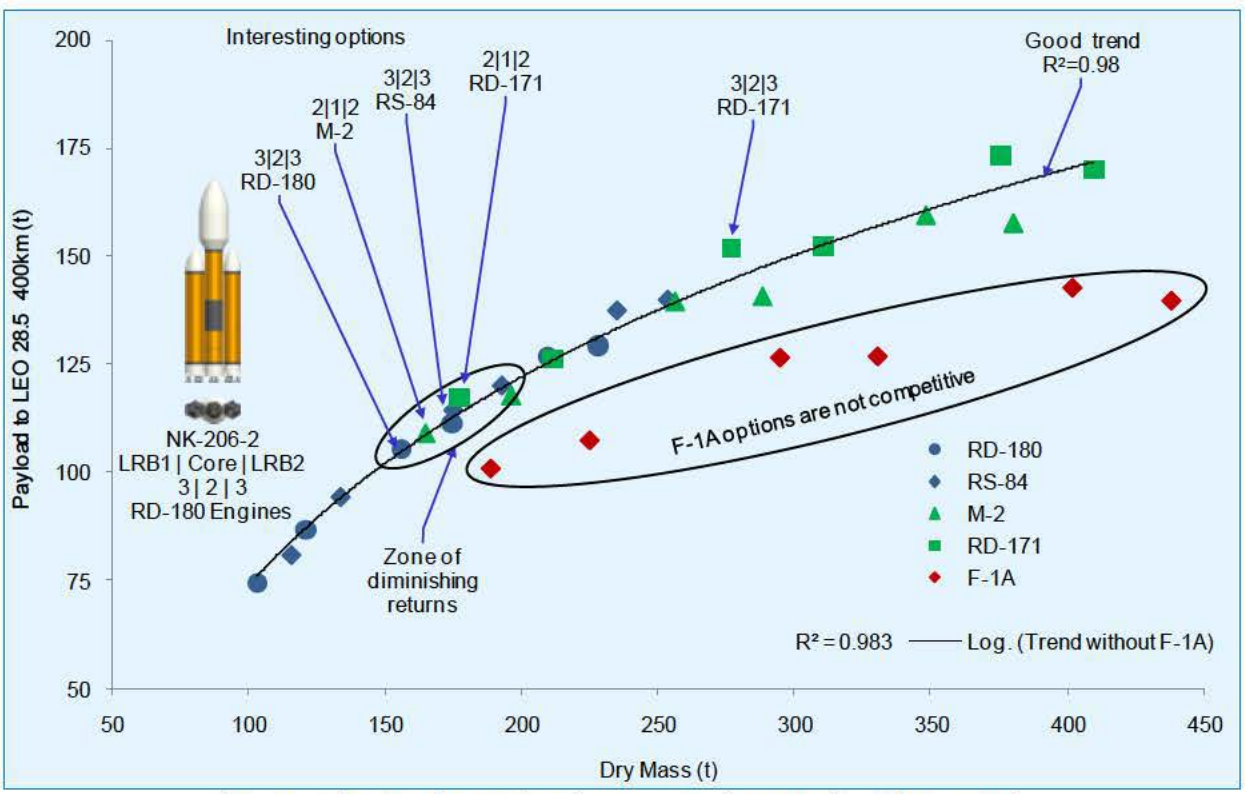


Figure 4-5 Three Body LOX/RP First Stage Engine Quantity and Type Study

The RD-180 and its heritage engines have been in production in Russia and the former Soviet Union for decades and are used for the USAF on the Atlas V Evolved Expendable Launch Vehicle (EELV). There are concerns however with the rigor of their pedigree and long term stability of their supply chain. A domestic LOX/RP engine with a sea level thrust level about 4.5MN (1Mlbf) is therefore desired. We therefore recommend that a competitive LOX/RP first stage engine program be initiated for use on the SLS and as an upgrade to the Atlas V EELV. Figure 4-6 shows our recommended NK206-2 configuration and an engine upgrade configuration NK206-2EU using the RS-84 predicted performance as an example of a next generation engine (NGE).

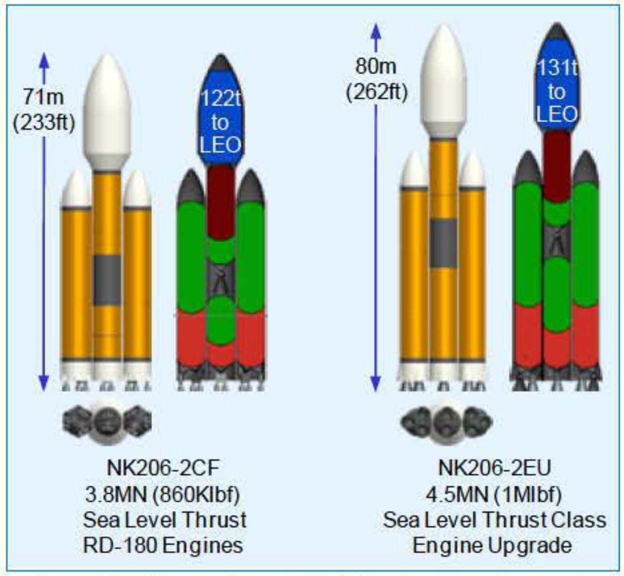


Figure 4-6 Domestic LOX/RP Engine Development

We recommend that the first stage LOX/RP engine quantity and thrust level be a high priority advanced trade with firm requirements for the NGE established by SLS System Requirements Review (SRR) and fed to the competitive domestic engine development program. The SLS can be initially deployed with tanks sized for the NGE but using current RD-180s until the NGE is available. This will likely require a separate thrust shell detail design but allows time for an incremental domestic LOX/RP engine development within available funding limits.

4.4. Dedicated LEO Variant

All of the configurations shown previously have been developed for exploration beyond Earth orbit (BEO) and have multi-purpose upper stages that can perform an Earth escape burn. As shown in Figure 4-7 we could develop a LEO only upper stage with more payload to LEO configuration (NK206-2LEO). The upper stage could be shortened for the LEO ascent burn propellant only and the dry mass associated with the escape burn propellant converted to LEO payload. This fact should be considered when comparing NK206-2 to other vehicles and external requirements and expectations.

The SLS primary mission is exploration beyond Earth orbit so we do not recommend developing the LEO only variant. For any LEO only SLS missions we recommend that the upper stage have propellant offloaded for the most available payload

4.5. Payload to LEO Growth Path

Sections 4.1 through 4.4 above describe how our recommended SLS configuration NK206-2 can be incrementally developed as a family and with cross feed, engine quantity and type

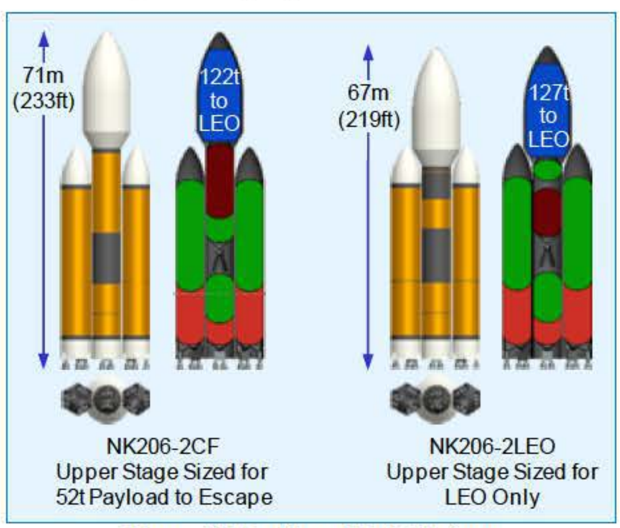


Figure 4-7 Dedicated LEO Variant

and by optimizing the upper stage. Figure 4-8 shows how all of these incremental developments provide a very robust LEO payload growth path. It is remarkable that one configuration family can be gracefully developed and deployed over time fulfilling mission payloads from 30-196t.

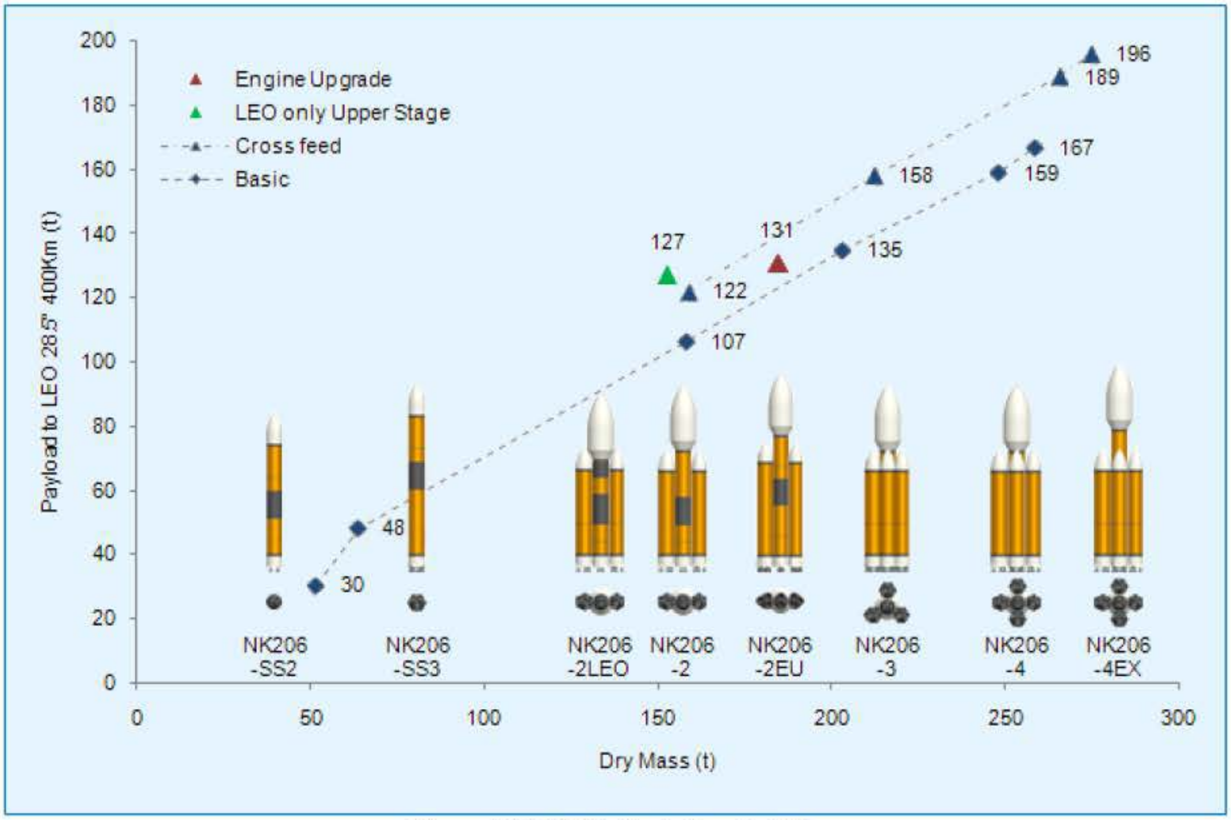


Figure 4-8 LEO Payload Growth Path

4.6. In-space Refueling Incremental Development Options

In Section 2 the dramatic payload and mission reliability benefits of refueling in LEO were discussed. Figure 4-9 shows three basic approaches for LEO refueling that can be developed incrementally as our exploration capabilities mature and payload to escape requirements grow. The first and easiest is to transfer propellant from payload tanks during ascent. This delivers a

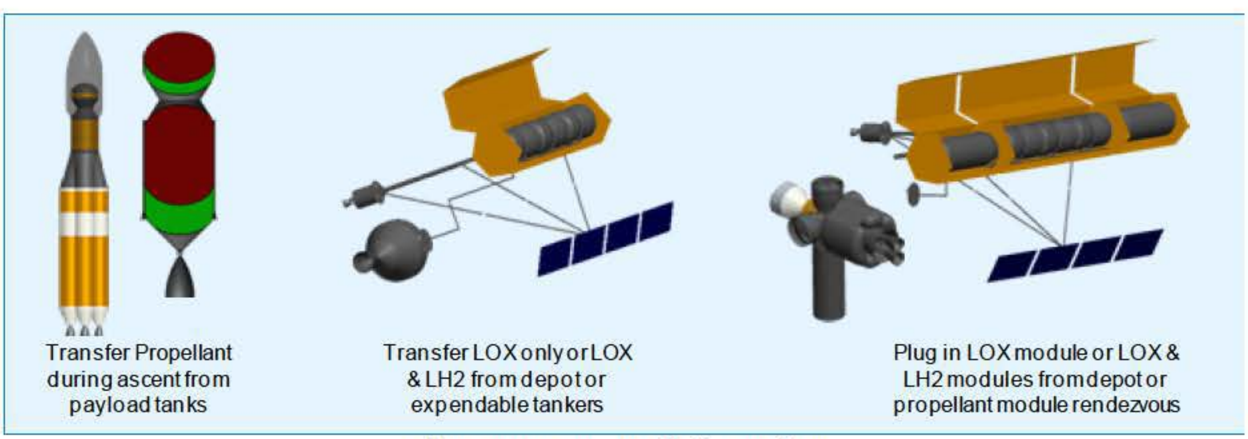


Figure 4-9 In-space Refueling Options

fully fueled upper stage to LEO where the payload tank is jettisoned and the stage rendezvous and docks with an exploration spacecraft. Since the propellant is transferred under load this approach can be used while micro-gravity propellant transfer methods are developed. The next approach is the transfer of propellant from a tanker or depot on orbit. This can start with LOX only and LH2 can be added later when long duration LH2 storage methods are matured. The last approach avoids micro-gravity propellant transfer by undocking full propellant tanks from a tanker or depot and plugging them into an exploration spacecraft. The tanks can be launched to LEO by the SLS or by commercial commodity contracts.

4.7. Dedicated In-space Cryo-propellant Stage Incremental Development

Multi-purpose upper stages were used on Apollo/Saturn and were planned for Constellation for Earth ascent and trans-lunar injection (TLI). The upper stage thrust level and TLI burn can be effectively performed by a single J-2X class engine. A single engine is acceptable due to the free return available from the moon if main propulsion is lost. Exploration beyond Earth orbit however requires main propulsion to return to Earth. The main propulsion system must therefore be very reliable and most likely requires redundancy to meet acceptable reliability.

(b) (3)

[redacted]

minute burn. These are all reasonable so three RL-10s is an acceptable engine selection for LENEO in-space cryo-propulsion stages (ISCPS). One J-2X is adequate for escape but is overkill for the destination burns with very short burn times that must be timed exactly. Two J-2Xs are much too big for an ISCPS and even for upper stage ascent. The NGE is a better fit with two engines having a reasonable thrust to weight and burn time during escape and reasonable thrust to weight and burn times with one engine out at the destination.

(b) (3)

[redacted]

We can begin human exploration with multi-purpose single J-2X upper stages for lunar missions with a free return. The multi-purpose J-2X stage is also useful for one way science missions and exploration cargo missions beyond Earth orbit. For human exploration LENEO missions an RL-10 based destination propulsion stage can be incrementally developed or for less demanding missions the Multi-purpose Crew Vehicle (MPCV) can use its service module propulsion. For future more demanding missions we recommend that a purpose built LOX/LH2 NGE be developed around 100Klbf thrust specifically optimized for long duration in-space operations and quiescent periods. Two of these engines are sized optimally for a dedicated ISCPS.

Relieving the ISCPS from ascent burn requirements simplifies the upper stage and opens up the design space for the ISCPS. Figure 4-12 shows a few of many concepts. An interesting concept (NISCPS01) sizes center core tanks for the escape burn and packages the destination propellant in saddle tanks. After the escape burn the center tanks convert to pressurized compartments that have some radiation shielding from the saddle tanks. The center compartments also provide access to the propulsion system for servicing. Another concept (NISCPS02) shows how propellant tanks can be plugged in to the ISCPS. The LH2 tanks are large due to low energy density and NISCPS03 illustrates how these large tanks might be used to advantage.

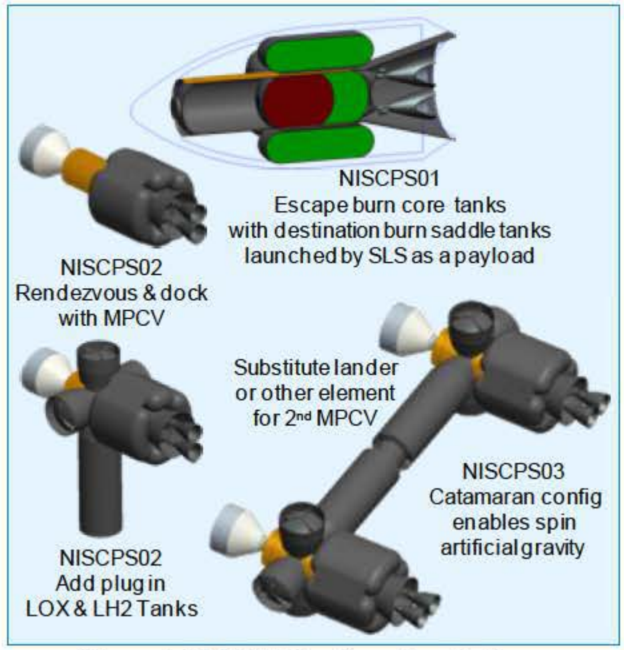


Figure 4-12 ISCPS Configuration Options

4.8. Payload to Escape Growth Path

Sections 4.6 and 4.7 describe how in-space capabilities can be grown incrementally for in-space refueling and advanced ISCPS capabilities. Figure 4-13 shows how the incremental in-space capability growth complements the incremental growth of the SLS. This two axis growth path provides a robust path forward for many decades of exploration beyond Earth orbit.

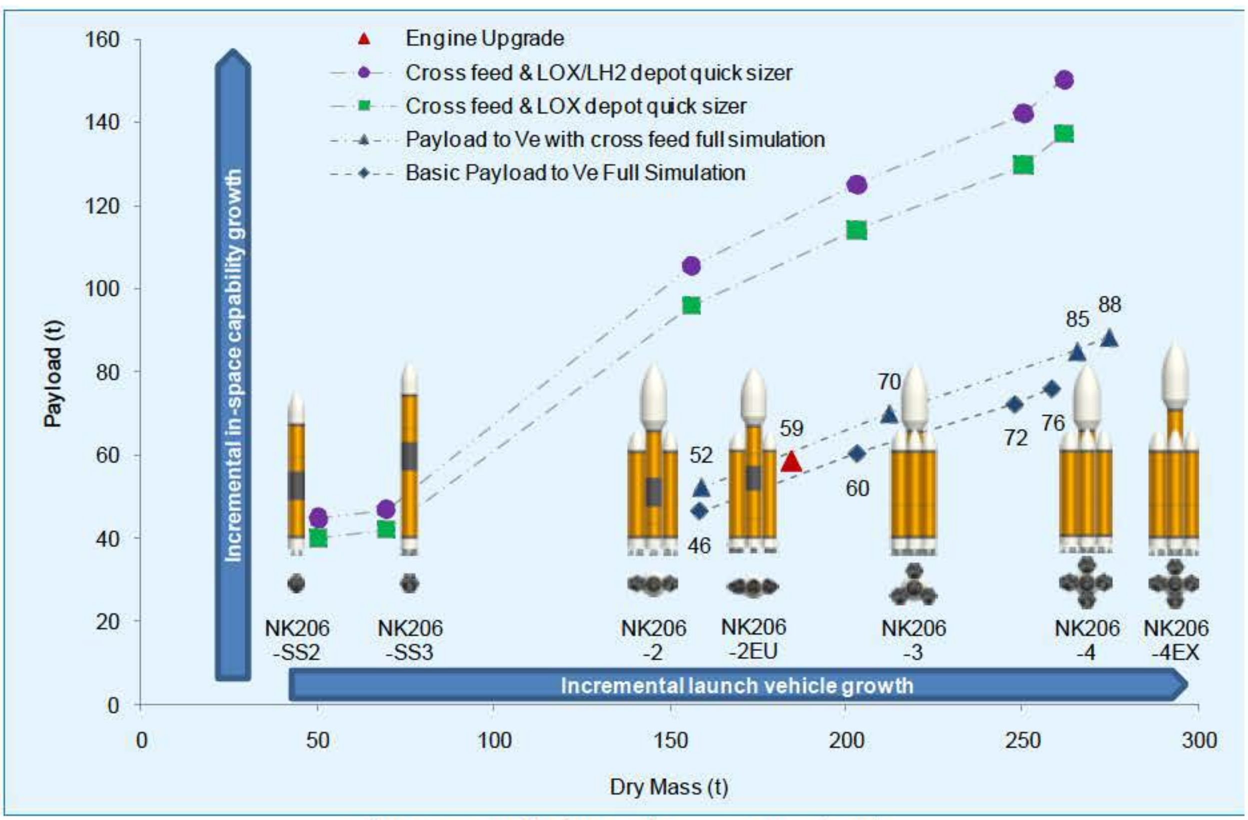


Figure 4-13 Payload to Earth Escape Growth Path

4.9. Production and Operations Growth Path

Section 3.5 discusses our production and operations plans. The matrix in Figure 4-14 shows how the production system shown in Figure 4-15 can be developed incrementally. For the –SS2 vehicle the light blue stations and mandrels are required. For –SS3 only mandrel 6 (M06) needs to be added. For -2 mandrels M07, M08 and M09 are added. The entire NK206 family can be produced with the blue and green stations and mandrels. Depending on production rate, stations 3 and 4 and mandrels M11 and M12 may be added.

	NK206 Family					
Sta/M#	-SS2	-SS3	-2	-3	-4	-4EX
Sta1	☑	☑	☑	☑	☑	☑
M01	☑	☑	☑	☑	☑	☑
M02	☑	☑	☑	☑	☑	☑
M08	☒	☒	☑	☑	☑	☑
M09	☒	☒	☑	☑	☑	☑
Sta2	☑	☑	☑	☑	☑	☑
M03	☑	☑	☑	☑	☑	☑
M04	☑	☒	☑	☑	☑	☑
M05	☑	☑	☑	☑	☑	☑
M06	☒	☑	☑	☑	☑	☑
M07	☒	☒	☑	☑	☑	☑
Sta3	☒	☒	☒	☒	?	?
M10	☒	☒	☒	☒	?	?
Sta4	☒	☒	☒	☒	?	?
M11	☒	☒	☒	☒	?	?

Figure 4-14 NK206 Family Production Station (Sta) and Mandrel Number (M#) Matrix

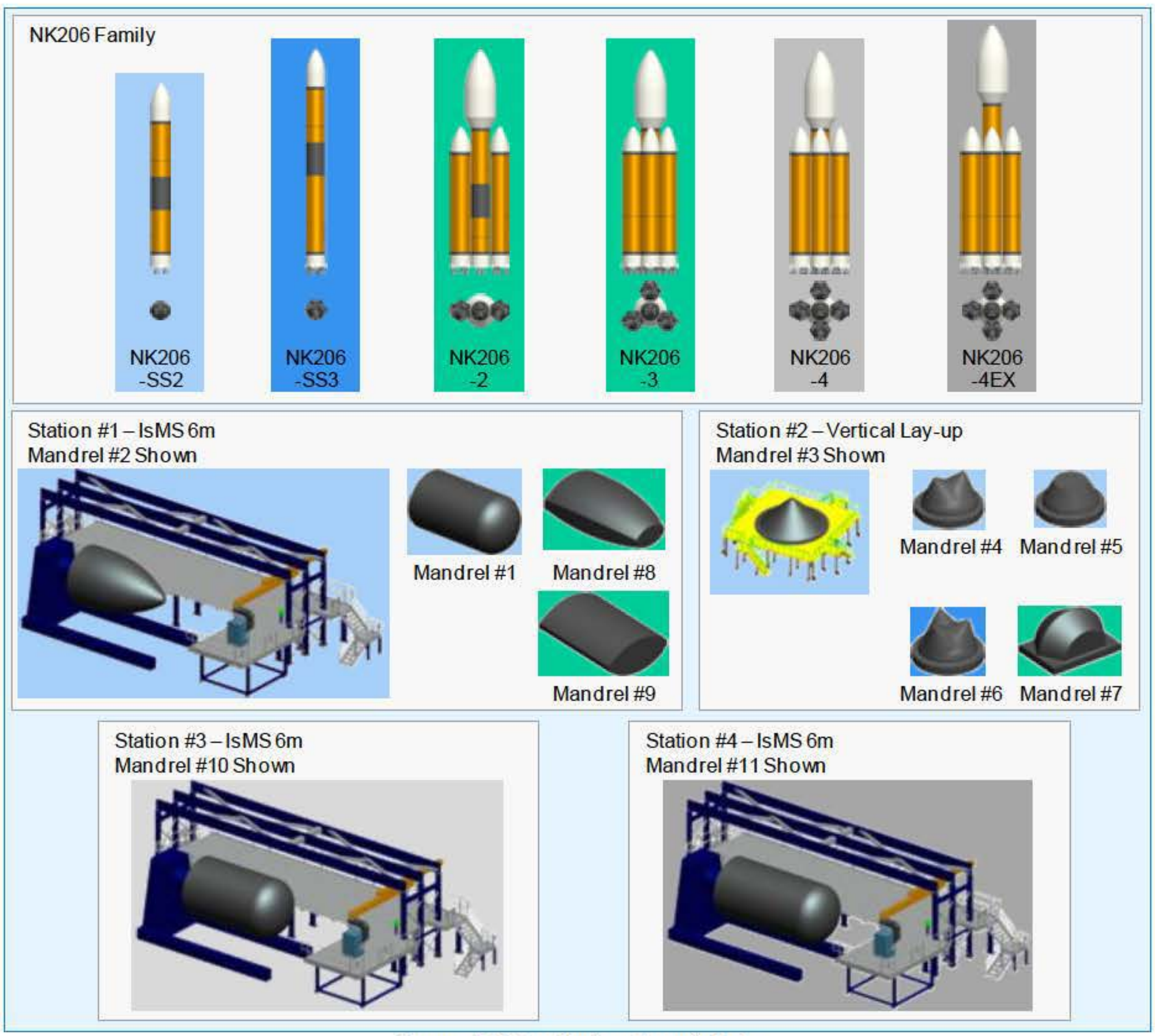


Figure 4-15 Production Growth Path

5. Capability Gap Analysis

We executed proprietary information agreements (PIAs) with the major launch vehicle subcontractors and rocket propulsion suppliers and conducted a two phase capability and technology gap and opportunity analysis. In the first phase we conducted a capability and technology push where we asked the suppliers to inform us on their capabilities and technologies that they believe we should be aware of as we explored the SLS configuration trade space. In the second phase we came back to the suppliers with a capability and technology pull where we shared our SLS study findings and assessed any gaps and additional opportunities. This was done at the vehicle configuration level. The University of Alabama Huntsville (UAHuntsville) Propulsion Research Center provided additional input and assessments. We protected all proprietary information and have not included any proprietary information in this report.

5.1. First Stage Main Engine Gaps and Opportunities *(SOW 5.1)*

Figure 5-1 is a non-proprietary excerpt from our first stage main engine catalogue compiled with input from Aerojet, Pratt and Whitney Rocketdyne and SpaceX. We obtained the RD-171 and and RD-180 data using the NPO Energomash web site. In all cases we used the supplier data as is in both our quick-sizer and full trajectory simulation analysis. The RD-171 and RD-180 are provided as modules including actuators but to be conservative we did not adjust their dry mass for engines only.

		First Stage Main Engine									
Parameter	units	SSME	RS-25E	F-1A	RD-180	RS-84	RD-171	M-2	RS-68A	RS-68B	RS-68C
Oxidizer		LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX	LOX
Fuel		LH2	LH2	RP-1	RP-1	RP-1	RP-1	RP-1	LH2	LH2	LH2
Thrust SL	klbf	416	416	1,800	860	1,049	1,627	1,700	702	702	702
Thrust SL	MN	1.85	1.85	8.01	3.83	4.67	7.26	7.56	3.12	3.12	3.12
Thrust vacuum	klbf	512	512	2,021	933	1,168	1,774	1,940	797	797	797
Thrust vacuum	MN	2.28	2.28	8.99	4.15	5.20	7.90	8.63	3.55	3.55	3.55
ISP SL	sec	367	367	271	311	301	309	283	363	363	368
ISP vacuum	sec	453	453	304	338	335	337	322	412	412	417
Chamber pressure	psia	2,994	2,994	1,161	3,722	2,800	3,560	2,000	1,560	1,560	1,560
Chamber pressure	MPa	20.6	20.6	8.0	25.7	19.3	24.5	13.8	10.8	10.8	10.8
Mixture ratio		6.04	6.04	2.27	2.72	2.70	2.60	2.40	5.97	5.97	5.97
Length	in	168	157	221	140	147	163	186	137	137	137
Length	m	4.27	3.99	5.61	3.57	3.73	4.15	4.72	3.47	3.47	3.47
Engine Diameter	in	96	94	144	124	101	158	138	91	91	91
Engine Diameter	m	2.44	2.38	3.66	3.14	2.57	4.02	3.51	2.32	2.32	2.32
Expansion ratio		70	70	16	37	36	37	30	22	22	22
Mass dry	klbs	7.78	8.10	19.9	12.1	13.0	20.9	12.5	15.1	15.1	12.7
Mass dry	t	3.53	3.68	9.03	5.49	5.90	9.50	5.68	6.88	6.88	5.77
Throttling	%	67%	67%	80%	47%	47%	56%	TBD	55%	55%	55%
Mass flow	t/s	0.51	0.51	3.01	1.25	1.58	2.39	2.72	0.88	0.88	0.86

Figure 5-1 First Stage Main Engine Catalogue

As shown in Figure 5-2 we did not identify any first stage main engine technology gaps.

There are two significant capability gaps, CG01 Domestic LOX/RP first stage engine and CG02 LRB to core cross feed subsystem.

Technology Gaps		
	None	
Capability Gaps		
CG01	Domestic LOX/RP engine 1m bf class	
CG02	LRB to core cross feed subsystem	
Technology Opportunities		
TO01	Reciprocating feed system for lower pressure LRB tanks	TLR3
Capability Opportunities		
CO01	Main propulsion modules	
CO02	Shared core & LRB subsystems	

Figure 5-2 First Stage Main Engine Technology & Capability Gaps & Opportunities

CG01: The F-1A, RS-84 and M-2 are all potential domestic substitutes for the Russian RD-171 and RD-180 engines but none exist today. The F-1A has low sea level ISP and thrust to weight and does not appear to be competitive. The RS-84 and M-2 are competitive but are preliminary design and conceptual design engines respectfully. Clearly the technology exits to produce a competitive domestic LOX/RP first stage engine and we recommend that a competitive development program be initiated with a down select to a single supplier at SDR or perhaps PDR. This should be coordinated with the USAF as a shared upgrade for the Atlas V.

CG02: LRB to core cross feed has useful precedent in the Atlas I and II engine manifold and Shuttle external tank disconnects. Atlas I and II jettisoned booster engines from a common manifold while the sustainer engine continued thrusting at full power. Atlas I had 3 failures in 11 launches that were unrelated to the engine staging. Atlas II had 63 out of 63 successful launches. The common LOX/RP engine manifold is exactly analogous to our recommended LRB tank to core tank to common engine manifold cross feed concept discussed in section 4.2. The Shuttle external tank to orbiter quick disconnects can serve as a reference design for the LRB to core tank quick disconnects. We believe therefore that the cross feed mechanical design has a sound legacy to draw on and is a straight forward development. The significant capability gap is the integrated cross feed pressurization and propellant transfer control system. This should be much easier with our recommended LOX/RP three body first stage than with LOX/LH2.We recommend that NASA lead an integrated ground test bed develop and certify the cross feed control system and that the engine manifold and tank quick disconnects be rigorously tested at the component level. This can all be done with confidence on the ground.

TO01: UAHuntsville has a proprietary reciprocating feed system that has the potential to provide the high pressure propellant required for pressure fed engines or the moderate pressure propellant required for pump fed engines from a small set of tanks that are sequentially filled, pressurized and vented. This could be phased in once the technology was proven as a preplanned improvement without changing engines. This technology could also enable pressure fed first stage engines that could be much simpler, more reliable and lower cost than pump fed engines since the large tanks are at low pressure with good mass fractions. Considerable investigation is required to validate the benefits and sort through an implementation path for first stage pressure fed engines. Contact Northrop Grumman or UAHunstville Propulsion Research Center for more information.

We identified two first stage main engine capability opportunities CO01 main propulsion modules and CO02 shared core and LRB subsystems.

CO01: Our recommended NK206-2 configuration has three engine LRBs and a two engine core. This offers the opportunity to integrate engines, their controllers and other components in three engine and two engine main propulsion modules. We believe there are considerable benefits to this approach in production flow, cost and reliability. This approach also facilitates the single stick variants within the NK206 configuration family. We believe this is a straight forward development that can be executed as part of the basic program.

CO02: Our recommended three body LRB and core configuration is similar to the Delta IV heavy but the LRBs are not independent stages. This offers the opportunity to share subsystems between the core and LRBs and maybe some with the upper stage. Avionics is an obvious potential shared subsystem and is in our baseline. The pressurization and feed subsystem and the thrust vectoring and control subsystem are other strong candidates. Again we believe this is a straight forward development that can be executed as part of the basic program.

5.2. Upper Stage Main Engine Gaps and Opportunities *(SOW 5.2)*

Figure 5-3 is a non-proprietary excerpt from our upper stage main engine catalogue compiled with input from the engine suppliers. The engine suppliers proposed a number of proprietary next generation engines. We have listed a LOX/LH2 upper stage NGE engine with target thrust level and ISP suitable for an In-space Cryo-propulsion Stage (ISCPS). As discussed in section 4.7 the J-2X is a good choice for an upper stage and escape burn stage and we recommend that its development be completed for the SLS.

Figure 5-4 shows that we have not identified any gaps for upper stage main engines. We have identified capability opportunity CO03 for a retractable nozzle. The J-2X has a large nozzle for high altitude and vacuum performance. The large nozzle drives a very long interstage which

		Upper Stage Main Engine				
Parameter	units	J-2X	RL10A-4	RL10B-2	RL10C-1	NGE
Oxidizer		LOX	LOX	LOX	LOX	LOX
Fuel		LH2	LH2	LH2	LH2	LH2
Thrust vacuum	k bf	294	22	25	23	100
Thrust vacuum	MN	1.31	0.10	0.11	0.10	0.44
ISP vacuum	sec	453	452	466	450	466
Chamber pressure	psia	1,337	611	637	624	TBD
Chamber pressure	MPa	9.22	4.21	4.39	4.30	
Mixture ratio		5.50	5.50	5.88	5.50	
Length	in	184	90	164	86	
Length	m	4.66	2.29	4.15	2.19	
Engine Diameter	in	116	45	84	58	
Engine Diameter	m	2.96	1.14	2.13	1.46	
Expansion ratio		92	84	285	130	
Mass dry	klbs	5.59	0.398	0.664	0.450	
Mass dry	t	2.54	0.18	0.30	0.20	
Throttling	%	NA	NA	NA	NA	
Mass flow	t/s	0.29	0.02	0.02	0.02	

Figure 5-3 Upper Stage Main Engine Catalogue

drives total SLS stack height and impacts the maximum bending moment. In addition the interstage does not have pressure relief like the tanks and has correspondingly higher axial inertial loads. A retractable J-2X nozzle may prove to be a good trade and more than offset its delta mass and cost by interstage mass and cost savings.

Technology Gaps		
	None	
Capability Gaps		
	None	
Technology Opportunities		
	None	
Capability Opportunities		
CO03	Retractable nozzle for shorter interstage	

Figure 5-4 Upper Stage Main Engine Technology & Capability Gaps & Opportunities

5.3. Other SLS Gaps and Opportunities *(SOW 5.3)*

Figure 5-5 shows that we have identified two other SLS capability gaps CG03 and CG04.

CG03: There have been two recent payload fairing separation subsystem failures impacting important missions. The SLS fairing is by far the largest fairing ever proposed and twice the diameter of the largest fairings currently in use. SLS payloads and missions will be correspondingly higher value. Very high reliability separation systems are therefore required with demonstrated reliability through extensive testing. Redundancy may also be required. In addition the SLS fairings require an operations approach that is practicable for their size. We believe this gap can be closed with existing technologies through an extensive certification and acceptance program. The thermal vacuum chamber at NASA's Glenn Research Center Plum Brook Station is suitable for development and certification testing but of course the test fairings will need to be assembled nearby or onsite. Recurring acceptance testing needs to be near or at Kennedy Space Center and will therefore have to be at ambient conditions. This will drive the subsystem design and certification approach. Fairing operations will require huge alignment fixtures or state of the art determinant assembly methods.

Technology Gaps		
	None	
Capability Gaps		
CG03	Reliable and operable separation subsystems for very large payload fairings	
CG04	Autonomous rendezvous and docking subsystems for very large high fluid mass fraction exploration elements	
Technology Opportunities		
TO02	Digital pyrotechnic separation system safe, arm and initiation subsystems	TLR7
TO03	Mechanical latch and release subsystems	TRL6
Capability Opportunities		
CO04	Composite cryo-tanks	
CO05	Composite common bulkheads	

Figure 5-5 Other SLS Technology & Capability Gaps & Opportunities

CG04: The former Soviet Union and Russia routinely conduct autonomous rendezvous and docking (AR&D) and we clearly have the technology. But AR&D has not been done for the very large elements with 90% fluid mass planned for future exploration beyond Earth orbit. The computing power, algorithms and appropriate sensitivity sensors exist but an integrated subsystem and operations design needs to be developed and the mechanical approach needs to be selected and developed. Much of this can be done via simulation and ground testing. The exploration mission planning should incrementally prove this capability backed up by continued ground simulation and specific testing.

Figure 5-5 also indentifies other SLS technology and capability opportunities.

TO02: Ensign Bickford Aerospace and Defense (EBA&D) informed us on the potential benefits of modern digital pyrotechnic safe, arm and initiation subsystems. This approach uses the vehicle bus instead of discrete wiring and should better enable built in testing (BIT) and reduce access requirements and the associated risk of induced damage during access.

TO03: Planetary Systems Corporation has deployed mechanical latch and release subsystems for many payloads and we believe there is potential to extend this technology to payload fairing separation subsystems. This may greatly simplify the certification and acceptance issues discussed in CG03 through demonstrated reliability and cycle testing of actual mission hardware.

CO04: We believe the technology exists for composite cryo-tanks and the production capability can be put in place for the 6m diameter LOX, RP and LH2 tanks required for our recommended NK206-2 configuration. Previous and current NASA programs have demonstrated the required technologies and the composites production know how exists in the aircraft industry. Composite construction is particularly suitable but is not required for our recommended NK206-2 configuration. The distributed loads and common diameters enable a common composites production infrastructure with shared low fixed costs for all elements and vehicles in the NK206 family.

CO05: Common bulkheads were successfully developed for the LOX/LH2 Saturn SII and SIVB stages and are used routinely on current LOX/RP and LOX/LH2stages. The reduction in stack height and shared pressure loads is attractive at the vehicle level but controlling metallic construction to tight enough tolerances for film adhesive bonded common bulkheads is daunting. Fortunately composites common bulkheads can be co-cured or co-bonded where uncured composite facesheets naturally conform to shape and wash out any film adhesive bonding critical tolerances. We believe all the technology exists for LOX/RP composite common bulkheads. For LOX/LH2 composite common bulkheads the technologies successfully tested on the Space Launch Initiative program composite LH2 cryo-tank program need to be applied to a common bulkhead application. A nested bulkhead similar to the Arianne 5 advanced upper stage might be a good compromise. We believe LOX/RP common bulkheads should be baselined for NK206-2 and the LOX/LH2 upper stage traded during preliminary design.

5.4. In-Space Propulsion Gaps and Opportunities *(SOW 5.4)*

Figure 5-6 indentifies three technology gaps, one capability gap, one technology opportunity and two technology opportunities for in-space propulsion.

Technology Gaps		
TG01	Long duration LH2 storage	TRL3
TG02	In-space cryo-transfer	TRL3
TG03	Long duration high reliability LOX/LH2 main propulsion system	TRL5
Capability Gaps		
CG05	In-space LOX/LH2 engine 100 klbf class	
Technology Opportunities		
TO04	Reciprocating feed system pressure fed engines with low main tank pressure	TLR3
Capability Opportunities		
CO06	Main propulsion servicing	
CO07	Dedicated in-space nozzles (plug, aerosp ke, etc)	

Figure 5-6 In-space Propulsion Technology & Capability Gaps & Opportunities

TG01: Long duration LH2 storage has been identified by many forums and reports as the number one technology need for exploration beyond Earth orbit. We agree and have supported this assessment in our reports, RFI responses and at the March 22, 2011 National Research Council panel on space propulsion. We believe a system solution can be found using our vacuum jacketed composite cryo-tank technology with current best technology multi-layer insulation and smart application of current technology cryo-coolers. The system solution should evaluate productive uses for boil off hydrogen that relax the LH2 storage requirements.

TG02: In-space cryo-transfer of LOX and LH2 is the second highest technology need for exploration beyond Earth orbit. Fortunately, as discussed in section 4.6, in-space refueling by cryo-transfer is not required for initial exploration and there is plenty of time to develop solutions. Micro-gravity propellant scavenging technologies are a candidate but may not be required. Current technology dedicated settling thrusters are always possible but will add mass and cost. Vehicle system level and operations solutions should be evaluated along with new technologies. Potential system level solutions include using attitude control system thrusters for propellant settling and another is to use a main propulsion start subsystem to settle propellant and apply a small acceleration for fluid transfer. Operationally required burns including ascent, depot orbit maintenance, escape, course correction, destination insertion and Earth return should all be taken advantage of to use their acceleration to settle and transfer propellant.

TG03: As discussed in Section 4.7 there is a compelling need for a highly reliable long duration in-space main propulsion system for human exploration beyond Earth orbit without a free return. We would categorize this as a capability gap if not for the very long in-space duration and extensive quiescent periods. Long term space vacuum and in-space radiation may alter the effectiveness of engine seals and other materials. Testing the vacuum effects is hard to do on the ground but could be done at the International Space Station (ISS). There is not any effective accelerating aging method for vacuum effects so a long exposure test at the ISS may be required. Deep space radiation effects cannot be duplicated at the ISS due to the Van Allen belts but can be tested on the ground using particle accelerators such as the Brookhaven National Laboratory cyclotron. Accelerated testing may be possible for radiation effects by increasing the radiation intensity. Combined effects testing can be done as a secondary objective for unmanned missions beyond Earth orbit. Fortunately there is time to incrementally develop and test long duration in-space reliability as our human exploration program beyond Earth orbit expands over the decades.

CG05: Also discussed in Section 4.7 is the need for a right sized in-space LOX/LH2 engine. This is clearly a capability gap to size the engine correctly if the technology gap described in TG03 above is treated separately.

TO04: The UAHuntsville reciprocating feed system described in technology opportunity TO01 for the first stage main engine has more potential for in-space cryo-propulsion systems (ISCPS). Once developed this technology can help solve technology gaps TG02 and TG03. Pressure fed instead of pump fed in-space propulsion should be more reliable and less vulnerable to the deep space environment. This technology merits a dedicated development program. We recommend that a task order be established under the Research and Technologies for Aerospace Propulsion Systems (RTAPS) program or other contracting mechanism.

CO06: In-space propulsion system servicing is an important opportunity for human exploration beyond Earth orbit. The exploration spacecraft can be configured to accommodate propulsion system servicing by packaging critical propulsion components in accessible pressurized compartments. Built in test and simulations should also be included so the crew can rehearse mission burns during transit or in destination orbit and take corrective servicing actions as required. This will greatly improve in-space propulsion system mission reliability and crew independence and confidence. We recommend that ISCPS system study program be established with multiple contractor participation similar to this SLS BAA study.

CO07: In-space propulsions systems may prefer alternative nozzles from launch vehicle nozzles and there is an opportunity to explore plug, aero-spike and other approaches. This may provide better plume expansion for in-space performance and may enable alternative general arrangements that facilitate in-space servicing or other desirable capabilities. This opportunity can also be included in the ISCPS system study recommended above

5.5. Other In-Space Element Gaps *(SOW 5.5)*

Figure 5-7 identifies two technology gaps, one capability gap, one technology opportunity and one capability opportunity for other in-space elements. These are not specific to the Space Launch System (SLS) but impact the payload requirements for the SLS and the multi-purpose upper stage versus dedicated in-space cryo-propulsion stage (ISCPS) trade discussed in Section 4.7.

Technology Gaps		
TG04	Galactic cosmic ray (GCR) radiation protection	TRL?
TG05	Solar particle event (SPE) detection and severity assessment	TRLX
Capability Gaps		
CG06	Solar radiation event radiation protection	
Technology Opportunities		
TO05	Nuclear thermal propulsion	TRL3
Capability Opportunities		
CO08	LOX/LH2 propellant depots	

Figure 5-7 Other In-space Technology & Capability Gaps & Opportunities

TG04: Crew galactic cosmic ray (GCR) radiation protection is a very serious gap for long duration human exploration beyond Earth orbit. Shielding methods can be effective against solar particle events (SPEs) but not against GCR radiation. Secondary radiation from spacecraft and shielding materials also need better understanding. In addition there are large biological uncertainties that limit the ability to evaluate risks and effectiveness of mitigation actions. The most effective method is to minimize human exposure by shortening the crew transit time. This will require much more LOX/LH2 than currently envisioned or nuclear propulsion. The very large payloads to escape velocity enabled by LOX/LH2 propellant refueling discussed in Section 4.5 Figure 4-8 may shorten transit times sufficiently. Decoupling crew and cargo delivery, developing a dedicated ISCPS, providing the additional LOX/LH2 for a more rapid crew transit and sending unmanned elements and cargo by minimum delta V transit will likely be necessary for high energy near Earth object (HENEO) missions. Other propulsion approaches such as nuclear thermal may be required to solve the GCR exposure issue long term. Fortunately we can begin human exploration beyond Earth orbit with shorter missions and develop means to solve GSR exposure over time. The NASA Human Research Program (HRP), Space Radiation Program Element (SRP) needs to be fully supported and we recommend that a rapid crew transit study be established with industry similar to this SLS study.

TG05: Solar particle event (SPE) forecasting and alerts with accurate hazard predictions is a gap for human exploration beyond Earth orbit. Current methods will likely over predict hazards driving unnecessary crew shelter requirements. Better correlation between solar observations and SPE radiation levels and timelines are needed. This approach will also benefit predicting SPE hazards to general Earth orbiting assets and ground systems helping society as a whole. Solar weather buoys could be deployed to measure SPE radiation levels directly but the quantity of buoys necessary to cover a sufficient solar radiation window is likely prohibitive. A limited number of buoys may be sufficient to correlate solar observations. This gap is clearly beyond the scope of the SLS but the SLS could be used to deploy new solar observatories and or solar weather buoys. Solar weather buoys could be deployed as secondary mission objectives during exploration mission transits.

CG06: Solar particle event (SPE) crew radiation protection is a capability gap that can be solved for human exploration beyond Earth orbit by configuring the spacecraft to utilize propellant, potable water and waste water as shielding for crew compartments. This can be concentrated for a storm shelter and or crew sleeping compartments.

TO05: Nuclear thermal propulsion technology development should be revived as a potential longer term solution to crew transit times and GCR radiation exposure. This beyond the scope of the SLS but the SLS can launch a nuclear thermal propulsion spacecraft and provide LH2 to LEO as a propellant.

CO08: LOX only depots can greatly expand an SLS multi-purpose upper stage payload to escape velocity as discussed in Section 4.5 Figure 4-8. LOX and LH2 depots expand the payload capacity by a further increment and can also provide LH2 for nuclear thermal propellant and as radiation shielding. As the technology gaps TG01 and TG02 are resolved at first LOX and then LOX/LH2 depots can be deployed. LOX and LH2 deliveries can be a commercial commodity buy or can be additional SLS non-critical missions.

5.6. In-Space Element Flight Demonstrations *(SOW 5.6)*

Figure 5-8 summarizes the recommended in-space cryo-propellant management flight demonstration we submitted as part of the Flagship Technology Demonstration request for information NNH10ZTT003L. This in-space demonstration properly supported by ground texting matures passive and active cooling methods for long term cryo-storage, cryo-fluid transfer, cryo-coupling mechanisms and cryo-propellant conditioning for engine start. Please refer to the NNH10ZTT003L RFI response for more details.

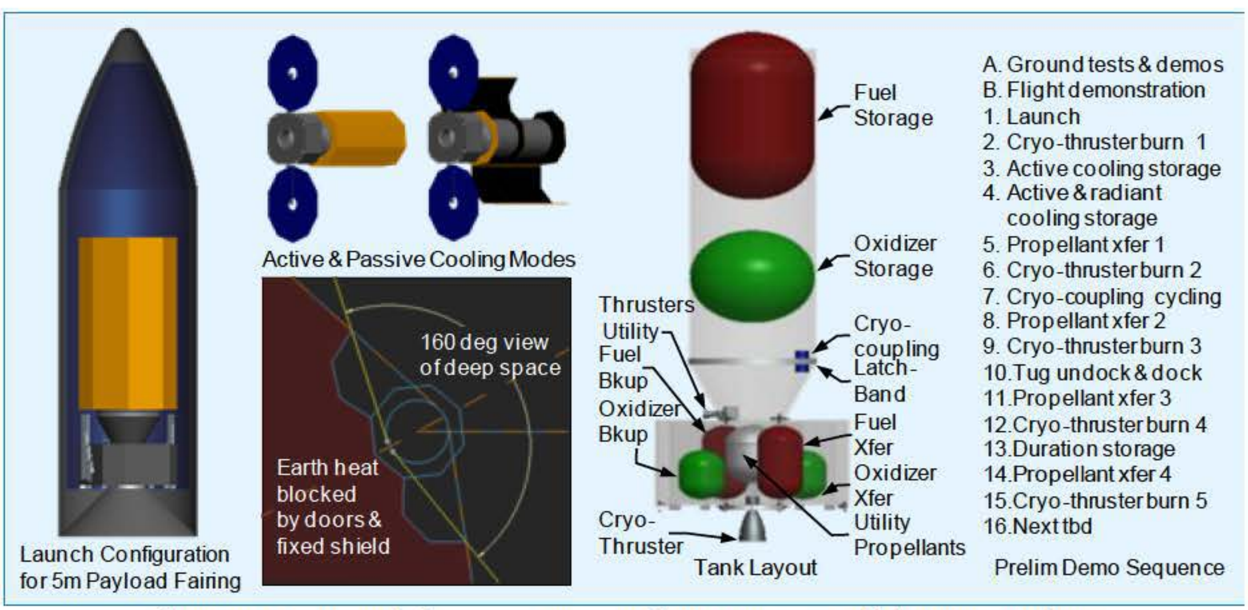


Figure 5-8 Recommended In-space Cryo-propellant Management Flight Demonstrations

6. Appendix

6.1. Study Technical Statement of Work (SOW)

The following Figures 6-1 through 6-5 are the technical paragraphs from our study contract SOW. We included them in this appendix to assist the reader in correlating the SOW paragraph numbers added in italics to relevant section heading of this report to SOW tasking.

2.0 HEAVY LIFT LAUNCH VEHICLE (HLLV) DECISION ATTRIBUTES

The contractor shall develop and define a set of decision attributes including variations to guide the definition and assessment of a wide variety of HLLV system options.

2.1 Document HLPT STUDY Ground Rules and Assumptions
The contractor shall receive and review government provided HLLV study ground rules and assumptions (GR&A) after contract award. The contractor shall recommend alternative GR&A and variations to GR&A to help guide a thorough exploration of HLLV options.

2.2 Define HLPT Study Figures of Merit
The contractor shall synthesize figures of merit (FOM) with variations to support the analytical assessment and comparison of HLLV options. FOMs shall be objective and supported by calculation with care to define significance and uncertainty.

2.3 Define Nominal Weighting Factors and Sensitivity Range
The contractor shall define a set of weighting factors suitable for adding FOMs together into objective summary scores for various HLLV options. The weighting factors shall reflect the relative importance between FOMs. The contractor shall determine a sensitivity range in the weighting factors to support sensitivity and uncertainty analyses.

Figure 6-1 SOW J-1-1 Paragraph 2

3.0 EXPLORATION ARCHITECTURE AND HLLV CONFIGURATION OPTIONS

The contractor shall define a sufficient set of exploration architectures and HLLV configuration options to thoroughly represent the available trade space. This set of options shall be a tool to capture technology opportunities (technology push) and later a tool to assess technology and capability gaps (technology pull).

3.1 Define Exploration Architecture Options
The contractor shall define a set of exploration architectures for human space exploration beyond earth orbit. Each architecture shall support a variety of missions over many decades and not be suboptimized for any single destination. The architectures shall consider options for inspace refueling, various propellants, advanced in-space propulsion, reusability, multifunctionality, affordability and commonality. This architecture definition shall be at a conceptual level and shall avoid false resolution details and cosmetic graphics.

3.2 Define HLLV Configuration Options for Alternative GR&AS
The contractor shall define a set of HLLV configuration options for the study GR&As established in Subsection 2.1 and the architectures defined in Subsection 3.1. This HLLV configuration definition shall be at the conceptual level. Each configuration shall be appropriately defined by a general arrangement computer-aided design (CAD) model and configuration description package (CDP). CAD models shall be at the conceptual definition level without false resolution or resource consuming details. The CDP shall be a MS PowerPoint briefing package with CAD screen shots, basic dimensions, major equipment tables, and a concept of operations diagram. Lengthy narratives are not desired.

3.3 Define Innovative or Nontraditional Processes or Technologies
The contractor shall identify innovative or non-traditional processes or technologies that can be applied to the heavy lift systems to dramatically improve its affordability and sustainability. Processes shall include business and organization concepts as well as technical design and manufacturing methods. Technologies shall include new or emerging as well as novel ways of applying existing technologies from other industries or applications.

3.4 Define Commonality Opportunities
The contractor shall identify aspects of a heavy lift system (including stages, subsystems, and major components) that could have commonality with other user applications, including NASA, Department of Defense (DoD), commercial, and international partners.

3.5 Define Incremental Development Options
The contractor shall identify incremental development testing, including ground and flight testing, of heavy lift system elements that could enhance the heavy lift system development. This shall include the synergism between nonrecurring development certification testing, recurring acceptance testing and flight operations vehicle health and performance monitoring

Figure 6-2 SOW J-1-2 Paragraph 3

4.0 HLLV STUDY CONFIGURATIONS ANALYSIS

The contractor shall analyze the set of exploration architectures and HLLV configuration options defined in Section 3 against the decision attributes defined in Section 2. Care shall be taken to keep this analysis at the conceptual level with all options analyzed to an equivalent depth without false resolution or unnecessary details. This analysis shall be used to help identify and prioritize the highest leverage capability gaps in Section 4.

4.1 Analyze Architecture Sensitivity to Weighting Factors
The contractor shall analyze the sensitivity of the architectures defined in Subsection 3.1 against the range in weighting factors established in Subsection 2.3.

4.2 Analyze Alternative GR&AS Impact
The contractor shall analyze the alternative HLLV configurations defined in Subsection 3.23.1 against the figures of merit and range in weighting factors established in Subsections 2.2 and 2.3.

4.3 Analyze Innovative or Nontraditional Processes or Technologies Impact
The contractor shall analyze the alternative HLLV configurations defined in Subsection 3.33.1 against the figures of merit and range in weighting factors established in Subsections 2.2 and 2.3.

4.4 Analyze Commonality Opportunities Impact
The contractor shall analyze the commonality opportunities defined in Subsection 3.43.23.1 against the figures of merit and range in weighting factors established in Subsections 2.2 and 2.3.

4.5 Analyze Incremental Development Options Impact
The contractor shall analyze the incremental development options defined in Subsection 3.53.1 against the figures of merit and range in weighting factors established in Subsections 2.2 and 2.3.

Figure 6-3 SOW J-1-3 Paragraph 4

5.0 HLPT CAPABILITY GAP ANALYSIS

The contractor shall identify capability gaps associated with the set of architectures and HLLV configurations defined and analyzed in Sections 3 and 4. For each capability gap the contractor shall identify specific areas where technology development may be needed. Items identified as requiring technology development shall be quantitatively evaluated using established metrics, e.g. NASA Technology Readiness Level (TRL), Capability Readiness Level (CRL), Manufacturing Readiness Level (MRL), Process Readiness Level (PRL).

5.1 First Stage Main Engine Gaps
The contractor shall identify capability gaps associated with the first-stage main engine functional performance and programmatic characteristics required to support each HLLV configuration defined and analyzed in Sections 3 and 4. The minimum set of functional performance characteristics identified shall include engine thrust, specific impulse (*Isp*), mixture ratio, mass, throttle range, and physical envelope. This assessment shall include Liquid Oxygen/Rocket Propellant (LOX/RP), Liquid Oxygen/Liquid Hydrogen (LOX/LH2), and Liquid Oxygen/Methane (LOX/CH4) main engine systems. The minimum set of programmatic characteristics identified shall include an estimated overall life-cycle cost (i.e., Design, Development, Test, and Evaluation (DDT&E), production and operations (fixed and variable) per engine cost), development schedule, and production rate. The contractor shall identify any impacts to overall life-cycle costs of the heavy lift system based on the engine studied.

5.2 Upper Stage Main Engine Gaps
The contractor shall identify capability gaps associated with the upper-stage main engine functional performance and programmatic characteristics required to support each HLLV configuration defined and analyzed in Sections 3 and 4. The minimum set of functional performance characteristics identified shall include engine propellants, thrust, *Isp*, mixture ratio, mass, throttle range, and physical envelope. The minimum set of programmatic characteristics identified shall include an estimated overall life-cycle cost (i.e., DDT&E, production and operations (fixed and variable) per engine cost), development schedule, and production rate. The contractor shall identify any impacts to overall life-cycle costs of the heavy lift system based on the engine studied.

Figure 6-4 SOW J-1-4 Paragraph 5

5.0 HLPT CAPABILITY GAP ANALYSIS (cont.)

5.3 Other HLLV Gaps
The contractor shall identify capability gaps associated with all other technical aspects of each HLLV configuration defined and analyzed in Sections 3 and 4, e.g., tanks, propellant and pressurization systems, integrated system health management, auxiliary propulsion systems, avionics and control systems, structures. The contractor shall identify test and integrated demonstrations to mitigate risk associated with the gaps.

5.4 In-Space Space Propulsion Gaps
The contractor shall identify capability gaps associated with the in-space space propulsion elements functional performance and programmatic characteristics required to support each configuration defined and analyzed in Sections 3 and 4. This assessment shall include LOX/H2 and LOX/CH4 propulsion systems. The minimum set of functional performance characteristics identified shall include propellant definition, thrust, *Isp,* mixture ratio, mass, throttle range (if any), and physical envelope. The minimum set of programmatic characteristics identified shall include an estimated overall life-cycle cost (i.e., DDT&E, production and operations (fixed and variable) per engine cost), development schedule, and production rate. The contractor shall identify any impacts to overall life-cycle costs of the heavy lift system based on the engines studied.

5.5 Other In-Space Element Gaps
The contractor shall identify capability gaps associated with all other technical elements of the in-space space propulsion elements defined and analyzed in Sections 3 and 4, e.g., tanks, propellant and pressurization systems, cryogenic fluid management, integrated system health management, auxiliary propulsion systems, avionics and control systems, structures, autonomous rendezvous and docking. The contractor shall identify test and integrated demonstrations to mitigate risk associated with the gaps.

5.6 In-Space Element Flight Demonstrations
The contractor shall identify what in-space space propulsion elements, if any, which should be demonstrated via space flight experiments.

Figure 6-5 SOW J-1-5 Paragraph 5 continued

REPORT DOCUMENTATION PAGE	*Form Approved* *OMB No. 0704-0188*

1. REPORT DATE *(DD-MM-YYYY)*
06-03-2011

2. REPORT TYPE
Final Technical Report

3. DATES COVERED *(From - To)*
Dec 2010 - Jun 2011

4. TITLE AND SUBTITLE
NASA Space Launch System – Systems Analysis and Trade Study

5a. CONTRACT NUMBER
NNM11AA11C

5b. GRANT NUMBER

5c. PROGRAM ELEMENT NUMBER

6. AUTHOR(S)
[redacted] (b) (4)

5d. PROJECT NUMBER

5e. TASK NUMBER

5f. WORK UNIT NUMBER

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
Advanced Programs & Technology

Northrop Grumman Systems Corporation
One Hornet Way
El Segundo, CA 90245-2804

8. PERFORMING ORGANIZATION REPORT NUMBER

9. SPONSORING / MONITORING AGENCY NAME(S) AND ADDRESS(ES)

NASA /Marshall Space Flight Center
Office of Procurement
Marshall Space Flight Center, AL 35812

10. SPONSOR/MONITOR'S ACRONYM(S)
NASA MSFC

11. SPONSOR/MONITOR'S REPORT NUMBER(S)

12. DISTRIBUTION / AVAILABILITY STATEMENT
ITAR Limited Distribution

13. SUPPLEMENTARY NOTES

14. ABSTRACT
This is the Northrop Grumman final report for the Space Launch System (SLS) Systems Analysis and Trade Study performed for the National Aeronautics and Space Administration (NASA) Marshall Space Flight Center. The SLS configuration trade space has been thoroughly explored with a compelling focus on sustainability. Key sustainability drivers are; incremental development within available funding profile, low production and operations fixed costs and SLS flexibility. Hundred of SLS configurations were assessed using the Northrop Grumman quick-sizer launch vehicle conceptual design tool. Promising concepts representing the entire trade space were then sized using a full trajectory multi-disciplinary optimization simulation. Computer aided design solid models were built to verify configuration geometry. Payload fractions, mission reliability and lifecycle costs were analyzed for a variety for human space exploration missions from low Earth orbit to Mars surface. Configurations were ranked using a Pugh relative rating method with a variety of weighting factors to test sensitivities for near and long term emphasis. Many good SLS configuration options were identified. Northrop Grumman recommends a three body liquid oxygen rocket propellant (LOX/RP) first stage with a liquid oxygen liquid hydrogen (LOX/LH2) multi-purpose upper stage. This configuration ranks well for all criteria and sensitivities. The major discriminator for the recommended configuration is the robust family of SLS vehicles that can be incrementally developed, using a single production system, with shared fixed costs, for a wide variety of exploration, science and national security missions. This SLS family will be sustainable for many decades of space exploration.

15. SUBJECT TERMS
Space Launch System (SLS), National Aeronautics and Space Administration (NASA), Marshall Space Flight Center, Sustainable, Exploration beyond Earth orbit, Northrop Grumman

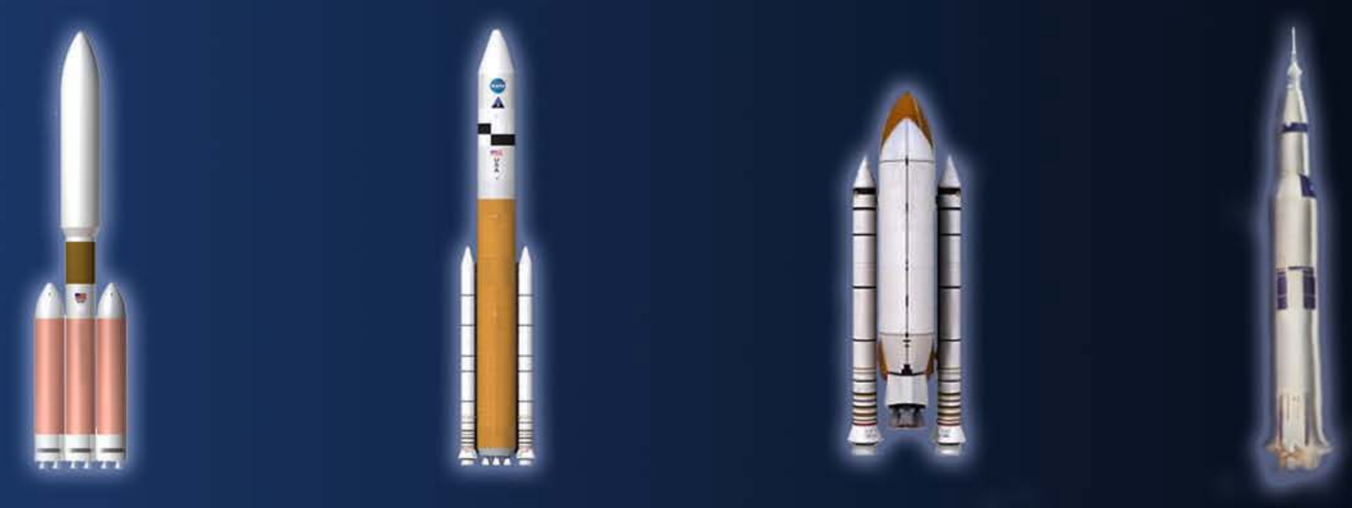
16. SECURITY CLASSIFICATION OF: U			17. LIMITATION OF ABSTRACT	18. NUMBER OF PAGES	19a. NAME OF RESPONSIBLE PERSON
					James Casper, CO
a. REPORT U	**b. ABSTRACT** U	**c. THIS PAGE** U	UU	79	**19b. TELEPHONE NUMBER** *(include area code)* 256-544-3294

Standard Form 298 (Rev. 8-98)
Prescribed by ANSI Std. Z39.18

ULA
United Launch Alliance

Heavy Lift and Propulsion Technology Final Report

NNM11AA16C | July 2011



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1.0 INTRODUCTION

United Launch Alliance (ULA) is pleased to support NASA on the Heavy Lift and Propulsion Technology (HLPT) Trade Study. With ULA's legacy of over 1,300 launches from the Atlas and Delta programs we hope that our study insights, merged in the marketplace of ideas coming from other HLPT contractors, will help NASA identify the best possible SLS concepts going forward.

1.1 Overview

The HLPT Study has been interesting because of the flexibility NASA provided to contractors write their own statements of work. Each contractor, based on their companies strengths and interests, have emphasized different aspects of the trade. Some have focused on vehicles or cost estimating, others on the system engineering process. Some contractors focused on broad tradespace, and still others focused more deeply on individual concepts. Taken together these contracts will have provided NASA a greater breadth of study than they could have received from a single constrained statement of work.

ULA's approach has been to have a strong vehicle focus, including vehicles in the tradespace that leveraged EELV synergies to help NASA affordability. Though the vehicle families introduced are EELV centric, we attempted to make a very fair assessment of their advantages and disadvantages relative to a Shuttle derived reference vehicle. Another area of focus has been cost estimating looking at vehicle evolutions rather than point designs. Rather than doing traditional bottoms-up estimates, we used a subcontractor, Advatech Pacific, to help us with parametric cost estimating due to the bread of the tradespace. We feel that this has been a very successful approach, creating a more even assessment across configurations, and allowing us to share cost estimates that might have otherwise been too competition sensitive if they had been based on traditional bottoms-up estimating methods. Finally, we included the propellant depot dimension into several of these scenarios to help NASA better define the architecture level implications of depots.

1.2 Affordability

During the course of the study an emphasis on helping SLS affordability has been a clear priority. The focus of our tradespace has been to try and leverage ULA affordability lessons learned to address this critical need. Our existing EELV program provides valuable insights, both positive, and negative to help address affordability.

The tenants of these affordability principles are:

1. <u>Share Engines and Engine Contractors with the Air Force</u>: Obvious opportunities exist for leveraging the same engines, or at least same engine suppliers to avoid NASA or the Air Force carrying the full overhead of these

producers. The loss of synergy between the AF and NASA with the retirement of SSME, makes this a very real issue for ULA. Though a the current time the impact is felt on the Air Force side, the same can be true for NASA if it chooses NASA unique engines vs. engines that share industrial base with the Air Force.

2. Eliminate Dedicated Facility Cost: Another large opportunity is to leverage the Decatur manufacturing facility, 30 minutes from NASA/MSFC with overhead paid for by the Air Force vs. restarting the dedicated Michoud facility. This is obviously true for 4-5m diameter stage configurations, but ULA also has provided separate data on Decatur's ability to support larger diameter production at reasonable cost. Choices between solids utilizing the dedicated Promontory Utah facility and liquid engines, which can share overhead with the Air Force is another example

3. Get Build Rates Up: The SLS system is burdened with the likelihood that affordability constraints will allow the SLS vehicle to fly only once or twice a year, far below the optimal production rate. By leveraging modular concepts, actual build rates can be raised into an efficient rate, even if the launch rate remains low. One concept shown has 7 Delta IV cores strapped together to make a single booster. If it was flown twice a year we would be producing 14 additional cores per year, which combined with very similar EELV cores, would put the Decatur factory at peak efficiency even while flying SLS at a suboptimal flight rate.

4. Pursue Commercial Development Environment to the Greatest Extent Practical: ULA recognizes that the SLS will be a NASA owned system, that will be supported by an array of contractors regardless of which configuration is selected. There still remain significant opportunities for NASA to move in the direction of commercial procurement and development to reduce cost of development and operation.

5. Graceful, Incremental Evolvability: Defining the SLS as a point design which is scarred by requirement 25 years in the future drives high peak funding requirements during the development phase, and builds the SLS around requirements that are certain to change. Why do we need a 10m diameter fairing? No elements exist that define that requirement. An oversized vehicle drives a high carrying cost that may reduce what missions can be funded nearer term. An evolvable design with incremental development steps keeps the system affordable, and support a smaller design team that is potentially sustained active over decades.

6. Get the Requirements Defined Early: The success stories of Atlas and Delta on hitting non-recurring budgets (eg Atlas III, EELV) have been on programs with well defined, stable requirements. Requirements stability has a huge impact on development schedule risk, and in turn on cost. SLS suffers from the extra challenge of large mission uncertainty. Solid definition of missions before embarking on full scale development of SLS may be the best investment NASA can make to restrain non-recurring, and recurring costs.

1.3 Trade Plan

ULA's approach to the study, is shown in figure 1.3-1. Leading up to TIM 1 we performed the vehicle trade looking at trading the best vehicle in each of three performance classes with multiple figures of merit. Each performance class (70t, 100t, 130t) vehicles was evaluated independently. At this stage of the study vehicles were treated as standalone developments rather than as part of larger architectures. They were compared on a basis of 2 flights per year rather than having their flight rate be tied to a mission model. During the course of the study we increased the scope of the vehicle trade to go into greater depth on the vehicles including structural concepts and operations concepts, given great interest on the part of the NASA customer community. This vehicle trade culminated in a Kepner Tregoe trade analysis.

Separately we formulated an overall mission model based on various NASA and industry sources looking at missions commonly discussed for Flexible Path. Though initially planned to be a parallel path from the Vehicle trade as shown on the flow chart, in reality this occurred later in the study.

ULA performed a depot assessment. This was descoped somewhat from the original plan due to the greater emphasis on Vehicle definition mentioned above. ULA provided reviews of depot concepts with the NASA customer, and build depot scenarios into three of six evolution scenarios evaluated subsequently

Six evolutions were defined, characterized by with lower capability vehicles (eg 70t) initially and evolving to larger performance class vehicles when the Asteroid mission appeared in the ULA mission model in 2025. These leveraged the same families of vehicles from the TIM 1 vehicle trade, though we modified some of the concepts based on lessons learned from earlier in the study. Also, for propellant depot scenarios, depots were introduced instead of larger heavy lifters to address these more strenuous missions. These six evolutions were evaluated against the mission model, with performance differences reflected in different numbers of flights, and associated cost differences. The evolution scenarios were evaluated in a separate Kepner Tregoe analysis, using architectures level figures of merit vs vehicle level figures of merit.

The final part of the study was gap analysis to evaluate technology needs in light of lessons learned from the study.

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2.0 VEHICLE TRADE

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2.8 Cost Trade

The cost methodology is the same as that shown in later section. In general costs in the architecture trade received more refinement than the vehicle trade. The estimates were based on 2010 dollars. For all vehicles we have assumed Pad 39 as the launch site. A 2 flight/year flight rate has been assumed for all vehicles.

Results from vehicle cost analysis are shown in figure 2.8-1 which summarize development and production cost, but do not include fixed costs such as fixed recurring (standing army), or launch pad and factory impacts.

	5m RP/LOX Derived Family					
	Dual RD-180 Engine Family			New Dual 1.25Mlb ORSCEngine Family		
Perf Class	70t	100t	130t	70t	100t	130t
Vehicle Description	3 Dual RD-180 5m Boosters + 4 RL-10 A4 US – No Solids	3 Dual RD-180 5m Booster + 1 J-2X US + 8 Gem-60 Solids	5 Dual RD-180 5m Booster + 1 J-2X US – No Solids	3 Dual 1.25 Mlb SC 5m Booster + 4 RL-10 A4 – No Solids	3 Dual 1.25 Mlb SC 5m Booster + 4 RL-10 A4 US + 8 GEM-60 SRBs	5 Dual 1.25 Mlb SC 5m Booster + 4 RL-10 A4 US – No Solids
Development Costs	$2,613,662,662	$4,161,000,000	$4,176,000,000	$6,814,000,000	$6,878,000,000	$6,948,000,000
Unit Production Costs	$366,856,500	$492,000,000	$537,000,000	$443,000,000	$545,000,000	$642,000,000

	Delta IV Derived Family				
	1 J-2X US Family		3 J-2X US Family		
Perf Class	70t	100t	70t	100t	130t
Vehicle Descript on	7 CBC + 1 J-2X – No Sol ds	7 CBC + 1 J-2X + 12 Solids	7 CBC + 3 J-2X – No Sol ds	7 CBC + 3 J-2X + 6 Atlas Sol ds	7 CBC + 3 J-2X + 12 Atlas Sol ds
Development Costs	$4,594,000,000	$4,679,000,000	$4,735,000,000	$4,792,000,000	$4,820,000,000
Unit Product on Costs	$637,000,000	$793,000,000	$780,000,000	$859,000,000	$937,000,000

	Shutte Derived Family		
Perf Class	70t	100t	130t
Vehicle Description	27.5ft Core, with 4 RS-25s, 4 Segment SRBs, No US	27.5ft Core, with 5 RS-25s, 5 Segment SRBs, No US	27 5ft Core, with 4 RS-25s, 5 Segment SRBs, New J-2X Upper Stage
Development Costs	$11,000,000,000	$11,600,000,000	TBD
Unit Production Costs	TBD	TBD	TBD

Shuttle Derived Costs from HEFT Not independently verified or normalized

Hardware Only – No Ground Systems
Single Unit Production Costs – No Impacts of Learning Curve

Figure 2.8-1 Vehicle Cost Trade Results

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3.0 MISSION MODEL

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4.0 PROPELLANT DEPOTS

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5.0 COST ANALYSIS

5.1 Cost Trade Methodology

To provide an independent and unbiased assessment of the costs associated with the concepts defined by ULA, Advatech Pacific, Inc. (Advatech) was issued a subcontract to develop parametric costs estimates for the concepts and architectures defined by ULA. This was a change for ULA since our normal process has been to develop bottoms up cost estimates for all of our work with the Air Force and NASA. The parametric approach was selected because it was more suited to study since the concepts were not defined in detail and this type of approach was responsive and could provide an apples to apples comparison at a reasonable cost. Though ULA scrutinized the results for internal consistency, we resisted the temptation to start inserting either known in-house estimates, or proprietary vendor quotes values, in place of the parametric values to keep the evaluation apples and apples, and to avoid bias from creeping into the results.

Advatech and its partner MCR LLC have developed an integrated, process-based, systems engineering approach that combines the technical performance, cost, schedule, and risk factors along with other programmatic data and factors that impact the project. This approach had been developed for the Air Force and incorporated into the development of the Integrated System and Cost Modeling (ISCM) Tool Suite which Advatech has used to perform a comprehensive assessment of proposed concepts in a matter of days compared to conventional processes that can take months for various studies supporting the Air Force. The existing modules of the ISCM Tool Suite are focused on launch vehicles, strategic missiles, re-entry vehicles and spacecraft so it was an ideal tool for supporting ULA in its study of Heavy Lift Launch Vehicle (HLLV) concepts.

Technical Discussion

The ISCM Tool Suite examines the nominal costs for Design, Development, Test, and Evaluation (DDT&E) based on historical programs, thereby providing insight into non-recurring and recurring costs. ISCM can also evaluate the impacts of new technology insertion on the system by assessing the impact of the Technical Readiness Level of the new technology on the development of the entire system. Technology Readiness Levels (TRLs) are a systematic metric that provides an objective assessment of the maturity of a particular technology and allows consistent comparison of maturity between different types of technology. The TRL scale developed by NASA has been cross referenced to a specific milestone in the acquisition process. For example, NASA TRL 5 is equivalent to successful Critical Design Review (CDR) and NASA TRL 6 is equivalent to Selection Verification Review (SVR).

Historical data has shown that programs incur cost and schedule growth as they mature through the development milestones. Programs with mature technology (TRL 7 or more) experience significantly lower cost growth than programs with less mature technology (i.e. TRL 6 or less). The cost and schedule growth that is incurred as the program matures is difficult to predict in the planning phases, resulting in point cost estimates that often significantly underestimate the true costs to mature a technology. Advatech's tool suite uses historical cost data, correlated with a program's technology maturity, to derive cost growth factors that predict realistic system development costs. The user can then assess the technology maturity and the related cost growth of a program and see the effects of cost growth on schedule, cost, and probability of success.

Once the DDT&E costs are evaluated the Concept of Operations (ConOps) needs to be evaluated to assess the operations and maintenance tasks. The ConOps describes the infrastructure, manpower, and launch processing functions needed to support the mission. The operations concept is driven by the mission requirements and technology selection. Certain details of the operations are specific to each application and should be customized by the user. In the Advatech ConOps module, the infrastructure (facilities, equipment, transportation vehicles, etc.), launch site crew, launch system support activities, and post launch operations are used to model pre and post launch processing, launch, and maintenance activities. The user defines high level mission constraints as they relate to operations.

Once the mission setup is specified, the user builds the mission operations from a template or load a previously saved custom mission operations file. When loading the mission operations from a template, the model uses vehicle information and mission setup data to dynamically build the template from a knowledge database. The knowledge database was created by launch system subject matter experts who have extensive practical and theoretical experience on launch vehicle and ICBM programs.

The launch operations are maintained in Microsoft® Project. The model maintains the operations tasks in the form of a work breakdown structure (WBS), with associated resources and schedule, labor requirements and flow time are defined for each step in the process flow. The WBS is intended to be used as a starting point for concept evaluation. In order to fairly evaluate concepts, the user must strive to represent the launch operations and labor requirements for each concept as closely as possible. The module evaluates using existing assets, modifying existing assets, or building new and provides insight into non-recurring and recurring costs.

Advatech uses cyclical methodology to perform studies and analyses designed to efficiently meet the objectives of, and support, our customers. The process begins with defining the primary objectives which include designing the methodology, identifying the appropriate skill sets and data to support the objective, analyzing the trade space, and/or reviewing information from the customer's technical portfolio. In most cases, the ISCM tool suite can be used to support the analysis and will produce accurate and rapid results. However, when functionality is required outside of the scope of ISCM, conventional tools such as Microsoft Excel will be used to integrate data provided from other sources with the results from ISCM. This is done to expedite the process, efficiently using available resources, and provide our customers with the accurate results.

5.2 DDT&E Cost Model

Advatech and MCR had previously built the Advanced Cost Model (ACM) with both a set of Cost Estimating Relationships (CERs) for a solid rocket motor (SRM) launch systems and for liquid rocket launch systems (LRLS). Since a HLLV may include either liquid or solid propulsion systems those CERs were available for application to launch systems to be studied here. In evaluating the HLLV alternatives, it was determined that the LRLS WBS and the associated CERs were most appropriate for the basic HLLV application which was a basic liquid rocket launch system with multiple booster cores and the addition of Solid Rocket Motors as strap-ons if additional thrust was required. The original LRLS CERs were built on 26 standard liquid rocket launch systems from the Atlas and Delta families making this a good starting point for this particular study since we were looking at evolutions of the Atlas and Delta systems (see Figure 5.2-1). For the HLLV some additional CERs were developed to cover an expanded WBS based on data provided by ULA. With additional data on the components of the liquid rocket launch systems more cost drivers could be added. (b) (3) [redacted] (b) (3)

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In the CER, Average Unit Cost or AUC is the dependent variable. The weight of each launch system is the independent variable.
The equation form is:

$$AUC_i = aW_i^b \tag{1}$$

Where, W = Weight of the LRLS element, a is a constant, and b = the corresponding exponent.

The DDT&E cost module in ACM is composed of eight separate spreadsheets within a Microsoft Excel file. The spreadsheets contain the following sets of information: 1) Input Mass, 2) WBS Trades, 3) CERs by WBS for both Development and Production, 4) Technical Maturity Schedule of Months to key milestones, 5) Risk, 6) Phasing, 7) Inflation Factors, and 8) Outputs. Figure 5.2-3 provides a flow chart of how the respective spreadsheets are linked in the cost model. The Input Mass spreadsheet allows the user to specify the mass in kilograms or the weight in pounds. The WBS Trades spreadsheet allows the user to collect weights from lower level components into the appropriate subsystems to align application to the CERs. A set of CERs by WBS is provided for both the Development effort and the Production effort that is generally experienced during program execution, the development effort for a system evolves until system verification review has been completed before production units are built. The key milestones tracked to measuring technical maturity are the contract award (CA), the preliminary design review (PDR), the critical design review (CDR), the system verification review (SVR), initial operational capability (IOC), and full operational capability (FOC). In the Risk spreadsheet, CERs are used to estimate the most likely level of Development or Production costs. The user then specifies a low and a high percentage based on his estimate of the risk associated with each element of the WBS. Based upon this triangular distribution, a mean and sigma are used to roll up the costs to higher levels of the WBS and allocate the associated risks to each of the WBS elements. The resulting costs estimated in base year (BY) $ are then phased across the period of performance and inflated to the then year dollar (TY) $ values for developing the S-curve shown in the Output spreadsheet. The selected total estimated value with risk is then spread across the period of performance to indicate how the budget for the program is expected to flow.

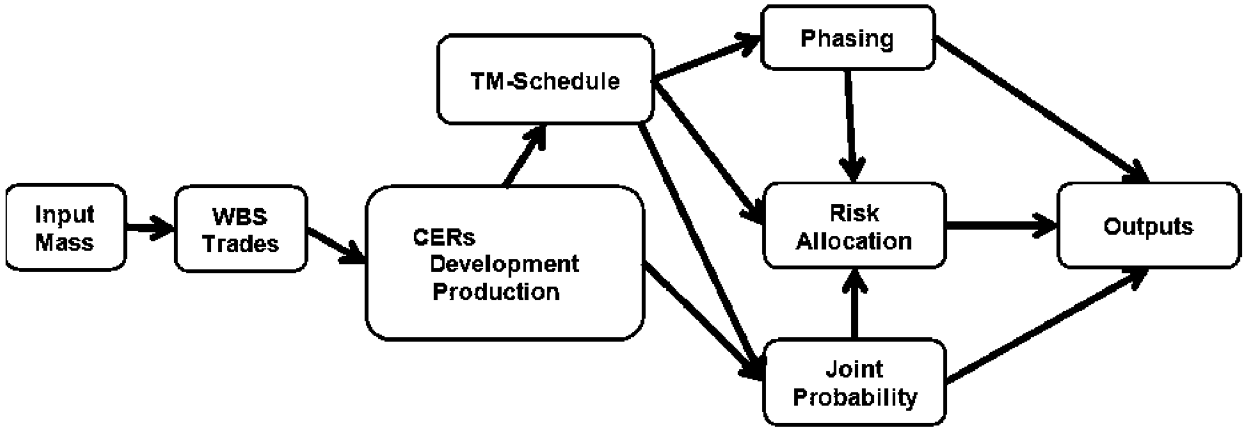

Figure 5.2-3 Reusable Launch Vehicle Cost & Schedule Flow Chart

For the ULA HLLV study development schedules were provided for the evolution of the vehicle families or a basic development period of four years was selected when calculating the cost estimates for the vehicles. Specific risk factors were determined based on the vehicle configurations so the features for Risk Allocation and the Joint Probability assessment were not used.

5.3 Calibration

To calibrate the cost models for the specific application mass data was provided by ULA so that the models in the ISCM Tool Suite could be calibrated specifically for this application. Verifying that the mass information produced by the ISCM Tool Suite was important since the models in ISCM would be used to determine mass inputs for the cost models when ULA defined the new configurations and Architecture.

CERs were primarily calibrated to Atlas V cost data provided by ULA, as well as specific engine models; each a value was changed such that the entry of the weight of a given component into its CER would yield a theoretical first unit cost corresponding to the figure in the proprietary data supplied. The CER exponent values were left unchanged.

Calibration was executed with Atlas data, as only top level data was available for the Delta platform. Initially, CERs were calibrated solely for the Atlas platforms, to the cost of the 26 elements that comprise the vehicle. These recalibrated CERs were then tested on the Delta platform, and were found to predict Delta costs within 1.5%. The fairly close fit suggests that the calibrated CERs are valid for both platforms. In addition, calibration factors were generated for each engine, using data provided by ULA as well as other sources.

These CERs are constructed to predict theoretical first unit costs, rather than an average unit cost. Each CER was calibrated such that subject to a learning curve, the average unit cost for a 7 unit lot buy was equal to the calibration target. Each component has its own learning curve slope, with a slope of 1.0 – no learning – for propellant, and 0.90 to 0.95 for all other CERs. For instance, a component with an actual cost of $1,000,000 and a learning curve slope of 0.95 would be assumed to have a first unit cost of $1,053,238, and an average unit cost of $969,246 for a 14 unit lot buy.

5.4 Concept of Operations Model

During the evaluation of launch vehicles and assessment of the total life-cycle costs the operations and maintenance (O&M) tasks are often not addressed even though they represent a significant portion of the total cost. This is true for all existing concept of operations (ConOps) used today including:

- Horizontal assembly and integration, transport to the launch pad, erect, and launch,
- Vertical assembly and integration, transport to pad and launch, or
- Assembly and integration on the pad and launch.

Advatech and MCR have developed a ConOps Module that is included in the ISCM Tool Suite and a unique approach to evaluating the O&M tasks associated with launch vehicle concepts, the impacts of introducing new technologies, and performing trades providing a comparison of various concepts. For launch vehicles the module describes the infrastructure, manpower, and launch processing functions needed to support the mission and is driven by the mission requirements and technology selection.

The ConOps module was developed to model and analysis the process flow, facilities requirements, and manpower needs required to support the launch operations for expendable and reusable launch vehicles or combinations of the two. The level of detail was selected to ensure that all technical aspects of the launch operations are addressed and risks can be identified. This level of detail supports verification that the launch operations meet all mission requirements, identify associated life-cycle costs, and supports identification of risk mitigation plans early in the development and/or acquisition process. This is critical because numerous studies have identified that the later a change is made in the program the more significant the impact on cost, schedule, and potentially the ability of the system to meet the mission requirements.

As the foundation for the ConOps Module detailed process flow diagrams for launch operations were created by Subject Matter Experts who have extensive practical and theoretical experience on launch programs. The information in these diagrams was assembled in a Knowledge Database that included O&M tasks, facilities, manpower information, and other related information.

The ConOps Module takes as inputs specific vehicle parameters defined by the physics based models of the ISCM Tool Suite and user defined high level mission constraints as they relate to operations. This information is used by the model to setup default infrastructure, process flow, and resource requirements that provide the user with a reasonable starting point for defining their concept of operations. The user can then tailor these requirements to their custom scenario performing specific analyses, evaluating impacts of changes in technologies and process flow, and completing exploration of defined trade spaces.

The user can build the mission operations or process flow for the launch vehicle from a template or load a previously saved custom mission operations file. When loading the mission operations from a template, the model uses vehicle information and mission setup data to dynamically build the template from the Knowledge Database including predessor information and identification of hazardous operations. The launch operations are maintained in Microsoft Project. This framework was selected because it is a widely used tool for managing task WBS, resources and schedules. This supports the capability to import existing schedules maintained within project by simply specifying the "Custom" selection and loading an existing file. The model maintains the operations tasks in the form of a WBS, with associated resources and schedule, labor requirements and flowtime are defined for each step in the process flow. Figure 4 represent roll up views of the WBS for a sample vehicle.

	Task Id	Phase	Work Breakdown Structure Elements	Duration	Predecessors
1	1	Site	− Minotaur I Operations	119.45 days?	
2	1.1	PAIF	+ Process Satellite	16.75 days	
13	1.2	Trnsfr Pyld-Int	Transport Satellite and Fairing to Integration	0.25 days	2
14	1.3	HIB	− Process and Assemble Upper Stack	104.95 days	
15	1.3.1	HIB	+ Process Missile Stages	43.75 days	
39	1.3.2	HIB	+ Integrated Buildup of Missile on Transporter	61.2 days	15
79	1.4	LVPF	− Process and Assemble Lower Stack	35.5 days	
80	1.4.1	LVPF	+ Process Stage 1	17.25 days	
88	1.4.2	LVPF	+ Process Stage 2	18.25 days	80
97	1.4.3	LVPF	+ Mount Stage 1	1.95 days	80
108	1.4.4	LVPF	+ Integrate Stage 2	4.5 days	97
117	1.5	Trnsfr-Pad	+ Launch Vehicle Transport	0.5 days	14,79
120	1.6	Ground Ops	− Conduct Ground Launch Operations	14 days?	117
121	1.6.1	Ground Ops	Plan Mission	14 days	
122	1.6.2	Ground Ops	+ Integrate Lower and Upper Assembly Stacks	4.7 days?	
128	1.6.3	Ground Ops	+ Exercise Range interfaces	0.5 days	127
130	1.6.4	Ground Ops	Secure Environmental Protection	0.25 days	
131	1.6.5	Ground Ops	Mission Simulation Test	2 days	127
132	1.6.6	Ground Ops	Safe Pad	0.25 days	131
133	1.6.7	Ground Ops	Initial Dress Rehearsal	0.5 days	132
134	1.6.8	Ground Ops	Final Dress Rehearsal	1 day	133

Figure 5.4-1 Roll-up of Sample Vehicle

As the process flow is created the ConOps module automatically assigns labor resources to top level tasks. Again this is a starting point and the user can customize the information to the specific application. The module includes military, civilian and contractor labor codes at different experience/qualification levels. Military rates were derived from the Office of the Under Secretary of Defense (OUSD) Standard Composite Rates. Civilian Rates were taken from Salary Table 2007-GS from the US Office of Personnel Management. Contractor Rates were taken from skill searches on www.salary.com for both East coast and West coast regions. It is assumed the crew is dedicated to the site year around and the crew is scaled by the model as facilities or work schedules are modified. The ConOps Module provides summary level information for the labor and can also provide detailed information as required (See Figure 5.4-2).

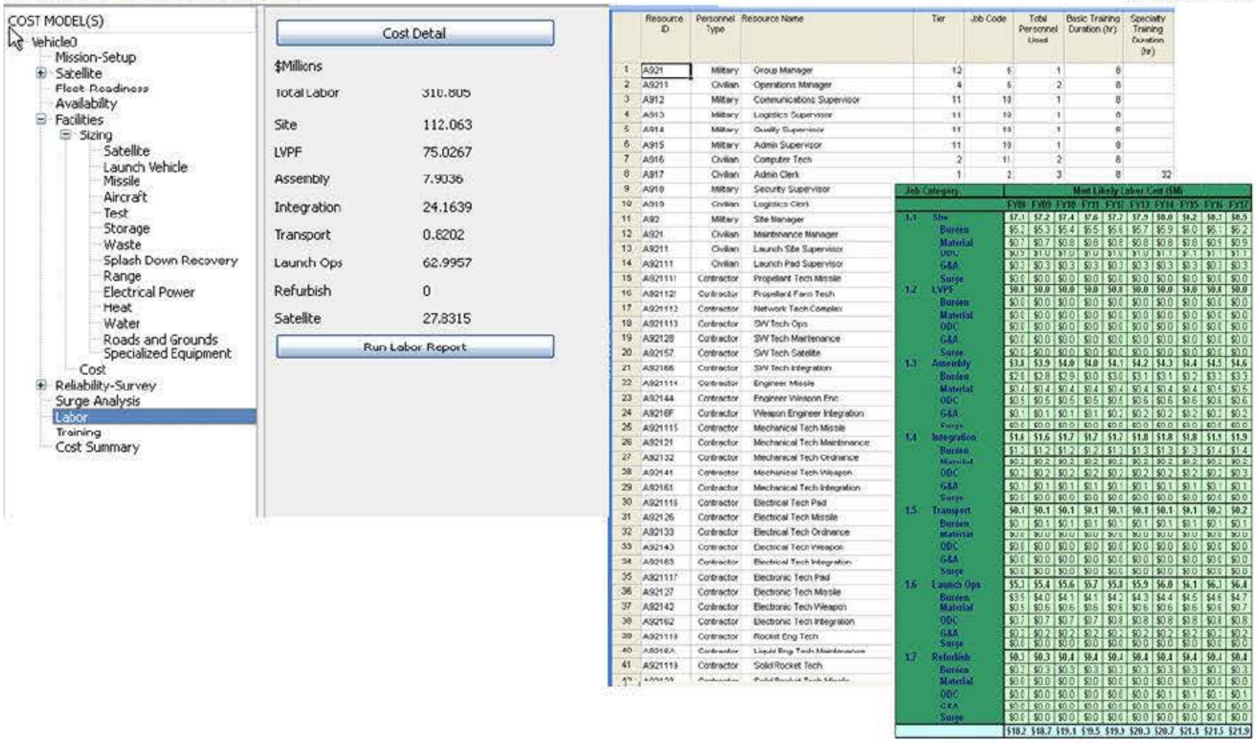

Figure 5.4-2 ConOps Module Labor Information

The ConOps Module also provides information about the facilities required to support launch operations. Based on the mission requirements defined by the user and information from the physics based models, the ConOps Module defines required facilities and estimated sizes for the facilities. This information includes nonrecurring costs for building the facilities and recurring costs for maintaining the facilities. Costs are based on the Department of Defense (DoD) Facilities Pricing Guide and Directorate of Engineering Support Historical Air Force Construction Cost Handbook. Figure 5.4-3 shows the Facilities Graphical User Interface (GUI) in the ConOps Module. The user can select whether new facilities are required, existing facilities are going to be used, or if existing facilities require refurbishment, Depending on the selection, the module will calculate the associated nonrecurring and recurring costs. In the case of refurbishment, the costs for refurbishment are input by the user. Similar to the labor costs, the ConOps Module provides summary and detailed information as required.

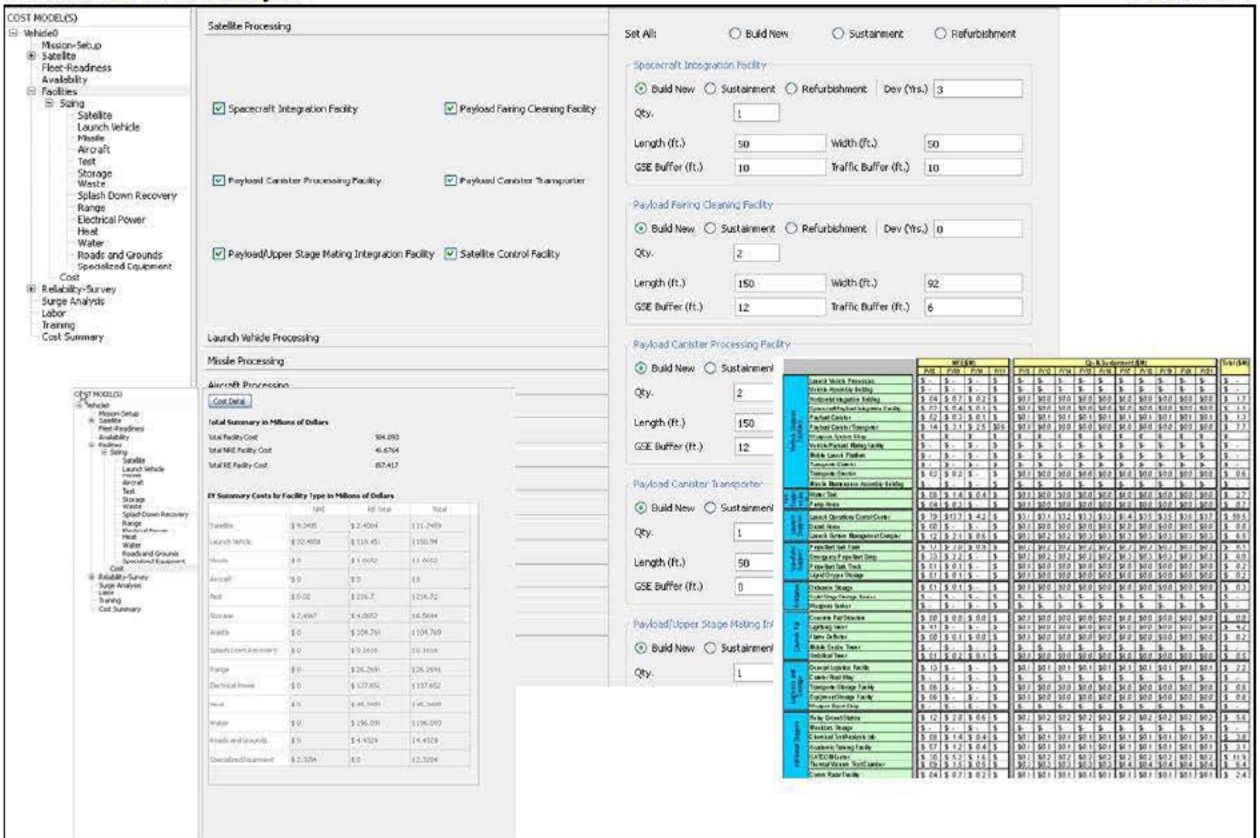

Figure 5.4-3 Facilities Information

As stated previously the initial information provided by the module is intended to be used as a starting point for concept evaluation. The user can then modify the information to represent the actual launch operations for the specific vehicle or vehicles. In order to fairly evaluate concepts, the user must strive to represent the launch operations and labor requirements for each concept as closely as possible. By utilizing the documentation features in the ConOps module the user can complete a model of the process flow including life-cycle costs so that details of the analysis are fully exposed and can be verified. This feature also allows for transition of information to new engineers and analysts minimizing the impact of personnel turnover. Proper development of the model supports evaluation of the impacts of introducing new technologies, proposed concept of operations, and vehicle configurations being proposed to support NASA HLLV mission requirements.

5.5 Analysis Approach

To complete the parametric study for ULA, Advatech used the configurations and architectures defined by ULA and modeled them in the DDT&E and ConOps Models. The results were then reviewed by subject matter experts at Advatech and ULA to identify inconsistencies that required further evaluation.
In the case of the DDT&E Model risk factors were incorporated for each subsystem and modified as appropriate to provide realistic cost estimates. Any change in the default

risk information was supported by specific rationale such as the transition from the RS-25E engine from the RS-25D engine assumed no risk since the RS-25E engine only represented the restart of production of the RS-25D engine and evolution to the 1.25 Mlb thrust LOX/RP engine from the existing RD-180 engine assumed minimal risk since it was just a scaled up version of the RD-180. The companies that manufacture engines thoroughly understood the design. In a few cases actual cost provided by ULA were used. These costs were based on previous studies performed by ULA

Since specifics of the concept of operations were not defined the existing launch operations for the Space Shuttle, Delta IV Heavy, and Atlas V vehicles were reviewed with representatives from United Space Alliance (USA) and ULA at Kennedy Space Center and Cape Canaveral Air Force Base. It was assumed that NASA facilities at Pad 39 would be used for launching the HLLV. Since the facilities existed the modifications required were estimated as a total cost of fabricating the facility new. Labor requirements were developed and then compared to System Acquisition Reports (SARs) to verify the reasonableness of the estimates.

5.6 Results Overview

During the completion of the parametric cost study, over 30 vehicle configurations were evaluated and used to develop the final estimates for six (6) scenarios developed by ULA. During the evaluation of these concepts some basic ground rules and assumptions were used throughout:

- Point estimates were done in base year 2010 dollars
- Sand charts escalated based on projected inflation rates
- Development costs for all RL-10 configurations are paid for by Air Force Space and Missile Command Center and not included here
- Development costs for the 1.25 Mlb booster engine assumes minimal risk
- Development costs for RS-25E engine assumes no risk since it is conversion of SSME
- No sustainment or modifications of test facilities included in estimate
- Production for each vehicle assumed to be 5 years to support identification of long lead items
- NASA facilities at the Pad 39 Complex would be used
- Modifications and sustainment of facilities at KSC included
- Labor costs – assumes a standing army for the full year
- All obsolescence issues would be addressed prior to initiation of HLLV Program
- Similar Concept of Operations (ConOps) assumed for each family
    - Horizontal processing initially
    - Erect Vertical in VAB and finish assembly on the MLP
    - Transport to pad
- National Mission Model used to develop rate impacts and learning curve information for EELV and 5M RP/LOX evolutions
- Short time on pad (3-5 days), requires significant automation
- Cost for propellant launches included at $6M per metric ton

Elements not included in the life cycle cost elements include NASA/MSFC oversight and insight cost in both the non-recurring and recurring phases, and NASA/KSC oversight and insight apart from the direct launch operations costs.

The six scenarios evaluated were grouped into three families:

1. 5m RP/LOX Derived Evolution – This family was based on the Atlas 551 Heavy Lift configuration utilizing three common cores and evolved into a vehicle with five common cores with varying booster engines and upper stages.
2. Delta Derived Evolution – This family was based on the Delta IV Heavy Lift Launch Vehicle and evolved from the standard three common core configuration to a seven body evolution with varying upper stages.
3. Shuttle Derived Evolution – This was the baseline configuration being evaluated by NASA which was used to compare the other two families against.

As part of the first two families a scenario was introduced using a propellant depot which included commercial launches to refill the depot as required to support the overall mission requirements.

The results of the parametric study are discussed in the following section.

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7.0 GAP ANALYSIS

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8.0 STUDY CONCLUSION

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