governmentattic.org
"Rummaging in the government's attic"

Description of document: Two (2) National Aeronautics and Space Administration (NASA) reports: Heavy Lift and Propulsion Technology Systems Analysis and Trade Study (2011):

1. Generated under Contract NNM11AA17C awarded by NASA George C. Marshall Space Center (MSFC) to United Space Alliance (USA)
2. Generated under Contract NNM11AA15C awarded by NASA MSFC to Space Exploration Technologies (SPACEX)

Requested date: 12-November-2018

Release date: 06-September-2019

Posted date: 11-November-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

The governmentattic.org web site ("the site") is a First Amendment free speech web site, and is noncommercial and free to the public. The site and materials made available on the site, such as this file, are for reference only. The governmentattic.org web site and its principals have made every effort to make this information as complete and as accurate as possible, however, there may be mistakes and omissions, both typographical and in content. The governmentattic.org web site and its principals shall have neither liability nor responsibility to any person or entity with respect to any loss or damage caused, or alleged to have been caused, directly or indirectly, by the information provided on the governmentattic.org web site or in this file. The public records published on the site were obtained from government agencies using proper legal channels. Each document is identified as to the source. Any concerns about the contents of the site should be directed to the agency originating the document in question. GovernmentAttic.org is not responsible for the contents of documents published on the website.

-- Web site design Copyright 2007 governmentattic.org --

National Aeronautics and Space Administration

Headquarters
Washington, DC 20546-0001

September 6, 2019

Office of Communications

FOIA: 19-MSFC-F-00097

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-00097 and was for:

> A copy of each Report generated under Contract NNM11AA17C awarded by NASA MSFC to United Space Alliance and Contract NNM11AA15C awarded by NASA MSFC to Space Exploration Technologies.

Please be advised that the search for responsive records has concluded and a total of 257 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:

United Space Alliance report (199 pages):

_133_ page(s) are being released in full (RIF);[1]
_15_ page(s) are being released in part (RIP);
_51_ page(s) are withheld in full (WIF).

Space Exploration Technologies (58 pages):

_5_ page(s) are being released in full (RIF);[2]
_15_ page(s) are being released in part (RIP);
_38_ page(s) are withheld in full (WIF).

---

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

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 and private party names.

Fees for processing this request are less than $50.00 and are not being charged in accordance with 14 CFR §1206.504(f).

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
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

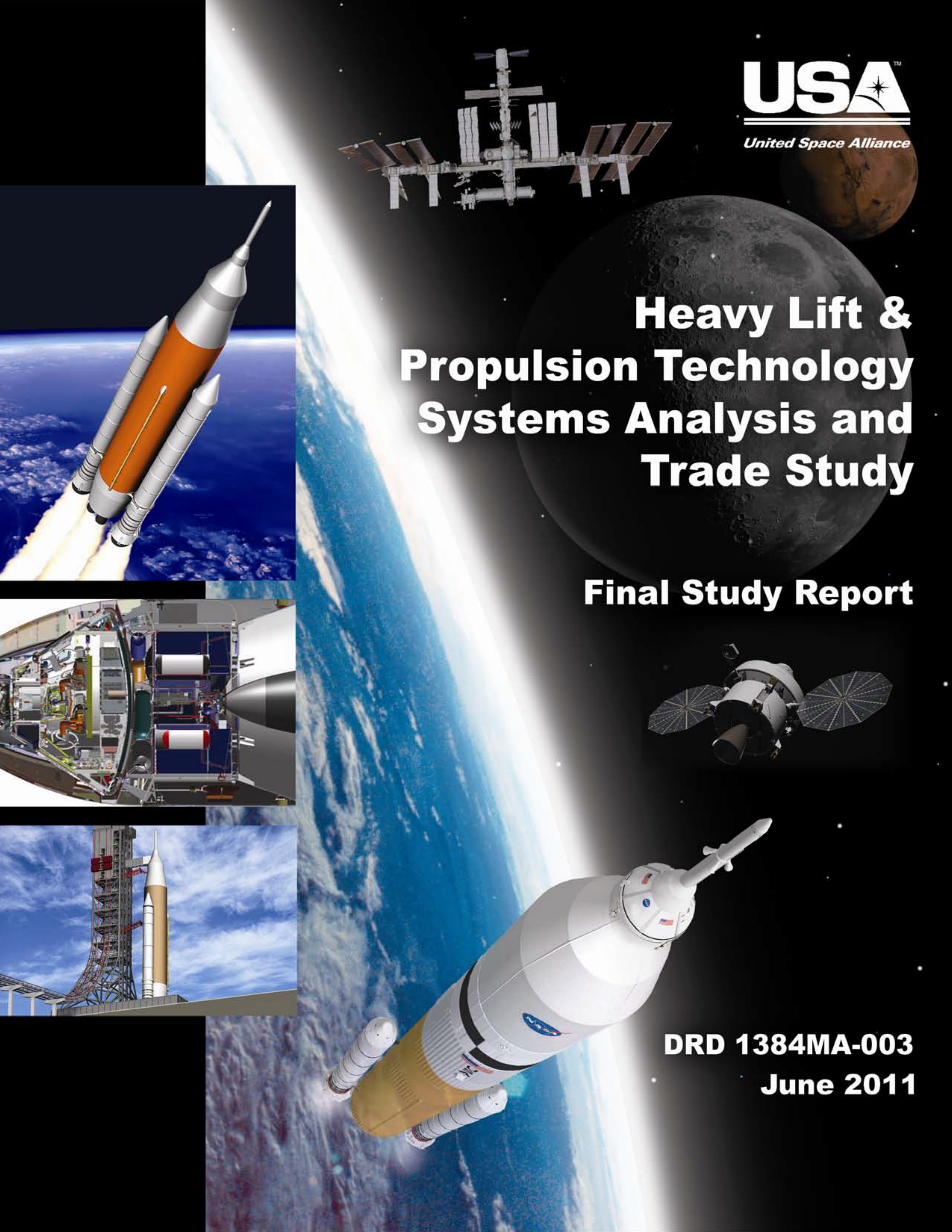
Additionally, you may contact the Office of Government Information Services (OGIS) at the National Archives and Records Administration to inquire about the FOIA mediation services it offers. The contact information for OGIS is as follows: Office of Government Information Services, National Archives and 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



USA™

United Space Alliance

Heavy Lift & Propulsion Technology Systems Analysis and Trade Study

Final Study Report

DRD 1384MA-003
June 2011

TABLE OF CONTENTS

TABLES

FIGURES

Figure **Page**

1.0 BACKGROUND, EXPERIENCE, AND SUMMARY

1.1 Introduction

United Space Alliance, LLC., (USA) is pleased to provide to the National Aeronautics and Space Administration (NASA) Marshall Space Flight Center (MSFC) the Heavy Lift and Propulsion Technology (HLPT) Systems Analysis and Trade Study (SATS) Broad Agency Announcement (BAA) Final Study Report under contract NNM11AA17C, DRD 1384MA-003, in support of the trade space analysis of potential Heavy Lift Launch (HLL) and space transfer vehicle concepts.

1.2 Background

USA has leveraged our 25-year legacy of integration and operations of the Space Shuttle vehicle in developing the best possible architecture, systems, and subsystem considerations and recommendations for the Space Launch System (SLS). Our understanding of the NASA technology-enabled Exploration goals is (a) a human mission to Beyond Earth destinations, a Near Earth asteroid in 2025, and (b) a human mission to Mars orbit and return by mid 2030 with a Mars surface mission to follow. In support of Exploration, within the next 2 decades, the development of an SLS that can provide launch services to multiple Customers for multiple reference missions will ensure its longevity. The SLS is not an end-state program; rather, it is a key element that enables NASA's Fiscal Year (FY) 2012-stated long-term Exploration Goals and Objectives (G&O). The SLS Program can be initiated using a complement of new and existing technology to provide the needed ascent heavy lift capability rapidly with minimal development cost. The result of the trade studies and analyses performed in this BAA indicate where existing and new technology should be applied. Development must start now to meet NASA's target need dates and be economically sustainable so its cost does not detract from new technology development funding for deep space exploration.

Government and Industry have demonstrated in the Life-Cycle Cost (LCC) Analysis (LCCA) discipline field that the longer you wait to consider operability, supportability, and sustainability the more expensive the changes become in terms of affordability. USA, as the Space Shuttle integrator and operator, has witnessed these factors and understands their impacts. USA understands these analyses results from experience on the Space Shuttle Program (SSP). This analysis is used as an important factor in conducting trade studies for evaluating requirements and design alternatives or operations concepts within a project's architecture when cost is a key driver. Concepts would be developed to the lowest level necessary (e.g., subsystem) to ensure that they are deemed feasible and to a level that will reduce the risk low enough to satisfy the architecture's affordability. Department of Defense (DoD) acquisition programs in the 1980s with a service life of about 30 years, similar to Space Shuttle and potentially SLS in the future, indicate that operations and support cost can represent over 70 percent of the program LCC. Those same studies show that 85 percent of decisions defining total LCC have been made by the end of system design, as depicted in Figure 1-1. Early, effective requirement, design, operability, and sustainability decisions can significantly reduce overall system LCC, consistent with and validated by the 1986 Packard Commission findings (President's Blue Ribbon Commission on Defense Management).

USA's past performance is based on Space Shuttle design, integration, processing, and operations experiences and lessons learned over the past 15+ years, leveraged with previous heritage company expertise. USA's scope executed under the BAA was based on research accomplished and submitted under the HLPT Request for Information (RFI) (RFI05042010PS30) in May 2010. USA's past architecture trades that are considered in context, but were not limited to the 1991 MSFC National Launch System Study, 2002 Future Shuttle Study, 2004 - 2005 Shuttle Derived Launch Vehicle Collaborative Industry Study, 2009 NASA Shuttle Derived Heavy Lift Launch Vehicle (HLLV) (Side Mount Versus In-Line) Study, and 2010 HLLV Industry Team Study.

In 2004, USA entered into an agreement among Boeing, Lockheed Martin, Alliant Techsystems, Inc. (ATK), and Rocketdyne (now Pratt & Whitney Rocketdyne) to participate on the Shuttle Derived Launch Vehicle Collaborative Industry Study Team. USA's accountability on that team was to trade vehicle configurations and their corresponding infrastructure needs against the concept of operations and then estimate the LCCs of the resultant architecture. The 18-month Study Team's results were pre-Exploration System Architecture Study (ESAS) and shaped USA's understanding that a cargo and crew HLLV was programmatically and technically feasible using a complement of existing and new innovations and technologies. The development cost and schedule of a new heavy lift launcher is proportional to the

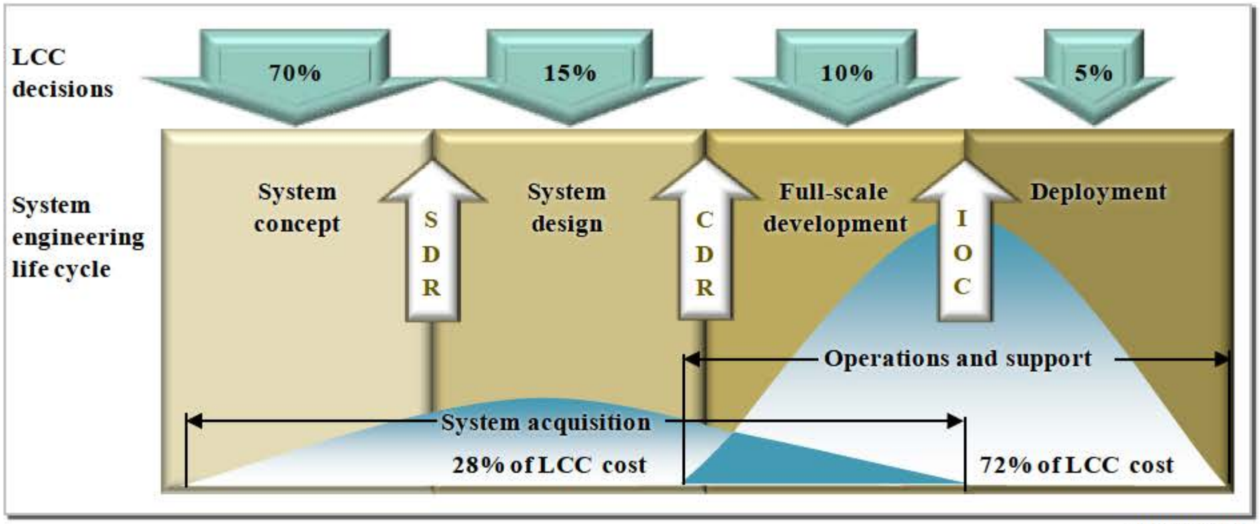


Figure 1-1. DoD study on 30-year program LCC considerations

amount of new technology infused, inserted, or transitioned. As an example, applying technology in select areas into the system development process using a progressive architecture approach and using building blocks in an evolving vehicle capability will provide the best probability to execute a program of record of 4 to 5 years. These results were used within the USA trade study as a benchmark for the HLPT SATS efforts, and this programmatic view was heavily considered within the trade space.

During the execution of the trade study, USA incorporated our Design for Operations (DFO) strategy, which provided a unique and high-payoff opportunity to apply our experience and lessons learned from past programs and evaluated the architectural design, model, and iterative maturation of integrated program processes. USA used its internal Knowledge Management System, which includes lesson learned databases, to support our trade study implementation decisionmaking. The collaborative effort between design, analysis, and operations to optimize and integrate design solutions on SLS will make the program affordable and sustainable.

1.3 Experience

USA was formed in 1996 by Rockwell International (purchased by Boeing in 1998) and Lockheed Martin Corporation to consolidate some 28 NASA contracts supporting the SSP. USA plays a central role in all of NASA's human space flight programs with capabilities in virtually every aspect of large-scale system integration, space operations, and engineering that were leveraged on developing this trade study. Our direct relevant experience included flight and ground systems and sustaining engineering, mission design and planning, ground operations and processing, space systems integration and program management, flight software, hardware and software development, and flight readiness verification.

Although USA has a distinguished integration and operations legacy, we have a broad Systems Engineering and Integration (SE&I) and Design, Development, Test, and Evaluation (DDT&E) capability within both hardware and software fields, particularly in regard to integrated performance and operability. Our availability of firsthand lessons learned and knowledge of Space Shuttle and International Space Station (ISS) integration and operations combined with our experience on both the Lockheed Martin Orion Crew Exploration Vehicle (CEV) and ATK Ares I Crew Launch Vehicle (CLV) First Stage (FS) DDT&E teams, along with logistical experience with unique low-production vehicles, provided MSFC insight into 25 plus years of human spaceflight operations. With a workforce of more than 5,000 employees experienced in all aspects of ground and flight operations, USA selectively used key personnel from its broad array of engineering capability and competency to perform the trade study activity.

1.3.1 Systems Engineering

USA is responsible for providing NASA SE&I support for development and management of requirements, configuration, and departure from the baseline at the system, project, and program levels on Space Shuttle. This capability exists at MSFC, Kennedy Space Center (KSC), and Johnson Space Center (JSC) and covers the entire human spaceflight vehicle and operations requirement set. These skills were applied to perform technical assessments; requirements refinement; design and operational risk assessments; development of alternatives; and identification of anticipated and contingency conditions and design for operability inputs to streamline Operations Requirements (con-ops).

The integrated Concurrent Engineering (CE) processes established within this asset provide unique design for operability for ground and mission operations to effect risk-based trade studies, optimize design solutions, and achieve sustainable program LCC targets. Significant trade studies conducted in the past include: Space Shuttle Upgrades across vehicle and infrastructure from a safety and supportability improvement focus; alternate Solid Rocket Booster (SRB) Thrust Vector Control (TVC) system; Advanced SRB; Flyback Booster; Ares I First Stage expendability versus reusability; and numerous studies for Ground Systems and integration within the Ground Operations Concept of Operations Study.

1.3.2 Design and Development

USA's design and development capabilities include both flight and ground elements. Design activities and support within this capability include flight (Shuttle SRB, X-33, Ares I, Orion, and Materials and Processes (M&P)) and ground systems (for Shuttle, Atlas, X-33, Orion, and Ares I, including Ares I-X).

USA's launch vehicle design and spacecraft vehicle design capabilities used on the integration of Shuttle and their elements (e.g., Orbiter, SRB), along with Orion CEV, Ares I CLV, and Ares I-X, include structural, mechanical, thermal, and electrical/electronic analysis tools; flight software development and analysis (Prime Contractor on Shuttle Primary Avionics Software System (PASS)); deceleration and recovery systems (land and water); separation system pyrotechnics; fluid systems (including high-pressure pneumatic and hydraulic systems); cryogenic and hypergolic propellants loading and storage; environmental control and life support; electric and electronic systems; avionics and range safety; low- and high-voltage power storage, control, and distribution; cable and circuit design using hard-line, fiber, and Radio Frequency (RF) components; command and control systems; instrumentation; modeling and simulation and imaging with modeling; M&P management; and analysis that covers structural, vibroacoustic, thermal, and Electromagnetic Interference (EMI).

It is critical that the safety aspect of reliability is present in the design and development decisions. USA's system safety is a vital process that has been demonstrated on current human spaceflight programs, as this experience and knowledge was applied on the SLS Program as a "check and balance" with the design architects and operations.

1.3.3 Integration and Operations

USA's core strength is our ability to integrate the HLL components, including propulsion, from numerous suppliers into a functional launch system from an integrator and operator perspective. Our expertise and experience in procuring and integrating multiple systems and designs at a system level provides an operations and supportability focus. USA understands the importance of reliable and safe flight-critical operations.

When considering LCCs, not only does the technical aspect provide a significant contribution, but the programmatics of the business model provide a large component of both the development and operations cost. We continuously improve program and business systems, infused with new innovations to maintain effective and affordable business practices during our normal course of program execution. USA's programmatic methodology maintains an open, Customer-centric relationship, ensuring the honest communication necessary to facilitate issue resolution fundamental to any program's success. This successful approach was continued on the Constellation (Cx) Ares I-X flight test and is being used today on both the Ares I CLV First Stage and Orion CEV contracts and was used on the HLPT SATS effort.

USA, as the SSP operations and program integrator and more directly to MSFC as the SRB vehicle integrator, establishes the foundation of our understanding of program integration across large, complex programs. Our internal capabilities extended during Cx has shown that integrated design, manufacturing, and resource planning tools have demonstrated efficiency in development and production programs. As an integrator, USA is heavily dependent on the ability to control the application of requirements, design products, assembly and test documents, and hardware usage throughout the life cycle of the program.

Also based on our experiences, another programmatic area that can drive affordability on human spaceflight programs is configuration change management. USA has hands-on experience and capability in this area based on lessons learned on both the SSP and the Constellation Program (CxP). To achieve efficiency without compromising integrity, USA developed a Configuration Change Control Express (CCCE) application to address this issue. CCCE is a suite of tools (Life Cycle Express, Change Management Express, Directive Management Express, and Action Management Express) that provides complete life-cycle tracking of engineering change proposals number assignment, change assessment, directive creation, board agendas and minutes, action tracking, and final change implementation action closeout verification. The knowledge of what it takes to configuration control human spaceflight data will prove invaluable for the SLS LCC and operability insight.

As detailed in the Ground Operations Concept of Operations Study (NNK06MB37C), USA's "Design for Operations" capability focus combines a unique flight and ground operations perspective with extensive design and analysis capability to achieve a balance of schedule, cost, and risk between the flight vehicle and the ground/flight infrastructure required for development and mission success. USA's existing human space flight experience provides state-of-the-art safety practices, reliability, redundancy, and fault-tolerance operations. While spacecraft normally receive the largest amount of attention on operability, launch vehicle systems' LCC will benefit enormously from an infusion of operability and sustainability considerations during the architecture definition, requirements development, and design phase.

USA's lessons lived and learned on effective and affordable recurring cost operations based on decades of planning, processing, and operating the Space Shuttle addresses launch vehicle equipment access points; vehicle system health monitoring and diagnostics; integrated vehicle processing; ground and mission facilities equipment; and infrastructure requirements. The benefits are high confidence in expected results, well-defined margins in all modes of operation, and schedule success.

USA's Flight Operations has supported NASA Mission Operations Directorate (MOD) since our company's inception on both Space Shuttle and ISS. USA has led and supported a Flight Design Operations Trade Study since 2007 with MOD. The purpose of that study was to develop a process to support preflight design of the trajectory and reconfigurable flight software parameters (i.e., I-Loads) for CxP-to-ISS missions. USA's clean-sheet approach to designing the new process is by assembling basic building blocks necessary to complete a mission design in the most affordable, operable, and reliable manner. Thus, the results are easily extensible to different launch vehicles and/or spacecraft, including SLS. In Space Shuttle mission operations, our payload staging analysis tool capabilities, including previous Centaur Upper Stage efforts and flight dynamics analysis tool capabilities, provided MSFC an independent flight vehicle system analysis capability for the In-Space propulsion trade space down-select process.

USA's knowledge capture and management database were used to the greatest extent possible to make informed decisions during the SATS activities.

Based on USA's past performance, breath of company experience, and select expertise chosen to work the HLPT SATS activities, USA's broad capabilities across systems engineering, design and development, and integration and operations were used to the fullest extent in this trade study.

1.4 Executive Summary

USA submits this HLPT SATS Final Study Report to NASA's SLS Program at MSFC in support of the trade space analysis of potential heavy lift launch and space transfer vehicle concepts. USA has leveraged our 25-year legacy of integration and operations of the Space Shuttle vehicle in developing the best possible architecture, systems, and subsystem considerations and recommendations for the SLS. USA's report traceability to NASA's BAA Technical Objectives and USA's Statement of Work (SOW) is documented in Appendix B.

USA's trade study approach and process is based on a robust SE&I foundation. As defined in Section 2.0, the methodology is considered thorough and complete. To guide USA's trade space boundaries, Groundrules and Assumptions (GR&A) were developed, that differed from NASA's boundaries, to provide guidance in the trade analyses as shown in Section 3.0. USA developed a broad set of trade study inputs based on different propulsion systems and integrated launch vehicle configurations, shown in Section 4.0, to have considerations of multiple different options. In Section 5.0, USA traded the Figures of Merit (FOMs) based on NASA's voice and USA's interpretation, decomposition, and allocation of the architecture, systems, and subsystems. USA's trade study results, as detailed in Section 6.0, consider six different trade trial cases, four as end-state configurations and two as evolvable configurations. In additional to vehicle trade results, In-Space systems, systems, and implementation trade results are documented in Section 6.0, Trade Results.

The Final Study Report does not just encompass the trade study results but also defines the end product "tool" that can help the NASA SLS Team in the final selection and provide justification of the architecture and vehicle configuration chosen. The Heavy Lift Launch Vehicle System Analysis Tool is an application that was designed to facilitate the trade study process using several tools, which, when integrated together end to end, enable a structured approach to systems analysis and decisionmaking. This tool has been submitted in association with this report deliverable as documented in Section 7.0.

As outlined in Section 8.0, to follow is a summary of the major and significant takeaways from this trade:

a. Collaboration with other NASA Programs, DoD and International Partners to share capability on missions, payloads, and infrastructure will reduce SLS nonrecurring and recurring costs, thus enabling an affordable and sustainable program

b. A "Progressive Architecture" approach and the use of building blocks in an evolving vehicle capability will provide an opportunity to infuse new technology with the least amount of risk and provide a flexible capability for specific mission and objective needs; i.e., an architecture that provides 100 mT of performance to LEO will meet the majority of missions and objectives through 2030

c. Multiple acquisition and procurement options are available to establish an executable strategy, as numerous different contracting mechanisms can solve the low flight rate affordability dilemma (e.g., ATK to provide motors and USA to integrate booster; There would be no learning curve as almost form, fit, and function identical to SSP [see USA unsolicited iSRB proposal]; Also, splitting the contract provides more buying leverage for the Government)

d. Heritage assets have a positive impact on near-term costs and the development timeline, although the operational requirements and processes will need to be reviewed and revised to reduce long-term LCC, as the architecture that meets schedule and cost in the near term may not provide the lowest long-term LCC

e. Delaying operability, supportability, and sustainability decisions in the DDT&E phase will result in a less affordable solution, and changes become more expensive in the long term

f. A new business operations model is needed in the context of recent NASA commercialization to reduce the Government and Contractor business costs; e.g., the lack of program-level taxonomy can impact affordability to the same degree as an architecture selection

2.0 STUDY APPROACH AND METHODOLOGY

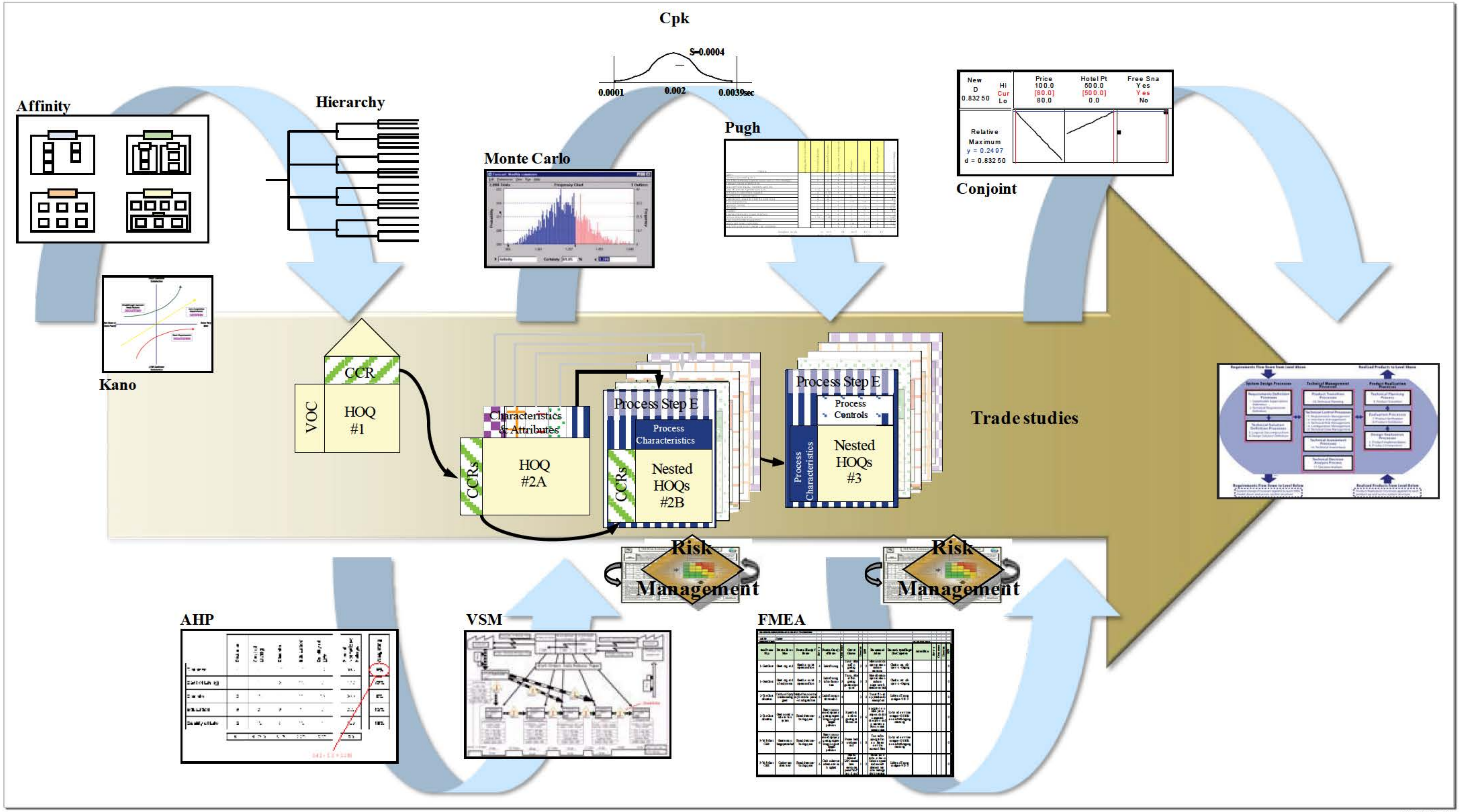
2.1 Trade Approach

USA's approach to the HLPT SATS involves starting with the MSFC Technical Objectives (BAA Section III), and uses proven processes and tools that provide a logic- and method-based trade comparison that has been successfully demonstrated on both the SSP and CxP.

The potential technical solutions (i.e., the architectures and the launch vehicle systems that comprise them) are many, and the benefits or penalties of each system architecture or component are often codependent. Evaluations of the technical solutions and the trade study outcome are also influenced by the background and work experiences of the various team members involved in the evaluations. For this reason, the team members from various organizations and work locations from within USA were selected to provide a well-balanced breadth of knowledge and occupational experiences. For USA, an integration and operations Point of View (POV) was selected as the study domain. USA's approach was not based on a launch vehicle or engine Original Equipment Manufacturer (OEM) POV but rather was focused on our SSP and CxP experience. As a result, USA's focus was on a "Shuttle-Derived" base architecture trade study. At an architectural level, the trade focused on three elements: vehicle and associated systems, ground systems and infrastructure, and mission systems and their infrastructure. Any configuration that was not evolvable, flexible, and extensible to capture all relevant mission objectives was removed from the trade space (e.g., Shuttle side mount). USA also assessed the direct versus progressive framework needed to reach the end-state capability within the cost and time constraints of the Program. Both destination-driven model and capability-driven model approaches were evaluated to determine which model could be used to balance cost, schedule, and performance enveloped by risk.

Our approach was to use a logical methodology within a CE environment in concert with the Analytical Hierarchy Process (AHP), Quality Function Deployment (QFD), and Pugh Matrix processes and tools with initial Customer inputs. These processes and tools are very well known in the Aerospace Industry, as well as throughout manufacturing industries, and allow for the evaluation of a broad range of factors, each with benefits, penalties, and interdependencies. Application of the AHP, Quality Function Deployment (QFD), and Pugh Matrix provides a logical methodology within the Requirement Analysis Cycle (RAC) and Design Analysis Cycle (DAC) to analyze alternatives that best achieve technical objectives. Using this method of solutions analysis, a numerical or quantitative value can be determined from a qualitative set of criteria to make ranking or grading of the different combinations of vehicle architectures possible.

Figure 2-1 depicts a notional trade study process flow used by USA for a process-based QFD.



HLPT-SATS_002

Figure 2-1. Trade study process

USA's SATS did not just consider the product (i.e., the vehicle and propulsion system) but also encompassed the implementation process to ensure that the complete life cycle is considered within the RAC and DAC. Significant uncertainties exist in development, production, operations, and disposal costs early in the requirement and conceptual design cycle. Early decisions within the feasibility phase of concept definition are critical to reduce downstream costs resulting from design changes late in the production or operational phases of the hardware life cycle.

Development and analysis support to the SLS Team began with understanding and defining the Voice of the Customer (VOC), including the technical objectives, GR&A to establish the boundaries and constraints, and top-level missions and objectives. These top-level qualitative goals and mission objectives form the basis for definition of specific Critical Customer Requirements (CCRs). CCRs are qualitatively defined by functional characteristics and attributes of the architecture, vehicle or system element, or system, which, in turn, are quantitatively defined by FOMs. Trade iterations are conducted with the Customer's inputs in a CE environment down to the system level.

After the completion of every trade analysis iteration within the architecture, vehicle or system, or system level, a gap analysis was conducted to determine the strengths/weaknesses against the VOC. The gap analysis was conducted to determine the risk assessment in quantitative scoring terms, which defined the cost, performance, and safety risks or identified the needed increase in performance or safety margin. Trades were performed to assess alternatives and identify gaps in capability and performance. The process is iterative, enabling refinement of alternatives and optimization of solution performance against defined requirements. Alternatives in the form of new or different technologies (i.e., design features) are developed, inserted back into the trade space, and assessed against the FOMs. Technology insertion is applied to the risk area, and the trade study process is iterated again. To facilitate the trade study analysis, USA developed and has delivered a desktop tool that facilitates decisionmaking and the analytical analysis of design options based on Customer evaluation criteria.

The following trade study definitions are provided to aid in the understanding of the study report.

a. Architecture - Ground systems, mission systems, and launch vehicle with associated onboard systems. Does not include payload in this context, as payload is framed by the missions and objectives definitions

b. Launch vehicle(s) - Comprised of Booster Stage and Core Stage, could also include Upper Stage (or Earth Departure Stage (EDS)) and/or In-Space elements

c. Propulsion system(s) - Engines and Propulsion Systems for Booster Stage, Core Stage, Upper Stage, and In-Space Stage

d. Block upgrade(s) - Changes to Initial Operational Capability (IOC) vehicle that increased reliability and/or decreased LCC without degradation in performance

e. Evolvability - Changes to IOC vehicle that provides a new increased performance capability that adds more missions and objectives into the architecture portfolio

2.2 Trade Process

USA's trade process is based on systems engineering principles that were first implemented at USA during Space Shuttle Upgrades Development and, since then, have been extensively used on CxP activities. The process starts with the VOC and derived inputs from the establishment of the GR&A and identification of missions and objectives. These top-level qualitative goals and mission objectives form the basis for definition of specific CCRs. CCRs are qualitatively defined by functional characteristics and attributes of the architecture, system, or subsystem, which, in turn, are quantitatively defined by the FOMs.

Examples of higher-order CCRs can be programmatic (e.g., business model using project alliance approach in lieu of conventional NASA/Contractor relationships) or technical (e.g., SLS defined as ascent vehicle to Low Earth Orbit (LEO) only with Core Stage and common interface and all other stages (i.e., Second/Upper and potentially Third)) and are mission specific based on Customer reference mission requirements.

Examples of functional performance characteristics and attributes are as follows:

a. Reduced payload cost derived from increased mass/volume that allows reductions in complexity of design, development, assembly, integration, and testing

b. Using a single launch to deploy several medium- to large-size payloads

c. Increased propellant loads to provide expanded access or access to multiple destinations

d. Increased power supplies and instrumentation

e. Increased payload life span with extra fuel, redundant systems, and additional shielding against harsh radiation exposure

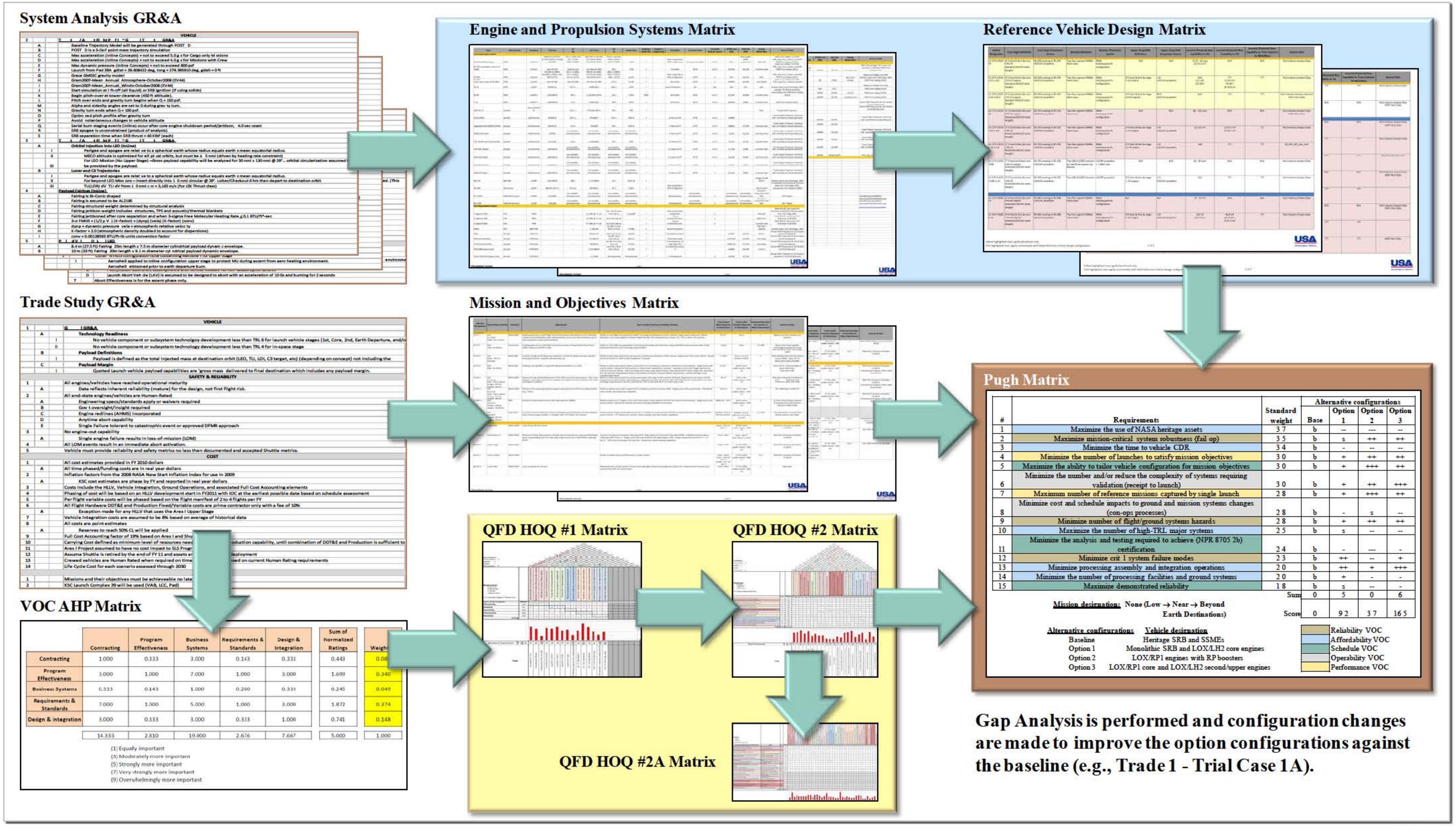
f. Reduction in mission timelines

g. Greater flexibility for extended science missions, including sample return

h. Ease of vehicle ground processing (reduced hazards due to lifts, hazardous commodities, complexity of vehicle propellant loading, etc.)

Trade iterations were conducted with the Customer's inputs in a CE environment to tier down from the architecture to the vehicle or system element (ground or mission) to the system levels. Our CE process included the capture of operator inputs during the requirements phase, including, for example, mission planning for normal and contingency operations, Ground Support Equipment (GSE) operation, nominal process flow, and in-field processing maintenance. Figure 2-2 shows the basic trade study flow process that was used and its iterative nature. With the foundation of VOC and governed by the trade GR&A and trade inputs, the quantifiable FOM and resultant trade trial cases provide architectural ranking against different alternative configurations.

Within the trade process, AHP and QFD are used to systematically determine relative importance and "build in" Customer wants and needs into the design based on the VOC, respectively.



System Analysis GR&A

			VEHICLE
2			T t /A t R M P f l ~G l T t GR&A
	A		Baseline Trajectory Model will be generated through POST D
	B		POST D is a 3-DoF point mass trajectory simulation
	C		Max acceleration (Inline Concepts) = not to exceed 5.0 g s for Cargo only M ssions
	D		Max acceleration (Inline Concepts) = not to exceed 4.0 g s for Missions with Crew
	E		Max dynamic pressure (Inline Concepts) = not to exceed 800 psf
	F		Launch from Pad 39A gdlat = 28.608422 deg, long = 279.395910 deg, gdalt = 0 ft
	G		Grace GM02C gravity model
	H		Gram2007-Mean Annual Atmosphere-October2008 (EV44)
	I		Gram2007-Mean_Annual_Winds-October2008 (EV44)
	J		Start simulation at l ft-off (all liquid) or SRB ignition (if using solids)
	K		Begin pitch-over at tower clearance (450 ft altitude).
	L		Pitch over ends and gravity turn begins when Q = 150 psf.
	M		Alpha and sideslip angles are set to 0 during grav ty turn.
	N		Gravity turn ends when Q = 100 psf.
	O		Optim zed pitch profile after gravity turn
	P		Avoid nstantaneous changes in vehicle attitude
	Q		Serial burn staging events (inline) occur after core engine shutdown period/jettison, 4.0 sec coast
	R		SRB apogee is unconstrained (product of analysis)
	S		SRB separation time when SRB thrust = 40 Klbf (each)
3			T t /A t R M P f l ~G l T t GR&A
	A		Orbital Injection into LEO (inLine)
		i	Perigee and apogee are relat ve to a spherical earth whose radius equals earth s mean equatorial radius.
		ii	MECO altitude is optimized for all pt cal orbits, but must be ≥ 5 nmi (driven by heating rate constraint)
		iii	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.
	B		Lunar and C3 Trajectories
		i	Perigee and apogee are relat ve to a spherical earth whose radius equals earth s mean equatorial radius.
		ii	For beyond LEO Miss ons – Insert directly into 1 0 nmi circular @ 29°. Loiter/Checkout 6 hrs then depart to destination orbit
		iii	TLI(LOR) dV TLI dV from 1 0 nmi c rc = 3,165 m/s (for J2X Thrust class)
4			Payload Fairings (inLine)
	A		Fairing is Bi-Conic shaped
	B		Fairing is assumed to be AL2195
	C		Fairing structural weight determined by structural analysis
	D		Fairing jettison weight includes structures, TPS and acoustic/thermal blankets
	E		Fairing jettisoned after core separation and when 3-sigma Free Molecular Heating Rate ≤ 0.1 BTU/ft²-sec
	F		3-σ FMHR = (1/2 ρ V) (K-factor) = (dynp) (vela) (K-factor) (conv)
	G		dynp = dynamic pressure vela = atmospheric relative veloc ty
	H		K-factor = 2.0 (atmospheric density doubled to account for dispersions)
	I		conv = 0.00128593 BTU/ft-lb units conversion factor
5			P l dV l ll L 1 LEO
	A		8.4 m (27.5 ft) Fairing 25m length x 7.5 m diameter cylindrical payload dynam c envelope.
	B		10 m (33 ft) Fairing 20m length x 9.1 m diameter cyl ndrical payload dynamic envelope.

		i	Aeroshell applied to inline configuration upper stage to protect MU during ascent from aero heating environment.
			Aeroshell ettisoned prior to earth departure burn.

	D	Launch Abort Veh cle (LAV) is assumed to be designed to abort with an acceleration of 10 Gs and burning for 2 seconds
7		Abort Effectiveness is for the ascent phase only.

Trade Study GR&A

			VEHICLE
1			G l GR&A
	A		Technology Readiness
		i	No vehicle component or subsystem technolgoy development less than TRL 6 for launch vehicle stages (1st, Core, 2nd, Earth Departure, and/o
		ii	No vehicle component or subsystem technology development less than TRL 4 for in-space stage
	B		Payload Definitions
		i	Payload is defined as the total injected mass at destination orbit (LEO, TLI, LOI, C3 target, etc) (depending on concept) not including the
	C		Payload Margin
		i	Quoted Launch vehicle payload capabilities are 'gross mass delivered to final destination which includes any payload margin.
			SAFETY & RELIABILITY
1			All engines/vehicles have reached operational maturity
	A		Data reflects inherent reliability (mature) for the design, not first flight risk.
2			All end-state engines/vehicles are Human-Rated
	A		Engineering specs/standards apply or waivers required
	B		Gov t oversight/insight required
	C		Engine redlines (AHMS) incorporated
	D		Anytime abort capability
	E		Single Failure tolerent to catastrophic event or approved DFMR approach
3			No engine-out capability
	A		Single engine failure results in loss-of-mission (LOM)
4			All LOM events result in an immediate abort activation.
5			Vehicle must provide reliabilty and safety metrics no less then documented and accepted Shuttle metrics.
			COST
1			All cost estimates provided in FY 2010 dollars
	A		All time phased/funding costs are in real year dollars
2			Inflation factors from the 2008 NASA New Start Inflation Index for use in 2009
	A		KSC cost estimates are phase by FY and reported in real year dollars
3			Costs include the HLLV, Vehicle Integration, Ground Operations, and associated Full Cost Accounting elements
4			Phasing of cost will be based on an HLLV development start in FY2011 with IOC at the earliest possible date based on schedule assessment
5			Per flight variable costs will be phased based on the flight manifest of 2 to 4 flights per FY
6			All Flight Hardware DDT&E and Production Fixed/Variable costs are prime contractor only with a fee of 10%
	A		Exception made for any HLLV that uses the Ares I Upper Stage
7			Vehicle integration costs are assumed to be 8% based on average of historical data
8			All costs are point estimates
	A		Reserves to reach 50% CL will be applied
9			Full Cost Accounting factor of 19% based on Ares I and Shu
10			Carrying Cost defined as minimum level of resources need oduction capability, until combination of DDT&E and Production is sufficient to
11			Ares I Project assumed to have no cost impact to SLS Progr
12			Assume Shuttle is retired by the end of FY 11 and assets ar deployment
13			Crewed vehicles are Human Rated when required on time sed on current Human Rating requirements
14			Life Cycle Cost for each scenario assessed through 2030
1			Missions and their objectives must be achieveable no late
2			KSC Launch Complex 39 will be used (VAB, LCC, Pad)

VOC AHP Matrix

	Contracting	Program Effectiveness	Business Systems	Requirements & Standards	Design & Integration	Sum of Normalized Ratings	Weight
Contracting	1.000	0.333	3.000	0.143	0.333	0.443	0.08
Program Effectiveness	3.000	1.000	7.000	1.000	3.000	1.699	0.340
Business Systems	0.333	0.143	1.000	0.200	0.333	0.245	0.049
Requirements & Standards	7.000	1.000	5.000	1.000	3.000	1.872	0.374
Design & Integration	3.000	0.333	3.000	0.333	1.000	0.741	0.148
	14.333	2.810	19.000	2.676	7.667	5.000	1.000

(1) Equally important
(3) Moderately more important
(5) Strongly more important
(7) Very strongly more important
(9) Overwhelmingly more important

Engine and Propulsion Systems Matrix

Reference Vehicle Design Matrix

Mission and Objectives Matrix

QFD HOQ #1 Matrix

QFD HOQ #2 Matrix

QFD HOQ #2A Matrix

Pugh Matrix

#	Requirements	Standard weight	Base	Option 1	Option 2	Option 3
			Alternative configurations			
1	Maximize the use of NASA heritage assets	3 7	b	--	---	--
2	Maximize mission-critical system robustness (fail op)	3 5	b	s	++	++
3	Minimize the time to vehicle CDR	3 4	b	-	--	--
4	Minimize the number of launches to satisfy mission objectives	3 0	b	+	++	++
5	Maximize the ability to tailor vehicle configuration for mission objectives	3 0	b	+	+++	++
6	Minimize the number and/or reduce the complexity of systems requiring validation (receipt to launch)	3 0	b	+	++	+++
7	Maximum number of reference missions captured by single launch	2 8	b	+	+++	++
8	Minimize cost and schedule impacts to ground and mission systems changes (con-ops processes)	2 8	b	-	s	--
9	Minimize number of flight/ground systems hazards	2 8	b	+	++	++
10	Maximize the number of high-TRL major systems	2 5	b	s	--	--
11	Minimize the analysis and testing required to achieve (NPR 8705 2b) certification	2 4	b	-	---	-
12	Minimize crit 1 system failure modes	2 3	b	++	--	+
13	Minimize processing assembly and integration operations	2 0	b	++	+	+++
14	Minimize the number of processing facilities and ground systems	2 0	b	+	-	-
15	Maximize demonstrated reliability	1 8	b	s	--	-
		Sum	0	5	0	6
		Score	0	9 2	3 7	16 5

Mission designation: None (Low → Near → Beyond Earth Destinations)

Alternative configurations	Vehicle designation
Baseline	Heritage SRB and SSMEs
Option 1	Monolithic SRB and LOX/LH2 core engines
Option 2	LOX/RP1 engines with RP boosters
Option 3	LOX/RP1 core and LOX/LH2 second/upper engines

- Reliability VOC
- Affordability VOC
- Schedule VOC
- Operability VOC
- Performance VOC

Gap Analysis is performed and configuration changes are made to improve the option configurations against the baseline (e.g., Trade 1 - Trial Case 1A).

HLPT-SATS_004

Figure 2-2. Trade study flow

After trade study iterations to the level desired was achieved, a gap analysis was conducted to determine the weaknesses against the VOC. The gap analysis provided quantification to the risk assessment. The risk assessment is conducted to define the cost, performance, and safety risks or to define the level of increase required in the performance or safety margin. Alternatives are assessed against the identified gaps in capability and performance. The gap analysis and risk assessment process are iterative after each run of the trade process, enabling refinement of alternatives and optimization of solution performance against defined requirements. Alternatives in the form of new or different technologies were developed and assessed against the FOMs. Technology insertion was applied to the risk area, and the trade study process is iterated again.

Based on years of Space Shuttle processing, testing, and checkout, examples of operability to ensure significant benefits in the reduction of recurring LCCs and promote efficiencies are as follows:

a. Ensure that access ports and doors are sized and located to permit replacement of parts and that hardware is accessible for preventive and corrective maintenance

b. Ensure a finite but complete set of payload interfaces of hardware and software (power-electrical, avionics-communication) for multiple different payload Customer requirements

c. Avoid multiple redundancy tolerance checks on onboard trajectory initialization data, and avoid the limitation of uplink content

While all of these guidelines are real experiences, they are critical in the definition of the early requirements development. If these requirements are not considered in the conceptual definition of the architecture, the resultant design characteristics will be inadequately weak in these areas, and the gap analysis in the corresponding risk assessment uncovered that weakness for recommendation correction.

Safety and reliability are integral in a design process; they drive associated costs and need to be considered in the trade space early. Similar processes and tools (i.e., Critical Item Review, Failure Modes Effects Analysis (FMEA), and Hazard Analysis), which are commonly used in design and development of vehicles and systems, were used in simplified form during the trade study gap analysis. In addition, other processes and tools like human factors assessments for designs were used to determine the risk to mission success based on the VOC.

Based on the acceptable level of risk assessed by the Customer and determined within its likelihood and consequence rating, risk was quantified, and alternatives were sought via new or upgraded technologies or different design characteristics. The technology was placed into the trade process and the trade study rerun. If the technology was deemed as an improvement to support the VOC, the technology insertion is considered as "buying down" the risk. For example, system reliability can be maximized through selection of components that provide the greatest operational integrity. It cannot be accomplished across the complete system due to affordability or operational constraints, but applied technology selection can provide the most benefit for cost trade.

Based on NASA Technology Program briefings, boiloff is one of the largest In-Space cryo propulsion technology challenges. Alternatives should not only consider mitigating this risk with advanced cryo Thermal Protection System (TPS) technology but also removing the cryogenics from the architecture, thus, removing the risk by identifying a more suitable In-Space propulsion energy source (e.g., nuclear, solar/electrical, combination).

2.3 AHP, QFD, and Pugh Matrix Tools

2.3.1 AHP Implementation

The AHP technique is most effective in determining the relative importance of evaluation criteria when assessing multiple candidate solutions. The process supports parametric analysis for advanced multivariate and modeling techniques (e.g., correlation, regression, decision matrix, Monte Carlo). For USA's trade study, a Pugh Matrix was applied to iterate combinations of candidate characteristics against a defined baseline to optimize capability or performance across a spectrum of CCR, characteristics and attributes, or FOM. It was determined that a classical AHP ranking scheme would be deployed to ensure clear separation between the FOMs. USA instituted the (1) equally important, (3) moderately more important, (5) strongly more important, (7) very strongly more important, and (9) overwhelmingly more important scoring breakdown.

Once the FOMs are defined against the CCRs, the AHP provides a method that decomposes the decisionmaking process into a series of relative comparisons that converts subjective assessments of relative importance into a set of overall scores or weights with accurate prioritization based on CCRs. The prioritized and ranked CCRs and functional characteristics and attributes start to define the basic architecture of the Customer's SLS system, as shown in Figure 2-3. The House of Quality (HOQ) provides a disciplined, structured analytical process based on Customer inputs.

A robust and logical methodology is required to develop requirements, manage the analysis of multiple alternatives, and enable the selection of solutions that best address the requirements and objectives. For example, in the HOQ #1, understanding the relative importance determined with AHP is essential for prioritizing design characteristics and attributes. The process also provides clear understanding of the significance of weighting applied to the CCRs. Weighting plays a key role in the performance outcome of each alternative. Weights are often derived from or driven by fundamental GR&A made early in the

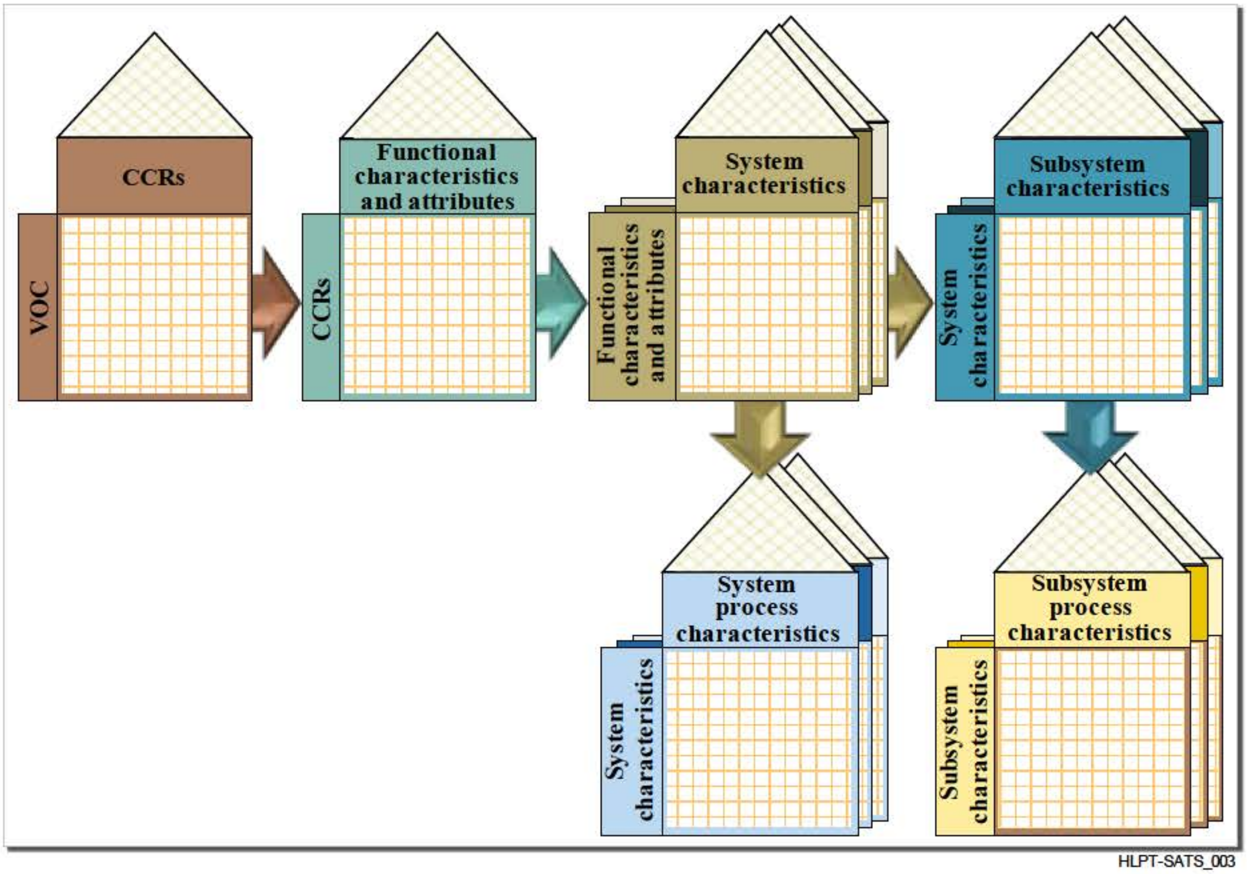


Figure 2-3. Product and Process QFD Process within the trade study

process. GR&A can impact both the definition of CCRs and the scoring of the FOM. Fundamental assumptions, such as "ship and shoot" versus "complete test and verify," can have significant influence on relative scoring for processing schedules and launch rates, labor resources, equipment and facility costs, and overall LCC for the program.

2.3.2 Customer *QFD Implementation*

USA employed an HOQ variant of the quality function deployment methodology for trade study analysis and requirement development in support of developing the trade space. The QFD technique is a recognized analytical Industry standard that provides a disciplined, structured process, employing a matrix-based approach to leverage Customer criteria to achieve performance objectives.

As previously mentioned, starting with the VOC and governed by the trade study GR&A, the Customer's voice is the starting input into the QFD implementation, where the CCRs are decomposed from the VOC and developed for both the functional performance characteristics and programmatic characteristics. As the trade study matures further into the trade space, the CCRs are further decomposed into lower and lower quantifiable FOM as characteristics and attributes. Each CCR had a target assigned to ensure quantification of each requirement. For HOQ #1, pair-wise comparisons between each VOC goal and each CCR was performed and scored on an 0 (none), 1 (weak), 3 (moderate), and 9 (strong) relationship scale. Then, correlations between each CCR were performed and scored on a strong positive, positive, none, negative, and strong negative scale to determine their interdependencies on each other.

2.3.3 Pugh Matrix Implementation

USA deployed a decision-matrix method to evaluate the alternative configurations. The Pugh Matrix is a quantitative technique used to rank the multidimensional options of an option set. It is frequently used in the Aerospace Industry, specifically in engineering for making design decisions when multidimensional entities have to be analyzed.

USA's decision matrix consists of establishing a set of criteria based on the HOQ results, upon which the potential options can be decomposed, scored, and summed to gain a total score that can then be ranked. Importantly, the criteria are weighted based on the QFD rankings to allow a priority selection process. The advantage of this approach to decisionmaking is that subjective opinions about one alternative versus another can be made more objective. Another advantage of this method is that sensitivity studies can be performed. The gap analysis uncovers the weakness, resulting in changes in the trade inputs, and then reevaluated to determine if a lower-ranked alternative would out rank a competing alternative with those trade input changes (e.g., Core Stage diameter size increase, a more mature Second/Upper Stage propulsion system).

3.0 TRADE SPACE BOUNDARIES AND CONSTRAINTS

In developing and documenting the trade study boundaries, USA evaluated the MSFC-defined BAA Technical Objectives to determine if the GR&A were impacting the objectives and started with the NASA SLS Team-provided GR&A that were made available at contract start. USA decomposed the grouping into either launch vehicle and propulsion configuration-specific or architecture-specific GR&A distinct groups. Overlaps were captured in both sets of GR&A as appropriate. USA then conducted a relative importance and sensitivity assessment to validate the rules and assumptions along with their insertion location in the trade process. Systems analysis GR&A were compared to previous USA architecture study GR&A, then adjusted accordingly. Additional rules and assumptions were extracted from new policies as they were issued (e.g., White House National Space Policy, etc.). Measured adjustments to the rules and assumptions (i.e., moving fixed constraints to variables in the trade space; e.g., propulsion system Technology Readiness Levels (TRLs)) were made to determine thresholds that affect the VOC. Trade iterations were conducted to determine the impact of those thresholds.

3.1 Systems Analysis GR&A

USA's GR&A that framed and bound the launch vehicle and propulsion configurations, called Systems Analysis GR&A, which are more technical in context, were reviewed and baselined prior to establishment of the vehicle alternative configurations or engine and propulsion systems were defined. Table 3-1 defines the established vehicle and propulsion systems analysis GR&A. Table 3-2 defines the established safety and reliability systems analysis GR&A.

Table 3-1. Vehicle and propulsion systems analysis GR&A

Vehicle and propulsion systems analysis GR&A
Trajectory/ascent flight profile - General trajectory GR&A
Max acceleration (inline concepts) = not to exceed 5.0 g's for cargo-only missions
Max acceleration (inline concepts) = not to exceed 4.0 g's for missions with crew
Max dynamic pressure (inline concepts) = not to exceed 800 psf
Launch from Pad 39A: gdlat = 28.608422 deg, long = 279.395910 deg, gdalt = 0 ft
Grace GM02C gravity model
Gram2007-Mean_Annual_Atmosphere-October2008
Gram2007-Mean_Annual_Winds-October2008
Start simulation at lift-off (all liquid) or SRB ignition (if using solids)
Begin pitchover at tower clearance (450 ft altitude)
Pitchover ends and gravity turn begins when Q = 150 psf

Table 3-1. Vehicle and propulsion systems analysis GR&A (continued)

Vehicle and propulsion systems analysis GR&A
Alpha and sideslip angles are set to 0 during gravity turn
Gravity turn ends when Q = 100 psf
Optimized pitch profile after gravity turn
Avoid instantaneous changes in vehicle attitude
Serial burn staging events (inline) occur after core engine shutdown period/jettison, +4.0 sec coast
SRB apogee is unconstrained (product of analysis)
SRB separation time when SRB thrust = 40 Klbf (each)
Trajectory/ascent flight profile - Trajectory GR&A
Orbital injection into LEO (inline)
Perigee and apogee are relative to a spherical Earth whose radius equals Earth's mean equatorial radius
MECO altitude is optimized for elliptical orbits but must be $\geq$75 n.mi. (driven by heating rate constraint)
For LEO mission (no Upper Stage) - Gross payload capability will be analyzed for 30 n.mi. x 130 n.mi. at 29. Orbital circularization assumed to be provided by the payload
Lunar and C3 trajectories
Perigee and apogee are relative to a spherical Earth whose radius equals Earth's mean equatorial radius
For beyond LEO missions, insert directly into 130 n.mi. circular at 29°. Loiter/checkout 6 hr then depart to destination orbit
TLI (LOR) dV: TLI dV from 130 n.mi. circ = 3,165 m/s (for J2X Thrust class)
Payload fairings (inline)
Fairing is biconic shaped
Fairing is assumed to be AL2195
Fairing structural weight determined by structural analysis
Fairing jettison weight includes: structures, TPS and acoustic/thermal blankets
Fairing jettisoned after core separation and when 3-sigma free molecular heating rate $\leq$ 0.1 BTU/ft^2-sec
3-σ FMHR = (1/2 ρ V3) (K-factor) = (dynp) (vela) (K-factor) (conv)
dynp = dynamic pressure; vela = atmospheric relative velocity
K-factor = 2.0 (atmospheric density doubled to account for dispersions)
conv = 0.00128593 BTU/ft-lb units conversion factor
Payload volume (inline) LEO
8.4 m (27.5 ft) fairing: 25 m length x 7.5 m diameter cylindrical payload dynamic envelope
10 m (33 ft) fairing: 20 m length x 9.1 m diameter cylindrical payload dynamic envelope
Payload volume (inline) beyond LEO
8.4 m (27.5 ft) fairing: 9 m length x 7.5 m diameter cylindrical payload dynamic envelope
10 m (33 ft) fairing: 9 m length x 9.1 m diameter cylindrical payload dynamic envelope
Launch Abort System (LAS) and Boost Protective Cover (BPC)
LAS+BPC mass = 16,005 lbm + 2,331 lbm (18,336 lbm total)
BPC jettison at 27 seconds after SRB separation for no Upper Stage concept
LAS jettison at 30 seconds after SRB separation for no Upper Stage concept

Table 3-1. Vehicle and propulsion systems analysis GR&A (continued)

Vehicle and propulsion systems analysis GR&A
BPC jettison at 27 seconds after Upper Stage ignition
LAS jettison at 30 seconds after Upper Stage ignition
LAS+CM+SM+vehicle adapter length = 70.7 ft
Inline aerodynamics
Three-DoF aero and base force generated by EV33 for HLLV 2.5 Stage February 2009 generated for Ares V
Weights and sizing (INTROS for inline) - General W&S GR&A
Dry mass margins: Based on MGA schedule consistent with SAWE recommended practices. Composite structures = 25 percent
Propellant density:
LOX: 71.04 lbm/ft^3
LH_2: 4.404 lbm/ft^3
RP: 50.50 lbm/ft^3
Unusable tank volume (ullage gas/manufacturing tolerance/loading accuracy/internal equipment and structures/cryo tank shrinkage): For all stage concepts: 0.04 (3 percent for gas volume and 1 percent for cryo shrinkage and internal tank equipment)
Miscellaneous secondary structures calculated as 5 percent of LVA primary structures for inline configurations
Vehicle sizing is considered closed when the payload capability is between the target payload and the target payload plus 0.1 percent
Propellant allocation
FPR is 1 percent of the total ideal dV for the mission
Final stage carries the entire FPR
Any excess FPR is not calculated as payload
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)
Residuals (InLine): Stage residuals mass (lbm) = 0.0631 * (usable propellant)^0.8469 (based on INTROS mass estimating relationship)
Start propellant: (side mount and inline)
Core Stage calculated based on engine startup transients
Air start stages: zero start propellant allocated
Boiloff:
0.32 percent LH_2 and 0.21 percent LOX per day based on full Upper Stage propellant load
Additional boil-off for suborbital burn of engine(s)
Other propellant note:
Ascent propellant includes all LOX in vertical portion of feedline
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

Table 3-1. Vehicle and propulsion systems analysis GR&A (concluded)

Vehicle and propulsion systems analysis GR&A
Structures (LVA) - General structural GR&A
Launch vehicle safety factors for new stages = 1.4 ultimate strength (consistent with NASA-STD-5001)
Closing criteria for INTROS – In LVA add:
Design liftoff acceleration = as flown in trajectory (POST) plus 0.1 g
Design max acceleration = as flown in trajectory (POST) plus 0.1 g
Design max dynamic pressure = as flown in trajectory (POST) plus 10 psf
Payload = as flown in trajectory (POST) plus 5,000 lb
For propellant tanks, use 50 psia MEOP (50 psid). Head pressure is in addition to ullage pressure
Pressure relief of flight loads on Core and Upper Stage
For tanks with pressure relief of flight loads use 30 psia (15.3 psid). No safety factor on pressure relief load
3 sigma dispersion estimation on angle of attack
Structural Buckling Knockdown Factor (SBKF) of 0.65
No proof analysis on tanks
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)
Structures (LVA) - General structural GR&A - Material properties assumptions:
Core Stage all Al-Li 2195
Upper stages: Al-Li 2195 pressurized structures
Shroud: Al-Li 2195
Aluminum 2219: For forgings, pipes, and plates requiring resistance to corrosion and contamination if utilized
Al-Li 2195: No limit on plate thickness
Al-Li 2050: Properties from manufacturer's Web site
Composites: IM7/8552 quasi-isotropic if utilized
Core Stage uses room temperature material properties
Upper stages use cryogenic material properties (if available)

Table 3-2. Safety and reliability systems analysis GR&A

Safety and reliability systems analysis GR&A
Orion/LAS/and service module design assumed with 95-percent Orion abort reliability applied to all vehicles (does not include launch vehicle failure environment)
LAS jettison assumed to occur 30 sec after final ascent stage engine start
Post jettison aborts are accomplished with service module for noncatastrophic failures
LAS is assumed to be designed to abort with an acceleration of 10 g's and burning for 2 seconds

3.2 Trade Study Ground Rules and Assumptions

USA's GR&A that framed and bound the architecture trade, called Trade Study GR&A, which are more programmatic in context, were reviewed and baselined prior to establishment of any trade space were defined. Tables 3-3 through 3-6 define the established Trade Study Ground Rules and Assumptions for each aspect of the trade study.

In addition to the established Trade Study GR&A, upon developing the FOMs for each level of trade, thresholds and/or targets were established as boundaries and included, but are not limited to, operational date(s), development (e.g., DDT&E) and operations costs by FY, flight rate, and technology and application readiness schedule thresholds. See Section 5.0 for a complete set of merits.

Table 3-3. Vehicle trade study GR&A

Vehicle trade study GR&A
General GR&A
Technology readiness
No vehicle component or subsystem technology development less than TRL 6 for launch vehicle stages (1st, Core, 2nd, Earth departure, and/or 3rd)
No vehicle component or subsystem technology development less than TRL 4 for In-Space Stage
Payload definitions
Payload is defined as the total injected mass at destination orbit (LEO, TLI, LOI, C3 target, etc.) (depending on concept) not including the burnout mass of the final stage
Payload margin
Quoted launch vehicle payload capabilities are "gross mass" delivered to final destination, which includes any payload margin

Table 3-4. Safety and reliability trade study GR&A

Safety and reliability trade study GR&A
All engines/vehicles have reached operational maturity
Data reflects inherent reliability (mature) for the design, not first flight risk
All end-state engines/vehicles are human rated:
Engineering specs/standards apply or waivers required
Government oversight/insight required
Engine redlines (AHMS) incorporated
Anytime abort capability
Single failure tolerant to catastrophic event or approved DFMR approach
No engine-out capability:
Single engine failure results in Loss of Mission (LOM)
All LOM events result in an immediate abort activation
Vehicle must provide reliability and safety metrics no less then documented and accepted Shuttle metrics

Table 3-5. Cost trade study GR&A

Cost trade study GR&A
All cost estimates provided in FY 2011 dollars
All time phased/funding costs are in real year dollars
Costs include the HLLV, vehicle integration, ground operations, and associated full cost accounting elements
Phasing of cost will be based on an HLLV development start in FY 2011 with IOC at the earliest possible date based on schedule assessment
Per flight variable costs will be phased based on the flight manifest of two to four flights per FY
All flight hardware DDT&E and production fixed/variable costs are Prime Contractor only with a fee of 10 percent
Exception made for any HLLV that uses the Ares I Upper Stage
Vehicle integration costs are assumed to be 8 percent based on average of historical data
All costs are point estimates
Reserves to reach 50 percent CL will be applied
Full cost accounting factor of 19 percent based on Ares I and Shuttle experience
Carrying cost defined as minimum level of resources needed to maintain production capability until combination of DDT&E and production is sufficient to sustain
Ares I project assumed to have no cost impact to SLS Program
Assume Shuttle is retired by the end of FY 2011 and assets are available for redeployment
Crewed vehicles are human rated when required on timeline and costed based on current human rating requirements
LCC for each scenario assessed through 2030

Table 3-6. Operations trade study GR&A

Operations trade study GR&A
Missions and their objectives must be achievable no later than 2030
KSC Launch Complex 39 will be used (VAB, LCC, pad)

4.0 TRADE INPUT CONSIDERATIONS

4.1 Missions and Objectives

In reviewing the NASA Exploration Mission Manifest documentation and assessing a realistic flight rate, USA recognizes that a multiple-Customer HLLV capability is needed to sustain the SLS Program. The HLL system must be able to service multiple Customers affordably and provide a service that shall be available to NASA, DoD, and Commercial and international payload communities for a wide range of reference missions. While the heavy lift capability's primary objectives are for NASA Exploration and Scientific missions, the heavy lift capability would provide large-mass and large-volume payload developers, providers, and operators with a launch platform never before available. In the past, payloads were constrained or complexities, such as folding mechanisms, were added into payloads due to launch vehicle mass and volume limitations. In short, the launch vehicle affects almost every other aspect of payload capability and operation.

With an architecture that has the flexibility and adaptability of the launcher with the payloads, the HLLV's potential Customers encompass NASA's Human and Robotic Exploration and Scientific communities, including the Space Telescope Science Institute, DoD, international communities, and Commercial satellite providers. The SLS provides the large vehicle shrouds and injection capability that the large telescope and observatory payloads require.

While it's primary role is to deliver large elements, in both mass and volume, to orbit to support human and robotic planetary exploration, the HLL system's main mission would be to deliver interplanetary vehicle modules for construction, delivery, and servicing of orbital construction support elements; consumable (fuel) supplies and transfer systems; and large-volume and large-mass satellites to LEO, Geosynchronous Earth Orbit (GEO), Lagrange (L) points, and Trans-Lunar Injection (TLI).

The SLS primary reference missions are destinations beyond LEO, with an operational capability, but not design, to go to the ISS if necessary. As a secondary reference mission, the SLS variants can be used in emergency conditions, as contingency, or to supplement the Commercial Crew and Cargo providers. The heavy lift launcher may deliver an oversized NASA scientific payload to LEO or injection to rendezvous with a Near Earth Object (NEO) or deliver a space-based radar or space-based laser satellite for DoD, an X-37 platform, or other classified satellite into a specific orbit. The primary reference mission is beyond LEO, i.e., flexible-path-type missions.

It was vital to assess the trades early and weigh the objectives and priorities of NASA. The trades include a strong consideration for benefits that can be realized across the Commercial Industry, DoD, and International Partners (IPs), thus taking advantage of and successfully providing heavy lift capability to a wide spectrum of missions. The MSFC HLLV Study, Missions Assessed, dated May 20, 2010, provided a very good starting point for reference missions. In addition to that list, a set of MSFC study guidelines that affected mission definition were assessed as follows:

a. The ultimate destination is Mars; the mission should lead towards developing Mars capability

b. Requirements generated from a compelling mission sequence (roadmap) within a modified flex path scenario

c. IOC schedule/compelling mission capability options

d. Crew capable; ISS not precluded as a future potential mission

e. Propellant transfer/depot capability not available for early missions

f. Cooperation between robotic and human

g. Orion is the crew vehicle

USA encompassed those stated missions and expanded the study space to include other potential Customers and their reference missions. These missions were evaluated and ranked in relative importance via AHP to ensure that system capability is progressive against the timeline.

As a starting point, the mission objectives included but were not limited to the following:

a. Enabling advanced on-orbit flight test demonstrations

b. Enabling the new capability in large GEO platforms

c. Placement of large observatories at L1 and L2 (Sun-Earth L points)

d. Ability to expand observatory aperture (up to 16 m+) for revolutionary advances in astronomy and astrophysics

e. Lunar and planetary sample return missions

f. Enabling crewed missions to lunar, NEOs, and Mars destinations

g. Delivery of large Orbital Replacement Units (ORUs) for ISS under contingency conditions

h. Backup vehicle for ISS crew/cargo support

Quantifiable performance measurements to successfully achieve these missions were established to define each mission. USA defined metrics including but not limited to destination, mass and volume to destination, and mission timeline from separation through insertion to destination. The missions and objectives matrix was developed and comprised candidate missions (with designations) with respective objectives. Missions were grouped into the following three destination categories:

a. Low Earth - Encompasses eight mission profiles

b. Near Earth - Encompasses seven mission profiles

c. Beyond Earth - Encompasses six mission profiles (one mission excluded)

Parameters considered per mission designation included the following:

a. Primary Customer and destination location

b. Objective(s) and executive concept of operations (con-ops) summary (including timeline)

c. Total useful mass required at destination

d. Total useful volume required at destination

e. Estimated number of launches to meet objective

The missions and objectives matrix was bounded and framed by the trade study GR&A.

Table 4-1, Table 4-2, and Table 4-3 depict the Low Earth, Near Earth, and Beyond Earth destinations, respectively, for the missions and objectives considerations that the architectures were assessed against.

Table 4-1. Low Earth destinations

Mission designation	Destination location	Customer	Objective(s)	Exec con-ops summary, including timeline	Total useful mass required at destination	Total useful volume required at destination	Estimated number of launches to meet objective(s)	Source of data
Low Earth								
LE-FT-1	LEO Incl 29.0° Orbit 241 x 241 km	NASA ESMD	Enabling advanced on-orbit flight test demonstrations with delivery of LEO inflatable module mission/inflatable habitat demonstrations, aeroassist demonstrations and In-Space propulsion system demonstrations	Ability to insert flight test payloads for NASA technology development and OCT activities. Single launch with ascent vehicle. Payloads could include Bigelow Inflatable Habitat BA 330, Sierra Nevada Dream Chaser, ATV, HTV or other COTS vehicles	25 mT+	Varies	1	Bigelow and Sierra Nevada Corp Web sites
LE-FT-2	GEO Orbit 35,700 km	Commercial	Enabling advanced on-orbit flight test demonstrations of Space Based Solar Power (SBSP) collection satellites in GEO	Ability to insert flight test payloads for Commercial technology demonstrations along with NASA-sponsored Space Solar Power Exploratory Research and Technology (SERT) program	Varies	Varies	1 to TBD	Space Solar Power Satellite Technology Development at the Glenn Research Center, NHTC2000-12067
LE-EO-1	LEO Orbit 705 km (circular)	NASA SMD	Insertion of large Earth observation platforms into LEO for global warming, weather tracking/predictions, long-life multiple payloads, etc.	Ability to place large observatory apertures for revolutionary advances in Earth sciences. Single launch with ascent vehicle. Payload has final insertion or orbit transfer capabilities if required	2 - 19 mT	22 m L x 12 m H (similar to JWST)	1	NOAA Satellite Data and Information Service (NSDIS), Space, KH-12 Advanced Crystal, 4/25/2007
LE-EO-2	GEO Orbit 35,700 km	NASA SMD	Enabling new capability in large Earth observation platforms in GEO	Ability to expand observatory aperture (up to16m+) for revolutionary advances in astronomy and astrophysics. Single launch with ascent vehicle. Payload has final insertion or orbit transfer capabilities if required. Examples are the 10 mT Single Aperture Far Infrared Telescope (SAFIR), 4.8 mT Advanced Technology Large-Aperture Space Telescope (ATLAST), Stellar Imager (SI), Generation-X X-ray telescope, Submillimeter Probe of the Evolution of Cosmic Structure (SPECS) spectrometer, and the Dark Ages Lunar Interferometer (DALI)	25 mT+	18.3 m x 8.4 m usable volume = 696 m3	1	Heavy Lift for a New Paradigm in Space Operations, AIAA 2010-2290
LE-ISS-1	LEO Incl 51.6° Orbit 278 to 460 km Perigee 347 km Apogee 360 km	NASA SOMD	Delivery of large Orbital Replacement Units (ORUs) (control moment gyros, solar arrays, radiators, solar rotary joints, future upgrades via additional large modules) to ISS under contingency conditions	Ability to deliver pressurized and unpressurized spares and cargo to ISS to achieve 2020 goal. Single launch with ascent vehicle. Payload has on-orbit transfer and rendezvous capabilities possible carrying ATV, HTV with COTS vehicles with emphasis on heavy and large cargo elements only HLV could deliver (73mT to LEO with 40 mT of useful cargo to ISS)	40 mT	12 m x 8.4 m usable volume = 493 m3	1	Deep Space Operations Enabled by HLLV, AIAA Space Ops Conference, April 2010-1906
LE-ISS-2	LEO Incl 51.6° Orbit 278 to 460 km Perigee 347 km Apogee 360 km	NASA SOMD	Delivery of ISS crew/cargo backup support and placement of ISS crew escape module (e.g., Orion)	Ability to deliver crew and/or cargo to ISS in contingency posture to achieve 2020. Single launch with ascent vehicle. Payload has orbit transfer and rendezvous capabilities	20.5 mT	12 m x 8.4 m usable volume = 493 m3	1	CxP CARD Req't CA1005-PO
LE-MSS-1	HEO Incl 63.4° Perigee 538 km Apogee 39,300 km	NRO	Insertion of space-based sensors with large apertures (SBIRS)	Ability to place up to four large (2.1m X 1.9m X 6m) sensor system aperture (4.8 mT) for national reconnaissance. Single launch with ascent vehicle. Payload has insertion and station-keeping capabilities (no transfer)	SBIRS (4) = 19 mT	18.3 m x 8.4 m usable volume = 696 m3	1	Air Force Infrared Space Systems Directorate Facts Sheet 8/2010, NASA/CP-2008-214588
LE-MSS-2	LEO Incl 40.0° Perigee 403 km Apogee 420 km	DoD-USAF	Insertion of military defensive systems, including small boost-phase intercept satellites, heavy kinetic energy satellites, or multiple USAF X-37 orbital test vehicles	Ability to place a large missile defensive system or multiple (two) reconnaissance vehicles for national defense. Single launch with ascent vehicle. X-37 vehicle has insertion, station-keeping, and orbit transfer capabilities	X37B(2) = 9.98 mT BPI Sys:50-XX mT	X37B(2) = 232 m^3 (8.9 m L x 4.5 m W x 2.9 m H)	1 to TBD	CBO Alternatives to Boost Phase Missile Defense, July 2004

Table 4-2. Near Earth destinations

Mission designation	Destination location	Customer	Objective(s)	Exec con-ops summary, including timeline	Total useful mass required at destination	Total useful volume required at destination	Estimated number of launches to meet objective(s)	Source of data
Near Earth								
NE-LO-1	Lunar orbit	NASA SOMD	Lunar flyby with free return	Insertion into lunar orbit with crewed spacecraft. Direct insertion without LEO loitering	Orion = 20.5 mT (20.2 T)	18.3 m x 8.4 m usable volume = 696 m3	1	NASA HLLV Summary of Studies, 11/30/10
NE-EML1-1	Earth-Moon L1	NASA SOMD	Placement of large observatories and deep-space preparation/In-Space assembly/deep-space escape staging point for long-range, large missions at L1 (Earth-Moon Lagrange point)	Insertion of Advanced Compoton Telescope (ACT), SAFIR, or Modern Universe Space Telescope (MUST) into L1. Single launch with ascent vehicle and Upper Stage as EDS. Energy required for insertion C3 = -1.7 km^2/s^2. EDS second burn provides final insertion. Payload has station-keeping capabilities	Total = 35 mT	Total = 405 m^3 (9 m mirror x 5 m H)	1	NASA HLLV Summary of Studies, 11/30/10
NE-LO-2	Lunar cargo	NASA SOMD	Placement of large supplies on the lunar surface	Simplified version of the Cx mission profile for lunar cargo mission. Single launch preferred. EDS second burn for insertion into lunar orbit. Energy required for insertion C3 = -1.8 km2/s2. Rendezvous and docking capability, as required, provided by payload	Total = 57 mT	17.2 m x 8.8 m usable volume = 860 m3	1	NASA HLLV Summary of Studies, 11/30/10
NE-LS-1	Lunar surface	NASA SOMD	Placement of crewed mission to the lunar surface	Similar to Apollo mission profile for lunar surface mission	Total = 66 mT Orion = 20.5 mT (20.2 T) Lander = 45.7 mT (45 T)	17.2 m x 8.8 m usable volume = 860 m3	1 to 2	NASA HLLV Summary of Studies, 11/30/10
NE-LO-3	Lunar orbit	NASA SOMD	Lunar sample return mission	Based on previous Lunar return missions (Luna 24) and future test missions for lander and crewed mission. Assume mass requirements are 65 percent of crewed mission	Total = 43 mT	17.2 m x 8.8 m usable volume = 860 m3	1	Luna 24, NASA NSSDC ID: 1976-081
NE-SEL2-1	Sun-Earth L2	NASA SOMD	Placement of large observatories at L2 (Sun-Earth Lagrange point)	Insertion of ATLAST (~16 mT) into L2. Single launch with ascent vehicle and Upper Stage as EDS. Energy required for insertion C3 = -0.7 km2/s2. EDS second burn provide final insertion. Payload has station-keeping capabilities	Total = 56 mT	15 m x 6.5 m usable volume = 498 m3	1	NASA ATLAST Mission Concept Study, 05/10
NE-LS-2	Lunar surface	NASA SOMD	Placement of crewed mission to the lunar surface	Similar to Cx mission profile for lunar surface mission. Multiple launches. First launch to parking orbit in LEO. Stationkeep. Second launch with arrays within 45 days. Two launched payloads rendezvous and dock in Earth orbit, checkout as integrated vehicle. EDS second burn for insertion into lunar orbit. Energy required for insertion C3 = -1.8 km2/s2. Rendezvous and docking capability, as required, provided by payload	Total = 66 mT Orion = 20.5 mT (20.2 T) Lander = 45.7 mT (45 T)	17.2 m x 8.8 m usable volume = 860 m3	2	NASA HLLV Summary of Studies, 11/30/10

Table 4-3. Beyond Earth destinations

Mission designation	Destination location	Customer	Objective(s)	Exec con-ops summary, including timeline	Total useful mass required at destination	Total useful volume required at destination	Estimated number of launches to meet objective(s)	Source of data
Beyond Earth								
BE-NEO-1	NEO GP2	NASA SOMD, NASA SMD	Rendezvous and proximity operations of crewed spacecraft (e.g., Orion) with NEO GP2. Launch date of December 2019 on a 304-day mission to obtain detailed characterizations of surface morphology, internal structure, mineral composition, topography, collisional history, density, particle size, etc.	Ability to explore another celestial body within reasonable distance from Earth. First two launches to parking orbit in LEO. Stationkeep. Launch with crew of four within 45 days. Vehicles rendezvous in Earth orbit; parking orbit checkout as integrated vehicle. Earth departure with total delta-V of 7.5 km/s to NEO rendezvous (3.5km/s for Earth/Moon departure and 3.0km/s for rendezvous). Mission duration is 304 days with a 10-day rendezvous at NEO	Total = 399 mT NTV (348 mT) habitat + shielding (51mT)	17.2 m x 10 m usable volume = 925 m3	3 to 4	NASA HLLV Summary of Studies, 11/30/10; CxP Advanced Programs Office 2006 Study Report
BE-NEO-2	NEO QJ142	NASA SOMD NASA SMD	Rendezvous and proximity operations of crewed spacecraft (e.g., Orion) with NEO QJ142. Launch date of April 2024 on a 200-day mission to obtain detailed characterizations of surface morphology, internal structure, mineral composition, topography, collisional history, density, particle size, etc.	Ability to explore another celestial body within reasonable distance from Earth. First two launches to parking orbit in LEO. Stationkeep. Launch with crew of four within 45 days. Vehicles rendezvous in Earth orbit; parking orbit checkout as integrated vehicle. Earth departure with total delta-V of 7.5 km/s to NEO rendezvous (3.5km/s for Earth/Moon departure and 3.0km/s for rendezvous). Total mission duration is 200 days with a 10-day rendezvous at NEO	Total = 383 mT NTV (333mT) habitat + shielding (50mT)	17.2 m x 10 m usable volume = 925 m3	3 to 4	NASA HLLV Summary of Studies, 11/30/10; CxP Advanced Programs Office 2006 Study Report
BE-NEO-3	NEO AO10	NASA SOMD NASA SMD	Rendezvous and proximity operations of crewed spacecraft (e.g., Orion) with NEO AO10, launch date of Sept 2025 on a 155-day mission to obtain detailed characterizations of surface morphology, internal structure, mineral composition, topography, collisional history, density, particle size, etc.	Ability to explore another celestial body within reasonable distance from Earth. First two launches to parking orbit in LEO. Stationkeep. Launch with crew of four within 45 days. Vehicles rendezvous in Earth orbit; parking orbit checkout as integrated vehicle. Earth departure with total delta-V of 7.5 km/s to NEO rendezvous (3.5km/s for Earth/Moon departure and 3.0km/s for rendezvous). Total mission duration is 155 days with a 10 day rendezvous at NEO	Total = 367mT NTV (318mT) habitat + shielding (49 mT)	17.2m x 10 m usable volume = 925 m3	3 to 4	NASA HLLV Summary of Studies, 11/30/10; CxP Advanced Programs Office 2006 Study Report
BE-NEO-4	NEO SM84	NASA SOMD, NASA SMD	Rendezvous and proximity operations of crewed spacecraft (e.g., Orion) with NEO SM84	VIOLATES TRADE STUDY GR&A. WILL NOT BE CONSIDERED AS FIRST OPPORTUNITY TO RENDEZVOUS WOULD NOT BE UNTIL JULY 2046				
BE-MARS-1	Mars Orbital (Mars Orbit)	NASA SMD	Placement of large robotic precursor mission to surface with sample return Constellation-Enabled Mars Mission Exhibiting New Technology (CEMMENT)	Mars Transport Vehicle (MTV) consists of EDS (110 mT), Earth Return Vehicle (ERV) and small lander (12 mT), and AeroSystem and large lander (28 mT). Aerocapture vehicle into a low-Mars orbit (atmospheric entry speed 7.4 km/sec). Would reenter 24 MT for landing and land 8 MT on Mars surface. Would return three 500-gram Mars samples to Earth	Total = 150 mT EDS (110 mT) large lander and AeroSys (28 mT) ERV and small lander (12 mT)	17.2m x 10 m usable volume = 925 m3	1 to 2	CEMMENT, 10/2008
BE-MARS-2	Mars Orbital (Mars Orbit)	NASA SOMD	Insertion of crewed spacecraft (e.g., Orion) into Mars orbit and return	MTV consists of Nuclear Thermal Rocket (NTR) propulsion stage (131 mT) with 3-25 klbf engines (ISP ~ 900 sec) and Core Stage propellant loading augmented with inline LH_2 tank for TMI maneuver (101 mT). Transit Habitat (TransHab) transports six crewmembers round trip from LEO to high-Mars orbit and return. Supports six crew for 400 days (plus 550 contingency days in Mars orbit). Crew direct entry in Orion at 12 km/s. NTR stage and payload elements are delivered to LEO and assembled via autonomous rendezvous and docking	Total = 283 mT MTV + TransHab (273 mT) Orion (10 mT)	17.2m x 10 m usable volume = 925 m3	3 to 4	Mars DRA Version 5.0, 02/2009

Table 4-3. Beyond Earth destinations (concluded)

Mission designation	Destination location	Customer	Objective(s)	Exec con-ops summary, including timeline	Total useful mass required at destination	Total useful volume required at destination	Estimated number of launches to meet objective(s)	Source of data
BE-MARS-3	Mars DRA 5.0 (Mars Surface)	NASA SOMD	Insertion of crewed spacecraft (e.g., Orion) into Mars orbit, descent to surface for sortie, and return	Reference BE-MARS-2 for MTV/TransHab/Orion details. Surface Habitation (SHAB) cargo vehicle consists of an NTR propulsion stage, aeroshell (43 mT), and the SHAB (64 mT) that is predeployed to Mars orbit, transports 6 crew from Mars orbit to the surface, and supports the crew for up to 550 days on the surface of Mars. Descent/Ascent cargo Vehicle (DAV) consists of an NTR propulsion stage, aeroshell (43 mT) and the DAV (63.7 mT) that is predeployed to Mars orbit, utilizes locally produced propellants (oxygen) from Mars atmosphere (methane transported from Earth), and transports six crew from the surface of Mars to high-Mars orbit	Total = 849 mT MTV + TransHab + Orion (283 mT) SHAB + Aeroshell (238 mT) DAV + Aeroshell (238 mT)	17.2 m x 10 m usable volume = 925 m3	8 to 9	Mars DRA Version 5.0, 02/2009

4.2 Engine and Propulsion Systems

The SLS mission profile can be broken down into the following distinct flight phases: ascent, insertion and transit. These phases can be accomplished with a variety of vehicle staging approaches, depending on engine selection and the resulting system architecture's performance. Ascent flight can be accomplished with either a Booster Stage (First Stage Booster) and Core Stage combined performance vehicle or a Core Stage-only performance vehicle. Insertion flight can be accomplished with a Second (and Third) Stage, also known as an Upper Stage, depending on the vehicle staging configuration. Transit or transfer can be accomplished with an In-Space Stage performance vehicle.

At a global level, current engine technology exists either in solid motor or liquid engine to support the expeditious development of the ascent flight phase for SLS. Current United States (U.S.) liquid engine capabilities, as defined by the current SSP Space Shuttle Main Engine (SSME) RS-25D (>390 klbf thrust) or the Delta IV Evolvable expendable Launch Vehicle (EELV) RS-68A (>650 klbf thrust) at sea level, are mature and have demonstrated reliability for a human-rated vehicle (or that expedite human rating certification). Industry evaluations of existing capabilities do not identify any near-term opportunities for significant performance growth with this or similar technology that would provide value to the SLS Program. SSME upgrades have been identified to further improve the current engines to reduce production and operating costs. RS-68 engines also have identified DDT&E modifications to improve operational efficiency and mission profile. Development costs to grow performance by 50 percent would be significant and would detract from the primary need to develop new engines to enable fast transit to NASA's interplanetary exploration objectives. Other engines considered to be on the fast track to human rating are those with previous NASA history, such as the Liquid Oxygen (LOX)/RP1 First Stage engine variant F1A or LOX/LH_2 Upper Stage engine variant J-2X, which were used during Apollo/Saturn programs.

The SLS Core Stage alone can provide the "off the pad" lift performance to insert the other stage(s) into LEO. The reference mission requirements dictate the launcher staging requirements. For LEO, no Upper Stage or a payload transfer vehicle would be required. For GEO or higher inclination LEO positions, an Upper Stage similar to the baselined Ares I Upper Stage with a LOX/LH_2 J-2X engine would be required. For Near Earth space, depending on the specific destination, an EDS with either a single LOX/LH_2 J-2X engine or three MB-100 engines would be required to leave Earth's dwell and inject the payload into the correct alignment and desired position. A transfer vehicle could be added if the reference mission required for orbital positioning and In-Space thrusting (if the payload does not have the inherent capability). For Beyond Earth space destinations, once a parking orbit is obtained, an EDS, along with an

Upper Stage or transfer vehicle, would be required to align and position the vehicle to its final mission objective destination.

In reviewing the technology maturation of powerplants and accompanying engines, propulsion technology development trade space for In-Space propulsion has numerous performance, cost, and schedule benefits. While not a single technology "out-of-the-box" will satisfy all requirements, combining them would provide significant merits. As an example, when integrating a nuclear thermal-electric powerplant with a Variable Specific Impulse Magnetoplasma Rocket (VASIMR) engine, the VASIMR's unique ability to trade between power usage, Specific Impulse (Isp), and thrust in flight; the combination of nuclear powerplant complimented with large solar arrays for in-situ power capture; a suitable and robust power storage system; and ion thruster engines provides a very positive initial trade quick look.

In assessing "quickest" to IOC for the "lowest" DDT&E costs, the combination of SSP/Cx heritage SRBs and demonstrated reliable, liquid-fuelled engines provides a mature technology that is well understood and has demonstrated reliability to support the ascent phase for the SLS mission. To achieve lower LCCs, engine and propulsion system requirements and processes must be changed from SSP to achieve the lower operational budget targets.

In assessing the engine and propulsion systems required, USA partnered with Pratt-Whitney Rocketdyne (PWR) and Aerojet to acquire both technical and programmatic data so a comprehensive trade could be accomplished. USA separated out the engine and propulsion systems by stage: First Stage, Core Stage, Second/Upper Stage, and In-Space Stage, depending on the vehicle configuration requirements.

An engine and propulsion systems matrix was developed and comprised candidate engines and propulsion systems with respective performance parameters. Engines and propulsion systems were grouped into three classifications.

Parameters considered per engine or propulsion system included in the following:

a. Type, manufacturer, propellant, weight

b. Sea-level and vacuum thrust, sea-level and vacuum Isp

c. Readiness, impacts to stage, availability, production rate, estimated number of engines per vehicle

d. DDT&E cost, production cost, existing market base

Engine and Propulsion Systems were bounded and framed by the systems analysis GR&A.

4.2.1 Core Stage

Early benchmarking conducted by USA indicates that liquid engines that could perform to ascent performance requirements, clustered in either four or five engines, limited by vehicle diameter and potential thrust structure requirements, currently exist within the engine market. While smaller thrust class LOX/RP engines, such as the SpaceX Merlin variants, are newer, they do not produce the thrust required without significantly increasing the number of engines and, correspondingly, reducing the reliability (e.g., Falcon 9 Core Stage Main Propulsion System (MPS)). A large part of the open trade space was whether the RS-68 or RS-25 core engines and their respective production capacities are optimal engines based on the LCCs. From a performance perspective, the recent Industry Team developed engine characteristics with variable power levels based on existing test data to determine the performance impact to the vehicle system.

In reviewing the proposed programmatic data for the Core Stage engine cost and schedule parameters that are available, the SSME RS-25D (residual Government inventory) and RS-68A (Delta IV) manufacturing are available immediately to satisfy the early vehicle timeline requirements. The RS-68B first flight engines would be available approximately at the same time as the RS-25E due to existing work already accomplished under the advanced Ares V engine development. A small amount of engine development cost on integrating the RS-68A would be required, and the recurring cost per engine set (five RS-25E versus four RS-68B) would be significantly more (approximately 90 percent per engine; approximately $165M per flight set) for the SSME derivative.

Table 4-4 depicts the Core Stage considerations that the architectures were assessed against.

Table 4-4. Core Stage engine systems

Type	Manuf	Propellant	SL thrust	SL Isp (sec)	Vac thrust	Vac Isp (sec)	Wt (lb)	Readiness (NASA TRL)	Availability	Prod rate	Est. reqd no. of engines per launch	DDT&E cost	Prod cost	Existing market base	Source of data
Core Stage engine system															
RS-25D (SSME heritage)	PWR	LO_2/LH_2	394,812 lb (104.5 percent): 217,750 lb (67 percent); 416,169 lb (109 percent)	364 (104.5 percent); 313 (67 percent); 368 (109 percent)	490, 603 lb (104.5 percent) 313, 454 lb (67 percent); 511,970 lb (109 percent)	453 (104.5 percent); 451 (67 percent); 453 (109 percent)	7,750	9	Now using Shuttle MPS configuration	15 engines available	4 or 5	[redacted]	(b) (4) [redacted]	NASA SSP	NASA HLLV GR&A, 11/15/09; PWR_eng_HLLV_industry_teamR4; PWR Price Catalog, 2/4/11
RS-25E (expendable version of SSME)	PWR	LO_2/LH_2	same RS-25D	same RS-25D	512 klbm at 109 percent	450	7,750	6	72 months	10/yr	4 or 5	[redacted]	[redacted]	Currently none	NASA HLLV GR&A, 11/17/09; PWR_eng_HLLV_industry_teamR4; PWR Price Catalog, 2/4/11
RS-68A	PWR	LO_2/LH_2	702,000 lb (108 percent): 663,000 lb (102 percent); 330,986 lb (57 percent)	359 (108 percent); 357 (102 percent); 318 (57 percent)	796,815 lb (108 percent): 757,815 lb (102 percent); 425,500 lb (57 percent)	408 (108 percent); 409 (102 percent); 409 (57 percent)	15,145	9	Now using Delta IV MPS configuration	10/yr	4	[redacted]	[redacted]	USAF	PWR_eng_HLLV_industry_teamR4; PWR Price Catalog, 2/4/11
RS-68B (man-rated variant of RS-68A)	PWR	LO_2/LH_2	same RS-68A	same RS-68A	797 klbm at 108 percent	412	15,145	6	60 months	10/yr	4	[redacted]	[redacted]	USAF/NASA	NASA HLLV GR&A, 11/17/09; PWR_eng_HLLV_industry_teamR4
RS-76	PWR	$LO_2/RP1$	900,000 lb	308	1,000,000 lb	342	8,720	5	Out of production	N/A	N/A	[redacted]	[redacted]	N/A	Andrews Space and Technology, 2001
RS-84	PWR	$LO_2/RP1$	1,064,000 lb	301	1,130 klbm at FPL	324	17,919	5 (PDR)	60 months	10/yr	4 or 5	[redacted]	[redacted]	Currently none	Encyclopedia Astronautica, NASA PDR package, The Boeing Company, www.boeing.com; PWR Price Catalog, 2/4/11
F-1A	PWR	$LO_2/RP1$	1,800,000 lb	271	2,020,500 lb	304	19,876	6	60 months	10/yr	3 or 4	[redacted]	[redacted]	Currently none	PWR Price Catalog, 2/4/11
LR87-AJ-11	Aerojet	Aerozine 50 $/N_2O_4$		252	553 klbm at FPL	304	4,780	9	N/A	N/A	N/A	[redacted]	[redacted]	Titan	Purdue AAE Propulsion
AJ26 (ORSC)	Aerojet	$LO_2/RP1$	339,900 lb	298	379,900 lb	331	2,985 lb	9	4 years to ATP	15/yr	4 or 5	[redacted]	[redacted]	N/A	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)
Upgraded AJ26 (500 klb) (ORSC)	Aerojet	$LO_2/RP1$	506,814 lb	312	552,000	339	4,160 lb	7	5 years to ATP	15/yr	3 or 4	[redacted]	[redacted]	Currently none	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)
500Klb GG Engine	Aerojet	$LO_2/RP1$	500,000	N/A	N/A	N/A	N/A	5	5 years to ATP	15/yr	3 or 4	[redacted]	[redacted]	Currently none	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)
1M LOX/ RP dual powerhead	Aerojet	$LO_2/RP1$	1,013,628	312	1,104,000 lb	339	8,320	5	5 years to ATP	15/yr	2 or 3	[redacted]	[redacted]	Currently none	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)

Table 4-4. Core Stage engine systems (concluded)

Type	Manuf	Propellant	SL thrust	SL Isp (sec)	Vac thrust	Vac Isp (sec)	Wt (lb)	Readiness (NASA TRL)	Availability	Prod rate	Est. reqd no. of engines per launch	DDT&E cost	Prod cost	Existing market base	Source of data
1M ORSC booster	Aerojet	LO_2/ kerosene	1,000,000 lb	N/A	N/A	N/A	N/A	5	5.5 years to ATP	15/yr	2 or 3	(b) (4) [redacted]	[redacted]	Currently none	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)
1MLB GG Engine	Aerojet	LO_2/ kerosene	1,000,000 lb	N/A	N/A	N/A	N/A	5	5.5 years to ATP	15/yr	2 or 3	[redacted]	[redacted]	Currently none	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)
2M ORSC Booster	Aerojet	LO_2/ kerosene	2,000,000 lb	N/A	N/A	N/A	N/A	5	6 years to ATP	15/yr	1	[redacted]	[redacted]	Currently none	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)
2Mlb GG Engine	Aerojet	LO_2/ kerosene	2,000,000 lb	N/A	N/A	N/A	N/A	5	6 years to ATP	15/yr	1	[redacted]	[redacted]	Currently none	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)
RD-170	NPO-EM	LO_2/RP	1,632,000 lb	309	1,777,000 lb	337	19,351	9	N/A	N/A	N/A	N/A	N/A	Energia LB and Zenith 2	Andrews Space and Technology, 2001
RD-180	RD Amross	LO_2/RP	868 klbm at FPL	311	933,000 lb	337	12,081	9	Now using Atlas V MPS configuration	N/A	N/A	N/A	N/A	Atlas III and V	Encyclopedia Astronautica & Andrews Space and Technology, 2001; www.pw.utc.com literature (2009)
RP1.25M	PWR (RD Amross)	LO_2/RP	1,250,000	N/A	N/A	N/A	N/A	5	Developmental	N/A; developmental	3	Not available; developmental	Not available; developmental	Develop in conjunction with DoD	HEFT report
RP-2.0M	PWR (RD Amross)	LO_2/RP	2,000,000	N/A	N/A	N/A	N/A	5	Developmental	N/A; developmental	3	Not available; developmental	Not available; developmental	Develop in conjunction with DoD	HEFT report

4.2.2 First-Stage Booster

The existing Reusable Solid Rocket Motor (RSRM) for Shuttle provides significant thrust (3,100 klb) with demonstrated performance and reliability. Continuous development and improvement, in addition to innovative evaluation and analysis tools, over the life of the Shuttle have resulted in a significant understanding of this technology. The recent significant recurring cost increase and stack integration costs of an open-grained segmented solid do have disadvantages. The infrastructure impacts of using the SRB configuration, which would require little or no facility and equipment modifications, is advantageous. Although perceived as resource intensive to produce and assemble, studies are being undertaken to significantly reduce production and integration costs to best serve the SLS Program.

Monolithic Solid Rocket Motors (MSRMs) in the thrust range of 1M lbf are not a current capability and would require development and infrastructure costs. Recent work by Aerojet on the Vega launch vehicle for the European Space Agency (ESA) has demonstrated a large MSRM with a thrust range of 800 klbf. Currently, no Industry facility exists to pour a Solid Rocket Motor (SRM) of larger than 800 klbf. MSRMs have significant transportation and handling limitations due to the inability to effectively inspect for propellant grain damage following shipment to the launch site. Options to multiple-batch cast from a number of separate mixes is possible but introduces several critical risks, including but not limited to, increased opportunity for foreign object contamination and susceptibility to mix-to-mix variability within the same motor. Development of a continuous cast capability for a motor this size would mitigate some of the risks associated with multiple batch production but would be an expensive infrastructure to develop and sustain. Depending on the number of MSRM required, per flight costs will increase to cover additional separation systems and for TPS closeout for each individual booster. If continuous casting could be put in place at the launch site, two of the problems would be mitigated, thereby enabling an effective cost trade. If cost effective against flight rate, evolution to MSRB may become a viable "technology insertion" if segmented solids are chosen to expedite IOC.

In reviewing the proposed programmatic data for the First Stage engine cost and schedule parameters that are available, the SRB RSRM is a known entity, while the MSRM elements present a higher uncertainty.

Table 4-5 depicts the First Stage Booster considerations that the architectures were assessed against.

Table 4-5. First-Stage booster systems

Type	Manuf	Propellant	SL thrust	SL Isp (sec)	Vac thrust	Vac Isp (sec)	Wt (lb)	Readiness (NASA TRL)	Availability	Prod rate	Est. reqd no. of engines per launch	DDT&E cost	Prod cost	Existing market base	Source of data
First Stage Booster System															
Four-segment RSRB	ATK	PBAN	-	-	3.1 Mlb at T + 1 sec	~ 267; burn time 126 sec	1,252	9	Now using Shuttle configuration	N/A	2	N/A	N/A	NASA SSP	NASA 074-99, Garry Lyles HLV Overview 11/30/2010, Andrews Space and Technology, 2001
Five-segment RSRB	ATK	PBAN	-	-	3.5 Mlb at T + 1 sec	Burn time 126 sec	N/A	7	3 years In development for Ares I	N/A	2	N/A	N/A	NASA Ares I	NASA 069-07, Garry Lyles HLV Overview 11/30/2010
Five-segment RSRB	ATK	HTPB	-	-	4.7 Mlb at T + 1 sec	Burn time 108 sec	N/A	5	6 years In development for Ares I	N/A	2	N/A	N/A	Currently none	NASA 309-07, Garry Lyles HLV Overview 11/30/2010
SRMU	ATK	88 percent HTBP	-	-	1,700,000	~ 286; burn time 137.8 sec	770	9	Recent production	N/A	N/A	N/A	N/A	Titan IVB, three segment	Andrews Space and Technology, 2001
AJ62	Aerojet	ANB-3745	-	-	281,213 lbf	Burn time 93 sec	102	9	In production, Atlas V	N/A	N/A	(b) (4) [redacted]	[redacted]	Atlas V	Aerojet SRM Propulsion for SLS Studies (1/14/10) (Bossard)
P80 Avio monolithic	Aerojet	HTPB 1912	-	-	677,800 lbf	~280; burn time 110 sec	209	5	Preliminary study	N/A	N/A	[redacted]	[redacted]	Avio (Italy)	Aerojet SRM Propulsion for SLS Studies (1/14/10) (Bossard)
P110 Avio monolithic	Aerojet	HTPB 1912	-	-	795,000 lbs	~287; burn time 114.3 sec	254	7	In development, first Vega flight 2011	N/A	N/A	[redacted]	[redacted]	Avio (Italy)	Aerojet SRM Propulsion for SLS Studies (1/14/10) (Bossard)
P230 SRM (segmented)	Aerojet	HTPB 1814	-	-	1,572,000 lbf	~275; burn time 129 sec	594	9	In production, Ariane 5	N/A	N/A	[redacted]	[redacted]	Ariane 5	Aerojet SRM Propulsion for SLS Studies (1/14/10) (Bossard)
P235, P250 SRM (segmented)	Aerojet	HTPB 1814	-	-	>1.5 Mlbf	N/A	N/A	5	In development	N/A	4	[redacted]	[redacted]	N/A	Aerojet SRM Propulsion for SLS Studies, Appendix B (1/14/10) (Bossard)

4.2.3 Second/Upper Stage

Development efforts for the proposed Cx J-2X Upper Stage engine have been progressing and have been reported to be 38 months from initial unmanned operational capability or 58 months from manned certification. As it is likely that SLS will require an Upper Stage or orbital insertion stage to reach LEO; continued development of an air start capability and upgrades to an existing engine provides best value to the Government. As indicated above the Industry has not identified any near-term opportunities for significant performance growth with this or similar liquid engine technology that would provide value in time to support the SLS DDT&E Program. Smaller engines of similar technology levels, such as the MB-100, would be available within 5 years to support SLS development by incorporating modifications to allow throttling and variable mix ratios. The MB-100 is a test-stand-ready engine developed by Mitsubishi/Boeing as part of a class of affordable modular liquid rocket engines. Modifications to these two engines provide a relatively low development cost and leverage existing technologies. This would optimize the distribution of SLS DDT&E funds and allow NASA to focus on technology development of In-Space or transfer engines.

In reviewing the proposed programmatic data for the Second/Upper Stage engine cost and schedule parameters that are available, the J-2X "to go" development is 38 months for nonhuman rating and 58 months for human rating based on POR since it has already started development, while the MB-100 engine would take much longer (68 months) to develop (approximately 15-percent longer timeline for the human-rated version). The nonrecurring development cost of the MB-100 engine is approximately 10 percent more than the J-2X, and the recurring cost of one J-2X is about the same as three MB-100 engines.

Table 4-6 depicts the Second/Upper Stage considerations that the architectures were assessed against.

Table 4-6. Second/Upper Stage engine systems

Type	Manuf	Propellant	SL thrust	SL Isp (sec)	Vac thrust	Vac Isp (sec)	Wt (lb)	Readiness (NASA TRL)	Availability	Prod rate	Est. reqd no. of engines per launch	DDT&E cost	Prod cost	Existing market base	Source of data
Upper Stage Engine System															
J-2X (derived from Saturn V J-2)	PWR	LO_2/LH_2	N/A	N/A	238 klbm at 81 percent; 294 klbm at 100 percent	453	5,587	7 (post CDR)	3 years (unmanned) 5 years (manned)	10 engines/ year	1 for LO2/LH2 CS 4 - 5 for LO2/RP CS	(b) (4) [redacted]	[redacted]	Currently none	NASA CDR package, Garry Lyles HLV Overview, 11/30/2010; PWR Price Catalog, 2/4/11
RL10A4-3 (derived from RL10A4-2 and RL10B-2)	PWR	LO_2/LH_2	N/A	N/A	21 klbm at derated PL max: 24 klbm at FPL	N/A	N/A	N/A	N/A	N/A	4	[redacted]	[redacted]	N/A	NASA HLLV GR&A, 11/17/09
RL10A4	PWR	LO_2/LH_2	N/A	N/A	22,300 lb	451	370	9	Now using Atlas 3 and 4 Centaur config	10 engines/ year	4	[redacted]	[redacted]	Atlas 3 and 4 Centaur	Andrews Space and Technology, 2001; PWR Price Catalog, 2/4/11
RL10A-5	PWR	LO_2/LH_2	N/A	N/A	max: 24 klbm at FPL 21 klbm at derated PL	452	N/A	N/A	N/A	N/A	4	[redacted]	[redacted]	N/A	NASA HLLV GR&A, 11/17/09
RL10B-2	PWR	LO_2/LH_2	N/A	N/A	24,800 lb	466	664	7	Out of production	N/A	N/A	[redacted]	[redacted]	N/A	PWR Price Catalog, 2/4/11
RL10C	PWR	LO_2/LH_2	N/A	N/A	22,900 lb	450	410	7	N/A	10 engines/ year	4	[redacted]	[redacted]	AF/ULA	PWR Price Catalog, 2/4/11
NGE	PWR	LO_2/LH_2	N/A	N/A	35,000 lb	460	600	6	48 mos	10 engines/ year	4	[redacted]	[redacted]	Currently none	PWR Price Catalog, 2/4/11
AJ52 (solid)	Aerojet	ANB-3772	N/A	N/A	91,300 lb	burn time 48 sec	14,418	9	Full-scale development test completed	N/A	N/A	[redacted]	[redacted]	N/A	Aerojet SRM Propulsion for SLS Studies (1/14/10) (Bossard)
AJ92 (solid)	Aerojet	ANB-3783	N/A	N/A	277,807 lb	burn time 59.4 sec	63,844	5	In development, first flight 2012	N/A	N/A	[redacted]	[redacted]	N/A	Aerojet SRM Propulsion for SLS Studies (1/14/10) (Bossard)
LR91-AJ-11	Aerojet	LO_2/RP	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	[redacted]	[redacted]	N/A	
LOX/hyd augmented expander	Aerojet	LOX/LH_2	N/A	N/A	100,000 lb	462 sec	1,310	5	N/A	15/yr	2 or 3	[redacted]	[redacted]	N/A	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)
LOX/hyd augmented expander	Aerojet	LOX/LH_2	N/A	N/A	150,000 lb	461	2,100	5	N/A	15/yr	2	[redacted]	[redacted]	N/A	
LOX/hyd augmented expander	Aerojet	LOX/LH_2	N/A	N/A	200,000 lb	Developmental	Developmental	5	N/A	15/yr	1 or 2	[redacted]	[redacted]	N/A	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)
LOX/RP1 ORSC	Aerojet	LOX/RP1	N/A	N/A	100,000 lb	Developmental	Developmental	5	N/A	15/yr	2 or 3	[redacted]	[redacted]	N/A	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)
LOX/RP1 ORSC	Aerojet	LOX/RP1	N/A	N/A	200,000 lb	Developmental	Developmental	5	N/A	15/yr	1 or 2	[redacted]	[redacted]	N/A	"Liquid Engine Propulsion Technical Data," provided by Aerojet (Bossard)
MB-100	MB/PWR	LO_2/LH_2	N/A	N/A	100,000 lb	464	2,000	5	60-month DDT&E program, add 8 months for man rating		2 or 3	[redacted]	[redacted]	N/A	PWR_eng_HLLV_industry_teamR4

4.2.4 In-Space Stage

Based on benchmarking exercises conducted by USA, while the new technologies for Core Stage and Second/Upper Stage propulsion are not advancing on a timeline that will support an operational HLL system by 2025, the transfer vehicle In-Space propulsion systems have matured significantly over the past decade. These are considered "newer" technologies that are either in use on current satellites or in development and can be leveraged within the next 10 years to field an operational engine for this intended use. Most of these technologies have been funded by the U.S. Government, e.g., NASA's Propulsion and Cryogenics Advanced Development (PCAD) Project under the NASA Exploration Technology Development Program, and others have been supported by NASA under Space Act Agreements, e.g., Ad Astra Rocket Company's VASIMR engines.

With a technology insertion focus on In-Space Stage propulsion, the VASIMR propulsion system will be advanced today from a 50-kW engine in the development laboratory to a 100-kW human-rated engine. Two 100-kW VASIMR engines will comprise the VF-200 propulsion system that will be used on ISS for orbit boosts. With respect to a orbital transfer vehicle, a VASIMR-powered spacecraft for Beyond Earth Orbit (BEO) mission destinations will be much more efficient than traditional integrated chemical rockets at moving crew and cargo through space. A spacecraft requiring 34 mT through TLI would require over 60 mT of LOX/LH_2 propellant, while numerous VASMIR propulsion systems with a 1-MW solar array would be capable of delivering the same spacecraft transporting only about 8 mT of argon propellant.

Regardless of the technological solution, the In-Space engine requirements and characteristics that are critical are that the engine shall be restartable, capable of very high acceleration, and shall maintain a high level of reliability through the system's life cycle. With these governing requirements and characteristics, there are two major technology deployment approaches on In-Space engines: chemical propellant and in-situ resource. A detailed In-Space Stage propulsion system evaluation is described in Section 6.3.

Table 4-7 depicts the In-Space Stage considerations that the architectures were assessed against.

Table 4-7. In-Space Stage propulsion systems

Type	Manuf	Propellant	SL thrust	SL Isp (sec)	Vac thrust	Vac Isp (sec)	Life	Readiness (NASA TRL)	Availability	Prod rate	Est. reqd no. of engines per launch	DDT&E cost	Prod cost	Existing market base	Source of data
In-Space propulsion system															
Electric (Hall effect or ion)															
BPT-2000	Aerojet	Hall effect	N/A	N/A	92 - 142 mN	1,500 - 1,800	>2,500 kW-hrs	9	XFC flight qual	N/A	N/A	N/A	N/A	N/A	
HiVHAc	Aerojet	Hall effect	N/A	N/A	21 - 139 mN	1,200 - 2,900	>60,000 kW-hrs	9	XFC flight qual	N/A	N/A	N/A	N/A	N/A	
BPT-4000	Aerojet	Hall effect (Xe)	N/A	N/A	79 - 278 mN	1,100 - 2,000	>90,000 kW-hrs	9	Flying on AEHF-1	N/A	N/A	N/A	N/A	N/A	
XR-12 (TSAT)	Aerojet	Hall effect	N/A	N/A	200 - 800 mN	1,200 - 2,300	>240,000 kW-hrs	9	XFC flight qual	N/A	N/A	N/A	N/A	N/A	
XR-20 (HPPS)	Aerojet	Hall effect	N/A	N/A	200 - 1,300 mN	1,000 - 3,000	>400,000 kW-hrs	6	Developmental	N/A	N/A	N/A	N/A	N/A	
XR-50	Aerojet	Hall effect	N/A	N/A	400 - 3,200 mN	1,000 - 2,000	>1,000,000 kW-hrs	5	Developmental	N/A	N/A	N/A	N/A	N/A	
NEXT	Aerojet/ NASA	Hall effect (Xe)	N/A	N/A	42 mN/kW	4,200	N/A	6	In development with NASA	N/A	N/A	N/A	N/A	N/A	
NEXT 6.9 kW	Aerojet/ NASA	Hall effect (Xe)	N/A	N/A	235 mN	>4,100	N/A	6	In development with NASA	N/A	N/A	N/A	N/A	N/A	
12 kW HTPS	Aerojet	Hall effect (Xe)	N/A	N/A			N/A	6	N/A	N/A	N/A	N/A	N/A	N/A	
20 kW HIPEP	Aerojet	Hall effect (Xe)	N/A	N/A	540 mN	6,000 - 9,000	N/A	4	N/A	N/A	N/A	N/A	N/A	N/A	
3 kW HPPS	Aerojet	Hall effect (Xe)	N/A	N/A	90 mN/kW	3,000	N/A	6	USAF research	N/A	N/A	N/A	N/A	N/A	
20 kW HPPS	Aerojet	Hall effect (Xe)	N/A	N/A	46 - 91 mN/kW	1,000 - 3,000	N/A	6	USAF research	N/A	N/A	N/A	N/A	N/A	
601HP Thruster	Boeing	Xenon ion	N/A	N/A	18 mN	2,568	N/A	9	In use (satellite)	N/A	N/A	N/A	N/A	N/A	www.Boeing.com/defense-space/space/bss/factsheets/xips
702 Thruster	Boeing	Xenon ion	N/A	N/A	165 mN	3,800	N/A	9	In use (satellite)	N/A	N/A	N/A	N/A	N/A	www.Boeing.com/defense-space/space/bss/factsheets/xips
ESA DS4G	ESA	Xenon ion	N/A	N/A	<0.5 nM/cm2	<10,000	N/A	7	N/A	N/A	N/A	N/A	N/A	N/A	www.esa.int/ACT/pro/pp/DS4G
VF-200	Ad Astra	Argon ion	N/A	N/A	~5.5N at 200 kW	5,000	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	
Hypergolic chemical															
	Aerojet	N_2H_4/NTO	N/A	N/A	>100 lbf	>320	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	
Shuttle OMS	Aerojet	MMH/NTO	N/A	N/A			N/A	9	In use (Shuttle)	N/A	N/A	N/A	N/A	N/A	
Hydrogen/methane/other chemical															
ESEX Arcjet	Aerojet	Ammonia	N/A	N/A	0.5 N	800	N/A	9	N/A	N/A	N/A	N/A	N/A	N/A	
ESEX Arcjet	Aerojet	LOX/LH_2	N/A	N/A	0.5N	360 - 450	N/A	7	N/A	N/A	N/A	N/A	N/A	N/A	
	Aerojet	LOX/LCH_4	N/A	N/A	3,500 lbf	335 - 375	N/A	6	Developmental	N/A	N/A	N/A	N/A	N/A	
RS-18	PWR	LOX/LCH_4	N/A	N/A	5,500 lbf	350 - 395	N/A	6	Developmental	N/A	N/A	N/A	N/A	N/A	
R-4D-15DM HiPAT	Aerojet	N/A	N/A	N/A	100 lbf	328	N/A	8	N/A	N/A	N/A	N/A	N/A	N/A	
R-42DM BiProp	Aerojet	N/A	N/A	N/A	200 lbf	327	N/A	6	N/A	N/A	N/A	N/A	N/A	N/A	
AMBR	Aerojet	N/A	N/A	N/A	140 lbf	332	N/A	6	N/A	N/A	N/A	N/A	N/A	N/A	

4.3 Reference Vehicle Design

USA's Operations Integration Team has a strong understanding of the overall process in both the Expendable Launch Vehicle (ELV) and Reusable Launch Vehicle (RLV) operating environments, due primarily to its extensive history with the SSP. The Space Shuttle system is an example of a hybrid system with an expendable External Tank (ET), recoverable SRBs, and an Orbiter.

It is crucial that the selection between expendable and reusable systems is determined early, especially between Manufacturing and Ground Operations. This balance of priorities and operational objectives should continue through all DDT&E and mission operations phases, thereby ensuring the most effective long-term cost profile. At some point in the life cycle, it may become more affordable to move from reusable to expendable on specific launch vehicle elements. It should not be assumed that in all cases a reusable system would be the least costly until all operational factors are considered. Examples of metrics that were considered when decisions regarding expendable and reusable systems were made include the following:

a. Cost of manufacturing versus cost of refurbishment

b. Mission manifest flexibility due to scheduling limitations

c. Manufacturing or refurbishment preparation cycle time

d. Overall LCC of both systems

e. Integration, facilities, systems, and labor costs

f. Trade between performance and margin to support reuse

g. Certification and recertification requirements, processes and cost

h. Partial versus total recovery/reuse

i. Recovery by land, water, or a combination

The economics of each type of vehicle configuration were studied and traded with a realistic understanding of the expected operations and launch rate.

As experienced on the SSP, generic age-life values such as "10 years" or "100 missions" that are unsupported by analysis or test will, at a minimum, mislead the assessment on the front end or drive unsustainable program costs following implementation.

NASA's selection of vehicle architecture will necessarily impose significant impacts on systems and operations. Reusability implies recoverability. That aspect will drive system and component safety factors for structural integrity, achievable reuse cycles, typical component and system weight, and manufacturing and certification costs. While the reusability benefit is typically focused on affordability, retrieving hardware after use provides the user insight into the performance and behavior of the system, a benefit recognized on Shuttle Orbiter, SSME, and SRB but not on ET element. The specific architecture will also drive infrastructure. This will include considerations for initial handling as well as recovery systems. These could be in the form of recovery vessels or runway facilities and their systems, GSE, and operations personnel.

Some operations areas are more impacted by the type of launch system selected, at a vehicle level, than others. For example, Flight Operations processes, data inputs, and architecture are not affected by whether a system is new or refurbished, so the vehicle system type has very little impact. However, vehicle attributes, such as sizing of onboard software, uplink capability, system complexity, performance envelopes, and autonomous onboard flight operations can have a large Flight Operations impact for design, crew training, and mission complexity. In contrast, Ground Operations could be significantly affected by an RLV or hybrid and would need to consider skills required for integration, recovery, refurbishment, and testing of the system to be flight ready.

The Reference Vehicle Design (RVD) matrix was developed and comprised benchmark alternative configurations with designations. The RVDs were single-configuration vehicles for point solutions but could be integrated to provide an evolvable configuration path.

Parameters considered per vehicle designation included the following:

a. Core Stage definition (width, height, type, and number of engines) and Core Stage propulsion system

b. First-Stage Booster definition (width, height, type, and number of engines) and booster propulsion system

c. Second/Upper Stage definition (width, height, type, and number of engines) and Upper Stage propulsion system

d. Inserted mass to LEO, TLI, and/or transinjection to NEO/Mars

In-Space Stage was treated as an integrated payload with activation of the stage post-EDS burnout.

The RVDs were bounded and framed by the systems analysis GR&A.

Table 4-8 depicts the RVD alternative configurations that architectures were assessed against.

Table 4-8. RVD alternative configurations

Vehicle designation	Core Stage definition	Core Stage propulsion system	Booster definition	Booster propulsion system	Second/Upper Stage (EDS) definition	Second/Upper Stage (EDS) propulsion system	Inserted (payload) mass capability to LEO (mT)	Inserted (payload) mass capability to TLI or trans-injection for NEO/BEO (if noted) (mT)	Source data
LV-275-3.RS25D-2.4S	27.5-foot (8.4-m) dia core 3 RS-25D Standard LOX/LH$_2$ tanks (length)	RS-25D LOX/LH$_2$ propellant	Two four-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	N/A	N/A	55 - 62	N/A	HLLV Industry Analysis Data Trial Case 3, Alternative 1, Phase 1
LV-275-3.RS25-2.4 S-4.RL10	27.5-foot (8.4-m) dia core 3 RS-25 Standard LOX/LH$_2$ tanks (length)	RS-25D evolving to RS-25E LOX/LH$_2$ propellant	Two four-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	27.5-foot (8.4-m) dia stage 4 RL10 engines	RL10 LOX/LH$_2$ propellant	40 - 50	13 - 20	HLLV Industry Analysis Data and MSFC HLLV Data Trial Case 2, Baseline
LV-275-4.RS25-4S	27.5-foot (8.4-m) dia core 4 RS-25 Standard LOX/LH$_2$ tanks (length)	RS-25D evolving to RS-25E LOX/LH$_2$ propellant	Two four-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	N/A	N/A	45 - 52	N/A	HLLV Industry Analysis Data
LV-275-4.RS25-2.4S-1.J2X	27.5-foot (8.4-m) dia core 4 RS-25 Standard LOX/LH$_2$ tanks (length)	RS-25D evolving to RS-25E LOX/LH$_2$ propellant	Two four-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	27.5-foot (8.4-m) dia stage 1 J-2X engine	J-2X LOX/LH$_2$ propellant	118 - 125	40 - 43	HLLV Industry Analysis Data
LV-275-4.RS25D-2.5S	27.5-foot (8.4-m) dia core 4 RS-25 Stretched LOX/LH$_2$ tanks (length)	RS-25D LOX/LH$_2$ propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	N/A	N/A	73 - 77	N/A	HLLV Industry Analysis Data Trial Case 3, Alternative 1, Phase 2
LV-275-4.RS25E-2.5 S-1.J2X	27.5-foot (8.4-m) dia core 4 RS-25 Stretched LOX/LH$_2$ tanks (length)	RS-25D LOX/LH$_2$ propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	27.5-foot (8.4-m) dia stage 1 J-2X engine	1-J-2X LOX/LH$_2$ propellant	112 - 120	36 - 40	HLLV Industry Analysis Data
LV-275-5.RS25-5S	27.5-foot (8.4-m) dia core 5 RS-25 Stretched LOX/LH$_2$ tanks (length)	RS-25D evolving to RS-25E LOX/LH$_2$ propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	N/A	N/A	86 - 101	N/A	HLLV Industry Analysis Data MSFC HLLV Data
LV-275-5.RS25-2.5S-1.J2X	27.5-foot (8.4-m) dia core 5 RS-25D Stretched LOX/LH$_2$ tanks (length)	RS-25D evolving to RS-25E LOX/LH$_2$ propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	27.5-foot (8.4-m) dia stage 1 J-2X engine	1-J-2X LOX/LH$_2$ propellant	123 - 126	53 - 55	HLLV Industry Analysis Data Trial Case 2, Option 1
LV-275-5.RS25E-2. 5CP-1.J2X	27.5-foot (8.4-m) dia core 5 RS-25 Stretched LOX/LH$_2$ tanks (length)	RS-25E LOX/LH$_2$ propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	27.5-foot (8.4-m) dia stage 1 J-2X engine	1-J-2X LOX/LH$_2$ propellant	123 - 129	43 - 46	Trial Case 3, Alternative 1, Phase 3
LV-275-5.RS25E-2. 5S-2.J2X	27.5-foot (8.4-m) dia core 5 RS-25E Stretched LOX/LH$_2$ tanks (length)	RS-25E LOX/LH$_2$ propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	27.5-foot (8.4-m) dia stage 2 J-2X engine	2-J-2X LOX/LH$_2$ propellant	137 - 143	49 - 51	03_SATs_KO_Gov_Arch Family 3, Option 5 Trial Case 4, Baseline
LV-275-5.RS25E-2. 5CP-1.J2X	27.5-foot (8.4-m) dia core 5 RS-25 Stretched LOX/LH$_2$ tanks (length)	RS-25E LOX/LH$_2$ propellant	Two five-segment RSRMs Composite cases	PBAN Existing grain/fin configuration	27.5-foot (8.4-m) dia stage 1 J-2X engine	1-J-2X LOX/LH$_2$ propellant	127 - 132	46 - 48	Trial Case 3, Alternative 1, Phase 4

Table 4-8. RVD alternative configurations (continued)

Vehicle designation	Core Stage definition	Core Stage propulsion system	Booster definition	Booster propulsion system	Second/Upper Stage (EDS) definition	Second/Upper Stage (EDS) propulsion system	Inserted (payload) mass capability to LEO (mT)	Inserted (payload) mass capability to TLI or trans-injection for NEO/BEO (if noted) (mT)	Source data
LV-275-5.RS25E-2.5CP-2.J2X	27.5-foot (8.4-m) dia core 5 RS-25 Stretched LOX/LH_2 tanks (length)	RS-25E LOX/LH_2 propellant	Two five-segment RSRMs Composite cases	PBAN Existing grain/fin configuration	27.5-foot (8.4-m) dia stage 2 J-2X engine	2-J-2X LOX/LH2 propellant	137 - 143	47 - 50	Trial Case 3, Alternative 1, Phase 5
LV-275-5.RS25E-2.5CH-2.J2X	27.5-foot (8.4-m) dia core 5 RS-25 Stretched LOX/LH_2 tanks (length)	RS-25E LOX/LH_2 propellant	Two five-segment RSRMs Composite cases	HTPB New grain/fin configuration	27.5-foot (8.4-m) dia stage 2 J-2X engine	2-J-2X LOX/LH_2 propellant	146 - 151	49 - 50	Trial Case 3, Alternative 1, Phase 6
LV-275-5.RS25E-6.AJ110-21J2X	27.5-foot (8.4-m) dia core 5 RS-25 Stretched LOX/LH_2 tanks (length)	RS-25E LOX/LH_2 propellant	Six AJ110 Monolithic SRBs	AJ110 monolithic	27.5-foot (8.4-m) dia stage 1 J-2X engine	1-J-2X LOX/LH_2 propellant	131 - 140	47 - 50	Trial Case 4, Option 3
LV-275-5.RS25-2LRB	27.5-foot dia (8.4-m) core 5 RS-25 engines Stretched LOX/LH_2 tanks (length)	RS-25D evolving to RS-25E LOX/LH_2 propellant	Two LRB (LOX/RP) boosters with two RS-84 engines per booster	LOX/RP propellant 1.1 Mlbf class	N/A	N/A	65 - 80	N/A	HLLV Industry Analysis Data
LV-275-5.RS25-2.LRB-1.J2X	27.5-foot (8.4-m) dia core 5 RS-25 Stretched LOX/LH_2 tanks (length)	RS-25D evolving to RS-25E LOX/LH_2 propellant	Two LRB (LOX/RP) boosters	LOX/RP propellant	27.5-foot (8.4-m) dia stage 1 J-2X engine	J-2X LOX/LH_2 propellant	77 - 85	25 - 29	HLLV Industry Analysis Data
LV-275-3.RS68-2.4S	27.5-foot (8.4-m) dia core 3 RS-68 Standard LOX/LH_2 tanks (length)	RS-68 LOX/LH_2 propellant	Two four-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	N/A	N/A	61 - 69	N/A	Trial Case 3, Alternative 2, Phase 1
LV-275-3.RS68-2.5S	27.5-foot (8.4-m) dia core 3 RS-68 Stretched LOX/LH_2 tanks (length)	RS-68 LOX/LH_2 propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	N/A	N/A	68 - 75	N/A	
LV-275-4.RS68-2.5S	27.5-foot (8.4-m) dia core 4 RS-68 engines Stretched LOX/LH_2 tanks (length)	RS-68A evolving to RS-68B LOX/LH_2 propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	N/A	N/A	81 - 85	N/A	HLLV Industry Analysis Data Trial Case 3, Alternative 2, Phase 2
LV-275-4.RS68-2.5S-3.MB100	27.5-foot (8.4-m) dia core 4 RS-68 engines Stretched LOX/LH_2 tanks (length)	RS-68 LOX/LH_2 propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	27.5-foot (8.4-m) dia stage 3 MB-100 engine	3 MB-100 LOX/LH_2 propellant	131 - 135	44 - 46	
LV-275-4.RS68B-2.5CP-3.MB100	27.5-foot (8.4-m) dia core 4 RS-68B Stretched LOX/LH_2 tanks (length)	RS-68B LOX/LH_2 propellant	Two five-segment RSRMs Composite cases	PBAN Existing grain/fin configuration	27.5-foot (8.4-m) dia stage 3 MB-100 engine	3 MB-100 LOX/LH_2 propellant	133 - 138	46 - 48	Trial Case 3, Alternative 2, Phase 3
LV-275-4.RS68B-2.5CH-3.MB100	27.5-foot (8.4-m) dia core 4 RS-68B Stretched LOX/LH_2 tanks (length)	RS-68B LOX/LH_2 propellant	Two five-segment RSRMs Composite cases	HTPB New grain/fin configuration	27.5-foot (8.4-m) dia stage 3 MB-100 engine	3 MB-100 LOX/LH_2 propellant	143 - 149	50 - 51	Trial Case 3, Alternative 2, Phase 4

Table 4-8. RVD alternative configurations (continued)

Vehicle designation	Core Stage definition	Core Stage propulsion system	Booster definition	Booster propulsion system	Second/Upper Stage (EDS) definition	Second/Upper Stage (EDS) propulsion system	Inserted (payload) mass capability to LEO (mT)	Inserted (payload) mass capability to TLI or trans-injection for NEO/BEO (if noted) (mT)	Source data
LV-275-4.RS68B-2.5S-1.J2X	27.5-foot (8.4-m) dia core 4 RS-68B Stretched LOX/LH$_2$ tanks (length)	RS-68B LOX/LH$_2$ propellant	Two five segment RSRMs Steel cases	PBAN Existing grain/fin configuration	27.5-foot (8.4-m) dia stage 1 J-2X engine	J-2X LOX/LH$_2$ propellant	117 - 125	42 - 45	Trial Case 4, Option 1 HLLV Industry Analysis Data
LV-275-6.RS68-2LRB	27.5-foot dia (8.4-m) core 6 RS-68 engines Stretched LOX/LH$_2$ tanks (length)	RS-68A evolving to RS-68B LOX/LH$_2$ propellant	Two LRB (LOX/RP) boosters with two RS-84 engines per booster	LOX/RP propellant 1.1 Mlbf class	N/A	N/A	82 - 88	N/A	HLLV Industry Analysis Data MSFC HLLV Data
LV-275-5.RS68-2.LRB-1J2X	27.5-foot dia (8.4-m) core 6 RS-68 engines Stretched LOX/LH$_2$ tanks (length)	RS-68A evolving to RS-68B LOX/LH$_2$ propellant	Two LRB (LOX/RP) boosters	LOX/RP propellant	27.5-foot (8.4-m) dia stage 1 J-2X engine	J-2X LOX/LH$_2$ propellant	120 - 127	39 - 44	HLLV Industry Analysis Data
LV-330-3.RS84	33-foot (10-m) dia core 3 RS-84 LOX/RP	RS-84 LOX/RP propellant	N/A	N/A	N/A	N/A	21 - 24	N/A	Trial Case 3 Alternative 4, Phase 1
LV-330-3.RS84-2.4S	33-foot (10-m) dia core 3 RS-84 LOX/RP	RS-84 LOX/RP propellant	Two four-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	N/A	N/A	63 - 71	N/A	Trial Case 3 Alternative 3, Phase 1
LV-330-3.RS84-2.5S	33-foot (10-m) dia core 3 RS-84 LOX/RP	RS-84 LOX/RP propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	N/A	N/A	84 - 88	N/A	Trial Case 3 Alternative 3, Phase 2
LV-330-3.RS84-2.5S-3.MB100	33-foot (10-m) dia core 3 RS-84 LOX/RP	RS-84 LOX/RP propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/fin configuration	33-foot (10-m) dia stage 3 MB-100 engines	3 MB-100 LOX/LH$_2$ propellant	135 - 141	42 - 45	
LV-330-3.RS84-2.5CP-3.MB100	33-foot (10-m) dia core 3 RS-84 LOX/RP	RS-84 LOX/RP propellant	Two five-segment RSRMs Composite cases	PBAN Existing grain/fin configuration	33-foot (10-m) dia stage 3 MB-100 engines	3 MB-100 LOX/LH$_2$ propellant	138 - 143	46 - 49	Trial Case 3 Alternative 3, Phase 3
LV-330-3.RS84-2.5CH-3.MB100	33-foot (10-m) dia core 3 RS-84 LOX/RP	RS-84 LOX/RP propellant	Two five-segment RSRMs Composite cases	HTPB New grain/fin configuration	33-foot (10-m) dia stage 3 MB-100 engines	3 MB-100 LOX/LH$_2$ propellant	147 - 154	48 - 51	Trial Case 3 Alternative 3, Phase 4
LV-330-3.RS84-2.RS84	33-foot (10-m) dia core 3 RS-84 LOX/RP	RS-84 LOX/RP propellant	2 RS-84 LOX/RP propellant	RS-84 LOX/RP propellant	N/A	N/A	53 - 56	18 - 19	Trial Case 3 Alternative 4, Phase 2
LV-330-2.RS84-2.RS84-3.AJ100	33-foot (10-m) dia core 3 RS-84 LOX/RP	RS-84 LOX/RP propellant	2 RS-84 LOX/RP propellant	RS-84 LOX/RP propellant	33-foot (10-m) dia stage 3 AJ100K ORSC engines	3 AJ100K LOX/RP1 propellant	72 - 75	25 - 26	Trial Case 3 Alternative 4, Phase 3 Trial Case 5, Baseline
LV-330-3,AJ2M-2.AJ2M-3.AJ100	33-foot (10-m) dia core 3 AJ2Mlbf LOX/RP	AJ 2Mlbf LOX/RP propellant	2 AJ2M ORSC LOX/RP propellant	2 AJ2M ORSC LOX/RP propellant	33-foot (10-m) dia stage 3 AJ100K ORSC engines	3 AJ100K LOX/RP1 propellant	97 - 101	34 - 36	Trial Case 3 Alternative 4, Phase 4

Table 4-8. RVD alternative configurations (continued)

Vehicle designation	Core Stage definition	Core Stage propulsion system	Booster definition	Booster propulsion system	Second/Upper Stage (EDS) definition	Second/Upper Stage (EDS) propulsion system	Inserted (payload) mass capability to LEO (mT)	Inserted (payload) mass capability to TLI or trans-injection for NEO/BEO (if noted) (mT)	Source data
LV-330-4.AJ2M	33-foot (10-m) dia core 4 AJ2M LOX/RP	AJ 2Mlbf LOX/RP propellant	N/A	N/A	33-foot (10-m) dia stage 3 AJ100K ORSC engines	3 AJ100K LOX/RP propellant	79 - 87	28 - 31	
LV-330-4.AJ2M -2.J2X	33-foot (10-m) dia core 4 AJ2M LOX/RP	AJ 2Mlbf LOX/RP propellant	N/A	N/A	33-foot (10-m) dia stage 2 J-2X engines	J-2X LOX/LH_2 propellant	84 - 92	30 - 32	Trial Case 5, Option 1
LV-330-5.RS68-4S-2.J2X	33-foot (10-m) dia core 5 RS-68 engines Stretched LOX/LH_2 tanks (length)	RS-68A evolving to RS-68B LOX/LH_2 propellant	Two four-segment RSRMs Steel cases	PBAN Existing grain/ fin configuration	33-foot (10-m) dia stage 2 J-2X -288 engines	J-2X LOX/LH_2 propellant	125 - 134	43 - 47	Trial Case 4, Option 2
LV-330-6.RS68-5S	33-foot (10-m) dia core 6 RS-68 engines Stretched LOX/LH_2 tanks (length)	RS-68A evolving to RS-68B LOX/LH_2 propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/ fin configuration	N/A	N/A	103 - 109	N/A	HLLV Industry Analysis Data MSFC HLLV Data
LV-330-6.RS68-2.5S-2.J2X	33-foot (10-m) dia core 6 RS-68 Stretched LOX/LH_2 tanks (length)	RS-68A evolving to RS-68B LOX/LH_2 propellant	Two five-segment RSRMs Steel cases	PBAN Existing grain/ fin configuration	33-foot (10-m) dia stage 2 J-2X engines	J-2X LOX/LH_2 propellant	145 - 150	49 - 53	HLLV Industry Analysis Data
LV-330-6.RS68-2.5S-2.J2X	33-foot (10-m) dia core 6 RS-68 Stretched LOX/LH_2 tanks (length)	RS-68A evolving to RS-68B LOX/LH_2 propellant	Two LRB (LOX/RP) boosters	RS-XX w/ ORSC LOX/RP propellant 2 1.25 Mlbf class ea	33-foot (10-m) dia stage 2 J-2X -288 engines	J-2X LOX/LH_2 propellant	139 - 142	48 - 50	03_SATs_KO_Gov_Arch
LV-330-5.XX(1.25)-0	33-foot (10-m) dia core 5 XX RP engine Stretched LOX/RP tanks (length)	RS-XX w/ORSC LOX/RP propellant 1.25 Mlbf class	N/A	N/A	N/A	N/A	49 - 54	N/A	HLLV Industry Analysis Data MSFC HLLV Data
LV-330-5.1.25-2.1.25-2AJ100	33-foot (10-m) dia core 5 1.25 Mlbf Stretched LOX/LH_2 tanks (length)	1.25 Mlbf LOX/RP1 propellant	Two LRB (LOX/RP) boosters	RS-XX w/ORSC LOX/RP propellant 2 1.25 Mlbf class	33-foot (10-m) dia stage 2 AJ 100K LOX/RP1	2 LOX/RP1 AJ 100K	72 - 75	25 - 27	Trial Case 2, Option 3
LV-330-5.XX(1.25)-0.00-1.J2X	33-foot (10-m) dia core 5 RS-84 Stretched LOX/RP tanks (length)	RS-84 (1.25 Mlbf) w/ORSC LOX/RP propellant	N/A	N/A	33-foot (10-m) dia stage 1 J-2X engine	J-2X LOX/LH_2 propellant	110 - 116	39 - 42	
LV-330-5.XX(2.0)-0	33-foot (10-m) dia core 5 XX RP engine Stretched LOX/RP tanks (length)	RS-XX w/GG LOX/RP propellant 2.0 Mlbf class	N/A	N/A	N/A	N/A	61 - 65	N/A	HLLV Industry Analysis Data MSFC HLLV Data
LV-330-5.XX(2.0)-0.00-1.J2X	33-foot (10-m) dia core 5 RS-XX Stretched LOX/RP tanks (length)	RS-XX (2.0 Mlbf) w/GG LOX/RP propellant	N/A	N/A	33-foot (10-m) dia stage 1 J-2X engine	J-2X LOX/LH_2 propellant	101 - 109	37 - 39	Trial Case 2, Option 2

Table 4-8. RVD alternative configurations (concluded)

Vehicle designation	Core Stage definition	Core Stage propulsion system	Booster definition	Booster propulsion system	Second/Upper Stage (EDS) definition	Second/Upper Stage (EDS) propulsion system	Inserted (payload) mass capability to LEO (mT)	Inserted (payload) mass capability to TLI or trans-injection for NEO/BEO (if noted) (mT)	Source data
LV-330-5.XX(2.0)-0.00-1.J2X	33-foot (10-m) dia core 5 RS-XX Stretched LOX/RP tanks (length)	RS-XX (2.0 Mlbf) with GG LOX/RP propellant	N/A	N/A	33-foot (10-m) dia stage 4J-2X-288 engine	J-2X LOX/LH_2 propellant	118 - 122	38 - 40	MSFC HLLV Data
LV-330-6.XX(2.0)-0.00-2.J2X	33-foot (10-m) dia core 6 2Mlbf GG Stretched LOX/RP tanks (length)	RS-XX (2.0 Mlbf) with GG LOX/RP propellant	N/A	N/A	33-foot (10-m) dia stage 2 J-2X engine	J-2X LOX/LH_2 propellant	130 - 137	44 - 47	Trial Case 5, Option 2
LV-330-6.XX(2.0)-4.AJ110-2.J2X	33-foot (10-m) dia core 6 2Mlbf GG Stretched LOX/RP tanks (length)	RS-XX (2.0 Mlbf) with GG LOX/RP propellant	Six AJ110 monolithic SRBs	AJ110 monolithic	33-foot (10-m) dia stage 2 J-2X engine	J-2X LOX/LH_2 propellant	149 - 157	52 - 57	

5.0 FIGURES OF MERIT

To determine the quantitative benefits of each trade characteristic and attribute, USA developed and documented the trade study FOMs at each step in the assessment. USA identified the FOMs, prioritized the FOMs using AHP and QFD processes, developed FOM weighting and prioritized the weights using AHP and QFD, and conducted FOM sensitivity assessments by adjusting the relative importance of the metrics within their groupings, depending on the Pugh Matrix results. The FOMs are categorized by trade level as either VOC, CCRs, or characteristics and attributes.

5.1 Voice of the Customer

The development and analysis support to the MSFC SLS Team began with defining the VOC. USA's initial approach on VOC was based on the BAA NNM10ZDA001K statement: "NASA is examining the trade space of potential HLL and space transfer vehicle concepts. The focus is on affordability, operability, reliability, and commonality with multiple end users (NASA, DoD, Commercial, IPs, etc.) at the system and subsystem levels."

USA reviewed and assessed the focus areas as described in the BAA and concluded that adjustments to the Customer's needs were appropriate based on numerous statements made by NASA Headquarters (HQ) and MSFC from the Exploration Enterprise Workshop to the approval of the 2010 NASA Authorization Act timeline. Based on that review, USA developed a set of VOCs that characterized the SLS goals. USA's identified VOCs were baselined as Affordability, Schedule, Operability, Performance, and Reliability. Each VOC was defined in detail to ensure that a mutual understanding was established so that consistent scoring would result.

a. Affordability - Achieving the VOC within the Office of Management Budget (OMB) FY budget constraints. Includes commonality enabler

b. Performance - Heavy lift capability to satisfy the maximum number of reference missions. Includes extensibility enabler

c. Reliability - Measures required to achieve NASA reliability and safety standards. Includes mission assurance, quality, and safety

d. Schedule - Quickest (operational) flight launch milestones

e. Operability - Ability (availability) and ease to process and operate the architecture (vehicle and ground/mission systems) with minimum resources and reduced schedule for multiple Customers, missions, and payloads. Includes flexibility enabler

USA conducted the AHP assessment for VOC with the mindset of NASA, the flight system design center, not as a Contractor operator, to help highlight the implications and impacts of initial weighting early in the design definition phase. The results, as shown in Table 5-1, of relative importance were generated:

Affordability: 47 percent

Performance: 25 percent

Reliability: 14 percent

Schedule: 10 percent

Operability: 4 percent

With the Customer focus on cost, the Affordability FOM is of significantly greater importance than Schedule, Operability, or Reliability FOMs.

Table 5-1. VOC prioritization

	Affordability	Schedule	Operability	Performance	Reliability	Sum of normalized ratings	Weighting
Affordability	1.000	5.000	7.000	3.000	5.000	2.371	0.474
Schedule	0.200	1.000	5.000	0.333	0.333	0.493	0.099
Operability	0.143	0.200	1.000	0.143	0.200	0.183	0.037
Performance	0.333	3.000	7.000	1.000	3.000	1.226	0.245
Reliability	0.200	3.000	5.000	0.333	1.000	0.727	0.145
	1.876	12.200	25.000	4.810	9.533	5.000	1.000

General observations on USA's VOC priorities are as follows:

a. Affordability - Highest Most Important Requirement (MIR). Twice as important as the second-highest-ranked VOC FOM

b. Performance - Second-highest ranked behind MIR. Customer wants to hold on to lift capability to capture the most missions, limiting reliability impact. Justifies the evolution of capability to every increasing performance. Tradable with schedule

c. Reliability - Indicates importance, setting a minimum threshold that is nonnegotiable. Safety threshold was established at Shuttle ascent probability level. Buying more reliability/safety is negotiable

d. Schedule - Low MIR but not as tradable as other FOMs. Schedule breakpoint to have an operational capability by 2017. NASA FOM was more likely driven by political rationale (influenced by Congressional VOC). If the political constraint changes, FOM would be tradable

e. Operability - Reflects perception of Customer operations' influences on life-cycle affordability and sustainability

To validate USA's VOC results, USA extracted the NASA FOMs from NASA's "Preliminary Report Regarding NASA's SLS and Multi-Purpose Crew Vehicle (MPCV), Pursuant to Section 309 of the NASA Authorization Act of 2010 (P.L. 111-267)," dated January 2011. Within the report to Congress, NASA's stated FOMs were identified as

Affordability: 55 percent

Schedule: 25 percent

Performance: 10 percent

Programmatic: 10 percent

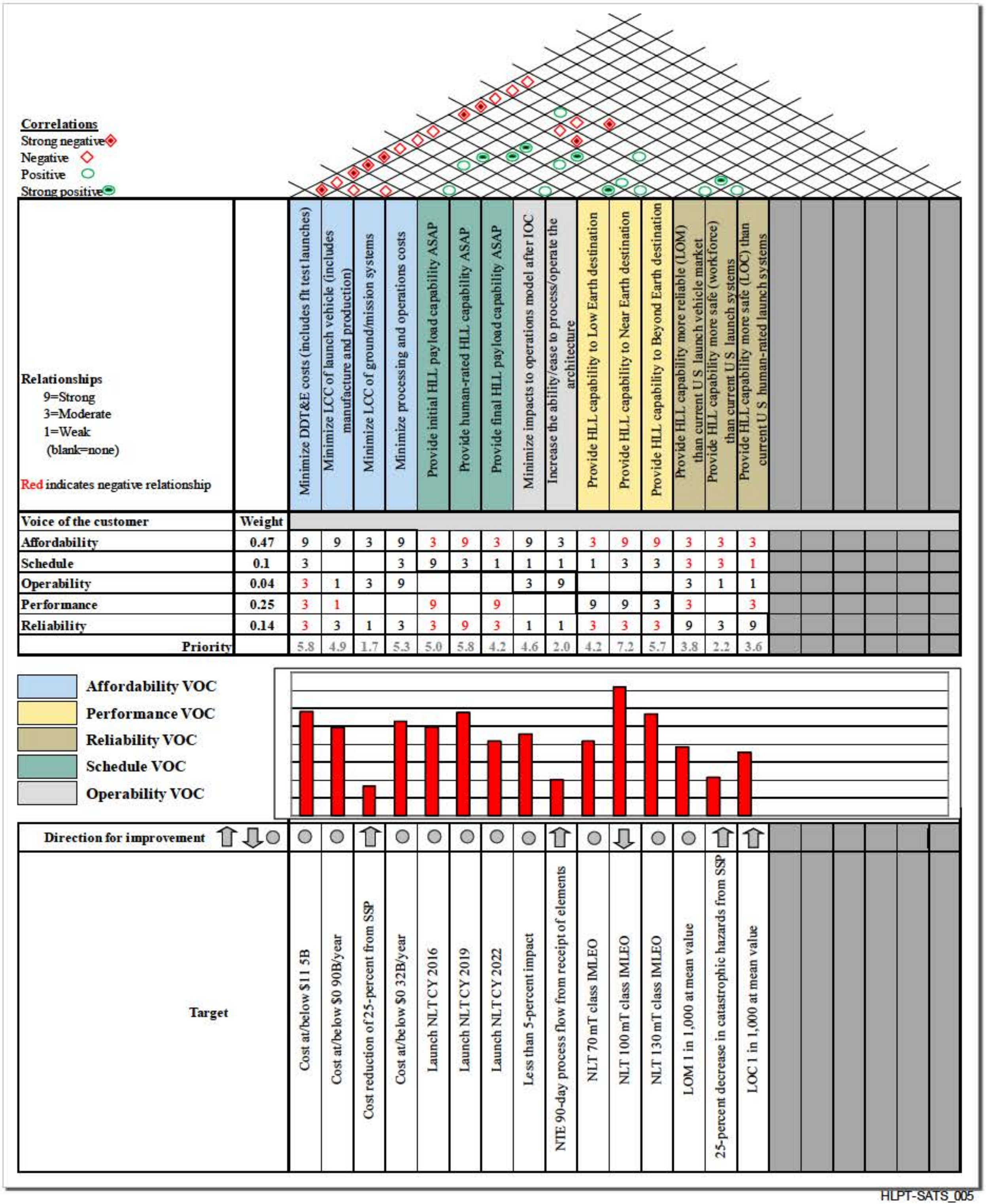
In performing a gap analysis and comparing the USA and NASA VOC FOMs, it was felt that the USA VOC priority was reasonable and no adjustments would be necessary based on the observations and USA's understanding of NASA's goals.

USA also recognizes that the VOC results may change during later program phases, such as reliability may be more important as one approaches the operational phase. However, these results are appropriate at this concept definition phase.

5.2 Customer Critical Requirements

In the development of the CCRs within the QFD HOQ #1, USA started with the AHP results for the VOC to capture the Customer's needs and the priority weighting of each need. For each VOC FOM, CCRs were defined to determine how success in meeting the requirements will be measured, translating the Customer's terms into process or product terms, and then identifying the delivery of those Customer needs. Once the CCRs were defined, the necessary performance targets for each requirement were established. These targets were determined by reviewing White House, Congressional, or NASA documentation (e.g., the Presidential budget, authorization bills). The interrelationship matrix was developed by evaluating the relationships between the VOCs (i.e., Customer needs) and CCRs to determine the relative importance of each CCR. To complete the evaluation of the CCRs, USA developed the correlation matrix to compare the CCRs to determine if they are in conflict with each other, leveraging each other, or have no effect on each other. Figure 5-1 depicts the matrix and CCRs. Development of the CCRs was based on USA's application of the Customer's needs.

An immediate observation with respect to the VOC-to-CCR relationship is the Operability FOM against the "Provide a HLL capability more reliable Loss of Mission (LOM) than current U.S. launch vehicle market" interdependency, where the relationship is positive but depends on how tightly coupled the "ease to operate" and how inherently reliable the system is.



Voice of the customer	Weight	Minimize DDT&E costs (includes flt test launches)	Minimize LCC of launch vehicle (includes manufacture and production)	Minimize LCC of ground/mission systems	Minimize processing and operations costs	Provide initial HLL payload capability ASAP	Provide human-rated HLL capability ASAP	Provide final HLL payload capability ASAP	Minimize impacts to operations model after IOC	Increase the ability/ease to process/operate the architecture	Provide HLL capability to Low Earth destination	Provide HLL capability to Near Earth destination	Provide HLL capability to Beyond Earth destination	Provide HLL capability more reliable (LOM) than current U S launch vehicle market	Provide HLL capability more safe (workforce) than current U S launch systems	Provide HLL capability more safe (LOC) than current U S human-rated launch systems
Affordability	0.47	9	9	3	9	3	9	3	9	3	3	9	9	3	3	3
Schedule	0.1	3			3	9	3	1	1	1	1	3	3	3	3	1
Operability	0.04	3	1	3	9				3	9				3	1	1
Performance	0.25	3	1			9		9			9	9	3	3		3
Reliability	0.14	3	3	1	3	3	9	3	1	1	3	3	3	9	3	9
Priority		5.8	4.9	1.7	5.3	5.0	5.8	4.2	4.6	2.0	4.2	7.2	5.7	3.8	2.2	3.6
Direction for improvement ⇧ ⇩ ○		○	○	⇧	○	○	○	○	○	⇧	○	⇩	○	○	⇧	⇧
Target		Cost at/below $11 5B	Cost at/below $0 90B/year	Cost reduction of 25-percent from SSP	Cost at/below $0 32B/year	Launch NLT CY 2016	Launch NLT CY 2019	Launch NLT CY 2022	Less than 5-percent impact	NTE 90-day process flow from receipt of elements	NLT 70 mT class IMLEO	NLT 100 mT class IMLEO	NLT 130 mT class IMLEO	LOM 1 in 1,000 at mean value	25-percent decrease in catastrophic hazards from SSP	LOC 1 in 1,000 at mean value

HLPT-SATS_005

Figure 5-1. QFD HOQ #1

Observations on the correlation matrix were in the areas of minimizing DDT&E costs, providing final HLL capability as soon as possible, and providing human-rated HLL capability as soon as possible.

5.2.1 Minimize DDT&E Costs

There were no strong positive correlations noted. There were strong negative correlations associated with the following:

a. *"Minimize LCC of launch vehicle"* assuming lack of prioritization of LCC driving operational considerations in a time- and money-constrained development environment in favor of achieving target performance, minimum reliability, and certification

b. *"Minimize processing and operations costs"* assuming the same as (a) above

c. *"Provide initial HLL payload capability (performance) As Soon As Possible (ASAP)"* assuming that schedule and budget constraints will drive compromise in the achievement of actual target performance for intermediate value that fits within top-level constraints

d. *"Provide human-rated HLL capability (performance) ASAP"* assuming that schedule and budget constraints will drive compromise in the achievement of full and sufficient human rating for intermediate configuration and performance that fits within top-level constraints

e. *"Provide HLL capability to Near Earth destination"* assuming the same as (c) above

f. *"Provide HLL capability to Beyond Earth destination"* assuming the same as (c) above, resulting, ultimately, in failure to ever achieve final mission performance configuration prior to termination of funding

5.2.2 Provide Final HLL Payload Capability ASAP

There were no strong negative correlations noted. There were strong positive correlations associated with the following:

a. *"Provide HLL capability to Low Earth destination"* assuming that technology and capability readiness is within 6 and 8 years, respectively

b. *"Provide HLL capability to Near Earth destination"* assuming that technology and capability readiness is within 6 and 8 years, respectively

5.2.3 Provide Human-Rated HLL Capability ASAP

There were no strong positive correlations noted. There were strong negative correlations associated with the following:

a. *"Provide HLL capability more reliable (LOM) than current U.S. launch vehicle market"*

b. *"Provide HLL capability more safe Loss of Crew (LOC) than current U.S. human-rated launch system"*

Notable takeaways and conclusions include the following:

a. Spending the effort upfront during "building block" DDT&E to reduce overall LCC and avoid future risks for both the vehicle and processing and mission support systems will provide the best affordability over the life cycle but has to be balanced because the more cost and schedule put into DDT&E, the later the systems will be able to deliver payload

b. Selecting hardware with proven reliability provides the best opportunity for reducing DDT&E costs and meeting the highest priority of HLL capability soonest

An observation, as an experienced operator of complex human-rated launch systems, is that USA's perception of NASA's importance on the Operability FOM is very low if the affordability focus is across the entire life cycle and not only near-term DDT&E costs, where operability considerations are typically removed to reduce costs.

Interestingly, the lowest-priority scores, from the assumed perspective, all concern ground systems: reducing ground systems costs, ease/ability to process the vehicle, and a vehicle which results in a safer environment for the workforce, thus indicating that reducing the amount of resources required during processing drives the solution more than the ground systems. As a validation, USA's top-10-ranked CCRs and associated targets are aligned with affordability, schedule and performance of NASA's top three VOC FOMs.

Based on the priority-weighted scores of the CCRs, the CCRs were listed in order of importance. Table 5-2 and an evaluation of "what requirements" were passed to the next evaluation cycle to determine the architecture-level FOMs. Development of the CCRs was based on USA's application of the Customer's needs.

Table 5-2. Prioritized CCR list

Critical Customer requirements	Target	Priority
Provide HLL capability to Near Earth destination	NLT 100 mT class IMLEO	7.2
Minimize DDT&E costs (includes flt test launches)	Cost at/below $11.5B	5.8
Provide human-rated HLL capability ASAP	Launch NLT CY 2019 (operational)	5.8
Provide HLL capability to Beyond Earth destination	NLT 130 mT class IMLEO	5.7
Minimize processing and operations costs	Cost at/below $0.32B/year	5.3
Provide initial HLL payload capability ASAP	Launch NLT CY 2016 (operational)	5
Minimize LCC of launch vehicle (includes manufacture and production)	Cost at/below $0.90B/year	4.9
Minimize impacts to operations model after IOC	Less than 5-percent impact	4.6
Provide final HLL payload capability ASAP	Launch NLT CY 2022 (operational)	4.2
Provide HLL capability to Low Earth destination	NLT 70 mT class IMLEO	4.2
Provide HLL capability more reliable (LOM) than current U.S. launch vehicle market	LOM 1 in 1,000 at mean value	3.8
Provide HLL capability more safe (LOC) than current U.S. human-rated launch systems	LOC 1 in 1,000 at mean value	3.6
Provide HLL capability more safe (workforce) than current U.S. launch systems	25-percent decrease in catastrophic hazards from SSP	2.2
Increase the ability/ease to process/operate the architecture	NTE 90-day process flow from receipt of elements	2
Minimize LCC of ground/mission systems	Cost reduction of 25 percent from SSP	1.7

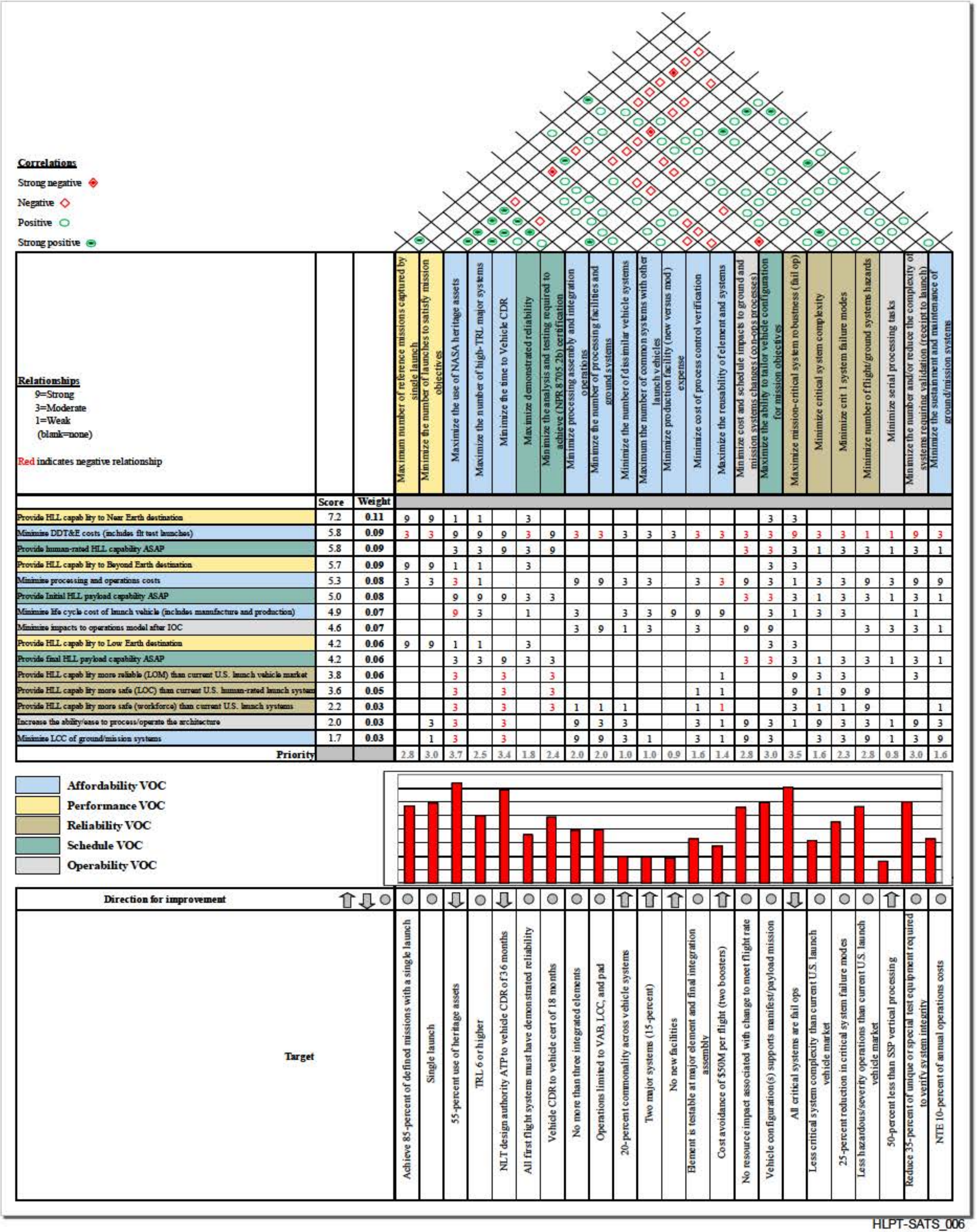
🟦 **Affordability VOC**
🟨 **Performance VOC**
🟫 **Reliability VOC**
🟩 **Schedule VOC**
⬜ **Operability VOC**

5.3 Characteristics and Attributes

In the development of the architecture characteristics and attributes requirements within the QFD HOQ #2, USA started with the QFD HOQ #1 results for the CCRs to capture the Customer requirements and the priority weighting of each requirement. For each CCR FOM, an architecture characteristic or attribute requirement was defined to further determine how success in meeting the Customer requirements will be measured, translating the Customer's terms into process or product terms, and then identifying the method of delivery of those Customer requirements. Once the architecture characteristic or attribute requirements were defined, the necessary performance targets for each requirement were established. These targets were determined by reviewing USA's developed integration and operational experiences and lessons learned from SSP and CxP. The interrelationship matrix was developed by evaluating the relationships between the CCR and architecture characteristic and attribute requirements to determine the relative importance of each. To complete the evaluation of the architecture characteristic or attribute requirements, USA developed the correlation matrix to compare the architecture characteristic or attribute requirements to determine if they are in conflict with each other, leveraging each other, or have no effect on each other. Figure 5-2 depicts the matrix and defined architecture characteristic or attribute requirements. Development of the architecture characteristics and attributes requirements was based on USA's application of the CCRs.

An immediate observation with respect to the CCR to architecture characteristics and attributes requirement relationship is *"minimize LCC of launch vehicle (includes manufacture and production)"* FOM against the *"maximize the reusability of element and systems"* FOM interdependency, which assumes that reusability is a cost savings/avoidance to manufacturing new elements. A second observation is *"provide HLL capability more safe (workforce) than current U.S. launch systems"* FOM against the *"maximize the reusability of element and systems"* FOM interdependency, which is a negative relationship due to hazards of open ocean recovery.

Observations on the correlation matrix were in the areas of maximizing the use of NASA heritage assets, maximizing the number of high-TRL major systems, and minimizing the time to vehicle Critical Design Review (CDR).



Correlations
Strong negative ◆
Negative ◇
Positive ○
Strong positive ●

Relationships
9=Strong
3=Moderate
1=Weak
(blank=none)

Red indicates negative relationship

	Score	Weight	Maximum number of reference missions captured by single launch	Minimize the number of launches to satisfy mission objectives	Maximize the use of NASA heritage assets	Maximize the number of high-TRL major systems	Minimize the time to Vehicle CDR	Maximize demonstrated reliability	Minimize the analysis and testing required to achieve (NPR 8705.2b) certification	Minimize processing assembly and integration operations	Minimize the number of processing facilities and ground systems	Minimize the number of dissimilar vehicle systems	Maximum the number of common systems with other launch vehicles	Minimize production facility (new versus mod) expense	Minimize cost of process control verification	Maximize the reusability of element and systems	Minimize cost and schedule impacts to ground and mission systems changes (con-ops processes)	Maximize the ability to tailor vehicle configuration for mission objectives	Maximize mission-critical system robustness (fail op)	Minimize critical system complexity	Minimize crit 1 system failure modes	Minimize number of flight/ground systems hazards	Minimize serial processing tasks	Minimize the number and/or reduce the complexity of systems requiring validation (receipt to launch)	Minimize the sustainment and maintenance of ground/mission systems
Provide HLL capability to Near Earth destination	7.2	0.11	9	9	1	1		3										3	3						
Minimize DDT&E costs (includes flt test launches)	5.8	0.09	3	3	9	9	9	3	9	3	3	3	3	3	3	3	3	3	9	3	3	1	1	9	3
Provide human-rated HLL capability ASAP	5.8	0.09			3	3	9	3	9								3	3	3	1	3	3	1	3	1
Provide HLL capability to Beyond Earth destination	5.7	0.09	9	9	1	1		3										3	3						
Minimize processing and operations costs	5.3	0.08	3	3	3	1				9	9	3	3		3	3	9	3	1	3	3	9	3	9	9
Provide Initial HLL payload capability ASAP	5.0	0.08			9	9	9	3	3								3	3	3	1	3	3	1	3	1
Minimize life cycle cost of launch vehicle (includes manufacture and production)	4.9	0.07			9	3		1		3		3	3	9	9	9		3	1	3	3			1	
Minimize impacts to operations model after IOC	4.6	0.07								3	9	1	3		3		9	9				3	3	3	1
Provide HLL capability to Low Earth destination	4.2	0.06	9	9	1	1		3										3	3						
Provide final HLL payload capability ASAP	4.2	0.06			3	3	9	3	3								3	3	3	1	3	3	1	3	1
Provide HLL capability more reliable (LOM) than current U.S. launch vehicle market	3.8	0.06			3		3		3							1			9	3	3			3	
Provide HLL capability more safe (LOC) than current U.S. human-rated launch system	3.6	0.05			3		3		3						1	1			9	1	9	9			
Provide HLL capability more safe (workforce) than current U.S. launch systems	2.2	0.03			3		3		3	1	1	1			1	1			3	1	1	9			1
Increase the ability/ease to process/operate the architecture	2.0	0.03		3	3		3			9	3	3			3	1	9	3	1	9	3	3	1	9	3
Minimize LCC of ground/mission systems	1.7	0.03		1	3		3			9	9	3	1		3	1	9	3		3	3	9	1	3	9
Priority			2.8	3.0	3.7	2.5	3.4	1.8	2.4	2.0	2.0	1.0	1.0	0.9	1.6	1.4	2.8	3.0	3.5	1.6	2.3	2.8	0.8	3.0	1.6
Direction for improvement	⇧ ⇩ ○		○	○	⇩	○	⇩	○	○	○	○	⇧	⇧	⇧	○	⇧	○	○	⇩	○	○	○	⇧	○	○
Target			Achieve 85-percent of defined missions with a single launch	Single launch	55-percent use of heritage assets	TRL 6 or higher	NLT design authority ATP to vehicle CDR of 36 months	All first flight systems must have demonstrated reliability	Vehicle CDR to vehicle cert of 18 months	No more than three integrated elements	Operations limited to VAB, LCC, and pad	20-percent commonality across vehicle systems	Two major systems (15-percent)	No new facilities	Element is testable at major element and final integration assembly	Cost avoidance of $50M per flight (two boosters)	No resource impact associated with change to meet flight rate	Vehicle configuration(s) supports manifest/payload mission	All critical systems are fail ops	Less critical system complexity than current U.S. launch vehicle market	25-percent reduction in critical system failure modes	Less hazardous/severity operations than current U.S. launch vehicle market	50-percent less than SSP vertical processing	Reduce 35-percent of unique or special test equipment required to verify system integrity	NTE 10-percent of annual operations costs

🟦 Affordability VOC
🟨 Performance VOC
🟫 Reliability VOC
🟩 Schedule VOC
⬜ Operability VOC

HLPT-SATS_006

Figure 5-2. QFD HOQ #2

5.3.1 Maximizing the Use of NASA Heritage Assets

There was a strong positive correlation noted with the following:

a. *"Maximize high TRL"* assuming heritage is high TRL

b. *"Minimize time to vehicle CDR"* assuming systems are understood and acceptable for intended use, thereby reducing time to acceptance

c. *"Maximize demonstrated reliability"* assuming that they come with that positive history

d. *"Minimize analysis and testing required to achieve human rating"* assuming that they are human rated or provide NASA-accepted data for basis of rating

e. *"Minimize production facility (new or mod) expense"* assuming production capability exists to deliver

It should be noted that a stronger positive is given to assets used in the manner in which they were originally applied; i.e., SSME (RS-25D) run within SSP specifications and parameters. Still positive, but less so, is the application of heritage assets used outside of the demonstrated performance envelope or used as the basis for derivative systems; e.g., RS-25E operating at altered ratio or percentage limits.

There was a strong negative correlation noted with the following:

a. *"Maximize the number of common systems with other launch vehicles"* based on existing U.S. launch vehicle inventory

b. *"Minimize serial processing tasks"* with segmented solids serial processing as a primary example

5.3.2 Maximize the Number of High-TRL Major Systems

There were no strong negative correlations noted. There were strong positive correlations noted with the following:

a. *"Minimize time to vehicle CDR"* assuming high TRL provides improved basis of acceptance for application

b. *"Maximize demonstrated reliability"* assuming demonstrated reliability is how high TRL is achieved

c. *"Minimize the analysis and testing needed to achieve human rating"* assuming demonstrated reliability in intended application is a fundamental basis for human rating

5.3.3 Minimize the Time to Vehicle CDR

There was a strong positive correlation with "*minimize the analysis and testing required to achieve human rating*" assuming that the analysis and testing drives the time to CDR.

There was a strong negative correlation with "*maximize critical system robustness (fail op)*" assuming a historical pattern of trading "good enough" or even a accepting a waiver on schedule over extending a schedule to achieve the desired robustness.

Notable takeaways and conclusions are that "heritage assets" have a positive impact on near-term costs and the development timeline, although the operational requirements and processes may need to be adjusted to have an affordable and sustainable program. Based on the selection and actual application, heritage systems with heritage requirements and processes may create negative to strong negative impacts to operability and programmatic LCCs. High-TRL systems provide positive impacts on both cost (existing market segment and supply chain) and schedule (availability), but a system that has a high TRL gives no indication as to complexity, number of Crit 1 failure modes, producibility, operability, etc. Selecting an architecture and elements that require less site processing and integration, whether considered production or operations, will minimize overall program costs, improve timelines, and enable operational flexibility. Based on the priority-weighted scores of the architecture characteristics and attributes requirements, the architecture requirements were listed in order of importance in Table 5-3 along with an evaluation of what requirements were passed to the next evaluation cycle to determine the system-level FOMs. Development of the architecture requirements was based on USA's application of the Customer's needs.

Table 5-3. Architecture attributes and characteristics requirements

Architecture attributes and characteristics requirements	Target	Priority
Maximize the use of NASA heritage assets	Fifty-five-percent use of heritage assets	3.7
Maximize mission-critical system robustness (fail ops)	All critical systems are fail ops	3.5
Minimize the time to vehicle CDR	NLT design authority ATP to vehicle CDR of 36 months	3.4
Minimize the number of launches to satisfy mission objectives	Single launch	3
Maximize the ability to tailor vehicle configuration for mission objectives	Vehicle configuration(s) supports manifest/payload mission	3
Minimize the number and/or reduce the complexity of systems requiring validation (receipt to launch)	Reduce 35 percent of unique or special test equipment required to verify system integrity	3
Maximum number of reference mission captured by single launch	Achieve 85 percent of defined missions with a single launch	2.8
Minimize cost and schedule impacts to ground and mission systems changes (con-ops processes)	No resource impact associated with change to meet flight rate	2.8
Minimize the number of flight/ground systems hazards	Less hazardous/severity operations than current U.S. launch vehicle market	2.8
Maximize the number of high-TRL major systems	TRL 6 or higher	2.5
Minimize the analysis and testing required to achieve (NPR 8705.2b) certification	Vehicle CDR to vehicle cert of 18 months	2.4
Minimize crit 1 system failure modes	Twenty-five-percent reduction in critical system failure modes	2.3
Minimize processing assembly and integration operations	No more than three integrated elements	2
Minimize the number of processing facilities and ground systems	Operations limited to VAB, LCC, and pad	2
Maximize demonstrated reliability	All first flight systems must have demonstrated reliability	1.8
Minimize cost of process control verification	Element is testable at major element and final integration assembly	1.6

🟦 **Affordability VOC**
🟨 **Performance VOC**
🟫 **Reliability VOC**
🟩 **Schedule VOC**
⬜ **Operability VOC**

Table 5-3. Architecture attributes and characteristics requirements (concluded)

Architecture attributes and characteristics requirements	Target	Priority
Minimize critical system complexity	Less critical system complexity than current U.S. launch vehicle market	1.6
Minimize the sustainment and maintenance of ground/mission systems	NTE 10 percent of annual operations costs	1.6
Maximize the reusability of element and systems	Cost avoidance of $50M per flight (2 boosters)	1.4
Minimize the number of dissimilar vehicle systems	Twenty-percent commonality across vehicle systems	1
Maximum the number of common systems with other launch vehicles	Two major systems (15 percent)	1
Minimize production facility (new versus mod) expense	No new facilities	0.9
Minimize serial processing tasks	Fifty percent less than SSP vertical processing	0.8

🟦 **Affordability VOC**
🟨 **Performance VOC**
🟫 **Reliability VOC**
🟩 **Schedule VOC**
⬜ **Operability VOC**

6.0 TRADE RESULTS

With the identification of the QFD HOQ #2 FOMs, including the priority ranking of the architecture characteristic and attribute requirements, USA assembled a set of trades at the architecture, launch vehicle stages, In-Space Stage, systems, and implementation.

6.1 Architecture Trade

In reviewing the top six prioritized Architecture FOMs, two are derived from the Affordability VOC, one from Reliability, one from Performance, one from Schedule, and the last from Operability VOC. "*Maximize the use of NASA heritage assets*," with a goal of 55-percent use of heritage assets, ranked first as it positively impacts near-term affordability and near-term schedule coupled with a community that understands the operability and reliability behavior of the hardware and software. "*Maximize mission-critical system robustness*," with a goal of all critical systems are fail ops, ranked second as it positively impacts reliability along with reducing processing validation and reducing the operational workarounds required. "*Minimize the time to vehicle CDR*," with a goal of no later than design authority Authority to Proceed (ATP) to vehicle CDR of 36 months, ranked third as it positively impacts near-term affordability and near-term schedule to achieve soonest entrance into the qualification program. "*Minimize the number of launches to satisfy mission objectives*," with a goal of accomplishing as many missions as possible with a single launch, ranked fourth as it positively impacts performance along with mission success reliability. "*Maximize the ability to tailor vehicle configuration for mission objectives*," with a goal of having a vehicle configuration(s) that supports the manifest and payload mission, ranks fifth as it positively impacts the operational schedule along with reducing mission planning and processing cycle times. "*Minimize the number and/or reduce the complexity of systems requiring validation from receipt to launch*," with a goal of reducing the validation 35 percent of unique or special test equipment required to verify system integrity, ranked sixth as it positively impacts operability with increased utilization and validation flexibility along with reduction in test and checkout schedules and costs.

As USA reviewed and assessed the priorities, it is recognized that they are significantly influenced by the Congressional schedule and budget constraints; therefore, any deviations from those constraints will change the architecture relationships.

Across all architecture requirements, significant architectural considerations and impacts on the SLS Program are as follows:

a. Reliability

    1. "*Minimize ground and flight hazards*" - Reduce processing costs and increase safety

2. *"Maximize mission-critical system robustness"* - Minimize vehicle ground verification required prior to each flight and increase flight mission success

b. Affordability

1. *"Minimize time to vehicle CDR"* - Reduce DDT&E costs and achieve IOC quicker
2. *"Maximize the use of NASA heritage systems or similar high TRL systems"* - Significant understanding of system supply chain health and cost and reduce certification costs by qual testing at system levels

c. Schedule

1. *"Maximize tailorability of vehicle configuration to ensure flexibility and extensibility"* - Minimize reconfiguration of ground/mission systems
2. *"Minimize analysis and testing required to achieve certification"*

d. Operability

"Reduce number or complexity of systems requiring process validation"

e. Performance

"Minimize number of launches or maximize number of missions that can be accomplished with single launch"

At an architectural level, these five VOC groupings are all interrelated. They have "soft" dependencies on each other and can drive design solutions. Finding the balance within the priorities will provide an architecture that will be more stable to environment changes that are political, budgetary, or technical.

6.1.1 Architecture Considerations Impact on Mission Systems

Based on the assessment of the SLS impacts related to Shuttle Derived Heavy Lift Launch Vehicle (SDHLLV) concepts and their respective concept of operations, it was determined that minimal impact to mission systems specifically related to launch vehicle, regardless of the evolving or end-state configurations proposed, would be realized. Regardless of architecture, it is critical that the evolvability, flexibility, and extensibility of the mission systems must be designed into the architecture with the end-state configuration in mind. Affordability can be improved by cost avoidance through sharing overlapping skills and tasks with other NASA programs (specifically MPCV Orion). Examples of joint development of design methodologies and techniques with both the launch vehicle and spacecraft are (a) ascent/abort integration and technical support for flight performance system integration groups, (b) Range Safety Panel and trajectory working groups, (c) technical panels for Guidance, Navigation, and Control (GNC), loads, thermal, and performance, (d) Natural Environments Day of Launch (DOL) Working

Group, (e) First-/Second-Stage design and optimization techniques for Nominal and Abort trajectories, and (f) Mission planning, analysis, and software model verification.

Architectural consideration impacts on mission systems are minimal when considering SDHLLV concepts and collaboration with other programs is a must if affordability is a key decision point.

6.1.2 Architecture Considerations Impact on Ground Systems

Based on the assessment of the SLS impacts related to SDHLLV concepts and their respective concept of operations, it was determined that architecture selection has a large impact on ground systems, including ground elements of support and test equipment, facilities, and other SLS-specific infrastructure.

The SLS Program cannot afford nor sustain a unique ground system for ground processing and operations at KSC with a flight rate of two to four launches per FY. This operations model, similar to Shuttle, has proven to be costly and not life-cycle efficient. To achieve a reasonable cost target for operations and sustainment, the market distribution of infrastructure costs must be maximized to cost-share initial and sustainment LCC over multiple users. One approach to this operations model is the Universal Launch Complex (ULC) concept, Figure 6-1, for SLS versus a dedicated launch complex. The ULC contains common infrastructure and unique infrastructure to accommodate SLS launch vehicle. The ULC contains the Universal Mobile Launcher (UML); Universal Ground Control System (UGCS); Vehicle Assembly

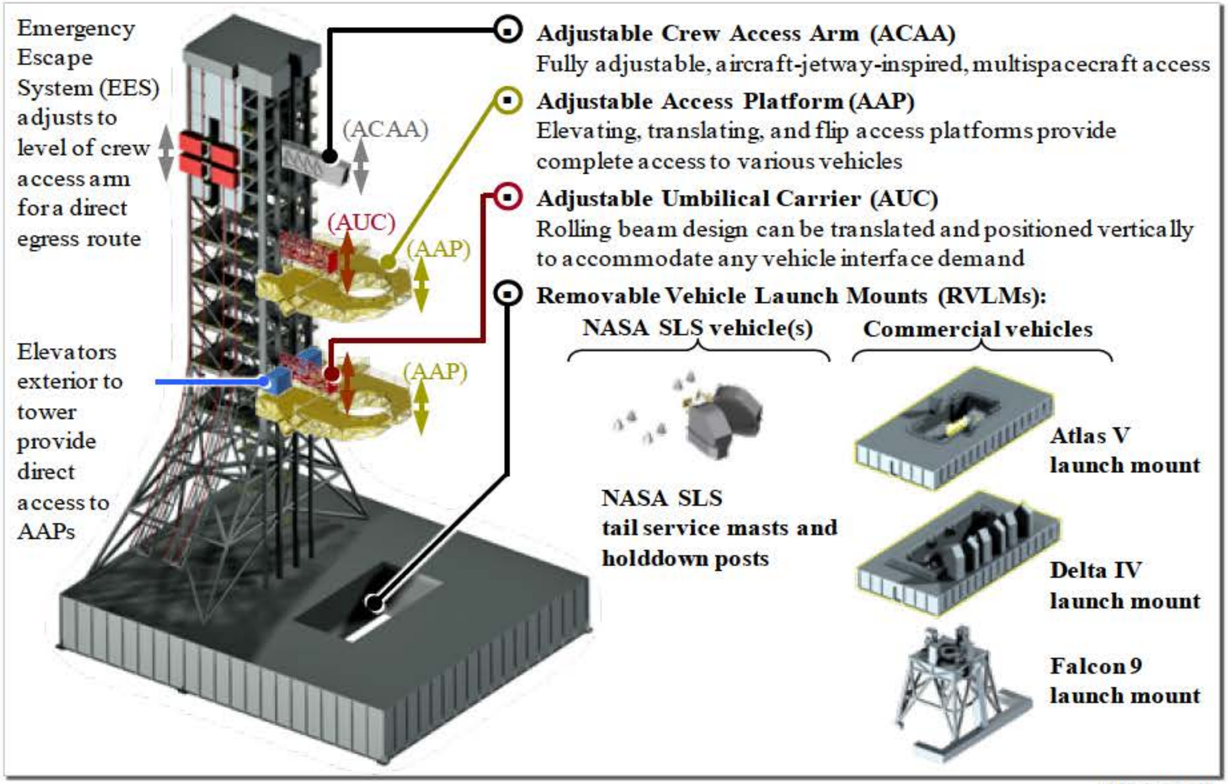


Figure 6-1. Universal Mobile Launcher (UML)

Building (VAB) with minimal modifications and access to SLS via UML so no dedicated VAB platforms are required; Pads A and B with multiple propellant servicing systems; and Crawler modified to support a min of 16.8 Mlb capacity. The UGCS uses a vehicle data server with interfaces to the vehicle console (located anywhere) across a vehicle data bus and ground control system vehicle interface controller to service SLS or other vehicles. The data server requires almost no reconfiguration, thus avoiding the costly expense of planning and preparation for each flight. In providing an upgraded and sustainable launch processing solution for SLS, NASA will then realize significant LLC avoidance with the commercial sector utilization of LC-39 while providing human launch services infrastructure.

Architectural consideration impacts on ground systems are significant when considering SDHLLV concepts, and, again, collaboration with other programs is a must if affordability is an important key decision point.

6.1.3 Architecture Trade Results

Any architecture selected has two fundamental elements, a system (vehicle, ground, and mission) and implementation (business model to operate the system). Based on current FY budget forecast constraints, development of an end-state launch vehicle with full BEO capability within the shortest duration as possible is not feasible. By using a capability-driven model within the implementation framework, a launch vehicle capability that is increasing in extent and the increasing availability of Exploration system payloads (i.e., deep space vehicles, surface landers) provides a balance of near-term capability. Although initially performance limiting, the core vehicle can grow into a long-term Exploration system enabler. Named "Progressive Architecture," the architecture employs a block-type vehicle system evolution with increasing performance capability to satisfy new mission objectives and provides a continuous development flow, resulting in an affordable and sustainable framework. In assessing the missions and comparing the vehicle capability required, a launch vehicle configuration that will deliver 100 mT Inserted Mass to Low Earth Orbit (IMLEO) will satisfy most mission objectives. Tailorability by adding an additional Core Stage engine to a core baseline vehicle, with associated increase in propellant consumables for NEO or BEO destinations, would reduce recurring costs for hardware costs. If your requirement is 130 mT, but 95 percent of your missions are below 100 mT, having flexibility to be extensible and tailorable on the architecture would enable NASA to optimize the configuration for that specific mission. This architecture and its implementation makes it possible to vary your budget profile, as you can increase or decrease the capability thresholds to achieve your FY cost targets. This is a unique implementation approach of the Design to Cost (DTC) methodology.

In developing the missions and objectives for SLS, there is much exploration and scientific interest in a HLL capability, both domestically and internationally. Based on the low flight rate required to support

NASA's Exploration missions in the future, there are not be enough flights per year to have a recognizable benefit to the projected LCC for the capacity that NASA has developed. Enabling mission payload partners and ground segment infrastructure cost-sharing will significantly reduce LCC. More missions, including international payloads, must be acquired to increase the flight rate to stabilize the workforce and retain proficiencies. Commonality in launcher ground system (ground segment) infrastructure between Commercial providers to LEO destinations and SLS to LEO/NEO/BEO destinations will allow cost-sharing distribution.

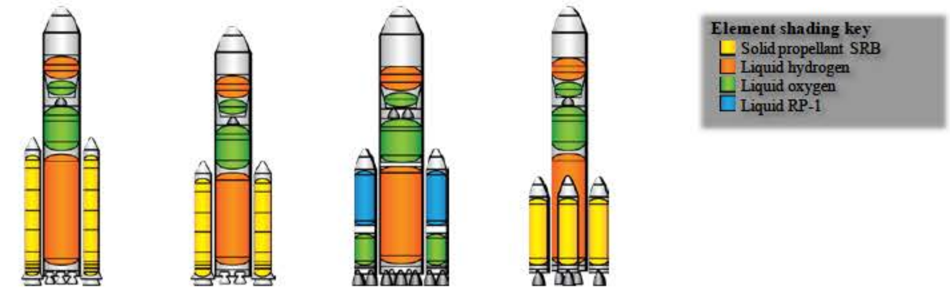
USA has conducted and participated in numerous studies over the life of the SSP SRB program that addressed the question of reusability and the technical benefit and costs associated with this approach. The most recent NASA study conducted, Ares I First Stage Expendability Trade Study, dated September 9, 2007, assessed both Ares I and Ares V expendable versus reuse options and concluded that, "It is not Life Cycle Cost effective to adopt expendable over reusable Ares I FS and Ares V Boosters." In each study, the technical benefits of retrieval and inspection of the hardware has been recognized. The ability to verify the flight performance by the design team versus relying solely on limited instrumentation and flight performance metrics has driven safety enhancements into the design and reduced program risk. However, there are costs associated with the retrieval and disassembly process and the flight hardware required for retrieval, such as parachutes, ordnance and logic circuits. The cost trade is driven significantly by the length of the program, the number of flights, and/or the value of the assets for follow-on programs. Recognizing the potential desire or need to use a nonreusable SLS booster based on budgetary constraints or a very limited flight manifest, USA has made an assessment of the cost deltas between the continued retrieval and reuse of the SLS booster versus a single-flight approach. Due to the limited number of flights and cost constraints associated with the startup of the SLS Program, with respect to near-term affordability, implementing a nonrecoverable booster concept for the SLS booster would be preferred. This allows for the elimination of the decelerator subsystem, retrieval and recovery, and refurbishment activities currently performed for the Space Shuttle SRBs. This includes maintenance and occupancy of the Parachute Refurbishment Facility (PRF), hangar AF, and the marine vessels. With respect to long-term affordability, if segmented solids remain in the evolving architecture beyond six flights, a recoverable booster concept would provide LCC benefits as the additional new manufacturing and production costs to replenish the expended assets would make reuse more affordable.

The technology insertion methodology that NASA uses will be pivotal to achieving the LCC desired. New technology and designs can reduce LCC, by improving operability as an enabler to cost reductions and cost avoidance, but only after the technology is matured. Immediately relying on new technology (low TRL) adds development cost, programmatic risk, and time to achieving initial launch capability. An architecture that can leverage and balance new technology and designs to allow tailorability, flexibility, and extensibility lowers operational delta costs. Be wary of "shiny new technology," as maturing it has a price and schedule. There are many lessons of failure when programs develop technology and design concurrently where the design requires the technology to be successful. Once the technology has been demonstrated and the risks understood by the receiving technical community, insert the technology at specific insertion points, either at a block upgrade milestone or in Pre-Planned Product Improvement (P^3I) points. Inserted technology should provide affordability benefits without degrading performance or capability or compromise safety and reliability.

6.2 Launch Vehicle Stages Trade

6.2.1 Trade 1 – Trial Case 1 Results

It was determined early that an end-state configuration trade should be conducted to assess different sensitivities in architecture that affected the launch vehicle, ground, and mission elements. In Trade 1 - Trial Case 1, the focus was on the sensitivity of the different SLS boosters that could be integrated. A Shuttle-derived five-segment solid booster, Shuttle-like four-segment solid booster, liquid oxygen and kerosene boosters, and solid monolithic boosters were extracted from the Reference Design Vehicle Matrix, as shown in Figure 6-2. Note that the performance to IMLEO is into a 30x130nm at 28.5° insertion orbit. Performance range is based on DOL conditions and variations in operational concept.



Vehicle designation	LV-275-5 RS25-5S-1 J2X	LV-275-4 RS25-4S-1 J2X	LV-330-6 RS68-2 2RP-2 J2X	LV-275-5 RS68-4MS-1 J2X	
Geometry	27.5-ft-dia. stretched height	27.5-ft-dia. standard	33-ft-dia. standard	27.5-ft-dia. stretched	
Booster	2x 5-segment PBAN	2x 4-segment PBAN	2x 2 x 1.25 Mlbf LOX/RP-1	4x 795 Klbf monolith SRB	RSRM metal case expended, migrate to expendable by flight 5
Core engine	5x RS-25D	4x RS-25D	6x RS-68A	5x RS-68A	RS-25D migrates to RS-25E by flight 4, RS-68A migrates to RS-68B for human rating
Upper Stage engine	1x J-2X	1x J-2X	2x J-2X	1x J-2X	
IMLEO	123 – 129 mT	110 – 118 mT	136 – 142 mT	118 – 124 mT	

HLPT-SATS_008

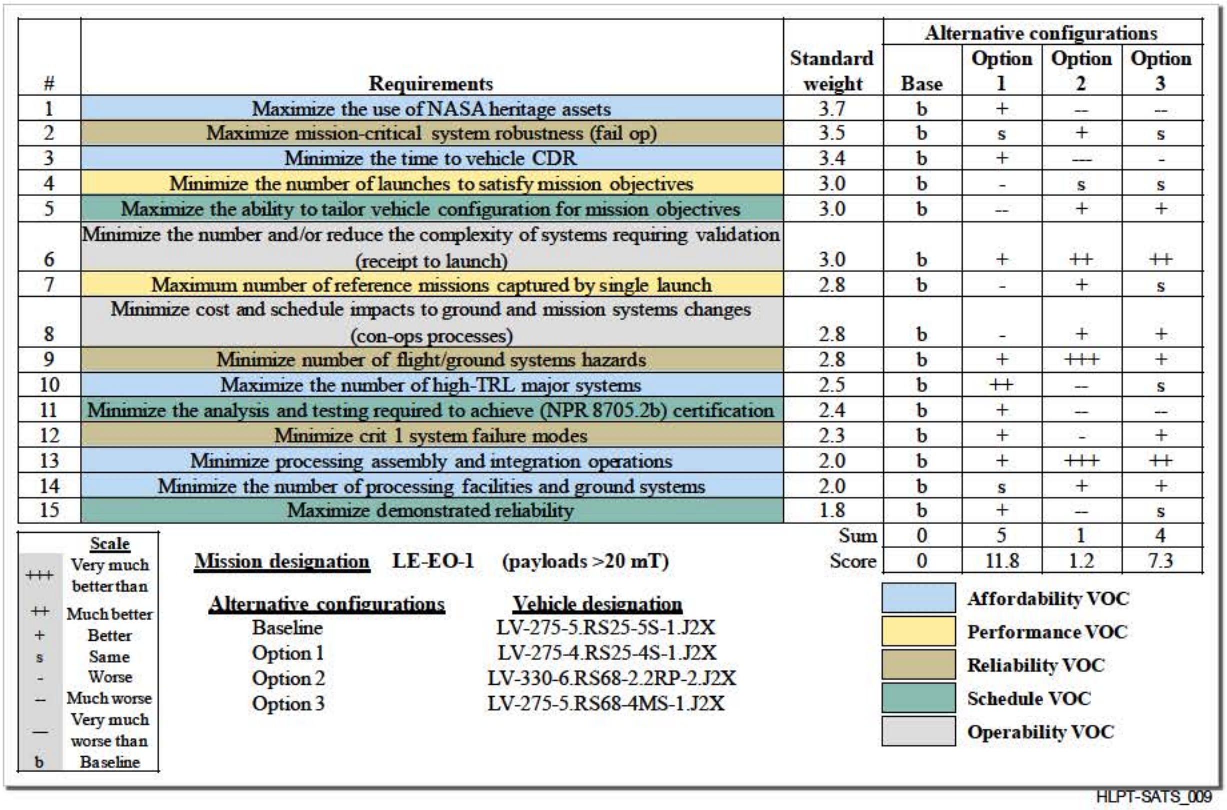
Figure 6-2. Trade 1 — Trial Case 1, sensitivities to booster configurations

To address the Reliability FOM for all trial cases, USA conducted the reliability analyses using the probabilistic risk analysis technique to determine nominal reliability with both engine out and no engine out conditions and an active Advanced Health Monitoring System (AHMS) to determine the loss of vehicle (LOV) estimates. As a benchmark, we used the most recent Space Shuttle LOC/V probability of 1 in 260 (ascent flight phase only based on SPRA Iteration 3.2). As applicable to the configuration assessed, the mean 1 per "n" at level 3 was calculated for SRB, RSRM, payload fairing, primary structure, avionics, software, propellant tanks, intertank, skirts, main propulsion system, hydraulic actuation system, and auxiliary propulsion system. Level 3 was rolled up to level 2. As applicable to the configuration assessed, the mean 1 in "n" at level 2 was calculated for Booster Stage, Core Stage engines, Core Stage structures and systems, in-line payload carrier, Upper Stage engines, and Upper Stage structures and systems. Total vehicle LOV was taken at level 2 rollup. Assumptions on system functionality were assumed, as necessary.

For Trade 1 – Trial Case 1, the LOV probability for the Baseline, Option 1, Option 2, and Option 3 were 1 in 224, 1 in 238, 1 in 132, and 1 in 263, respectively.

Once the architectures were defined and characterized, the different vehicles were traded in a decision matrix as shown in Table 6-1.

Table 6-1. Trade 1 – Trial Case 1 decision matrix



#	Requirements	Standard weight	Alternative configurations			
			Base	Option 1	Option 2	Option 3
1	Maximize the use of NASA heritage assets	3.7	b	+	–	–
2	Maximize mission-critical system robustness (fail op)	3.5	b	s	+	s
3	Minimize the time to vehicle CDR	3.4	b	+	—	-
4	Minimize the number of launches to satisfy mission objectives	3.0	b	-	s	s
5	Maximize the ability to tailor vehicle configuration for mission objectives	3.0	b	–	+	+
6	Minimize the number and/or reduce the complexity of systems requiring validation (receipt to launch)	3.0	b	+	++	++
7	Maximum number of reference missions captured by single launch	2.8	b	-	+	s
8	Minimize cost and schedule impacts to ground and mission systems changes (con-ops processes)	2.8	b	-	+	+
9	Minimize number of flight/ground systems hazards	2.8	b	+	+++	+
10	Maximize the number of high-TRL major systems	2.5	b	++	–	s
11	Minimize the analysis and testing required to achieve (NPR 8705.2b) certification	2.4	b	+	–	–
12	Minimize crit 1 system failure modes	2.3	b	+	-	+
13	Minimize processing assembly and integration operations	2.0	b	+	+++	++
14	Minimize the number of processing facilities and ground systems	2.0	b	s	+	+
15	Maximize demonstrated reliability	1.8	b	+	–	s
		Sum	0	5	1	4
		Score	0	11.8	1.2	7.3

	Scale
+++	Very much better than
++	Much better
+	Better
s	Same
-	Worse
–	Much worse
—	Very much worse than
b	Baseline

Mission designation **LE-EO-1** **(payloads >20 mT)**

Alternative configurations	**Vehicle designation**
Baseline	LV-275-5.RS25-5S-1.J2X
Option 1	LV-275-4.RS25-4S-1.J2X
Option 2	LV-330-6.RS68-2.2RP-2.J2X
Option 3	LV-275-5.RS68-4MS-1.J2X

🟦	**Affordability VOC**
🟨	**Performance VOC**
🟫	**Reliability VOC**
🟩	**Schedule VOC**
⬜	**Operability VOC**

HLPT-SATS_009

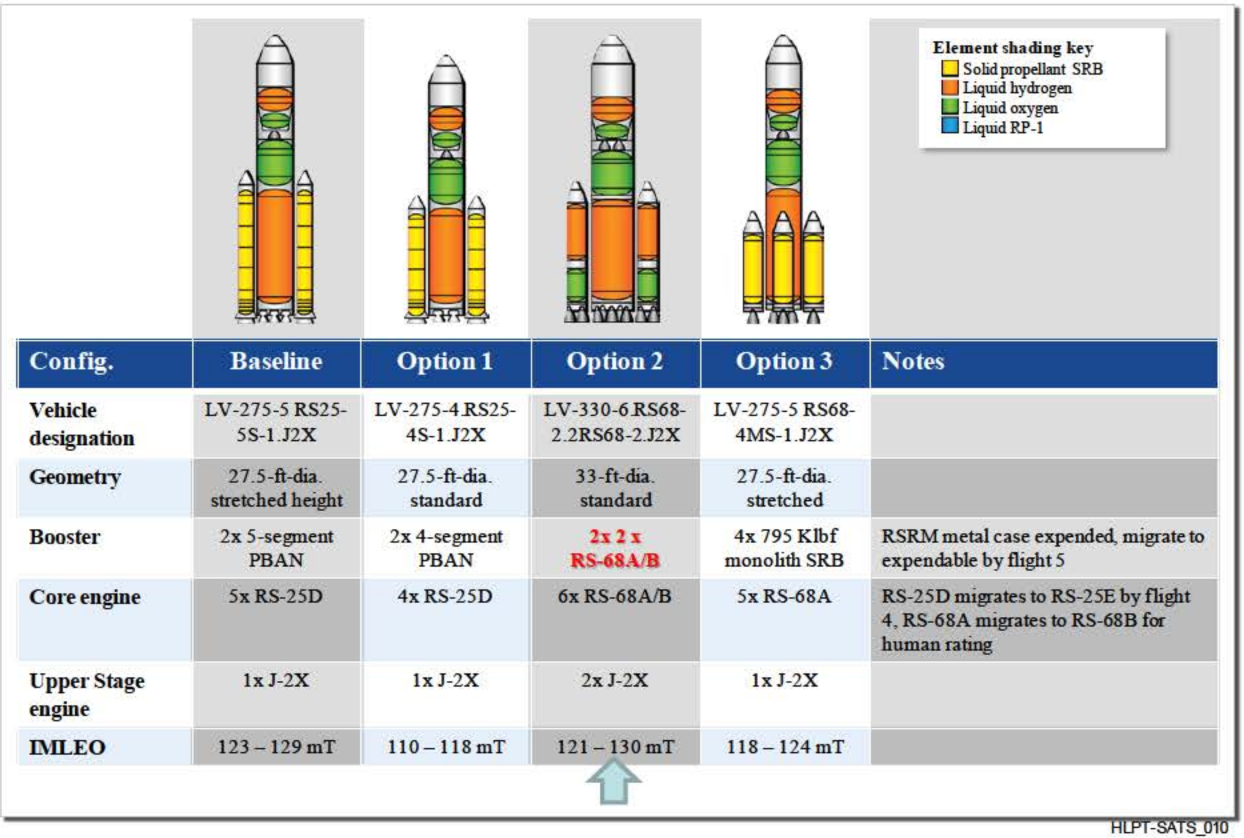
A key point when evaluating the sums and scores of the decision matrix, the results are against the Baseline identified in a pairs-wise comparison, so you should not carry sums or scores from one trade space to the next trade space.

In assessing the results, Option 1 against the Baseline, the Shuttle-like architecture traded very well because of the much lower DDT&E costs and the quickness to field the system within the first 5 years after ATP. Liquid booster configurations have great processing advantages over the life-cycle affordability, but DDT&E costs within the first 5 years are not feasible based on the current FY 2012 budget constraints.

The takeaways from Trade 1 - Trial Case 1, are that an initial HLL capability within 5 years after ATP to LEO is critical to SLS Program viability. Quickly fielding of a system, very SSP heritage with four-segment Polybutadiene Acrylonitrile (PBAN) solid boosters and standard height/diameter core tank than that of the SSP ET, provides near-term early launch capability to LEO as compared to the baseline configuration. Hazards, verifications and inspections, special access, and special tooling ALL significantly impact operational LLCs in labor, schedule time, sustaining, and maintenance. These affect the affordability of the vehicle, ground systems, missions systems, and operations. While heritage SRB motors, four-segment PBAN RSRM, family provides "untouched" propulsion system performance at 3.1 Mlbf each, LLC due to the limited market segment for that technology and the inherent hazards (i.e., open grain) along with additional integration operations (i.e., more end items to integrate together) is not optimizing the long-term costs.

6.2.1.1 Trade 1 – Trial Case 1A Results

When reviewing the results of Trade 1 – Trial Case 1 of the end-state configuration trade, a gap analysis was conducted to determine if "weaknesses" in the architecture could be identified. In Option 2, by removing the unproven and undemonstrated 1.25 Mlbf LOX/RP1 engines on the Booster Stage and replacing them with the same engine on the Core Stage, an engine that has a proven history, existing supply chain and market segment, and a demonstrated reliability would increase the engine commonality within the integrated vehicle and thus increase the supply chain production rate per flight, lowering the recurring costs. Figure 6-3 depicts the change, in red shadow font, to Option 2.



Config.	Baseline	Option 1	Option 2	Option 3	Notes
Vehicle designation	LV-275-5 RS25-5S-1.J2X	LV-275-4.RS25-4S-1.J2X	LV-330-6.RS68-2.2RS68-2.J2X	LV-275-5 RS68-4MS-1.J2X	
Geometry	27.5-ft-dia. stretched height	27.5-ft-dia. standard	33-ft-dia. standard	27.5-ft-dia. stretched	
Booster	2x 5-segment PBAN	2x 4-segment PBAN	**2x 2 x RS-68A/B**	4x 795 Klbf monolith SRB	RSRM metal case expended, migrate to expendable by flight 5
Core engine	5x RS-25D	4x RS-25D	6x RS-68A/B	5x RS-68A	RS-25D migrates to RS-25E by flight 4, RS-68A migrates to RS-68B for human rating
Upper Stage engine	1x J-2X	1x J-2X	2x J-2X	1x J-2X	
IMLEO	123 – 129 mT	110 – 118 mT	121 – 130 mT	118 – 124 mT	

HLPT-SATS_010

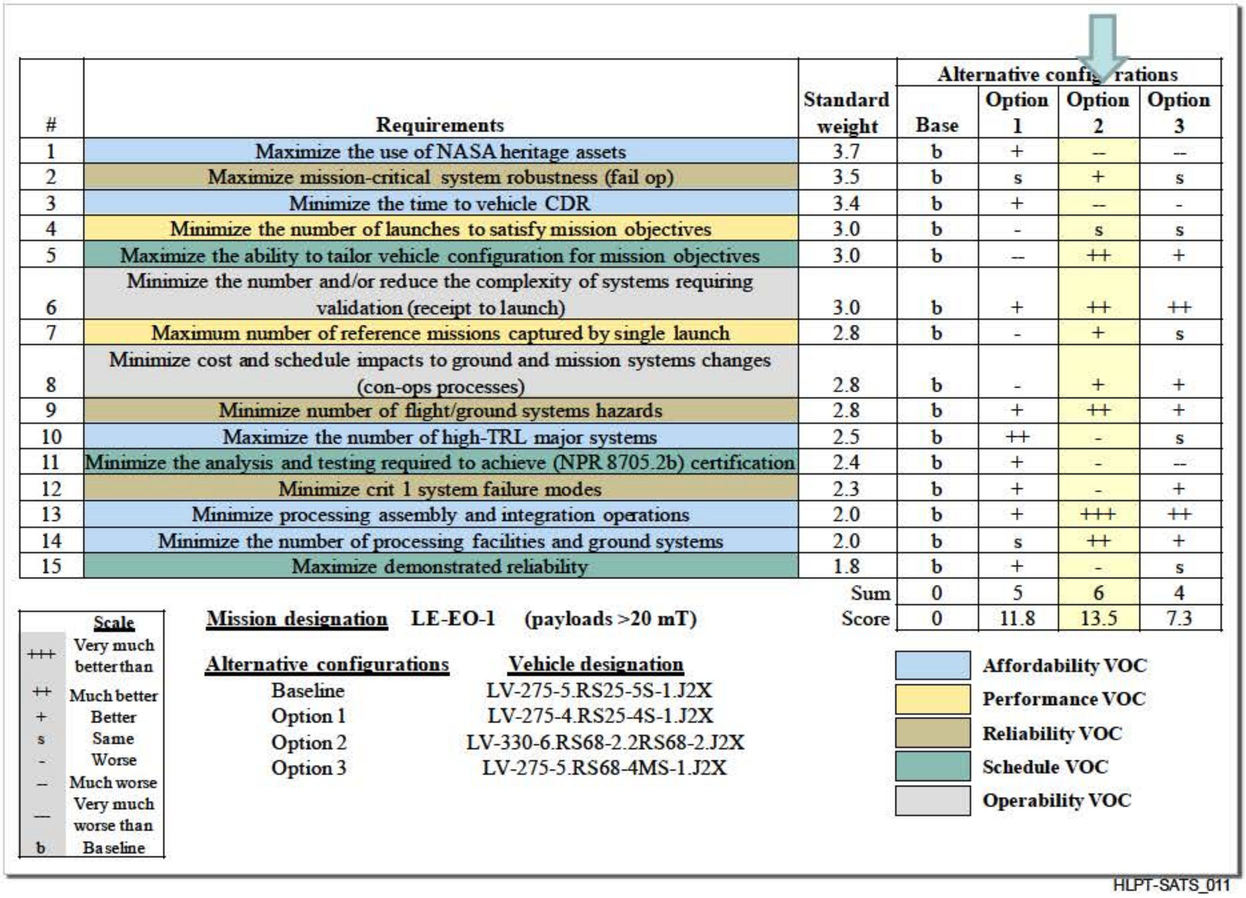
Figure 6-3. Trade 1 – Trial Case 1A, sensitivities to liquid booster configuration with known Core Stage engine

For Trade 1 – Trial Case 1A, the LOV probability for the modified configuration Option 2 was 1 in 168.

Once the changed architecture was defined and characterized, the Option 2 configuration was traded in the decision matrix as shown in Table 6-2.

In assessing the results, inserting a higher TRL engine and not having two different fuel propellants helps the liquid booster concept score better, but it still requires a unique-stage DDT&E activity. Changes in Option 2 configuration, booster engine, to a common, well understood, and currently available production-capacity propulsion system significantly improves the overall benefit as compared against the Baseline and Option 1 configurations.

Table 6-2. Trade 1 – Trial Case 1A decision matrix



#	Requirements	Standard weight	Alternative configurations			
			Base	Option 1	Option 2	Option 3
1	Maximize the use of NASA heritage assets	3.7	b	+	--	--
2	Maximize mission-critical system robustness (fail op)	3.5	b	s	+	s
3	Minimize the time to vehicle CDR	3.4	b	+	--	-
4	Minimize the number of launches to satisfy mission objectives	3.0	b	-	s	s
5	Maximize the ability to tailor vehicle configuration for mission objectives	3.0	b	--	++	+
6	Minimize the number and/or reduce the complexity of systems requiring validation (receipt to launch)	3.0	b	+	++	++
7	Maximum number of reference missions captured by single launch	2.8	b	-	+	s
8	Minimize cost and schedule impacts to ground and mission systems changes (con-ops processes)	2.8	b	-	+	+
9	Minimize number of flight/ground systems hazards	2.8	b	+	++	+
10	Maximize the number of high-TRL major systems	2.5	b	++	-	s
11	Minimize the analysis and testing required to achieve (NPR 8705.2b) certification	2.4	b	+	-	--
12	Minimize crit 1 system failure modes	2.3	b	+	-	+
13	Minimize processing assembly and integration operations	2.0	b	+	+++	++
14	Minimize the number of processing facilities and ground systems	2.0	b	s	++	+
15	Maximize demonstrated reliability	1.8	b	+	-	s
		Sum	0	5	6	4
		Score	0	11.8	13.5	7.3

	Scale
+++	Very much better than
++	Much better
+	Better
s	Same
-	Worse
--	Much worse
---	Very much worse than
b	Baseline

Mission designation **LE-EO-1** **(payloads >20 mT)**

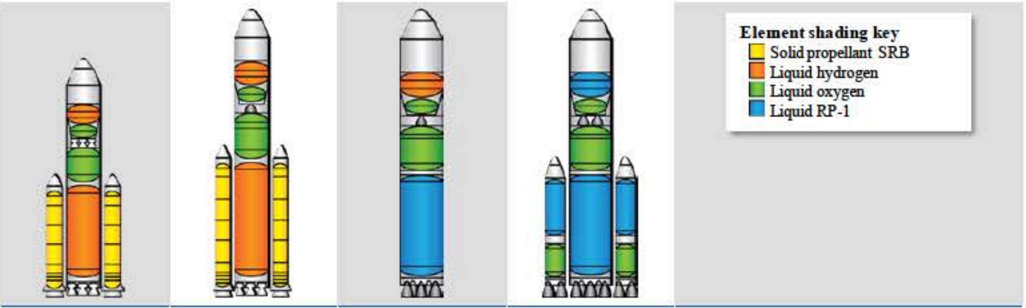
Alternative configurations	Vehicle designation
Baseline	LV-275-5.RS25-5S-1.J2X
Option 1	LV-275-4.RS25-4S-1.J2X
Option 2	LV-330-6.RS68-2.2RS68-2.J2X
Option 3	LV-275-5.RS68-4MS-1.J2X

Affordability VOC
Performance VOC
Reliability VOC
Schedule VOC
Operability VOC

HLPT-SATS_011

6.2.2 *Trade 2 – Trial Case 2 Results*

Trade 2 – Trial Case 2 was developed as an end-state configuration trade. The focus of Trade 2 was on the sensitivity of a LOX/RP1 Saturn-like vehicle configuration as compared to the Shuttle-derived configurations that scored well in Trade 1 and an all LOX/RP1 hybrid configuration. The Shuttle-based configurations and Saturn-based configurations are shown in Figure 6-4.



Config.	Baseline	Option 1	Option 2	Option 3	Notes
Vehicle designation	LV-275-3 RS25-4S-4.RL10	LV-275-5.RS25-5S-1.J2X	LV-330-5.RP2.0-0-1.J2X	LV-330-5.RP1.25M-2.2RP1.25M-2.RP100K	
Geometry	27.5-ft-dia. standard height	27.5-ft-dia. stretched	33-ft-dia. standard	33-ft-dia. standard	
Booster	2x 4-Seg Steel Case PBAN	2x 5-Seg Steel Case PBAN	None	2x 2 x 1.25 Mlbf LOX/RP-1	RSRM metal case expended, migrate to expendable composite RSRM by flight 5
Core engine	3x RS-25D	5x RS-25D	5, RP 2.0 Mlbf	5x RP 1.25 Mlbf	RS-25D migrates to RS-25E by flight 4
Upper Stage engine	4x RL10	1x J-2X	1x J-2X	2x 100 Klbf LOX/RP-1	
IMLEO	40 – 50 mT	123 – 126 mT	118 – 122 mT	72 – 75 mT	

HLPT-SATS_012

Figure 6-4. Trade 2 – Trial Case 2, sensitivities to an RP Saturn-like configuration as compared to a Shuttle-based architecture and hybrid

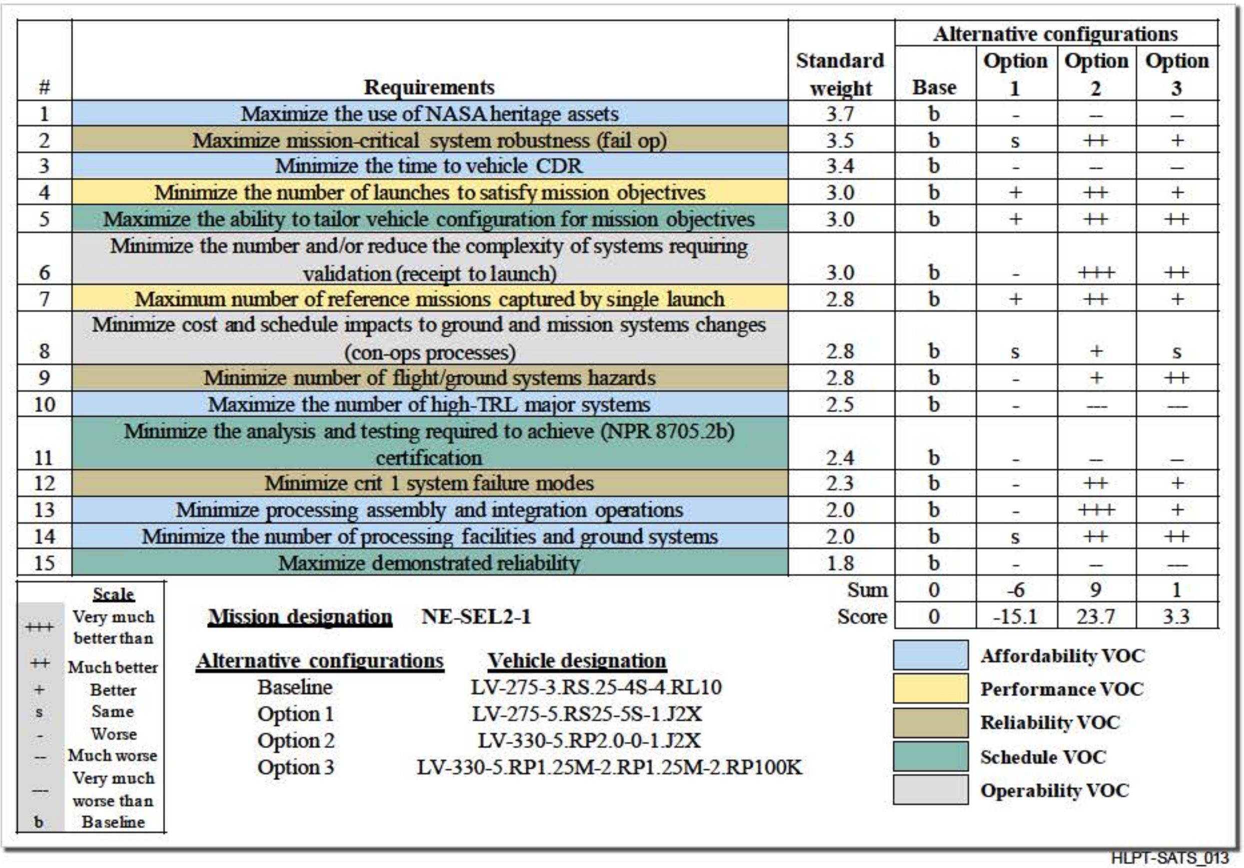
For Trade 2 – Trial Case 2, the LOV probability for the Baseline, Option 1, Option 2, and Option 3 were 1 in 182, 1 in 224, 1 in 170, and 1 in 112, respectively.

Once the architectures were defined and characterized, the different vehicles were traded in a decision matrix, as shown in Table 6-3.

In assessing the Option 2 results, the ability to tailor the vehicle, including the 33-foot-diameter shroud interface to accommodate multiple payload requirements, and one less element to integrate and operate, no boosters, depicts an operations-friendly configuration. This configuration is as close to "ship and shoot" as possible and can be operated with a reduced processing footprint.

Reducing the number of integrated elements (i.e., no boosters), with an all liquid system, which is end-item delivered with fully integrated propulsion systems, an engine cluster with propulsion system, MPS, and trust vector control, improves affordability due to processing and operations simplicity (less process validation and less processing restrictions) as compared to the Baseline configuration. The lack of initial performance of the Baseline to achieve the 70-mT performance target allows the alternative configuration options to score better on the performance FOMs.

Table 6-3. ***Trade 2 – Trial Case 2 decision matrix***



#	Requirements	Standard weight	Alternative configurations			
			Base	Option 1	Option 2	Option 3
1	Maximize the use of NASA heritage assets	3.7	b	-	--	--
2	Maximize mission-critical system robustness (fail op)	3.5	b	s	++	+
3	Minimize the time to vehicle CDR	3.4	b	-	--	--
4	Minimize the number of launches to satisfy mission objectives	3.0	b	+	++	+
5	Maximize the ability to tailor vehicle configuration for mission objectives	3.0	b	+	++	++
6	Minimize the number and/or reduce the complexity of systems requiring validation (receipt to launch)	3.0	b	-	+++	++
7	Maximum number of reference missions captured by single launch	2.8	b	+	++	+
8	Minimize cost and schedule impacts to ground and mission systems changes (con-ops processes)	2.8	b	s	+	s
9	Minimize number of flight/ground systems hazards	2.8	b	-	+	++
10	Maximize the number of high-TRL major systems	2.5	b	-	---	---
11	Minimize the analysis and testing required to achieve (NPR 8705.2b) certification	2.4	b	-	--	--
12	Minimize crit 1 system failure modes	2.3	b	-	++	+
13	Minimize processing assembly and integration operations	2.0	b	-	+++	+
14	Minimize the number of processing facilities and ground systems	2.0	b	s	++	++
15	Maximize demonstrated reliability	1.8	b	-	--	---
		Sum	0	-6	9	1
		Score	0	-15.1	23.7	3.3

Scale	
+++	Very much better than
++	Much better
+	Better
s	Same
-	Worse
--	Much worse
---	Very much worse than
b	Baseline

Mission designation **NE-SEL2-1**

Alternative configurations	**Vehicle designation**
Baseline	LV-275-3.RS.25-4S-4.RL10
Option 1	LV-275-5.RS25-5S-1.J2X
Option 2	LV-330-5.RP2.0-0-1.J2X
Option 3	LV-330-5.RP1.25M-2.RP1.25M-2.RP100K

🟦	Affordability VOC
🟨	Performance VOC
🟫	Reliability VOC
🟩	Schedule VOC
⬜	Operability VOC

HLPT-SATS_013

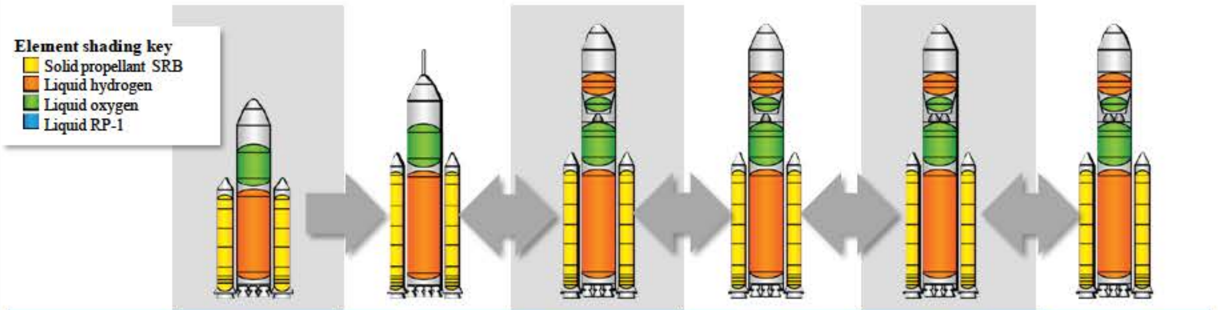
Lower TRL systems, regardless of NASA heritage status, brings the lack of domain knowledge in product manufacturing, qualification testing, production, processing, and operations, that results in extended schedule and additional cost.

In addition to propellant load increases, the 33-foot-diameter core tank provides added mission operations tailorability for oversized volume payloads for some mission objectives captured. This benefit does require additional DDT&E cost for infrastructure changes in manufacturing, transportation, and ground systems, mainly GSE.

6.2.3 Trade 3 – Trial Case 3 Results

Based on the results from Trial Case 1 and the gap analysis producing Trial Case 1A, along with the results from Trial Case 2, Trade 3 – Trial Case 3 was developed as an evolvable family configuration trade. The general concept of operations is for the Block 0 vehicle to be nonhuman rated, whereas the Block 1 vehicle and subsequent configurations are human rated. The phase arrows depict the migration from one capability to the next. A double arrow indicates that the configuration can be tailored for flight in either increasing or decreasing capability. For this trade, USA exaggerated the evolution phases to determine what architecture or vehicle changes via technology steps impact the FOMs the most. In a practical evolution within family, it would not be recommended to plan for this many flight-designated block vehicles. Technology changes from one phase to the next are identified in blue font.

The first set of in-family configurations were based on heritage SRB and SSME configurations, as depicted in Figure 6-5.



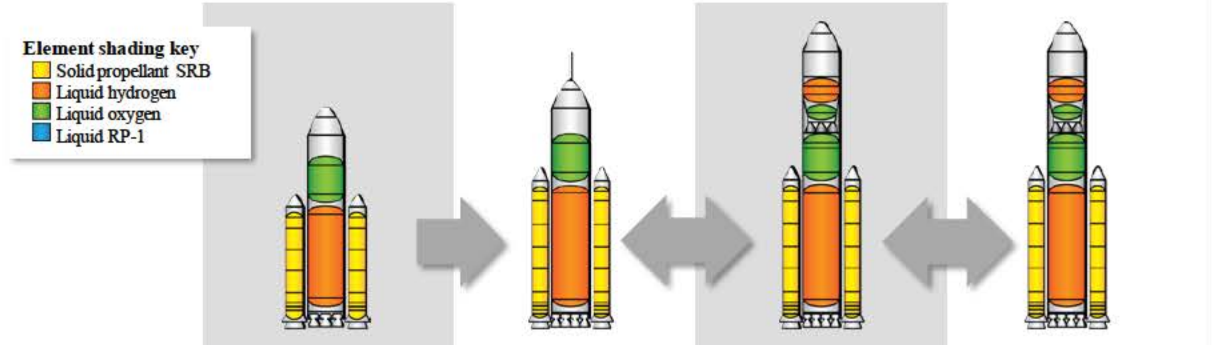
Config.	Phase 1 Block 0	Phase 2 Block 1	Phase 3 Block 2	Phase 4 Block 3	Phase 5 Block 4	Phase 6 Block 5
Booster	4-segment steel case PBAN	5-segment steel case PBAN	5-segment steel case PBAN	5-segment comp case PBAN	5-segment comp case PBAN	5-segment comp case HTPB
Core tank	27.5-ft-dia. standard height	27.5-ft-dia. stretched	27.5-ft-dia. stretched	27.5-ft-dia. stretched	27.5-ft-dia. stretched	27.5-ft-dia. stretched
Core engine	3x SSME (RS-25D)	4x SSME	5x RS-25E	5x RS-25E	5x RS-25E	5x RS-25E
Second Stage	N/A	N/A	1x J-2X	1x J-2X	2x J-2X	2x J-2X
IMLEO	55 – 62 mT	73 – 77 mT	123 – 129 mT	127 – 132 mT	137 – 143 mT	146 – 151 mT
Mission	LEO	LEO	LEO/NEO	NEO/BEO	NEO/BEO	NEO/BEO
	≤2 flights	≤2 flights	≤2 flight	>2 flights	>2 flights	>2 flights

HLPT-SATS_014

Figure 6-5. Trade 3 – Trial Case 3, Alternative Configuration 1 – Baseline, sensitivities to technology steps required in a heritage SRB and SSME configuration-evolvable launch vehicle family

For Trade 3 – Trial Case 3, Alternative Configuration 1 – Baseline, the LOV probability for the configurations range from 1 in 272 down to 1 in 205. In the early phases while you are attempting to demonstrate the reliability of the system, a design solution with no engine out capability drops the probability of 1 in 272 to 1 in 190. Subsequent block vehicle's LOV probability drops accordingly.

The second set of in-family configurations were based on heritage SRB and high TRL LOX/LH_2 Core Stage engine configurations, as depicted in Figure 6-6.



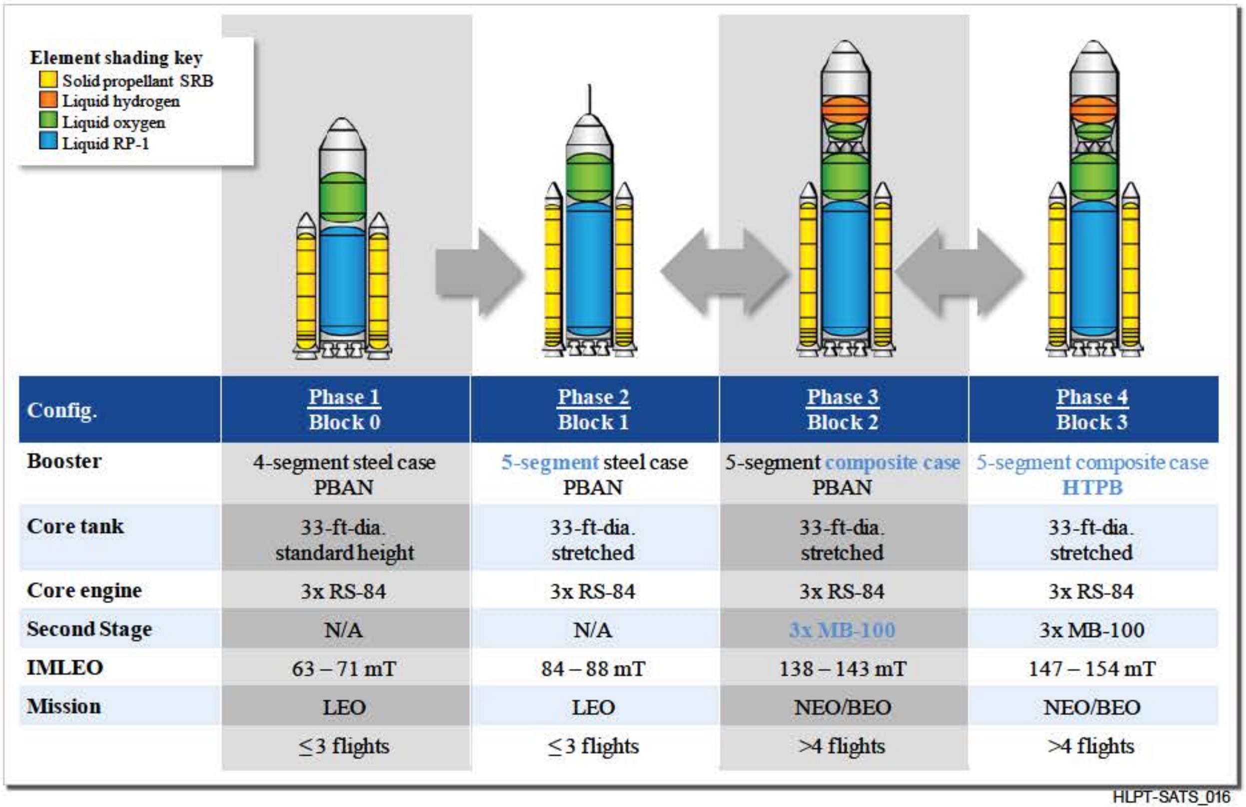
Config.	Phase 1 Block 0	Phase 2 Block 1	Phase 3 Block 2	Phase 4 Block 3
Booster	4-segment steel case PBAN	5-segment steel case PBAN	5-segment composite case PBAN	5-segment composite case HTPB
Core tank	27.5-ft-dia. standard height	27.5-ft-dia. stretched	27.5-ft-dia. stretched	27.5-ft-dia. stretched
Core engine	3x RS-68A	4x RS-68A	4x RS-68B	4x RS-68B
Second Stage	N/A	N/A	3x MB-100	3x MB-100
IMLEO	61 – 69 mT	81 – 85 mT	133 – 138 mT	143 – 149 mT
Mission	LEO	LEO	LEO/NEO/BEO	LEO/NEO/BEO
	≤3 flights	≤3 flights	>4 flights	>4 flights

HLPT-SATS_015

Figure 6-6. Trade 3 – Trial Case 3, Alternative Configuration 2–- Option 1, sensitivities to technology steps required in a heritage SRB and LOX/LH$_2$ core engine configuration-evolvable launch vehicle family

For Trade 3 – Trial Case 3, Alternative Configuration 2 – Option 1, the LOV probability for the configurations range from 1 in 290 down to 1 in 126.

The third set of in-family configurations were based on heritage SRB and low TRL LOX/RP1 Core Stage engine configurations, as depicted in Figure 6-7.



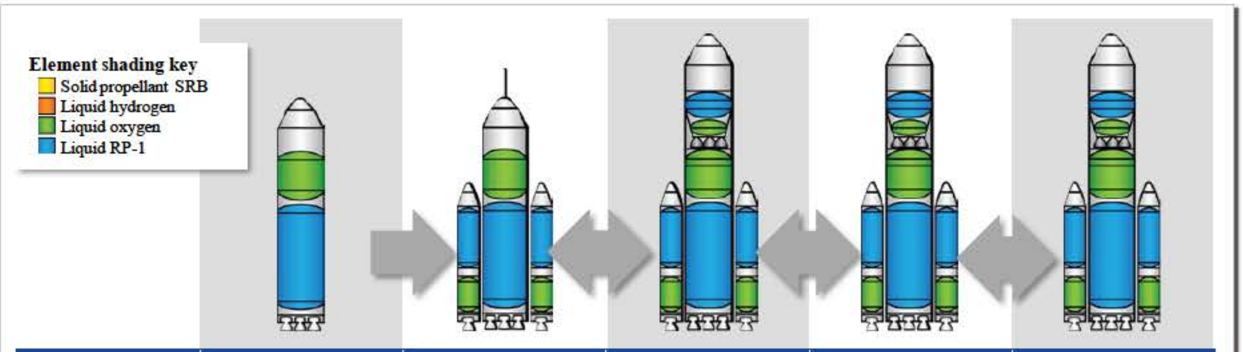
Config.	Phase 1 Block 0	Phase 2 Block 1	Phase 3 Block 2	Phase 4 Block 3
Booster	4-segment steel case PBAN	5-segment steel case PBAN	5-segment composite case PBAN	5-segment composite case HTPB
Core tank	33-ft-dia. standard height	33-ft-dia. stretched	33-ft-dia. stretched	33-ft-dia. stretched
Core engine	3x RS-84	3x RS-84	3x RS-84	3x RS-84
Second Stage	N/A	N/A	3x MB-100	3x MB-100
IMLEO	63 – 71 mT	84 – 88 mT	138 – 143 mT	147 – 154 mT
Mission	LEO	LEO	NEO/BEO	NEO/BEO
	≤3 flights	≤3 flights	>4 flights	>4 flights

HLPT-SATS_016

Figure 6-7. Trade 3 – Trial Case 3, Alternative Configuration 3–- Option 2, sensitivities to technology steps required in a heritage SRB and LOX/RP1 core engine configuration-evolvable launch vehicle family

For Trade 3 – Trial Case 3, Alternative Configuration 3 – Option 2, the LOV probability for the configurations range from 1 in 282 down to 1 in 114.

The fourth set of in-family configurations were based on LOX/RP1 Core Stage engines and LOX/RP1 engines in the booster configurations, as depicted in Figure 6-8.



Config.	Phase 1 Block 0	Phase 2 Block 1	Phase 3 Block 2	Phase 4 Block 3	Phase 5 Block 4
Booster	None	2x RS-84	2x 1 x AJ2M ORSC	2x 1 x AJ2M ORSC	2x 1 x AJ2M ORSC
Core tank	33-ft-dia. standard height	33-ft-dia. standard	33-ft-dia. standard	33-ft-dia. standard	33-ft-dia. standard
Core engine	3x RS-84	3x RS-84	3x RS-84	3x RS-84	3x AJ2M ORSC
Second Stage	N/A	N/A	3x AJ100K ORSC	3x AJ100K ORSC	3x AJ100K ORSC
IMLEO	21 – 24 mT	53 – 56 mT	72 – 75 mT	97 – 101 mT	134 – 140 mT
Mission	LEO	LEO	LEO/NEO	NEO/BEO	NEO/BEO
	>2 flights	>4 flights	>4 flights	>4 flights	>4 flights

HLPT-SATS_017

Figure 6-8. Trade 3 – Trial Case 3, Alternative Configuration 4 – Option 3, Sensitivities to technology steps required in a LOX/RP1 core engine with RP boosters configuration-evolvable launch vehicle family

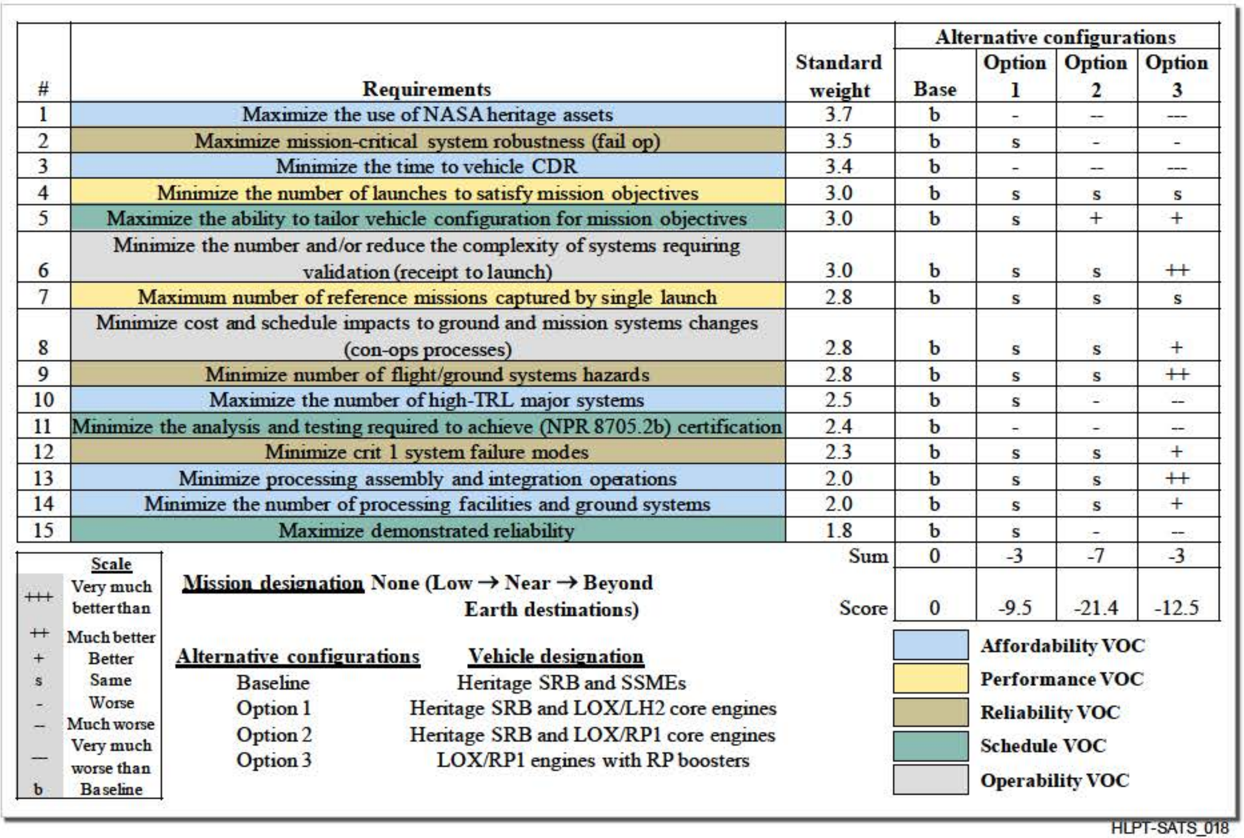
For Trade 3 – Trial Case 3, Alternative Configuration 4 – Option 3, the LOV probability for the configurations range from 1 in 224 down to 1 in 120.

Once the architectures were defined and characterized, the different vehicles were traded in a decision matrix, as shown in Table 6-4.

With its understood performance and limitations, along with inserting new technology incrementally, the Baseline depicts a strong in-family architecture. Correctly grouping the technology insertions to reduce the number of phases will be the implementation cornerstone.

The Baseline-evolvable family configuration scores very strong as pairs-wise compared against the alternative configuration options. The Baseline Block 1, 2, and 4 demonstrate the highest benefit per capability/cost step based on performance increase as compared to development risk posture and cost. The Baseline configuration provides earliest flight capability and minimizes DDT&E required based on significant use of existing NASA heritage assets, including ground and mission systems. Limitations on flexibility and extensibility as vehicle tailoring for specific mission profiles are not as favorable as Options 2 and 3. Heritage assets do come with the penalty of existing processing and operational requirements and processes that tend to be more restrictive and less efficient. To address the life-cycle affordability, when using heritage assets, it will become very important to perform a bottoms-up assessment of how the system is built, acceptance tested, installed, integrated tested and verified, and operated.

Table 6-4. Trade 3 – Trial Case 3 decision matrix



#	Requirements	Standard weight	Alternative configurations			
			Base	Option 1	Option 2	Option 3
1	Maximize the use of NASA heritage assets	3.7	b	-	--	---
2	Maximize mission-critical system robustness (fail op)	3.5	b	s	-	-
3	Minimize the time to vehicle CDR	3.4	b	-	--	---
4	Minimize the number of launches to satisfy mission objectives	3.0	b	s	s	s
5	Maximize the ability to tailor vehicle configuration for mission objectives	3.0	b	s	+	+
6	Minimize the number and/or reduce the complexity of systems requiring validation (receipt to launch)	3.0	b	s	s	++
7	Maximum number of reference missions captured by single launch	2.8	b	s	s	s
8	Minimize cost and schedule impacts to ground and mission systems changes (con-ops processes)	2.8	b	s	s	+
9	Minimize number of flight/ground systems hazards	2.8	b	s	s	++
10	Maximize the number of high-TRL major systems	2.5	b	s	-	--
11	Minimize the analysis and testing required to achieve (NPR 8705.2b) certification	2.4	b	-	-	--
12	Minimize crit 1 system failure modes	2.3	b	s	s	+
13	Minimize processing assembly and integration operations	2.0	b	s	s	++
14	Minimize the number of processing facilities and ground systems	2.0	b	s	s	+
15	Maximize demonstrated reliability	1.8	b	s	-	--
		Sum	0	-3	-7	-3
		Score	0	-9.5	-21.4	-12.5

	Scale
+++	Very much better than
++	Much better
+	Better
s	Same
-	Worse
--	Much worse
---	Very much worse than
b	Baseline

Mission designation **None (Low → Near → Beyond Earth destinations)**

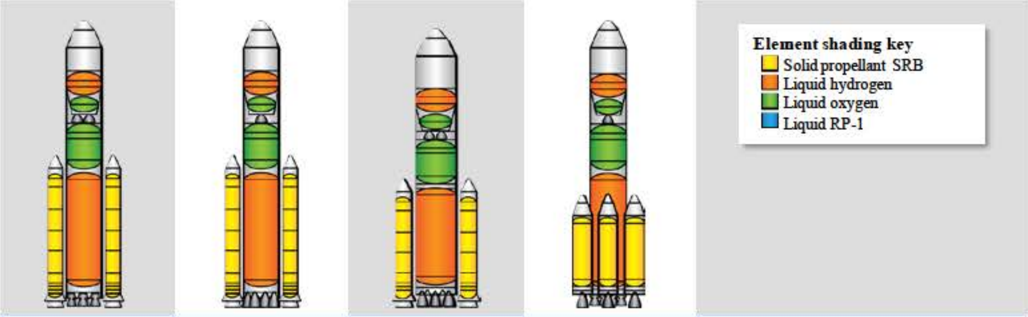
Alternative configurations	**Vehicle designation**
Baseline	Heritage SRB and SSMEs
Option 1	Heritage SRB and LOX/LH2 core engines
Option 2	Heritage SRB and LOX/RP1 core engines
Option 3	LOX/RP1 engines with RP boosters

🟦	**Affordability VOC**
🟨	**Performance VOC**
🟫	**Reliability VOC**
🟩	**Schedule VOC**
⬜	**Operability VOC**

HLPT-SATS_018

6.2.4 Trade 4 – Trial Case 4 Results

Trade 4 – Trial Case 4 was developed as an end-state configuration trade. The focus of Trade 4 was on the sensitivity of segmented solids vs. monolithic solids. As shown in Figure 6-9, two five-segment and two four-segment heritage solids were compared against a known large monolith solid booster.



Config.	Baseline	Option 1	Option 2	Option 3	Notes
Vehicle designation	LV-275-5 RS25-5S-2.J2X	LV-275-4 RS68-5S-1.J2X	LV-330-5 RS68-4S-2.J2X	LV-275-5 RS25-6MS-1.J2X	
Geometry	27.5-ft-dia. stretched height	27.5-ft-dia. stretched	33-ft-dia. standard	27.5-ft-dia. stretched	
Booster	2x 5-segment steel case PBAN	2x 5-segment steel case PBAN	2x 4-segment steel case PBAN	6x P110 SRB monolith	
Core engine	5x RS-25E	4x RS-68B	5x RS-68B	5x RS-25E	
Upper Stage engine	2x J-2X	1x J-2X	2x J-2X	1x J-2X	
IMLEO	137 – 143 mT	117 – 125 mT	125 – 134 mT	131 – 140 mT	

HLPT-SATS_019

Figure 6-9. Trade 4 – Trial Case 4, sensitivities to segmented solids versus monolithic solids

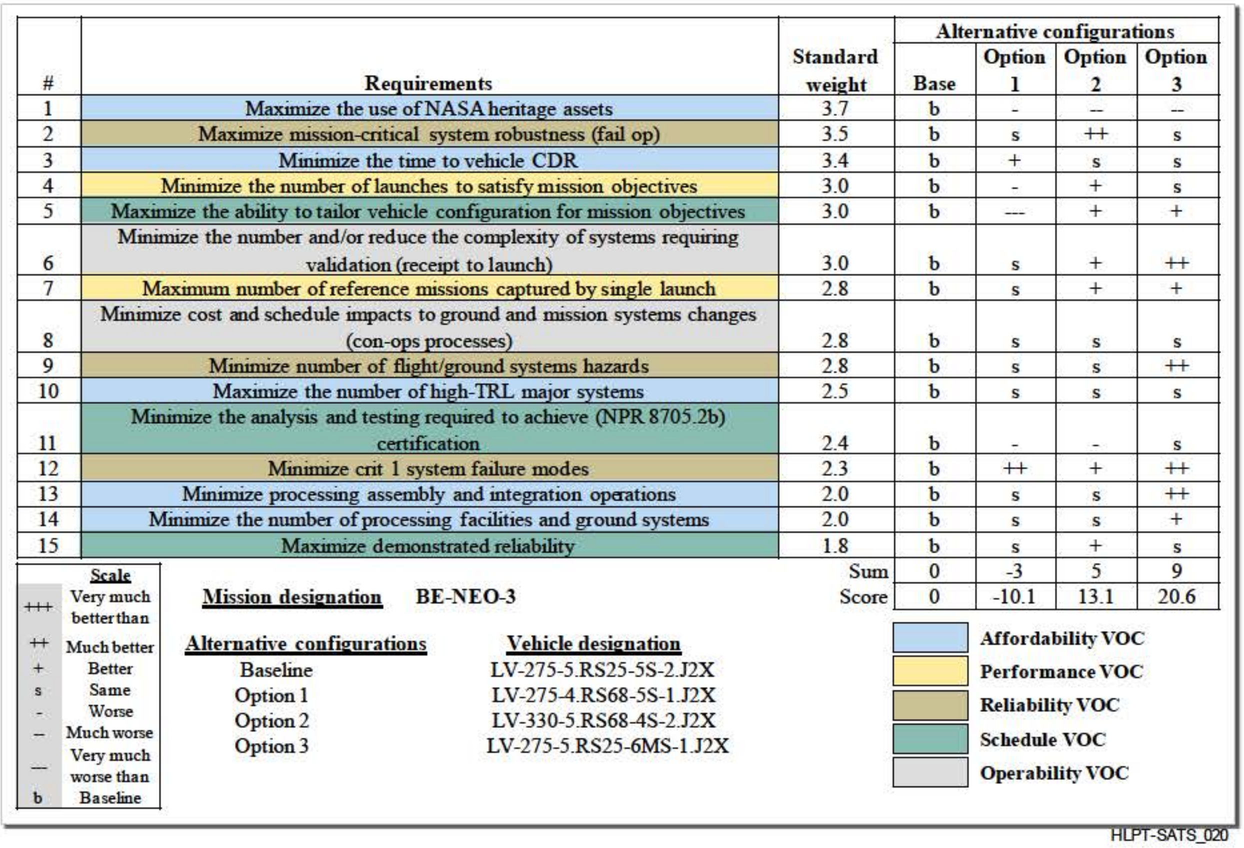
For Trade 4 – Trial Case 4, the LOV probability for the Baseline, Option 1, Option 2, and Option 3 were 1 in 208, 1 in 233, 1 in 248, and 1 in 261, respectively.

Once the architectures were defined and characterized, the different vehicles were traded in a decision matrix, as shown in Table 6-5.

In assessing Option 3 with the monolith solids against the other segmented solid booster configurations, reducing the complexity of the booster validation and reduction in the processing hazards depicts monolithic booster vehicles as a viable configuration. While not as performance capable as the current SSP SRBs, large monolith boosters demonstrate very positive LLC affordability predictions. Allowing tailorability by adding or removing boosters from a core configuration to "right-size" performance and align cost against mission requirements is an enabler.

The expendable versus recoverable affordability trade turns around at two to three flights per year instead of five to six flights per year for SSP SRBs when monolith boosters are recovered and refurbished or scrapped for reusable parts onsite at KSC. Monolith boosters are considered more "operations friendly" than segmented solids due to reduced operations, reduced hazards, reduced required inspections, and less failure modes. The development and production risk is low, as the P110 can be "up scaled" from production P80 variant that are designed and built for the Arianespace Vega rocket, a joint project by the Italian Space Agency and the ESA. This risk will be partially mitigated by the time NASA requires the capability.

Table 6-5. Trade 4 – Trial Case 4 decision matrix



#	Requirements	Standard weight	Alternative configurations			
			Base	Option 1	Option 2	Option 3
1	Maximize the use of NASA heritage assets	3.7	b	-	--	--
2	Maximize mission-critical system robustness (fail op)	3.5	b	s	++	s
3	Minimize the time to vehicle CDR	3.4	b	+	s	s
4	Minimize the number of launches to satisfy mission objectives	3.0	b	-	+	s
5	Maximize the ability to tailor vehicle configuration for mission objectives	3.0	b	---	+	+
6	Minimize the number and/or reduce the complexity of systems requiring validation (receipt to launch)	3.0	b	s	+	++
7	Maximum number of reference missions captured by single launch	2.8	b	s	+	+
8	Minimize cost and schedule impacts to ground and mission systems changes (con-ops processes)	2.8	b	s	s	s
9	Minimize number of flight/ground systems hazards	2.8	b	s	s	++
10	Maximize the number of high-TRL major systems	2.5	b	s	s	s
11	Minimize the analysis and testing required to achieve (NPR 8705.2b) certification	2.4	b	-	-	s
12	Minimize crit 1 system failure modes	2.3	b	++	+	++
13	Minimize processing assembly and integration operations	2.0	b	s	s	++
14	Minimize the number of processing facilities and ground systems	2.0	b	s	s	+
15	Maximize demonstrated reliability	1.8	b	s	+	s
		Sum	0	-3	5	9
		Score	0	-10.1	13.1	20.6

Scale	
+++	Very much better than
++	Much better
+	Better
s	Same
-	Worse
--	Much worse
---	Very much worse than
b	Baseline

Mission designation **BE-NEO-3**

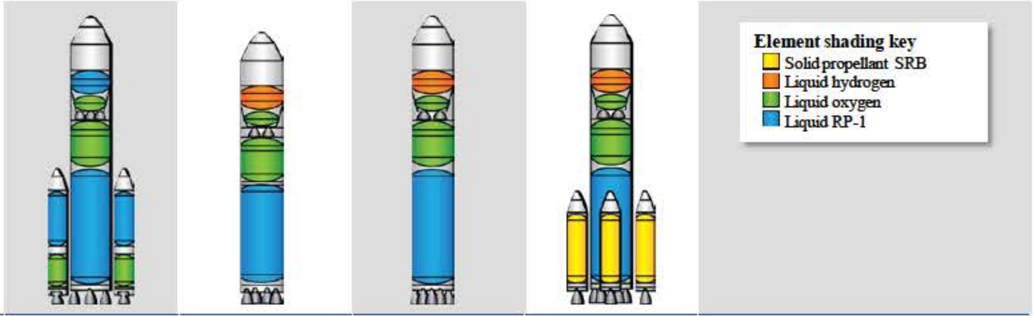
Alternative configurations	Vehicle designation
Baseline	LV-275-5.RS25-5S-2.J2X
Option 1	LV-275-4.RS68-5S-1.J2X
Option 2	LV-330-5.RS68-4S-2.J2X
Option 3	LV-275-5.RS25-6MS-1.J2X

🟦	**Affordability VOC**
🟨	**Performance VOC**
🟫	**Reliability VOC**
🟩	**Schedule VOC**
⬜	**Operability VOC**

HLPT-SATS_020

6.2.5 *Trade 5 – Trial Case 5 Results*

Trade 5 – Trial Case 5 was developed as an end-state configuration trade. The focus of Trade 5 was on the sensitivities amongst LOX/RP1 vehicles. While all Core Stage engines were LOX/RP1 propellant propulsion systems, different Second/Upper Stages and Booster Stages were added, as shown in Figure 6-10.



Config.	Baseline	Option 1	Option 2	Option 3	Notes
Vehicle designation	LV-330-3.RS84-2.2RS84-3.AJ100K	LV-330-4.RP2.0M-0-2.J2X	LV-330-6.RP2.0M-0-2.J2X	LV-330-6.RP2.0M-4MS-2.J2X	
Geometry	33-ft-dia. stretched height	33-ft-dia. standard	33-ft-dia. stretched	33-ft-dia. stretched	
Booster	2x 2 RS-84	None	None	4x P110 SRB monolith	
Core engine	3x RS-84	4x RP 2.0 Mlbf GG	6x RP 2.0 Mlbf GG	6x RP 2.0 Mlbf GG	
Upper Stage engine	3x AJ100K ORSC	2x J-2X	2x J-2X	2x J-2X	
IMLEO	66 – 75 mT	84 – 92 mT	130 – 137 mT	145 – 153 mT	

HLPT-SATS_021

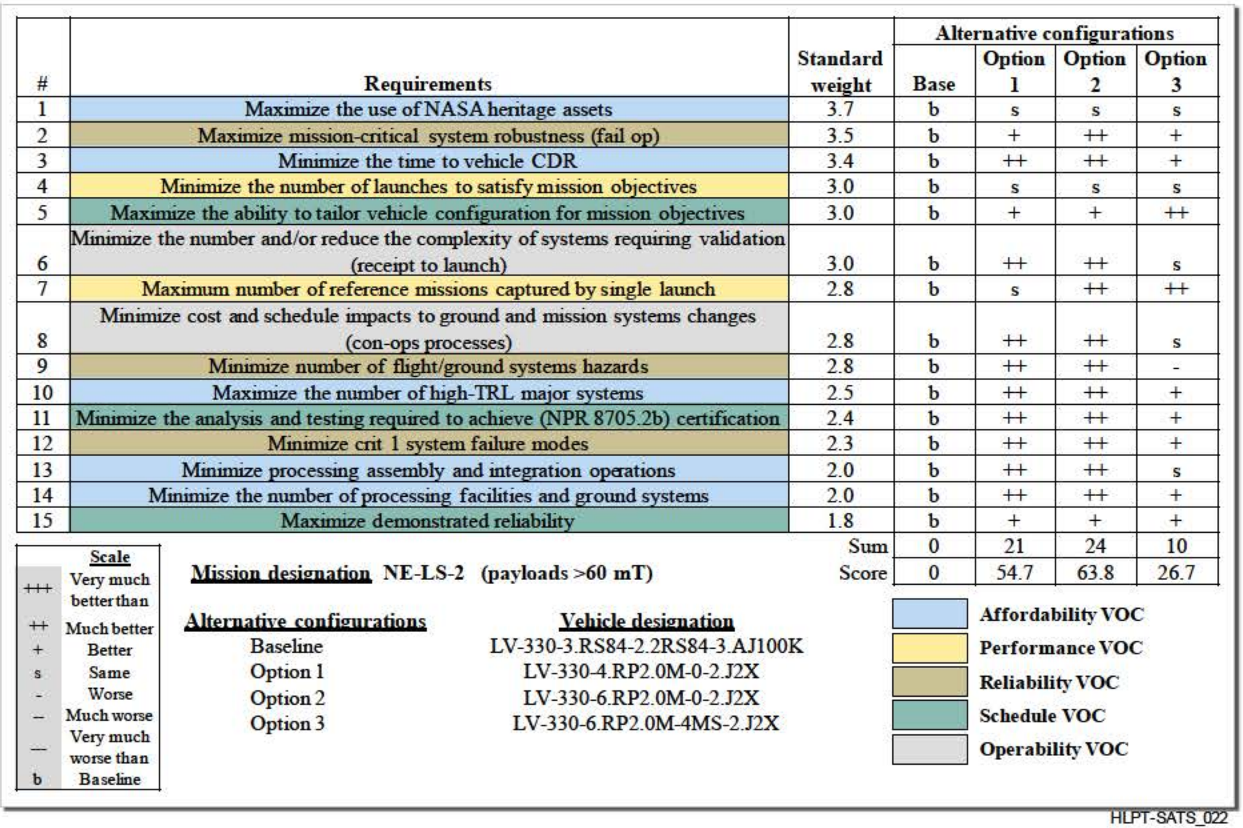
Figure 6-10. Trade 5 – Trial Case 5, sensitivities between difference Core Stage RP engine configurations

For Trade 5 – Trial Case 5, the LOV probability for the Baseline, Option 1, Option 2, and Option 3 were 1 in 103, 1 in 181, 1 in 157, and 1 in 234, respectively.

Once the architectures were defined and characterized, the different vehicles were traded in a decision matrix, as shown in Figure 6-6.

The ability to tailor the vehicle, including the 33-foot-diameter shroud interface, and one less element to integrate and operate without the boosters, depicts an operations friendly configuration on Options 1 and 2. All liquid propulsion and minimized integrated elements make Options 1 and 2 very favorable against the Baseline configuration, although significant DDT&E schedule and cost are associated with the development of the new LOX/RP 2.0 Mlb_f Gas Generator (GG) and/or 100K Mlb_f ORSC class engines for Options 1, 2, and 3. Two integrated stage developments on Options 1 and 2 in lieu of three integrated stage developments for the Baseline directs costs to two distinct element developments (i.e., core and upper) versus three (Booster, Core, and Upper) proves advantageous when assessing and reducing the near-term DDT&E costs.

Table 6-6. ***Trade 5 – Trial Case 5 decision matrix***



#	Requirements	Standard weight	Alternative configurations Base	Option 1	Option 2	Option 3
1	Maximize the use of NASA heritage assets	3.7	b	s	s	s
2	Maximize mission-critical system robustness (fail op)	3.5	b	+	++	+
3	Minimize the time to vehicle CDR	3.4	b	++	++	+
4	Minimize the number of launches to satisfy mission objectives	3.0	b	s	s	s
5	Maximize the ability to tailor vehicle configuration for mission objectives	3.0	b	+	+	++
6	Minimize the number and/or reduce the complexity of systems requiring validation (receipt to launch)	3.0	b	++	++	s
7	Maximum number of reference missions captured by single launch	2.8	b	s	++	++
8	Minimize cost and schedule impacts to ground and mission systems changes (con-ops processes)	2.8	b	++	++	s
9	Minimize number of flight/ground systems hazards	2.8	b	++	++	-
10	Maximize the number of high-TRL major systems	2.5	b	++	++	+
11	Minimize the analysis and testing required to achieve (NPR 8705.2b) certification	2.4	b	++	++	+
12	Minimize crit 1 system failure modes	2.3	b	++	++	+
13	Minimize processing assembly and integration operations	2.0	b	++	++	s
14	Minimize the number of processing facilities and ground systems	2.0	b	++	++	+
15	Maximize demonstrated reliability	1.8	b	+	+	+
		Sum	0	21	24	10
		Score	0	54.7	63.8	26.7

Scale

+++	Very much better than
++	Much better
+	Better
s	Same
-	Worse
–	Much worse
—	Very much worse than
b	Baseline

Mission designation **NE-LS-2 (payloads >60 mT)**

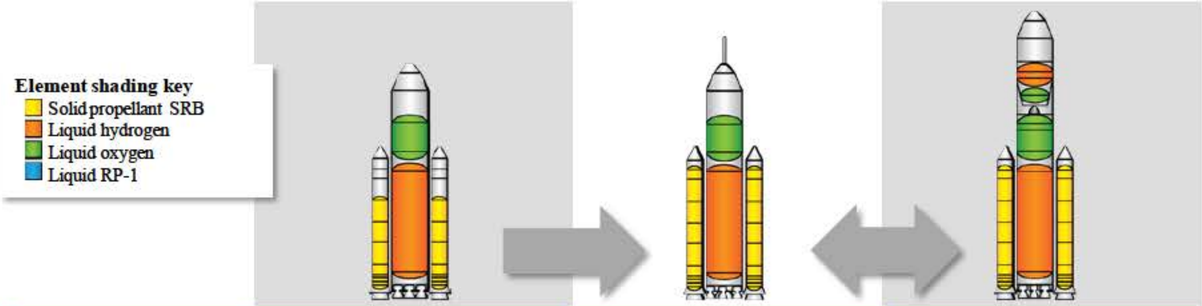
Alternative configurations	**Vehicle designation**
Baseline	LV-330-3.RS84-2.2RS84-3.AJ100K
Option 1	LV-330-4.RP2.0M-0-2.J2X
Option 2	LV-330-6.RP2.0M-0-2.J2X
Option 3	LV-330-6.RP2.0M-4MS-2.J2X

🟦	**Affordability VOC**
🟨	**Performance VOC**
🟫	**Reliability VOC**
🟩	**Schedule VOC**
⬜	**Operability VOC**

HLPT-SATS_022

6.2.6 Trade 6 – Trial Case 6 Results

Based on the results from Trial Cases 1, 1A, 2, evolvable family Trial Case 3, Trial Cases 4, and 5, Trade 6 – Trial Case 6 was developed as an evolvable family configuration trade. These evolvable vehicle families have the technology steps logically grouped and therefore are more streamlined block changes. Again, Block 0 would be IOC, not human rated, and Blocks 1, 2, etc would be human rated Arrows depict capacity extensibility so the mission can be tailored. A double arrow indicates that the configuration can be tailored for flight in either increasing or decreasing capability. Technology changes from one phase to the next are identified in blue font. The first set of in-family configurations were based on heritage SRB and SSME configurations, as depicted in Figure 6-11.



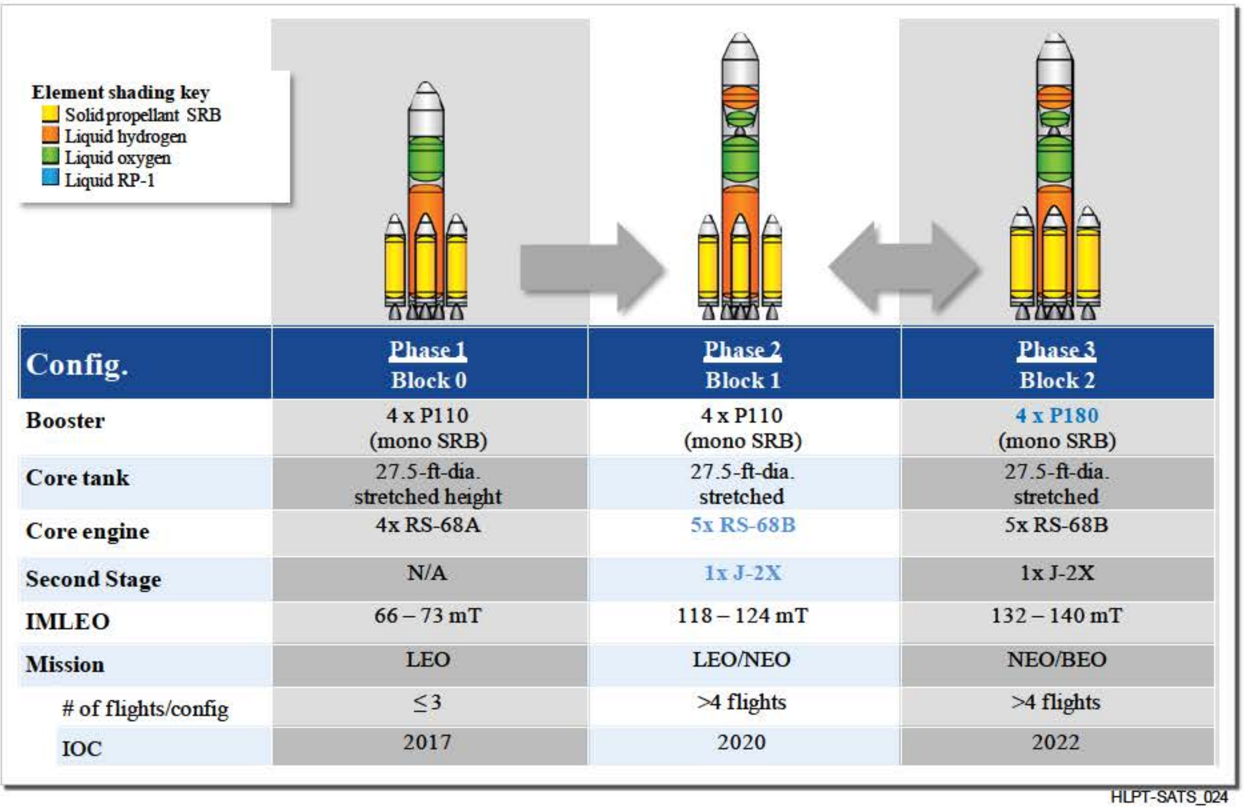
Config.	Phase 1 Block 0	Phase 2 Block 1	Phase 3 Block 2
Booster	4-segment steel case PBAN + spacer	5-segment steel case PBAN	5-segment steel case PBAN
Core tank	27.5-ft-dia. stretched height	27.5-ft-dia. stretched	27.5-ft-dia. stretched
Core engine	3 SSME (RS-25D, Shuttle-like MPS)	5 RS-25D/E (new MPS)	5 RS-25E
Second Stage	N/A	N/A	1 J-2X
IMLEO	55 – 62 mT	75 – 81 mT	123 – 126 mT
Mission	LEO	LEO	NEO/BEO
# of flights/config	≤2	>2	>2
IOC	2016	2019	2022

HLPT-SATS_023

Figure 6-11. Trade 6 – Trial Case 6, Alternative Configuration 1 – Baseline, sensitivities to evolution-increasing performance capability required in a Heritage SRB and SSME configuration-evolvable launch vehicle family.

For Trade 6 – Trial Case 6, Alternative Configuration 1 – Baseline, the LOV probability for the phase configurations were 1 in 265, 1 in 242, and 1 in 224.

The second set of in-family configurations were based on monolithic SRB and LOX/LH_2 core engine configuration evolvable launch vehicle family, as shown in Figure 6-12.



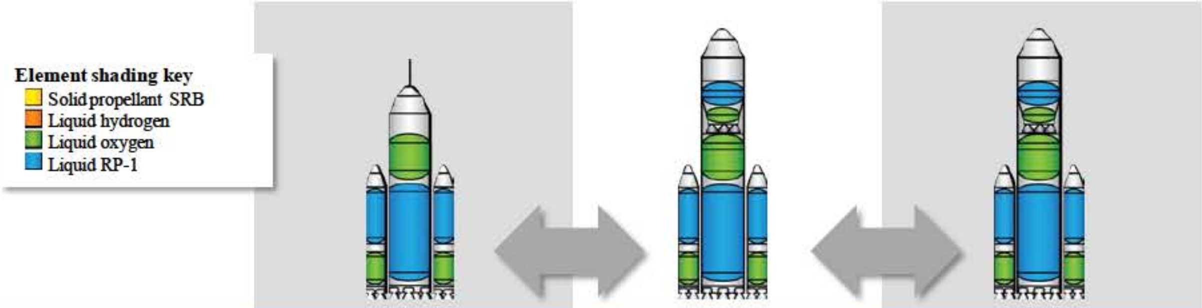
Config.	Phase 1 Block 0	Phase 2 Block 1	Phase 3 Block 2
Booster	4 x P110 (mono SRB)	4 x P110 (mono SRB)	4 x P180 (mono SRB)
Core tank	27.5-ft-dia. stretched height	27.5-ft-dia. stretched	27.5-ft-dia. stretched
Core engine	4x RS-68A	5x RS-68B	5x RS-68B
Second Stage	N/A	1x J-2X	1x J-2X
IMLEO	66 – 73 mT	118 – 124 mT	132 – 140 mT
Mission	LEO	LEO/NEO	NEO/BEO
# of flights/config	≤3	>4 flights	>4 flights
IOC	2017	2020	2022

HLPT-SATS_024

Figure 6-12. Trade 6 – Trial Case 6, Alternative Configuration 2 – Option 1, sensitivities to evolution-increasing performance capability required in a monolithic SRB and LOX/LH$_2$ core engine configuration-evolvable launch vehicle family

For Trade 6 – Trial Case 6, Alternative Configuration 2 – Option 1, the LOV probability for the phase configurations were 1 in 295, 1 in 263, and 1 in 263. Increasing the performance of the monolith solid rockets did not introduce any new failure modes.

The third set of in-family configurations were based on LOX/RP1 engines with LOX/RP1 booster configuration-evolvable launch vehicle family in Figure 6-13.



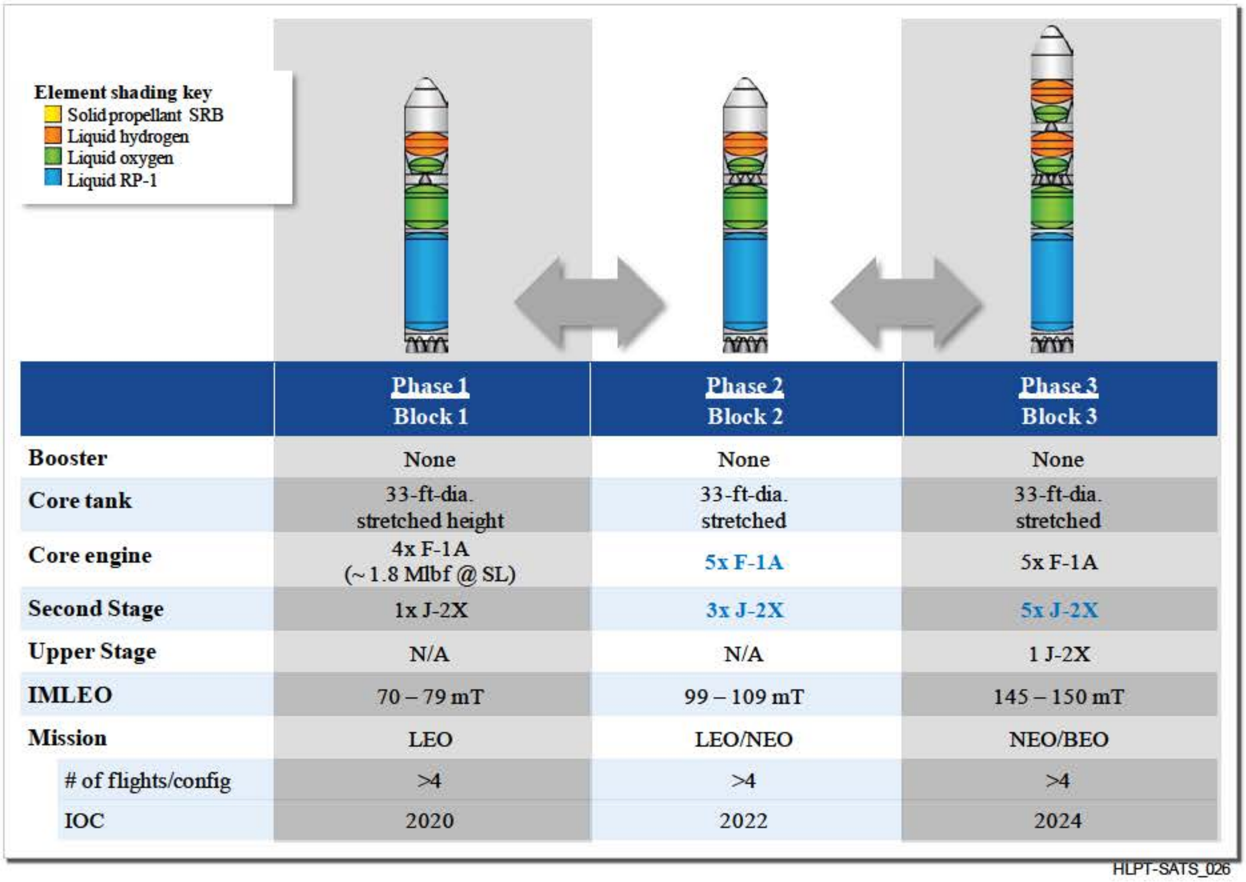
	Phase 1 Block 1	Phase 2 Block 2	Phase 3 Block 3
Booster	2x 2 x RS-84	2x 2 x RS-84	2x 2 x RS-84
Core tank	33-ft-dia. standard height	33-ft-dia. standard	33-ft-dia. standard
Core engine	5x RS-84	5x RS-84	7x RS-84
Second Stage	N/A	3x AJ100K ORSC	3x AJ100K ORSC
IMLEO	98 – 103 mT	127 – 131 mT	141 – 148 mT
Mission	LEO	NEO/BEO	NEO/BEO
# of flights/config	>4	>4	>4
IOC	2019	2022	2024

HLPT-SATS_025

Figure 6-13. Trade 6 – Trial Case 6, Alternative Configuration 3 – Option 2, sensitivities to evolution-increasing performance capability required in a monolithic SRB and LOX/LH_2 core engine configuration-evolvable launch vehicle family

For Trade 6 – Trial Case 6, Alternative Configuration 3 – Option 2, the LOV probability for the phase configurations were 1 in 189, 1 in 138, and 1 in 78. The affect of the liquid engine reliability drives the total vehicle level probability very low.

The fourth set of in-family configurations were based on LOX/RP1 engines, without boosters, configuration-evolvable launch vehicle family, as shown in Figure 6-14.



	Phase 1 Block 1	Phase 2 Block 2	Phase 3 Block 3
Booster	None	None	None
Core tank	33-ft-dia. stretched height	33-ft-dia. stretched	33-ft-dia. stretched
Core engine	4x F-1A (~1.8 Mlbf @ SL)	5x F-1A	5x F-1A
Second Stage	1x J-2X	3x J-2X	5x J-2X
Upper Stage	N/A	N/A	1 J-2X
IMLEO	70 – 79 mT	99 – 109 mT	145 – 150 mT
Mission	LEO	LEO/NEO	NEO/BEO
# of flights/config	>4	>4	>4
IOC	2020	2022	2024

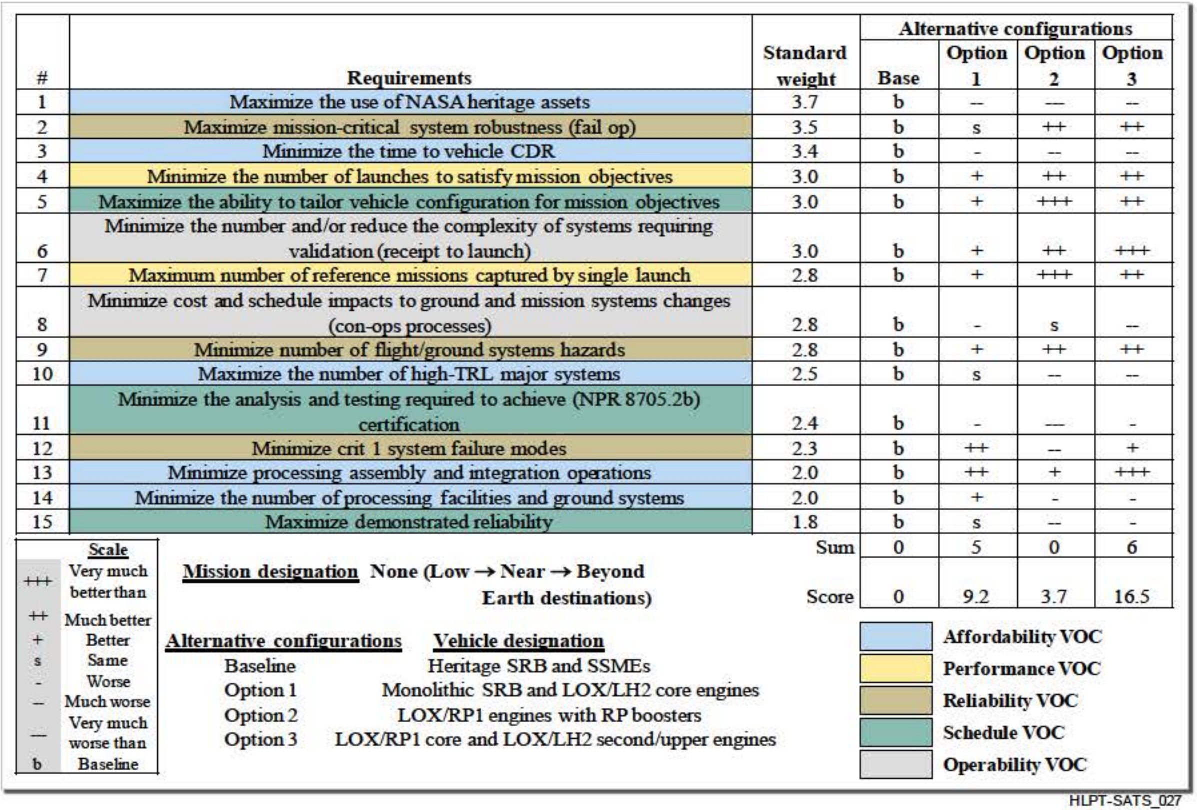
Figure 6-14. Trade 6 – Trial Case 6, Alternative Configuration 4 – Option 3, sensitivities to evolution-increasing performance capability required in a LOX/RP1 core and LOX/LH$_2$ second engines configuration-evolvable launch vehicle family

For Trade 6 – Trial Case 6, Alternative Configuration 4 – Option 3, the LOV probability for the phase configurations were 1 in 192, 1 in 142, and 1 in 62. With the addition of a third stage and three more liquid engines in the stack, the Block 3's total vehicle level probability drops significantly.

Once the architectures were defined and characterized, the different vehicles were traded in a decision matrix, as shown in Table 6-7.

All liquid propulsion and minimized integrated elements make Option 3 a favorable evolvable family configuration against the Baseline configuration. When installing an understood Core Stage propulsion system, like the F-1A based on manufacturing two demonstration engines and test stand results, makes this "unflown" engine a very good candidate as compared to the 2.0 Mlbf GG traded engine, which is feasible but without demonstrated performance or reliability. The three stage vehicle introduces a new challenge on processing with respect to LLC (recurring) and development cost to design a third stage. This architecture from a processing flow standpoint is not any different than the three elements with

Table 6-7. ***Trade 6 – Trial Case 6 decision matrix***



#	Requirements	Standard weight	Alternative configurations Base	 Option 1	 Option 2	 Option 3
1	Maximize the use of NASA heritage assets	3.7	b	--	---	--
2	Maximize mission-critical system robustness (fail op)	3.5	b	s	++	++
3	Minimize the time to vehicle CDR	3.4	b	-	--	--
4	Minimize the number of launches to satisfy mission objectives	3.0	b	+	++	++
5	Maximize the ability to tailor vehicle configuration for mission objectives	3.0	b	+	+++	++
6	Minimize the number and/or reduce the complexity of systems requiring validation (receipt to launch)	3.0	b	+	++	+++
7	Maximum number of reference missions captured by single launch	2.8	b	+	+++	++
8	Minimize cost and schedule impacts to ground and mission systems changes (con-ops processes)	2.8	b	-	s	--
9	Minimize number of flight/ground systems hazards	2.8	b	+	++	++
10	Maximize the number of high-TRL major systems	2.5	b	s	--	--
11	Minimize the analysis and testing required to achieve (NPR 8705.2b) certification	2.4	b	-	---	-
12	Minimize crit 1 system failure modes	2.3	b	++	--	+
13	Minimize processing assembly and integration operations	2.0	b	++	+	+++
14	Minimize the number of processing facilities and ground systems	2.0	b	+	-	-
15	Maximize demonstrated reliability	1.8	b	s	--	-
		Sum	0	5	0	6
		Score	0	9.2	3.7	16.5

Scale

+++	Very much better than
++	Much better
+	Better
s	Same
-	Worse
--	Much worse
---	Very much worse than
b	Baseline

Mission designation **None (Low → Near → Beyond Earth destinations)**

Alternative configurations	**Vehicle designation**
Baseline	Heritage SRB and SSMEs
Option 1	Monolithic SRB and LOX/LH2 core engines
Option 2	LOX/RP1 engines with RP boosters
Option 3	LOX/RP1 core and LOX/LH2 second/upper engines

🟦 **Affordability VOC**
🟨 **Performance VOC**
🟫 **Reliability VOC**
🟩 **Schedule VOC**
⬜ **Operability VOC**

HLPT-SATS_027

boosters on Option 2. Schedule is a driving factor in Option 3, if and only if the IOC is not a driving constraint, which will allow some FY distribution of costs.

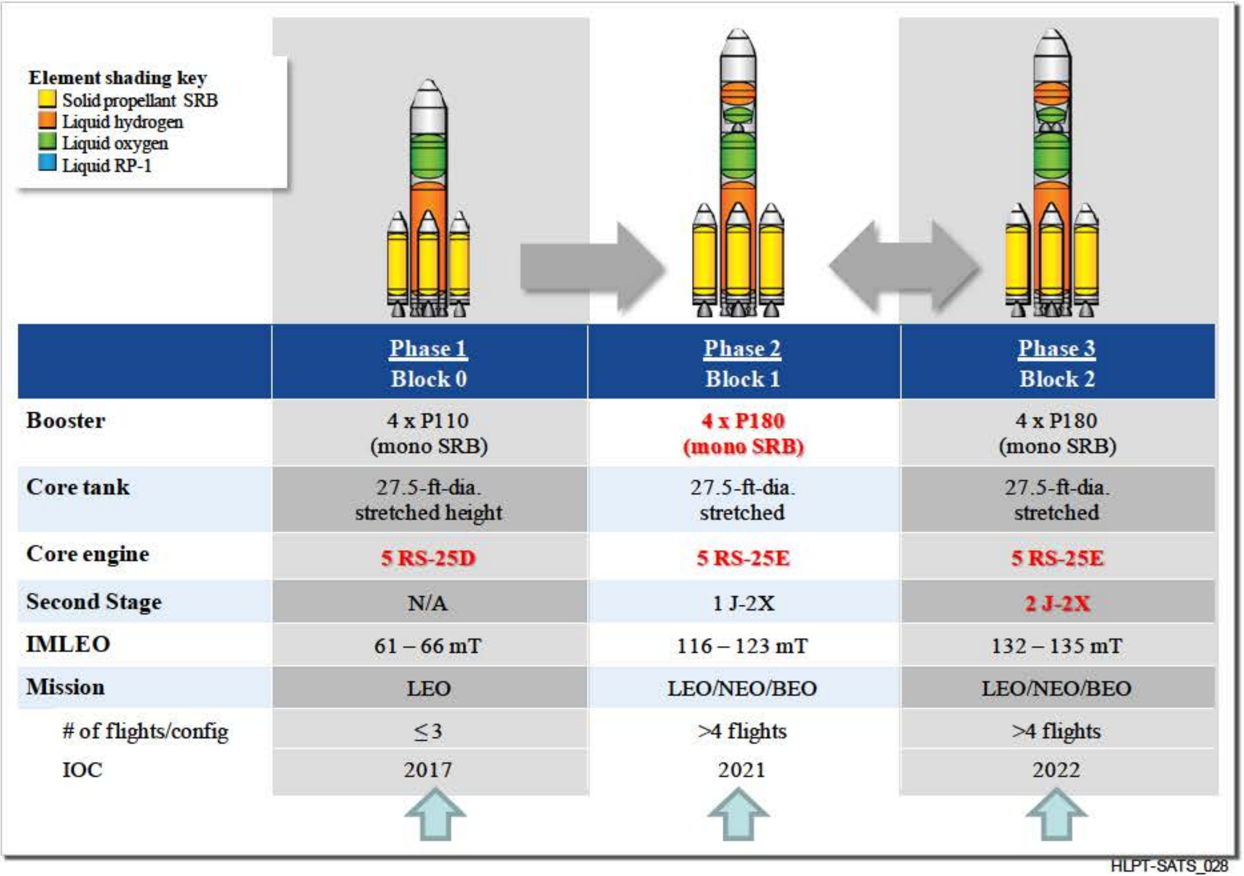
The benefit of the monolithic solids in Option 1 is in the Affordability and Reliability FOMs as compared to the Baseline, as this clearly depicts a reusable element with significant cost avoidance relative to segmented solids. In addition, with Option 1, you can avoid significant overhead and transportation costs if monolithic boosters are manufactured and refurbished at KSC.

Tailorability of Option 1 to fly four or six boosters and ease of processing of Option 3 depict overarching benefits toward affordability. Liquid boosters in Option 2, while removing VAB hazards without segmented solids, due to immaturity of systems, do not score very well.

USA's parametric cost evaluastions of Trade 6, baseline and option comparisions, are documented in Appendix C.

6.2.6.1 Trade 6 – Trial Case 6A Results

When reviewing the results of Trade 6 – Trial Case 6 of the end-state configuration trade, a gap analysis was conducted to determine if "weaknesses" in the architecture could be identified. In Option 1, by removing the high TRL but yet unproven human-rated RS-68 Core Stage engine and replacing them with the known SSME (RS-25D) on the Core Stage, an engine that has a proven history, existing supply chain and market segment, and a demonstrated reliability, along with moving the higher performance monolith boosters earlier in the evolution to compensate for reduction in the Core Stage engine performance going away from the RS-68 to the RS-25, would provide near-term schedule and cost benefits. Figure 6-15 depicts the change, in red shadow font, to Option 1.



	Phase 1 Block 0	Phase 2 Block 1	Phase 3 Block 2
Booster	4 x P110 (mono SRB)	**4 x P180 (mono SRB)**	4 x P180 (mono SRB)
Core tank	27.5-ft-dia. stretched height	27.5-ft-dia. stretched	27.5-ft-dia. stretched
Core engine	**5 RS-25D**	**5 RS-25E**	**5 RS-25E**
Second Stage	N/A	1 J-2X	**2 J-2X**
IMLEO	61 – 66 mT	116 – 123 mT	132 – 135 mT
Mission	LEO	LEO/NEO/BEO	LEO/NEO/BEO
# of flights/config	≤3	>4 flights	>4 flights
IOC	2017	2021	2022

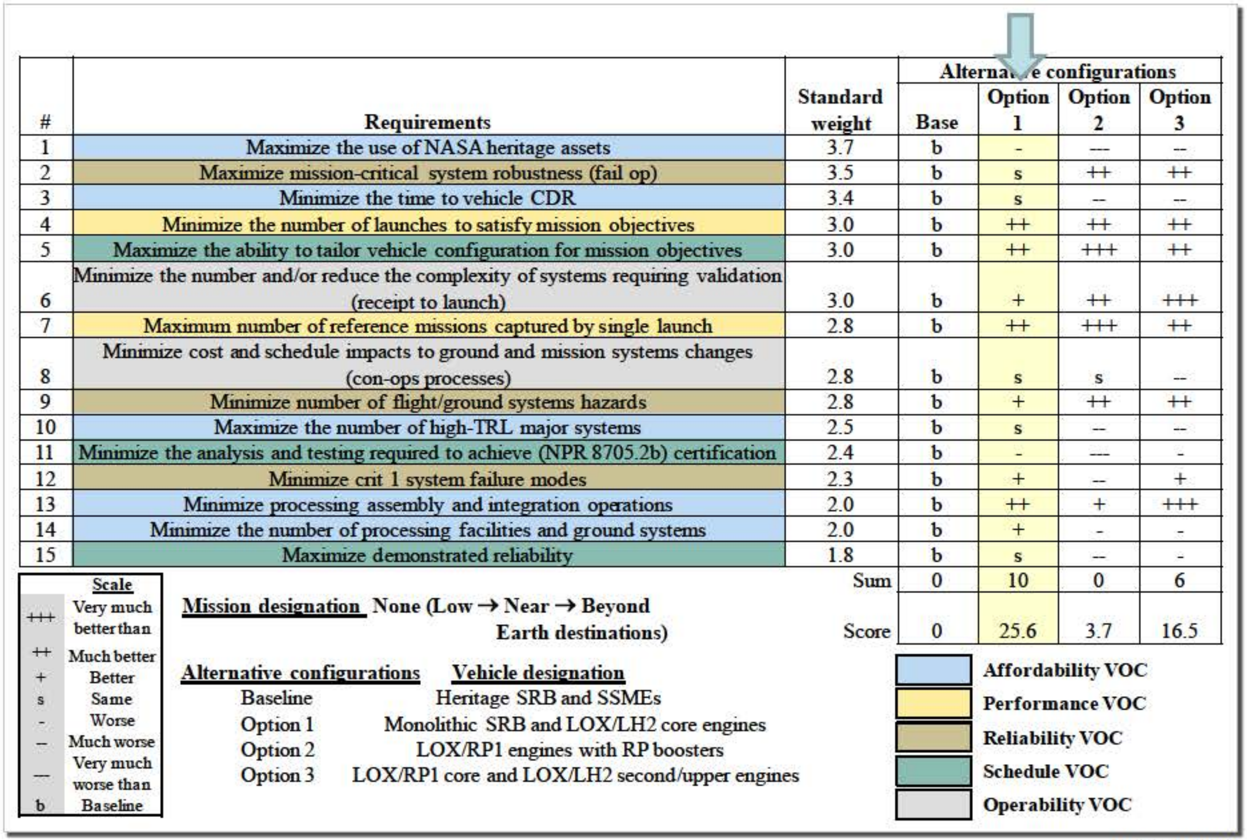
HLPT-SATS_028

Figure 6-15. Trade 6 – Trial Case 6A, Alternative Configuration 2 – Option 1, sensitivities to evolution-increasing performance capability required in a monolithic SRB and LOX/LH$_2$ core engine configuration-evolvable launch vehicle family

For Trade 6 – Trial Case 6A, the LOV probability for the modified configurations in Phases 1, 2, and 3 were 1 in 301, 1 in 276, and 1 in 230, respectively. This is mainly due to 5 RS-25D/E having a probability of 1 in 234 and 4 RS-68A/B having a problability of 1 in 221 with no engine-out considerations.

Once the changed architecture was defined and characterized, the Option 1 configuration was traded in the decision matrix, as shown in Table 6-8.

*Table 6-8. **Trade 6 – Trial Case 6A decision matrix***



#	Requirements	Standard weight	Alterna e configurations Base	 Option 1	 Option 2	 Option 3
1	Maximize the use of NASA heritage assets	3.7	b	-	---	--
2	Maximize mission-critical system robustness (fail op)	3.5	b	s	++	++
3	Minimize the time to vehicle CDR	3.4	b	s	--	--
4	Minimize the number of launches to satisfy mission objectives	3.0	b	++	++	++
5	Maximize the ability to tailor vehicle configuration for mission objectives	3.0	b	++	+++	++
6	Minimize the number and/or reduce the complexity of systems requiring validation (receipt to launch)	3.0	b	+	++	+++
7	Maximum number of reference missions captured by single launch	2.8	b	++	+++	++
8	Minimize cost and schedule impacts to ground and mission systems changes (con-ops processes)	2.8	b	s	s	--
9	Minimize number of flight/ground systems hazards	2.8	b	+	++	++
10	Maximize the number of high-TRL major systems	2.5	b	s	--	--
11	Minimize the analysis and testing required to achieve (NPR 8705.2b) certification	2.4	b	-	---	-
12	Minimize crit 1 system failure modes	2.3	b	+	--	+
13	Minimize processing assembly and integration operations	2.0	b	++	+	+++
14	Minimize the number of processing facilities and ground systems	2.0	b	+	-	-
15	Maximize demonstrated reliability	1.8	b	s	--	-
		Sum	0	10	0	6
		Score	0	25.6	3.7	16.5

Scale	
+++	Very much better than
++	Much better
+	Better
s	Same
-	Worse
--	Much worse
---	Very much worse than
b	Baseline

Mission designation None (Low → Near → Beyond Earth destinations)

Alternative configurations	**Vehicle designation**
Baseline	Heritage SRB and SSMEs
Option 1	Monolithic SRB and LOX/LH2 core engines
Option 2	LOX/RP1 engines with RP boosters
Option 3	LOX/RP1 core and LOX/LH2 second/upper engines

Affordability VOC
Performance VOC
Reliability VOC
Schedule VOC
Operability VOC

HLPT-SATS_029

In assessing the results, a higher Core Stage engine understanding and tailorability of the number of monolithic boosters allows flexibility in vehicle configurations to optimize for specific mission profiles.

Changing the RS-68A/B to RS-25D/E on the Core Stage and advancing the P180 monolithic solids forward in the evolution makes Option 1 a more favorable evolvable family configuration as compared against the Baseline configuration. The benefits are reduction in Core Stage propulsion system schedule and cost to certification, although the market segment for RS-25 is unique as compared to RS-68 family and commonality with Delta IV, thus more sensitive to supply chain volatility. With moving the higher performance boosters forward in the evolution, the architecture provides more performance sooner and more tailorability to "right size" cost against mission requirements.

6.3 In-Space Stage Results

Over the past decade, multiple In-Space propulsion systems, along with accompanying energy and power systems, have matured significantly. The current NASA Office of Chief Technologist (OCT) Technology Area (TA) TA-02 and TA-03 initiatives provide a very good groundwork as a point of departure for the trade. Once the data were reviewed, it was determined that to trade the In-Space Stage, a framework was established specifically for this phase of operations.

a. In-Space transfer operations to support an operational HLL system by earliest 2022 to No Later Than (NLT) 2025

b. Stage performance governing characteristics include the following:

   1. Engine shall be restartable, capable of very high acceleration and have high propellant efficiency

   2. Maintain a high reliability through the system's life cycle

USA researched and leveraged past technology advancements to accomplish this trade. Research in NASA's PCAD Project developments of the LOX/LCH_4 engines, NASA's Space Act Agreements for the development activity on the VASIMR propulsion systems, and NASA's SBTR Program for the development of the Electron Cyclotron Resonance propulsion systems to name a few.

Because of the mission operations constraints beyond Earth's dwell, this is the phase of the mission where technology developments can really pay off. Significant technology achievements and operational limitations make you look for a better design solution.

For technology achievements, USA broke them down into either chemical propulsion stages or in-situ resource stages (i.e., electric propulsion and power) "operational" classes. For chemical propellant engines, these are defined as engines that use a fuel propellant burned with an oxidizer propellant, expelled at a very high speed, to product thrust. They require both a fuel propellant and oxidizer propellant. There are three different categories of chemical propellants.

a. Solid propellants - Comprised of a fuel and oxidizer that is cured into a solid shape. Cannot be throttled in real time, although a predesigned thrust profile can be created by altering the interior propellant geometry

b. Liquid propellants - Comprised of a liquid fuel and liquid oxidizer. Higher efficiency than solid propellant engines and are capable of being throttled, shut down, and restarted. Although, they are very susceptible to the environment

c. Hybrid propellants - Comprised of a solid fuel and a liquid/gas oxidizer. The fluid oxidizer can make it possible to throttle and restart the motor just like a liquid-fueled rocket

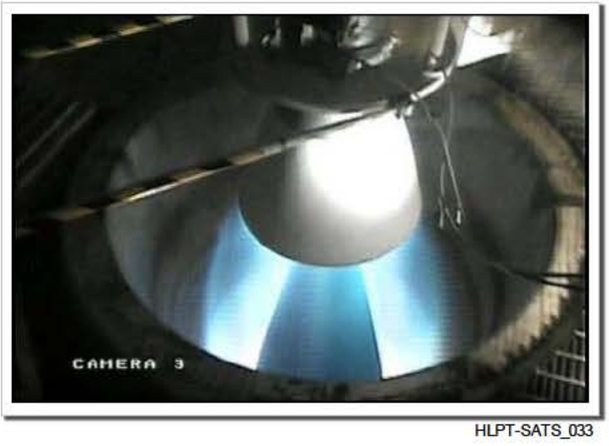
LOX/LCH$_4$ Engine

The development of an advanced liquid oxygen/liquid methane engine has been in work for a few years across many different thrust classes (3,500, 5,500, and 7,500 lbf). Multiple LOX/LCH$_4$ engine providers, such as Aerojet and ATK, have been developing first-generation flight development prototype designs and have tested and evaluated them both at sea level and in vacuum (high-area-ratio nozzle altitude). The altitude engine requires an igniter and injector propellant valves, an ablative chamber, and a columbium nozzle extension. An operational 3,500-lbf engine is anticipated to produce a Isp of 335 to 375 seconds in vacuum. With a TRL of 6 (technology demonstration), these designs could be developed further and used in an In-Space Stage or transfer vehicle.

The benefit of the LOX/LCH$_4$ engine is that the propellant is cryogenically stable and it is non-toxic. This means that the propellant can be stored for long durations in space, which would be required for an interplanetary reference mission, and can be ground processed with easier restrictions and limitations. As compared to other conventional chemical propellants; e.g., a hypergolic system, the propulsion system mass is a significantly lighter mass.

The disadvantages of the LOX/LCH$_4$ engine are due to the fuel consumption rate. The weight penalty for carrying a mission plus margin of propellant mass from Earth to stage initiation does not trade well as a figure of merit. This weight has to be carried during the launch and mission timeline without benefits to ascent. In addition, the engine wear (ablative chamber char) can severely limit the useful life of the engine.

Initial demonstration tests have shown performance levels that meet or exceed basic engine requirements for these class engines, as demonstrated during PCAD testing, as shown in Figure 6-16.



HLPT-SATS_033

Figure 6-16. NASA/PWR RS-18 engine in test cell

NTO and N$_4$H$_4$ Engine

The bipropellant Nitrogen Tetroxide (NTO) and Hydrazine (N$_4$H$_4$) engine has been used for many decades in reaction, maneuvering, and station-keeping propulsive systems. The engine class has understood performance characteristics, as the development and operational risks are well understood, but ground hazards associated with toxic propellants introduces processing risks and safety considerations. While on the ground, the propellant is stable at ambient temperatures/pressures and storable without special system designs, such as those required for cryogenics. The implementation challenge is to increase the performance at elevated temperatures in the combustion chamber. Recent testing in the Advanced Materials Bipropellant Rocket (AMBR) engine being developed with Glenn Research Center shows some promise, but the Capability Readiness Level (CRL) is very low. Processing the In-Space Stage with the bipropellant has a negative impact on the vehicle processing con-ops and adds significant cost to the ground support infrastructure in both facilities and GSE due to hazards associated with the handling of both fuel and oxidizer commodities. The weight penalty for carrying a mission plus margin of propellant mass from Earth to staging initiation is also a negative trade FOM as it will require complex solutions for thermally protective fuel storage tanks, pressurization system tanks, etc.

LOX/LH_2 Engine

The liquid oxygen/Liquid Hydrogen (LH_2) engine has been in service for decades, mainly for First Stage ascent flight. As an Upper Stage or transfer vehicle propulsion system, LOX/LH_2 has been used on the Centaur configurations and Saturn Upper Stages. The RL-10 family of engines provides a small engine thrust class propulsion, 14K lbf to 20K lbf, with respective Isp in vacuum of 360 to 450 seconds.

The benefits of the LOX/LH_2 chemical engine are the understood performance characteristics. The developmental and operational risks are understood, but the efficiency optimization required to increase performance and reduce lift cycle costs could compromise this position.

The disadvantages of the LOX/LH_2 chemical engine are that the propellant is cryogenically less stable and toxic. This impacts the vehicle processing con-ops and adds significant cost to the ground support infrastructure in both facilities and GSE. The weight penalty for carrying a mission plus margin of propellant mass from Earth to staging initiation is also a negative trade FOM.

Summary Assessment of LOX/LCH_4 Engine for In-Space Stage Propulsion

a. Propellant is cryogenically stable and it is less hazardous (nontoxic) than other fuel propellants

b. Propellant can be stored for long-durations in space, which would be required for an interplanetary reference mission and can be ground processed with easier restrictions and limitations

c. As compared to other conventional chemical propellants (e.g., hypergolic system), the propulsion system is significantly lighter mass

d. The poor fuel consumption rate results in a weight penalty for carrying a mission plus margin of propellant mass from Earth to stage initiation and does not trade well as an FOM

e. Engine wear (ablative chamber char) can severely limit the useful life of the engine

Summary Assessment of Bipropellant NTO and Hydrazine (N_4H_4) Engine for In-Space Stage Propulsion

a. Understood performance characteristics, such as the development and operational risks, are understood, but ground hazards associated with toxic propellants introduces processing risks and safety considerations

b. The propellant is stable at ambient temperatures/pressures and storable without special system designs, such as those required for cryogenics

c. The technology challenge to increase performance at elevated temperatures in the combustion chamber (AMBR engine) is being developed with Glenn Research Center

d. The negative impact on the vehicle processing con-ops and adds significant cost to the ground support infrastructure in both facilities and GSE due to hazards associated with handling of both fuel and oxidizer commodities

e. The weight penalty for carrying a mission plus margin of propellant mass from Earth to staging initiation is also a negative trade FOM (e.g., fuel storage tanks, pressurization system tanks)

Summary Assessment of LOX/LH_2 Engine for In-Space Stage Propulsion

a. Understood performance characteristics, such as the development and operational risks, are understood, but the efficiency optimization required to increase performance and reduce lift cycle costs compromises the advantages

b. The propellant is cryogenically less stable and more hazardous (toxic) than other fuel propellants

c. The technology challenge is to mitigate propellant management (e.g., boil-off)

d. There is a negative impact on vehicle processing con-ops that adds significant cost to the ground support infrastructure in both facilities and GSE

e. The weight penalty for carrying a mission plus margin of propellant mass from Earth to staging initiation is also a negative trade FOM

In-situ resource engines are defined as engines that use either in-situ available energy (e.g., solar) and/or high-density or long-life energy (e.g., nuclear) as the engine power. Various electronic propulsion systems use electricity to ionize the atoms of a fuel propellant, creating a gradient to accelerate the ions to high exhaust velocities. Electronic propulsion systems, better known as ion thrusters, produce either electrostatic or electromagnetic forces to accelerate the propellant reaction mass (stream of ions). Propulsion efficiency and thrust are all inversely proportional to exhaust velocity, requiring large amounts of energy resulting in lower thrust, but fuel usage rates are extremely low. With current technology of electrical power, chemical, nuclear or solar, the maximum amount of power that can be generated limits the amount of thrust that can be produced to a small value. Power generation adds significant mass to the spacecraft, limiting the vehicle performance, so balancing Isp-Thrust profiles depending on mission requirements in a variable manner is desired. For ion thruster engines, there are two different categories of ion thrusters: electrostatic ion thrusters and electromagnetic thrusters.

Types of electrostatic ion thrusters include the following:

a. Gridded electrostatic ion thrusters

b. Hall effect thrusters

c. Field Emission Electric Propulsion (FEEP)

Types of electromagnetic thrusters include the following:

a. Pulsed Inductive Thrusters (PIT)

b. Magnetoplasmadynamic (MPD)/Lithium Lorentz Force Accelerator (LiLFA)

c. Electrodeless Plasma thrusters

d. Electrothermal thrusters

e. Helicon Double Layer thruster

Ion Thruster Engine

An ion thruster is a form of electric propulsion that creates thrust by accelerating ions. There are two different category types of ion thrusters, electrostatic and electromagnetic. Electrostatic ion thrusters accelerate the ions in the direction of the electric field. Electromagnetic ion thrusters use a magnetic force to accelerate the ions.

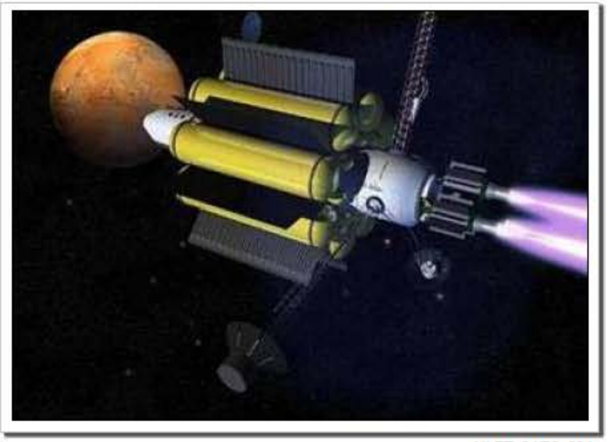
A plasma propulsion engine is a type of ion thruster that uses plasma in some or all parts of the thrust generation process. Though far less powerful than conventional chemical engines, plasma engines can operate at higher efficiencies and for longer periods of time. Plasma engines are well suited for long-distance interplanetary space missions.

The benefit of the ion thruster is that it produces a very high Isp (i.e., propellant efficiency). The high propellant efficiency is acquired through efficient propellant consumption management of the ion thruster propulsion system. The efficiency is obtained by high exhaust velocity. Power for the operation can be acquired by in-situ resources via solar arrays.

The disadvantages of the ion thruster are that the thrust created is very small compared to conventional chemical rockets. Ion thrusters consume large amounts of power, thus performance is ultimately limited by the available spacecraft power. When sizing space-qualified power sources, the accelerations given by these types of thrusters are of order 0.03 to 0.05 ft/sec^2.

The low thrust requires ion thrusters to provide continuous thrust for a very long time to achieve the needed change in velocity (delta-V) for a particular mission. If continuous operation is not feasible due to available power, burst firing of the thruster engine is an operational consideration. To achieve these delta-Vs, ion thrusters can be designed to last for periods of years. Overuse of the ion thruster can lead to propulsion plate or discharge chamber failure. Use of in-situ solar resources provides sufficient energy for short bursts, but a higher energy density is required for continuous operation.

One example of the electromagnetic ion thruster is the VASIMR engine. The concept of heating hydrogen or helium into plasma was a biproduct of the research accomplished on nuclear fusion. The VASIMR is intended to bridge the gap between high-thrust, low-Isp propulsion systems and low-thrust, high-Isp systems. VASIMR is capable of functioning in either mode. Using helium or hydrogen in lieu of argon would remove toxic material from the vehicle, thus improving cost affordability. As part of the Progressive Architecture, an In-Space Stage using a VASIMR engine provides the technology insertion to improve operability cost as shown in Figure 6-17.



HLPT-SATS_034

Figure 6-17. In-Space Stage with VASIMR propulsion system

In summary, ion thrusters are not feasible for launching spacecraft into orbit, but they are practical for In-Space propulsion applications, like maneuvering and positioning. Ion thrusters have many applications for In-Space propulsion. The best applications of the thrusters make use of the long lifetime when significant thrust is not needed. This includes but is not limited to orbit transfers, attitude adjustments, drag compensation for LEOs, and fine adjustments for more scientific missions. Ion thrusters are ideal for interplanetary and deep-space missions in which time is not crucial. The continuous thrust over a very long time can build up a larger velocity than traditional chemical rockets.

Nuclear Thermal-Electric Engine

In a nuclear electric engine, nuclear thermal energy is changed into electrical energy that is used to power one of the electrical propulsion technologies. The nuclear powerplant provides the energy requirements, converting heat to electricity, for the propulsion system. Numerous different reactor approaches have been used that which have demonstrated to be inherently safe. There are two different nuclear (fission reactor) thermal-electric powerplants that should be considered: the Stirling Convertor, which is a more efficient system and the Brayton-Cycle Engine, which is a more reliable system.

Choosing the style of reactor that provides high power without excessive mass for shielding to contain the high pressures should be a critical FOM. Early research in nuclear propulsion began with studies for nuclear thermal propulsion. NASA's Nuclear Engine for Rocket Vehicle Application (NERVA) project of the 1950's and 60's demonstrated a working design concept. The demonstrated performance of the engine was in the 190K lbf thrust class and Isp of 825 seconds in vacuum.

Harnessing mini nuclear powerplants with electric propulsion units, such as an ion-drive motor, is the next technology step. In the past several decades the attention has turned to using the nuclear reactor to drive a turbine to produce electricity, which is used to create plasma that is accelerated. An area that nuclear-powered system technology has developed has been under the Project Prometheus effort started in 2003 by NASA for long-duration space missions. This technology provides a 400-kW thermal reactor and a gas turbine to produce electric power.

The benefit of the nuclear thermal-electric system is that the overall gross lift-off mass of a nuclear rocket is about half that of a chemical rocket. When used as an Upper Stage, it approximately doubles or triples the payload capacity that can be carried to orbit. With the higher energy density of nuclear fuel as compared to chemical fuels, approximately 10^7 times, the resulting propellant efficiency (effective exhaust velocity) of the engine is at least twice as good as chemical engines.

The disadvantages of the nuclear thermal-electric system is the safety aspect of crew health and radiation exposure if a breach of containment occurs. Separation of crew and powerplant may not be feasible based on the vehicle design.

Summary Assessment of Ion Thruster Engine for In-Space Stage Propulsion

a. Produces very high Isp (i.e., propellant efficiency)

b. High propellant efficiency is acquired through efficient propellant consumption management of the ion thruster propulsion system by high exhaust velocity

c. Supplementary power for the operation can be acquired by in-situ resources via solar arrays

d. Thrust created is by very small acceleration (0.03 to 0.05 ft/sec^2) as compared to conventional chemical rockets. Trading Isp for thrust in variable operation is desired

e. Ion thrusters consume large amounts of power, thus performance is ultimately limited by the available spacecraft power unless supplemented with other available power sources

Summary Assessment of the Nuclear Thermal-Electric Powerplant

a. Nuclear thermal energy is changed into electrical energy that is used to power one of the electrical propulsion technologies

b. The overall gross lift-off mass of a nuclear rocket is about half that of a chemical rocket

c. With higher energy density of nuclear fuel as compared to chemical fuels, approximately 10^7 times, the resulting propellant efficiency (effective exhaust velocity) of the engine is at least twice as good as chemical engines

d. There is increased concern over the safety aspect of crew health and radiation exposure if a breach of containment occurs, where separation of crew and powerplant may not be feasible based on the vehicle design

6.3.1 In-Space Stage Impacts and Considerations on Ground Systems and Processing

Ground operations considerations for chemical propellant loading are as follows:

a. Ground systems to support fuel storage and delivery, potentially with remote monitoring and safing capability (system interfaces)

b. Pressurant process - Pressurant commodity storage and delivery ground systems, including monitoring

c. If there are hazardous commodities; hazard monitoring systems are necessary

d. Hazardous commodities typically require associated purge systems, storage, delivery, monitoring, and venting capability

e. Fuel loading and pressurization timing within the process flow may drive specialized access for personnel and equipment (e.g., at the pad)

f. Fuel selection may result in commodity “standby time” constraints, impacting contingency consecutive launch attempt capability

g. Commodity loading and pressurization system configuration could impact contingency consecutive launch attempt capability for top-off/repress (remotely operated T - 0 versus manual servicing (e.g., cart))

h. Contingency deservice capability, equipment, process monitoring

Ground Operations considerations on monitoring, potentially with launch count no-go criteria, include the following:

a. Manual (console operator) - Sensors, hardwire safing

b. Automated ground launch system - Sensors, SW control logic

Chemical systems that are cryogenic typically require structural TPS and have the following characteristics:

a. Susceptible to in-process collateral damage

b. Susceptible to environmental damage (hail, birds, etc.)

c. Repair processes typically involve stringent process controls, specialized application and inspection equipment, environmental controls during application (temperature, humidity, dew point), and extended cure times

d. Access to full acreage for repair is rarely convenient

Ground operations considerations with hypergolic systems include the following:

a. Hypergolic systems are always an issue on payloads due to extreme toxicity and corrosivity

b. Hypergolic systems require special material permitting, storage, and handling

c. Hypergolic systems require haz-mat containment/cleanup equipment

d. Hypergolic systems require specialized PPE and process controls; Self-Contained Atmosphere Protective Ensemble (SCAPE) operations are required

e. The long-term effect is elevated ground systems maintenance; QDs, seals, system decontamination, contaminated component processing procedures and facilities

Ground operations considerations on solid propellant systems include the following:

a. Solid propellants have unique hazards when handling the stage

b. Solid propellants require ordnance permitting, procedures, and hazard controls for storage and handling

c. Quantity/distance restrictions may impact processing facility constraints and operational uses

d. Open propellant grain operations, if any, involve extended safety hazard area clears and operational restrictions

Ground operations considerations with nuclear power/electric/ion systems include the following:

a. Historically (payloads), nuclear material arrives to the processing site fully encapsulated, imposing no special ground processing restrictions

b. Based on the form and quantity of the material involved, special hazard controls may apply for permitting, storage, handling, monitoring, etc., as these special requirements could drive the need and use of specialized ground processing equipment and monitoring equipment

c. The state of the system during processing; i.e., active versus passive, can impact the nature and extent of processing controls and support systems involved

"Critical" systems, such as the In-Space Stage, typically carry unique hazard control and elevated levels of redundancy/certification requirements and associated verifications. It is accepted that the addition of any system will result in increased processing, integration, testing, and verification regardless of the nature of the system. The specific type and implementation of the selected system can significantly affect the extent of the impact. All systems require some amount of attention, but a system that does not require any additional ground system infrastructure or processing will avoid costs and will not add schedule risk at KSC during launch processing.

6.3.2 In-Space Stage Impacts and Considerations on Mission Design and Flight Operations

Mission design groundrules and constraints, along with Flight Rules, should address whether loss of the high-Isp system results in LOC or loss of primary mission. This mission operations concept needs to be considered early as these constraints and rules drive the architecture design as failure scenarios of either propulsion system by mission phase is defined. In-Situ Resource Utilization (ISRU) propulsion generation can reduce propellant carried outbound but adds more failure modes. If ISRU is to be considered, the trade needs to address groundrules on how much propellant must be generated prior to committing to a crewed mission to a Beyond Earth destination. The institutional knowledge base for In-Space use of LOX/LH_2 propellant is much larger than LOX/CH_4, NTO/N_4H_4, or ion thrusters. Additional operational procedures and training will be required as there is an expectation that a "learning curve" on the operational phase would be realized.

6.3.3 In-Space Stage Trade Results

From a con-ops, once EDS is discarded, operational constraints on crew exposure limits, crew extended-duration proficiency, and consumable limitations drive the trade space. Crew and consumable limitations drive your answer in the trade space, and In-Space duration is the driving constraint to the selection. The development path of an In-Space Stage propulsion system should segregate out the propulsion system and energy/power system. In-Space Stage systems with chemical propellant require significantly larger operational processing and maintenance and introduction of new hazards, along with consumable limitations while in transit; therefore, harnessing innovative powerplants with electric propulsion units, such as an ion-drive motor, is the next technology step. The SLS Program will need to overcome mission needs of periodic high thrust versus periodic high Isp during the mission duty cycle. An in-situ or regenerating power system with very little active consumption of reactant mass provides the best trade solution for deep-space powered flight. A single energy source, such as the nuclear-powered system technology developed under the Project Prometheus effort for long-duration space missions, is best. That system provided a 400-kW thermal reactor and a gas turbine to produce electric power, but may be not feasible on the NASA OCT technology roadmap. Coupling multiple power systems together to control the energy more effectively or the combination of power sources, nuclear and solar (e.g., FAST array), trade well on both TRL and CRL metrics to achieve the operational readiness timeline. Power needs for variable systems (PIT or MPD) depend on the mission profile. Approximately 400 kW for missions not requiring expedited transit times and over 1 MW for crewed missions where transit time needs to be shortest are driving constraints on the In-Space Stage propulsion down select, thus development of the power generation, storage, and distribution is as important as the propulsion system itself.

6.4 System

Starting with the QFD HOQ #2 results, Architecture Characteristics and Attributes Requirements, USA conducted a QFD to develop the lower-level system requirements for QFD HOQ #2A. The system-level attributes and characteristics results were prioritized based on the scores, and the desired design feature and control feature enablers were determined. These characteristics and attributes are satisfied by either design features and/or control features. In order of priority importance, Table 6-9 depicts the enablers.

Table 6-9. QFD HOQ #2A system-level design and control enablers

System-level attributes and characteristics	Score	Design feature enabler	Control feature enabler
Downselect system designs without crit 1 failure modes	4.07	Design with one level of redundancy (min). No crit 1 in avionics/controls. Redundancy in flight software. Pressurant systems have one level redundancy (min) for function	Realistic probability of failure inclusive in crit 1 definition. FMEA should be used as a design tool (and not an afterthought book-keeping document)
Operational efficiency	3.88	Design in "testability" and maintenance. Keep accessible components (e.g., fuses) outside of avionics and electronic HW (STS-134 ALCA problem). Commonality of systems, components, GSE, tooling, required critical skills. Minimal "manufacture" at the launch site (ship and shoot). Minimize hazardous operations. Automation in system performance verifications, system/vehicle health monitoring	Establish human factors requirements for the design with specific consideration to access, assembly, and integration. Design standards should be used. Forced commonality between Contractors and between vehicle elements
Designed-in/built-in evolution capability	3.56	Standardized parts and interfaces. Design-in margin that supports upgrade. Design and certify hardware with end state in mind. Qualify parts enveloping environment (e.g., vibration). Certify subsystems with other subsystems in mind (modularity to avoid "domino affect" of changes)	Standardized component test requirements
Mission configurable performance (thrust) profile	2.91	Engines should have multiple or defined ranges of throttle settings (limit to keep certification costs down). Design-in additional booster (or booster segment) attach points. Design in variability to avoid Max Q violations. Main thrust structure designed to accommodate different combinations/number of engines. Gimbal configuration and capability	Define vehicle trajectory profile (aborts, control Max Q, thermal issues)
Configuration tailorable (up or down capability)	2.72	Standardized parts and interfaces. Supply chain support. Design and certify hardware with end state in mind. Launch facility able to accommodate up/down capability configuration changes. Modular approach expands capabilities by adding more of the same, as opposed to redesigning to increase capabilities	Define vehicle end state early in program. Build-in spare/expansion/margin capability

🟦 **Affordability VOC**
🟨 **Performance VOC**
🟫 **Reliability VOC**
🟩 **Schedule VOC**
⬜ **Operability VOC**

Table 6-9. QFD HOQ #2A system-level design and control enablers (continued)

System-level attributes and characteristics	Score	Design feature enabler	Control feature enabler
Initial high performance (>70 mT)	2.67	Selection of Core Stage propulsion	Define engine cluster with sufficient thrust to achieve lift of pad capability
Development testing	2.66	Common vendors, common test facilities, common pass/fail criteria	Realistic definitions of qual test requirements (nonreusable vehicle)
Certify by design (not by process)	2.57	Use designs that can be verified by the OEM prior to end item delivery	Establish design criteria to limit nonverifiable features
Redundancy	2.53	Size redundancy to support LOM/LOCV. Redundancy should be weighed against reliability. Test for redundancy is reliability versus consequence of failure/time to react to failure	Define requirements/standards for minimum levels of redundancy
Multiapplication/with purpose support equipment	2.33	Common vendors, establish standards for interfaces, software commonality	Design standards should be used. Forced commonality between Contractors and between vehicle elements
Number of critical processes	2.33	Make critical process definition be per engineering and not per contract or purchase order	Maintain realistic control on design tolerances. Definition of critical process. Critical processes should be controlled by the application of the end item and defined by engineering callout
Failure modes can be inspected or tested at delivered-item level (no repeating of inspection or testing already accomplished at the OEM)	2.30	Have built-in test capability. Include health monitoring systems. Design the vehicle/systems to include necessary test points/inspection access	Properly flow requirements from OEM to launch facility. Define what really needs to be retested due to transport/handling/installation or criticality. Identify appropriate level to perform test based on level of risk
High performance (>100 mT)	2.10	Selection of Core Stage propulsion	Define engine cluster with sufficient thrust to achieve lift of pad capability
Fail ops protects crew return capability (LOC)	2.01	Increase intact abort capability. Minimize catastrophic failure scenarios	Vehicle design/performance requirement

🟦 **Affordability VOC**
🟨 **Performance VOC**
🟫 **Reliability VOC**
🟩 **Schedule VOC**
⬜ **Operability VOC**

Table 6-9. QFD HOQ #2A system-level design and control enablers (concluded)

System-level attributes and characteristics	Score	Design feature enabler	Control feature enabler
Vertical processing cycle efficiency	1.98	Design in "testability" and maintenance. Commonality of systems, components, GSE, tooling, required critical skills. Stack and shoot design. Minimize serial operations. Eliminate repetitive testing for component and systems. Minimize hardwire interfaces using command buses. Do not carry commodities (especially hazardous) across major elements	Vehicle design/maintenance requirement
Requirements baseline	1.98	N/A	Establish a good set of requirements (no "TBDs"). Define targets and let the design process refine the criteria. Never go into CDR with TBDs
Simplify system designs	1.97	Keep it simple and robust	Have design requirements that satisfy the needs. Simple integration plan collaborated across the vehicle. Consistent requirements/procedures/policies allow for a simple design
Use common parts, components, and/or assemblies	1.94	Drive commonality of piece parts within each element and across elements	Forced commonality between Contractors and between vehicle elements. Vehicle-level system responsibility that spans elements. Vendors use common specs
High design and process margin	1.93	Develop good models supporting hardware design margins. Validate models by test	Design for high margins/factors of safety. Establish a minimum margin target over the design factor of safety

🟦 **Affordability VOC**
🟨 **Performance VOC**
🟫 **Reliability VOC**
🟩 **Schedule VOC**
⬜ **Operability VOC**

Launch vehicle risk and subsequent failures are normally associated to the complexity of the system and, therefore, related to the affordability of the system. Based on 27 years of vehicle failures, as shown in Figure 6-18, propulsion, GNC, and avionics were further evaluated.

Based on perceived launch vehicle risk and historic failures, which is an indicator of cost, key focus areas are as follows:

Risk-informed concentration

a. Propulsion

b. GNC

c. Avionics

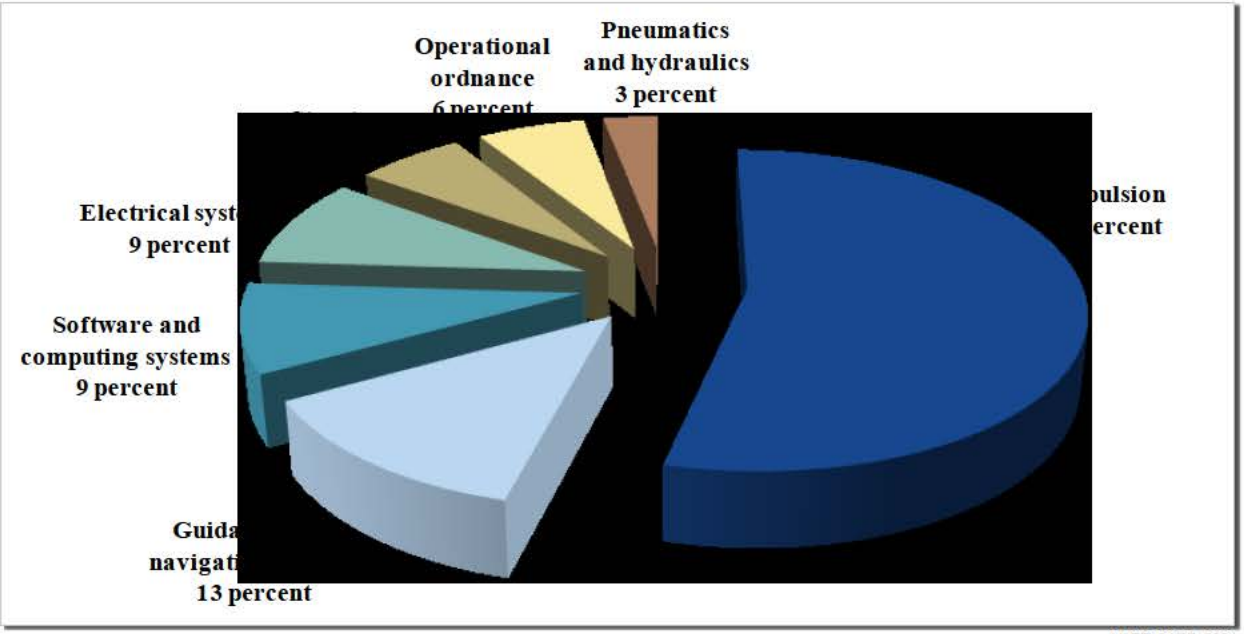
d. Software

e. Electrical

f. Crew systems

g. Separation systems

Based on this list, the top three were further evaluated by OEM specialists in their respective fields. Six specialist companies in their respective fields participated in USA's system-level trade. Feedback was integrated and consolidated into common themes by USA. On propulsion systems, Aerojet and



HLPT-SATS_030

Figure 6-18. 1980 to 2007 worldwide launch failure causes

Ref.: FAA Launch Vehicle Failure Mode Database, May 2007

Pratt Whitney Rocketdyne provided performance, schedule, and cost data. On the GNC system, Hamilton Sundstrand and Moog provided TVC concepts with performance and schedule data. On the Avionics system, L-3 Communications and Honeywell Space Systems provided Military-Off-the-Shelf (MOTS) and Commercial-Off-the-Shelf (COTS) thoughts, respectively, along with L-3 providing an Ares I First Stage assessment.

Propulsion System Trade Results

Industrial base sustainment is adversely affecting affordability. The anemic U.S. launch rates and U.S. propulsion demand is low. With the expendable market collapse in the late 90's, the industrial base sustainment of competitive segment markets eroded. The scarcity of new launch vehicle propulsion development programs in the Commercial market, DoD, or NASA is not developing the skills within the Industry. The lack of U.S. "staying power" to see new programs through and the loss of propulsion capability due to the aging aerospace work force is diminishing the U.S. Industrial base.. The U.S. market share is being lost to foreign competition, where foreign markets are effectively closed to the U.S. Propulsion Industry because of ITAR restrictions. The reliance on foreign suppliers of raw materials (e.g., carbon fiber) and products (e.g., RD-180) are also hampering the development and production environments.

In assessing the propulsion system and its interface to the vehicle, there are benefits in schedule and cost in an integrated propulsion system in terms of an "engine cluster" that would enable the Industry base to be more stable with a bigger base, improve accountability, reduce overlap with the tank providers, and would allow tailorability for mission-specific requirements easier. A propulsion system in an engine cluster concept, which would include the engines, MPS, primary/secondary structure, and TVC as a delivered end item for higher order integration, moving the propulsion system interface to tank, would allow for a more cost-effective design and operations solution. As an example, acceptance test verification, known as green run testing, at Stennis Space Center (SSC) could be accomplished with just the engine cluster, not requiring a large transportation and handling costs of the Core Stage tank.

GNC and Avionics Systems Trade Results

In evaluating the GNC and avionics systems, it was discovered that there are many parallels between GNC and avionics, including common themes and concerns. At a system level, there are four significant Affordability FOMs, as shown in Table 6-10. To achieve affordability across the SLS platform, the use of systems, subsystems, or components in multiple applications versus in a unique single application takes advantage of the market segment to benefit the SLS functional requirements. A single acquisition for avionics that would be used on Core Stage, boosters, if applicable, Second/Upper Stage, and maybe

Table 6-10. System current state versus recommendations to improve affordability

System recommendations	Current system state
Platform solutions for multiple applications (commonality within the SLS program and with other programs)	NASA requests off-the-shelf components, but OEM effort to work with supply chain to align the system requirements and architectures is seldom executable
• Orion CEV/MPCV is great example	
Achieve early risk reduction with fast-paced demonstrations	Rigorous development activities burdened by continuously evolving flight requirements
• Tailored rules of engagement	
EEE parts depot for the program	Each supplier acts independently and is burdened with lot charges and low quantity pricing
• Possibly a small business contract	
Increase redundancy to reduce the need for grade 1 EEE parts and extensive verification testing at system and component level	Single-fault-tolerant system with grade 1 parts (unique to NASA) and extensive human-rated verification
• Consider two-fault-tolerance as a trade with reduced verification • Use more Industry-standard practices (i.e., J-STD with space addendum)	

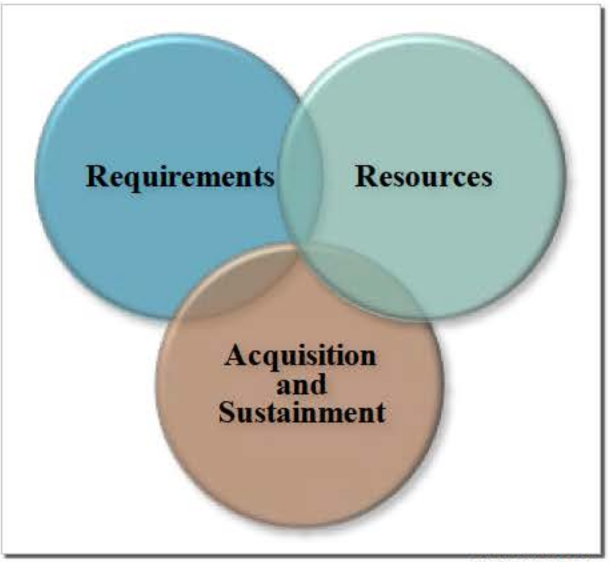
even In-Space Stage, would maximize the production rate and allow cost avoidance and reduction in manufacturing, assembly, and testing. Institute fast-paced demonstrations to gain confidence and reduce the assessed technical risk. Deploy a common Electrical, Electronic, and Electromechanical (EEE) parts procurement pool where a buying agent can leverage multiple design requirements into a single buy. Change the implementation approach on redundancy and align with Industry-standard practices for aerospace.

NASA and the design authority should champion the development of systems that promote commonality across and within stages and that are extensible to other programs. Where possible, modular architectures and platforms provide savings in ease of parts application and integration of those parts and distribution by increasing common parts usage and reducing assembly risk. The Aircraft Industry is currently switching over to electric actuation systems, replacing the need for hydraulic and pneumatic systems, which, in turn, provides weight savings, reduced processing and operating costs, and eliminates high-pressure hazards. Leverage the complete aerospace market segment diversity to achieve the Affordability FOM.

Only implement new technologies to reduce LCCs. Do not spread new technology over the vehicle, as technology insertion should be used only if it demonstrates a positive return on investment (ROI) over the life cycle. System LCC is only a component; consideration of integration and infrastructure costs need to be included in trade space.

6.5 Implementation

While the technical side of the acquisition equation is a significant contributor to affordability, how you implement the program will also contribute to the LCC in both developmental and operational phases. Acquisition and sustainment must stay linked to requirements and resources to ensure an affordable and sustainable program over the life cycle as depicted in Figure 6-19.



HLPT-SATS_031

Figure 6-19. Continued relationship of requirements, resources, and acquisition and sustainment over the life cycle is the key to affordability

When developing the requirements, attributes, characteristics, and resultant FOMs, a gap analysis was conducted to ensure that the team captured all FOMs that drove affordability and schedule. It was very apparent that implementation approaches and associated metrics that were not dependent on launch vehicle configurations or propulsion systems were not being captured in the trade space. Five implementation FOM categories, along with their associated approaches, were developed and defined, an AHP assessment was conducted to get the pairs-wise comparison priority importance, and the following relative importance was generated as shown in Table 6-11 and Table 6-12.

Table 6-11. Implementation FOM importance

FOM	Implementation category	Importance
1	Requirements and standards	37 percent
2	Program effectiveness	34 percent
3	Design and integration	15 percent
4	Contracting	9 percent
5	Business systems	5 percent

Table 6-12. Implementation FOM categories and approaches

Category	Approaches
Requirements and standards	• Initialization of a zero-based Requirements set - Question all requirements • Provide direct association and flow down of requirements - Avoid blanket requirements • Agile change control - Balance flexibility with minimizing late requirements • Use simplified Earned Value Management (EVM) at the subcontractor level • Common specifications and standards across program elements
Program effectiveness	• Program-level taxonomy • Common Prime Contractor PPMs - Eliminate or minimize functional redundancy across program activities • Collaborative teams (blended workforce) - Clearly defined, lowest possible level RAA • Use smaller, dedicated DDT&E teams with enhanced RAA and signature authority • Ensure the proper level and frequency of reporting
Design and Integration	• Integration of the integrators - Program, system, vehicle, and technology need to be balanced • Performance margin management schemes • Simplified integration with less labor • Follow a DFO maturity model - Con-ops drives the evolving design • Maximize use of modeling and simulation in lieu of physical hardware • Simplify the CoFR responsibility and accountability (both process and product) • Commonality of NASA parts and materials across program elements
Contracting	• Apply risk-based contracting with incentives • Alliance project (alliance contractual solution where KPIs are aligned) - Clearly define functional roles and responsibilities • Reduced number of prime OEM Contractors, make/buy, and GFE versus prime trades • Contractual delivery of end-item in "ready-to-install, -assemble, or -integrate" configuration • Eliminate duplication of engineering and M&P groups between OEM and operator
Business systems	• Common element to element business systems - Interoperability at data level and standardization of toolset interfaces • Support systematic integration of tools and data across and between systems with integrated PLM (vendor-to-supplier-to-provider-to- user) - Shared, single authoritative source • Integrated configuration management and requirements control

The Implementation FOMs affect the program "-ilities," specifically how the "-ilities" are interpreted and implemented, which, as a result, influences affordability. The "-ilities" are defined as follows:

a. Availability

b. Commonality

c. Maintainability

d. Operability

e. Producibility

f. Reliability

g. Reusability

h. Testability

i. Upgradeability

j. Usability

6.5.1 Requirements and Standards FOM Results

Based on years of SSP experience and Cx lessons, it was determined that (a) initialize a zero-based requirements approach to eliminate nonnecessary requirements by starting with basic functionality requirements that add value, (b) avoid legacy requirements development process by capturing only mandatory requirements by avoiding blanket requirements, (c) all requirements shall have clearly explanatory intent and rationale so that the designer can implement it by capturing design assumptions, technology limitations, real or presumed constraints, etc., (d) avoid "To Be Determined (TBD)" callouts by starting with quantified requirements (i.e., substantiated target) with accompanied "To be Revised (TBR)" callout to allow designers to start with known metric, (e) when using heritage hardware, when applying zero-based requirements approach in a "like but different" type function or limits, understanding the certification threshold(s) and basis of the certification will allow cleaner definition, and (f) accept and allow for equivalency assessment for heritage hardware.

Zero-Based Requirements Example

When requirements are done poorly, the issue adversely affects Affordability, Sustainability, and Operability FOMs. Definition and management of requirements forms the basis of any program. Inadequate definition, or ineffective management of the requirements baseline will result in poor performance against cost, schedule and technical objectives. As an example, the CxP requirements definition and management was assessed. In an effort to capture and leverage critical lessons learned from SSP, innumerable-system, and operational-level requirements were elevated to Level II, resulting in duplication, gaps, and conflicting criteria between requirements. Cradle was identified as the requirement management tool for CxP. However, organizations across the agency and program produced an extensive library of requirements documents outside of the tool, providing little to no association or traceability between requirement levels. The resulting requirement set was difficult to assess for applicability, labor intensive to change manage, and would have been difficult and expensive to fully verify. Numerous requirements existed at every level with no documented intent/rationale.

The solution(s) or recommendation(s) are to (a) begin requirement definition based on current mission objectives (legacy lessons learned should be leveraged to reduce risk and not allowed to impose undue constraints), (b) enforce fundamental SE&I requirement flow-down with full bidirectional traceability (top-level requirements define "what" with successive levels of detail to define "how"), and (c) each requirement *must* include a documented intent/rationale, ultimately, with associated linkage to substantiating basis of certification/acceptance; e.g., assumptions, analysis, test results, performance data, etc.

Summary: Sometimes good intentions have bad consequences, as SSP lessons learned were used to assist in the development of the CxP requirements. This resulted in duplication, gaps, and conflicting criteria between requirements. Fundamental SE&I methodologies would have reduced these weaknesses.

Another area of requirements and standards worth mentioning is requirement ownership and management should be consistent with program-defined Responsibility, Accountability, and Authority (RAA). Each requirement should have an assigned owner. Provide direct association and allocation/flowdown of requirements by (a) starting with G&O for the SLS program, (b) flowdown and decomposition of intent and rationale will provide operators a better understanding when deviations or departures from the certification baseline is required during the operational life cycle, and (c) keep requirements quantified to a lower level along with requirement intent to avoid requirement intent creep.

6.5.2 Program Effectiveness FOM Results

One of the largest challenges in the Aerospace Industry is data management. Using a simple technique of program-level taxonomy would solve the handling costs associated with large data sets. It is not about drawings and documents, but, rather, it is about data. If you cannot integrate the data, you cannot manage it.

It was determined that if you establish connectivity of requirements to design to manufacturing to assemble and integrate to operation in a data-driven taxonomy, the process will minimize program

Integration overhead by reducing the number of handoffs, translations, replications, and interpretations. In addition, it avoids large OEM launch site support that wants to ensure that requirements of their hardware are being properly implemented as intended, as all data, including as-built data, are available Program wide.

Deploying collaborative teams that use smaller, dedicated DDT&E teams facilitates better optimized design solutions. Establish a NASA and Contractor "blended" team that use the best of expertise from the competencies and proficiencies within both parties. Clearly defined, lowest-possible-level RAA at the Integrated Product Team (IPT) that carries signature authority as decisionmaking should reside at where the requirement is owned, IPTs should be responsible for products and not reviews, with accountability for schedule and budget, and empower to the lowest level of decisionmaking. Encourage decisionmaking.

Program-Level Taxonomy Example

When not implemented by a program office, the resultant issue affects Affordability, Commonality, Operability, Usability, and Availability FOMs. Lack of a program-level taxonomy, hierarchical structure, and number schema allows for introduction and use of multiple, unassociated numbering systems and information formats, significantly impeding effective and efficient information and data exchange, driving manual translation and verification processes and minimizing the effectiveness of automated business systems. As an example, Constellation's Ares I-X lacked an integrated program-level requirement structure or standards definitions. All OEMs provided element/systems hardware designs in their own format, structure and numbering schema. All vehicle designs were provided in "drawing" format, either as paper renderings or electronic Portable Document Format (PDF) documents, requiring manual review, interpretation and entry of associated data into operations center business systems to support processing, and Certification of Flight Readiness (CoFR) verification. Inconsistencies in design content and structure necessitated extensive labor hours to reconcile and approve prior to implementation. This process involved operations center, OEM, design center and level-II board personnel for hundreds of items ranging from verification of controlling installation drawing to part callouts, source, and pedigree requirements.

The solution(s) or recommendation(s) are to (a) establish and implement a program-level taxonomy for all elements, both flight and ground, early in the program, applying it contractually to all participants to address the full supply chain and life cycle, to include evolution and (b) implement the structure in conformance with Industry or DoD standards for electronic data exchange to maximize leveraging best practices inherent in current COTS business systems and IT.

Another area of program effectiveness is the contractual issuance of the NASA Insight and Oversight models. The NASA governance model is an Insight/Oversight driver. The ability to tailor NPR 7120, 7123, and 8705.2 will reduce SLS Program cost, similar to Commercial crew tailorability approach. Clearly identifying and separating the Insight and Oversight Teams' RAAs will avoid both NASA and Contractor overlap. On insight, embed the MSFC SLS Insight Team within the Contractor Team to provide constant communication of understanding and recommendations to the Oversight Team. On oversight, modify the current, near-continuous practice to a discrete practice, where oversight decisions and direction are substantially less and more focused based on risk. This technique is pictorially shown in Figure 6-20.

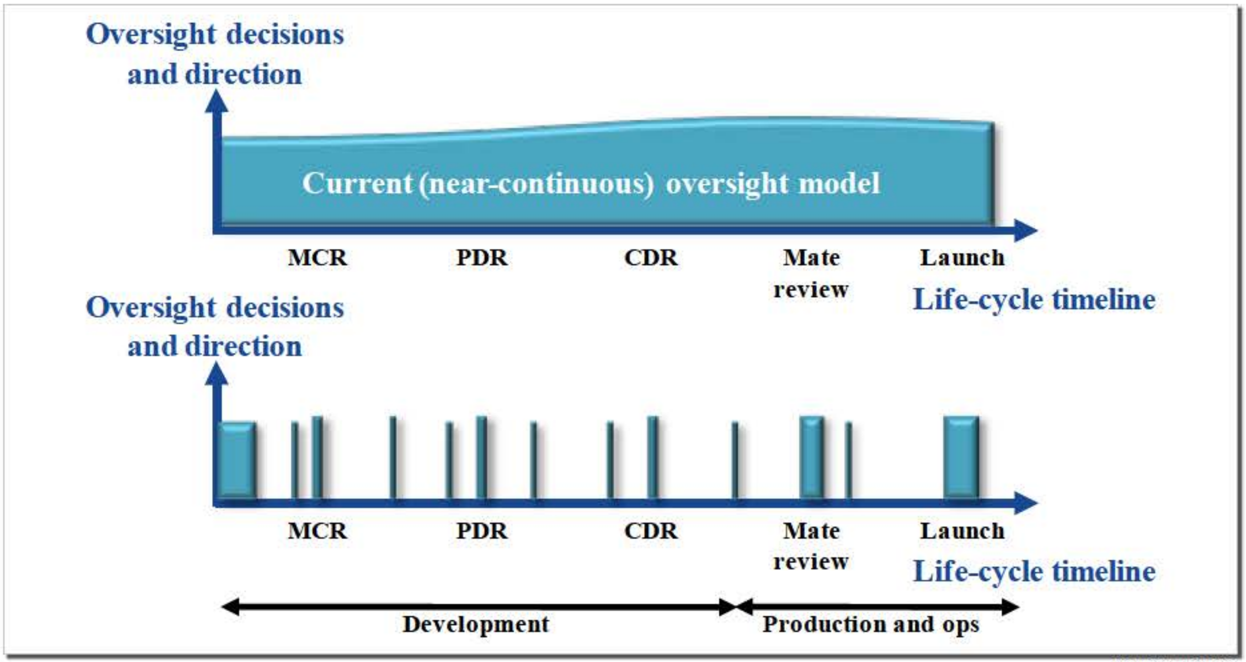


Figure 6-20. Current oversight versus proposed Oversight models

6.5.3 Design and Integration FOM Results

The lack of integration within a program can significantly impair a team's efficiency and cohesiveness. An integration of the integrators allows for a sufficient amount of SE&I planning and baseline framework to be established prior to release of requirements to OEM element providers and enables standardization and commonality across integrators and elements

SE&I is an enabler for affordability. The SE&I requirements analysis process is a key contributor to defining and optimizing affordability, as the Design To Cost (DTC) and Cost As an Independent Variable (CAIV) processes play a pivotal role. Based on lessons learned from SSP, changes that were planned for Ares I, and changes that were partially demonstrated on Ares I-X, instituting a sound SE&I foundation (e.g., requirements control) will provide cost avoidance opportunities (and you cannot afford not doing it) to achieve the FY budget targets.

As detailed in Section 1.2, the DFO Maturity Model will assist with the interoperability of systems across elements, as this evaluation should be accomplished early, not after each piece has designed its own and the corrections become too expensive to implement. The concept of operations and requirements relationship must be established to ensure that DFO is carried through the design cycle. Consideration of Operability, Maintainability, Testability and Usability FOMs when developing vehicle- and system-level requirements will reinforce the DFO Maturity Model. When trading the top-level FOMs, carry performance margin for the purpose of trading for increased operability.

When assessing the design certification, simplifying the CoFR responsibility and accountability in a CoFR streamlining initiative will eliminate CoFR overlap of sustainment and certification between design and operations centers. As an example, the design center and its OEM Contractor(s) are responsible and accountable for the design and certification thereof, to include operational departures from certification baseline. The operations center and its processing Contractor(s) are responsible and accountable for all of their processes and the proper implementation of the defined requirements as provided by the design center's certification.

DFO Maturity Model Example

Operational considerations in the design are proportional to operations costs. Specific Shuttle examples are factually to the point. When operations are not considered early in the RAC, the issue affects Operability, Reusability, and Availability FOMs. Design decisions made early in a program's development can significantly affect the operational efficiency of the delivered vehicle, and thus cost over the life cycle. The number, complexity, and evasiveness of the operations causes a proportional increase in labor and physical resources that has a direct affect on the recurring cost of the spacecraft and launch systems. There are many examples from the SSP where operability considerations were either not made or original assumptions differed significantly from actual implementation. The cost of a reusable, complex, multifunctional, aging Space Shuttle vehicle are additionally challenged by the following initial program decisions:

a. The Space Transportation System (STS) was optimized at the subsystem level and not at the integrated vehicle level

b. Every Orbiter function, whether used or not on a given mission, must be verified and checked out prior to flight

c. Numerous critical functions must be monitored and managed to avoid a catastrophic event

d. Reusability of aging complex systems requires ever increasing manual operations to maintain performance and safety

e. The design is difficult to evolve in response to actual or changing operational environments

f. STS design turnaround time was 160 hours. Actual turnaround time is 1296 hours minimum

g. STS design flight rate was 10 flts/veh/year. Actual flight rate 2.5 - 3 flts/veh/year maximum

h. STS planned recurring costs were $100/lb to LEO. Actual recurring costs are $10,000/lb to LEO

The solution(s) or recommendation(s) are (a) maximize integration at the architecture level; launch vehicle, spacecraft, ground systems) at the start of the design; (b) minimize the number and complexity of subsystems; (c) automate systems test and verification where practical to minimize manual operations, equipment wear, and damage; (d) baseline Level I/II con-ops at PRR, which will allow a realism assessment and manage the requirement changes thereafter; and (e) include manufacturing and operations participation in the DDT&E processes.

Another area of design and integration to consider is the streamlining of the design review. streamlining the design review approach will avoid or reduce large costs associated with DDT&E major milestones. Current NASA HSF major milestone design reviews are conducted with a large number of agency personnel. A wide range of materials equates to large number of candidate issues (e.g., RIDs). This is very expensive, requiring a large infrastructure, logistics, and technical disposition costs. Streamlining can avoid/reduce the costs of design reviews by up to one-third based on Orion CEV/MPCV data. This is accomplished by (a) expanding the use of subsystem design reviews, (b) focusing technical review on design products, (c) changing the identification and vetting of issues, and (d) limiting participation to highly qualify personnel with the identified ability, responsibility, and authority.

6.5.4 Contracting FOM Results

Appling risk-based contracting with an incentives approach provides a contractual mechanism to reward productivity. Deploying contract structures to incentivize profile for productivity with recognition of products and services that contribute directly to successful completion of mission objectives and incentivizing for process improvements that reduce overall cost with improvement or no negative impact to technical or schedule performance nor impact to risk baseline.

Establishing clear lines of accountability will assist in reduction of Contractor overlap. As an example, the contractual delivery of end item in a "ready-to-install, -assemble, or -integrate" configuration will avoid the overlap between OEM and operator. The delivery of an as-certified, as-built product end-item should complete contract obligation, thus eliminating RAA overlaps and duplication of costs in operations by delivery of certified, ready-to-assemble end items to the operations and processing center.

Establishment of a program-level-defined RAA to the centers that should flow to their contracts to eliminate duplication. Examples are Engineering, M&P, Non-Destructive Evaluation (NDE), and other support groups. End-item delivery must contain hardware and all the associated data, information, standards, etc., in compliance with program-level taxonomy.

On SSP, the Launch Support Services (LSS) cost to provide OEM knowledge domain onsite at KSC is significant. During "down periods" in the vehicle flow at KSC, the productivity and efficiency of the LSS workforce is low. A model used in the EELV market, known as an "operations campaign," is to bring OEM knowledge, if it is not local to the launch site, to either Cape Canaveral or Vandenberg AFB during a 1-to-2-week period just before launch for launch preparation and just after a launch for postflight activities. This technique allows for a very small OEM footprint at the launch site during the year, while still being able to provide expertise at the launch complex during terminal launch countdown and postflight analysis.

Risk-Based Contracting Example

Ideally, NASA would like to reward performance for productivity, products, and services that directly contribute to successful completion of mission objectives. Providing an incentive reward of fee for reaching a milestone date should be avoided. While Level of Effort (LOE) provides NASA flexibility in using the Contractor base, it is perceived to inflate the program costs.

When the contracting mechanism for incentivizing the Contractors are not thoroughly evaluated against their scope, the issue affects Supportability, Operability, and Availability FOMs. Contracting mechanisms have significant influence in achieving the near- and long-term program objectives for cost, schedule, and technical performance. Incentives must be devised to motivate Contractors to deliver the desired results at an affordable price and within acceptable risk.

As an example, the Space Shuttle contracting evolution is a good example. The cost-plus contracts provided the desired LOE support and flexibility for early development activities, modification periods, and changing mission manifests but were perceived to result in elevated workforce numbers and program costs. Cost-based incentives provided motivation for Contractors to reduce total operating budgets but were perceived to result in a trade of safety versus profit and increased programmatic risk.

The solution(s) or recommendation(s) are to (a) deploy contract structures to incentivize productivity; products and service that contribute directly to the successful completion of mission objectives, (b) incentivize process and performance improvements that reduce overall cost with no negative impact to the risk baseline, and (c) manage issuance of contract scope consistent with the defined RAA of the issuing agency/organization to avoid cost of duplication and potential conflicts during execution.

6.5.5 Business Systems FOM Results

Business systems should enhance the user's ability to manage a program. Too often, business systems become an ineffective tool in the program manager's ability to evaluate the program's performance, and therefore, corrective action is either misaligned against the real risk or goes unnoticed. Common element-to-element business systems with interoperability via standardization of toolset interfaces provide a complete programmatic health picture. Mandating the use of common, Industry standards for tools and data compatibility and complying with program-level taxonomy will allow interoperability at the data level. Interconnecting data access to support all processes and life-cycle phases, while minimizing the number and costs of disparate interfaces and information translations, provides the program manager in-depth insight into the program performance. This was demonstrated by USA on Ares I-X between the First Stage and ground processing, which resulted in a more responsive and efficient activity.

Compatibility Standards Example

Although the example was not a life-cycle driver on CxP because it was a flight test, the Ares I-X lack of interoperability of business systems is a good example of what to fix in your data stream. As a benchmark, USA evaluated the SSP business systems. Verification of system configuration is a large cost on SSP. Having business system compatibility between NASA and its SLS Contractors is crucial to have the ability to manage the data required to be efficient in the future, thus managing LCC. Even though the example appears to impact the programmatic, actually, the issue affects Commonality, Operability, Usability, and Availability FOMs, as the business systems "touch" all aspects of program execution. Legacy, paper document-based processes and lack of interoperability among the various program element business systems impedes effective communication, restricts visibility to essential information, delays dissemination of time-critical data, and drives up the cost of support and verification labor necessary to manage programmatic risk. As an example, the Ares I-X program was comprised of many diverse systems and processes operating without overall data standards and with limited interconnectivity. KSC developed and provided Collaborative Integrated Processing Solutions (CIPS), a standards-based Data Information Management System (DIMS) with interorganizational connectivity between NASA centers and Contractors to support processing operations, integration, and CoFR. The system included a collaborative working environment with electronic work instruction authoring and approval; paperless work execution with enhanced process controls via wireless Land Area Network (LAN), Integrated Supply Chain Management with task, material, and labor planning; closed-loop requirements and configuration management; Web-based training and processing portal with real-time metrics, information, and reports; and integrated data management and repository, including nonconformance.

CIPS provided major productivity improvements, including (a) reduction of paper documents (print/copy/distribute/file), (b) elimination of manual postexecution documentation quality reviews, (c) reduction of work plan update process from months to days, and (d) automated task and as-built versus as-designed status. Based on KSC USA Shuttle process data, the savings extrapolated to other centers produced projected annual program savings approaching $9M.

The solution(s) or recommendation(s) are to (a) establish, contractually obligate, and enforce interoperability standards; (b) select and implement proven market technologies to satisfy requirements at minimum cost; (c) take advantage of standardization based on common functions by avoiding multiple solutions for common need; (d) provide secure access to authoritative source information across the program, build once, use many; (e) minimize the data collection burden while improving product quality; and (f) leverage Industry best practices inherent in COTS technologies.

7.0 HLLV SYSTEMS ANALYSIS TOOL

Concurrent with the trade study activities, USA developed, designed, built, and will deliver a HLLV System Analysis Tool (HLSAT), along with a user's handbook, in association with the Final Study Report under DRD 1384MA-003. The tool will use common desktop browser software, be supported by multiple desktop operating systems, and will not require additional software licensing and specialized configurations. The tool will provide flexibility to allow updates to the criteria and weighting as shown in Figure 7-1.

Based on the current SLS study work accomplished to date by MSFC's SLS Team, the trade process, specifically, the HLSAT, was used with the current trade study data. In addition to the tool and user's handbook, provided on a separate Compact Disk (CD), the USA six trades and respective trial cases of architecture life-cycle evaluation of SLS are also provided in the tool database. These trial cases demonstrated the tool's validity, ensuring flexibility and extensibility of the tool capability. The output of the tool at each iteration is provided in the database, which couples the affordability, schedule, reliability (including safety), operability, and performance characteristics and attributes for the architectures evaluated.

Figure 7-1. HLSAT home page

8.0 SYSTEMS ANALYSIS AND TRADE STUDY CONCLUSIONS AND RECOMMENDATIONS

No single technical concept in the SLS HLPT trade study was clearly better than the other configuration on every measure of the Affordability, Performance, Schedule, Reliability, and Operability FOMs. USA's evaluation includes many options that yielded strengths and weaknesses that were very dependent on near-term or long-term considerations affecting affordability. In addition, scoring and the resultant weighting of the FOMs plays a key role in comparing alternatives. It was critical in the process to define and document our understanding; i.e., what we meant and why we meant it, as achieving a common understanding of terminology and importance was crucial. The timeline constraint of earlier than 2017 for initial launch capability results in significant negative impacts to any alternative that requires development beyond that predicted IOC for existing heritage systems, even outweighing Operability FOMs that ultimately drive LCC. Cost considerations with respect to dual development and infrastructure changes score very negatively and, therefore, limit architecture evolvability. These factors combine to indicate that major architecture changes, either evolution or changeover outside the initially selected configuration, are generally cost prohibitive, while evolution within a launch vehicle configuration family remained cost feasible. USA's product and results were reviewed independently by Davidson Technologies. Their study report review is documented in Appendix D. To follow are USA's conclusions and recommendation on the HLPT SATS results.

8.1 Conclusions

A "Progressive Architecture" that employs a block-type vehicle evolution with increasing performance capability using a capability-driven model methodology to satisfy new mission objectives appears to balance cost and schedule, both near-term and long-term, resulting in an affordable and sustainable framework. The implementation of that model will depend heavily on the NASA acquisition model deployed by the SLS Program and MSFC Procurement Offices. The use of NASA heritage assets, along with their maturity and knowledge domains, kept the initial DDT&E costs within the forecasted FY budgets, but LCC in the out years broke the FY budget thresholds. Heritage assets have a very positive impact on near-term costs and the development timeline. In general, high-TRL systems typically provide positive impacts on both cost and schedule. With heritage assets, to mitigate this long-term affordability risk, legacy requirements and processes associated with heritage assets need to be readdressed to improve efficiency and realign the asset usage to achieve sustainability. With the heritage assets, it is also imperative that a schedule for P^3I to improve affordability; i.e., increase supportability and decrease obsolescence, be implemented to sustain the SLS Program.

In reviewing the "hardware and software" aspect of the architecture trades, it became apparent that implementation of the MSFC business operations model to execute the program was as important as selecting the best architecture. One aspect of that model is the program integration approach. Instituting a very strong SE&I function from conceptual design through system operation will provide closed-loop engineering accountability. SE&I should be executed in an integrated manner across the entire SLS Program with a focus on system performance and margins, operations costs, and program schedule. To have the authority, SE&I must be directly accountable to the SLS Program manager. The second aspect of the model is the requirements strategy. Zero-based requirements were determined to be the highest-ranking implementation approach to improve affordability and schedule. The SLS Program should consider implementing and structuring organizationally to address requirements at the start of the program; i.e., initially, everyone should be system engineers working requirements. The third aspect of the model is technology insertion. To balance cost and schedule, especially near-term, technology insertion on the ascent launch vehicle stage elements must "fight its way" into the initial baseline. Technology insertion can provide risk buy-down on cost, schedule, and/or technical (e.g., performance, safty). It must be traded carefully to realize target gain without incurring undue costs or schedule impacts. It is clear that ascent boost technology is in place today, with many different performance and cost options, for a successful SLS, thus, development should be focused on higher architectural risk; e.g., In-Space Stage propulsion systems, In-Space Habitats, etc. New technologies can be applied at specific transition points when mature enough for acceptable risk to increase capabilities and performance and/or reduce LCC.

From the Operator standpoint and independent of a mandated availability consideration; i.e., time to develop and certify, and assuming reliability and performance being equal, a vehicle architecture that is "operationally friendly" and minimizes evasive operations, limits inspections and verifications, reduces processing hazards, and requires minimal special tooling or access, consistently scores better for operational affordability and sustainability with respect to full Program LCC. Consider operability early in the design and development phase, as historically, a low priority has been placed on operations considerations during the flight system DDT&E phase, but it is a significant LCC driver on the long-term Program.

8.2 Recommendations

Based on USA's analyses and study results, the following programmatic recommendations are offered for consideration:

a. To achieve an affordable and sustainable SLS Program, a stable DDT&E path and a new NASA programmatic implementation (operating model) will be required. NASA should challenge all "business as usual" cultural business rhythms and tailor existing policies from Level 0 down

b. Consider LCC influences early in the development cycle. LCC decisions must be made early in the SLS Program. Postponing decisions will adversely affect long-term affordability as the design becomes baselined. Minimizing DDT&E costs impacts the ability to implement innovations to reduce LCCs

c. To reduce the recurring and the allocated nonrecurring fixed costs on a per-flight basis, SLS needs to market distribute the costs by cost sharing or increasing the business base. Common infrastructure and processes with other programs within NASA, DoD, and IPs to share capability and reduce costs is needed. Cost sharing the Ground Segment with other users; e.g., NASA Commercial Crew, will address the nonrecurring cost burden on common and universal infrastructure. The NASA Exploration manifest alone will not support the SLS Program. To address the recurring costs, more than two flights per year are desired. To achieve this manifest, NASA's portfolio of payload Customers needs to be expanded beyond NASA Exploration. Do not optimize manufacturing capacity at four flights per year, as the potential launch manifest only shows one to two flights per year through 2025

d. Integration is essential for SLS Program's success. Building on a successful SE&I foundation will be critical to survive numerous changes both internal to the program and from external political influence. NASA has many options and technologies to "pull on" to succeed, and the SLS Program needs a very capable integration effort to pull those pieces together in a timely approach. Regarding the political aspect, USA believes that the "constraints" will change or additional "constraints" will be added over the life cycle, thus designing and developing only an end-state vehicle for one purpose would be taking on additional SLS programmatic risk. The selection of an architecture should not only consider the technical, budget, and schedule "winds of change" going forward but the political change, too

Based on USA's analyses and study results and in consideration of the recommended implementation changes above, the following technical and performance recommendations are offered for consideration:

a. Near-term solution: To optimize performance and balance cost and schedule requirements within the next 6 years, an SDHLLV configuration using heritage ET tank features and engines (RS-25D → RS-25E) for Core Stage and segmented solids (4-segments → 5-segments) for Booster Stage as defined in Trade 6 - Trial Case 6 scenario, alternative configuration 1 (baseline), is recommended

b. Midterm solution: To optimize operability and balance cost and schedule requirements over the life cycle, an SDHLLV using ET tank features and engines (RS-25D → RS-25E) for Core Stage and demonstrated large monolith solids (P110 → P180 size) for Booster Stage in combinations of four or six, depending on mission requirements as defined in Trade 6 - Trial Case 6A scenario, alternative configuration 2 (option 1), is recommended

c. Long-term solution: To optimize performance and operability, although with higher costs and longer schedules over the next 8 years, with lower life-cycle operations costs (both recurring and nonrecurring), a LOX/RP1 Core and LOX/LH_2 Second Stage "Saturn-like" vehicle using the F-1A and J-2X development work to date and developing new manufacturing techniques for the 33-foot diameter Core Stage tank as defined in Trade 6 - Trial Case 6, alternative configuration 4 (option 2), is recommended

The time-dependent, affordability-considered solutions above depict an in-family evolution with technology insertions to increase performance and lower LCC in the future. An out-of-family evolution from near-term segmented solid boosters to midterm monolith solid boosters is very feasible, assuming the early block vehicle has end-state design considerations for the out-of-family migration (e.g., Core Stage propulsion cluster has load bearing attachments for monolithic solid boosters). If the Phase 1, Block 0 or 1 vehicle, does not have these features, MSFC's acquisition options for Phase 2 will be limited.

APPENDIX A ACRONYMS

AAP	Adjustable Access Platform
ACAA	Adjustable Crew Access Arm
ACT	Advanced Compoton Telescope
AF	Audio Frequency
AHMS	Advanced Health Monitoring System
AHP	Analytical Hierarchy Process
AIAA	American Institute of Aeronautics and Astronautics
ALCA	Aft Load Controller Assembly
AMBR	Advanced Materials Bipropellant Rocket
ASAP	As Soon As Possible
ATK	Alliant Techsystems, Inc.
ATLAST	Advanced Technology Large-Aperture Space Telescope
ATP	Authority to Proceed
ATV	Automated Transfer Vehicle
AUC	Adjustable Umbilical Carrier
BAA	Broad Agency Announcement
BEO	Beyond Earth Orbit
BPC	Boost Protective Cover
BTU	Bus Thermal Unit
CARD	Constellation Architecture Requirements Document
CCCE	Configuration Change Control Express
CCR	Critical Customer Requirements
CD	Compact Disk
CDR	Critical Design Review
CE	Concurrent Engineering
CEMMENT	Constellation-Enabled Mars Mission Exhibiting New Technology
CEV	Crew Exploration Vehicle
CIPS	Collaborative Integrated Processing Solutions
CLV	Crew Launch Vehicle
CoFR	Certification of Flight Readiness
con-ops	Operations Requirements
COTS	Commercial-Off-the-Shelf
CRL	Capability Readiness Level
CS	Civil Servant
Cx	Constellation
CxP	Constellation Program
CY	Calendar Year
DAC	Design Analysis Cycle
DALI	Dark Ages Lunar Interferometer
DAV	Descent/Ascent Cargo Vehicle
DDT&E	Design, Development, Test, and Evaluation
DFMR	Design for Minimum Risk
DFO	Design for Operations
DIMS	Data Information Management System
DoD	Department of Defense
DoF	Degrees of Freedom

DOL	Day of Launch
DRA	Design Reference Analysis
DTC	Design to Cost
EDS	Earth Departure Stage
EEE	Electrical, electronic, and Electromechanical
EELV	Evolvable Expendable Launch Vehicle
EES	Emergency Escape System
ELV	Expendable Launch Vehicle
EMI	Electromagnetic Interference
ERV	Earth Return Vehicle
ESA	European Space Agency
ESAS	Exploration System Architecture Study
ESMD	Exploration Systems Mission Directorate
ET	External Tank
EVM	Earned Value Management
FMEA	Failure Modes Effects Analysis
FMHR	Free Molecular Heating Rate
FOM	Figures of Merit
FPL	Full Power Level
FPR	Flight Performance Propellant Reserve
FS	First Stage
FY	Fiscal Year
G&O	Goals and Objectives
GEO	Geosynchronous Earth Orbit
GG	Gas Generator
GNC	Guidance, Navigation, and Control
GR&A	Groundrules and Assumptions
GSE	Ground Support Equipment
HEFT	Human Exploration Framework Team
HEO	Highly Elliptical Orbit
HLL	Heavy Lift Launch
HLLV	Heavy Lift Launch Vehicle
HLPT	Heavy Lift and Propulsion Technology
HLSAT	HLLV System Analysis Tool
HLV	Heavy Lift Vehicle
HOQ	House of Quality
HQ	Headquarters
HTPB	Hydroxyl-Terminated Polybutadiene
HTV	H-II Transfer Vehicle
IMLEO	Inserted Mass to Low Earth Orbit
IOC	Initial Operational Capability
IP	International Partner
iPLM	Institutionalized Product Life Cycle Management
IPT	Integrated Product Team

Isp	Specific Impulse
ISRU	In-Situ Resource Utilization
ISS	International Space Station
JSC	Johnson Space Center
JWST	James Webb Space Telescope
KPI	Key Performance Indicators
KSC	Kennedy Space Center
L	Lagrange
LAN	Land Area Network
LAS	Launch Abort System
lb	pound(s)
LCC	Life-Cycle Cost
LCCA	Life Cycle Cost Analysis
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LiLFA	Lithium Lorentz Force Accelerator
LOC	Loss of Crew
LOCV	Loss of Crew/Vehicle
LOE	Level of Effort
LOM	Loss of Mission
LOR	Lunar Orbit Rendezvous
LOV	Loss of Vehicle
LOX	Liquid Oxygen
LSS	Launch Support Services
LVA	Launch Vehicle Assembly
M&P	Materials and Processes
MIR	Most Important Requirement
MLI	Multi-Layer Insulation
MOD	Mission Operations Directorate
MOTS	Military-Off-the-Shelf
MPCV	Multi-Purpose Crew Vehicle
MPD	Magnetoplasmadynamic
MPS	Main Propulsion System
MPTA	Main Propulsion Test Article
MSFC	Marshall Space Flight Center
MSRM	Monolithic Solid Rocket Motors
mT	metric Tons
MTV	Mars Transport Vehicle
MUST	Modern Universe Space Telescope
N_4H_4	Hydrazine
NASA	National Aeronautics and Space Administration
NDE	Non-Destructive Evaluation
NEO	Near Earth Object
NERVA	Nuclear Engine for Rocket Vehicle Application
NLT	No Later Than
NOAA	National Oceanic and Atmospheric Administration

NRO	National Reconnaissance Office
NSDIS	NOAA Satellite Data and Information Service
NTE	Not to Exceed
NTO	Nitrogen Tetroxide
NTR	Nuclear Thermal Rocket

OCT	Office of Chief Technologist
OEM	Original Equipment Manufacturer
OMB	Office of Management Budget
ORSC	Oxygen Rich Stage Combustion
ORU	Orbital Replacement Unit

P^3I	Pre-Planned Product Improvement
PASS	Primary Avionics Software System
PBAN	Polybutadiene Acrylonitrile
PCAD	Propulsion and cryogenics Advanced Development
PDF	Portable Document Format
PDR	Preliminary Design Review
PIT	Pulsed Inductive Thrusters
PL	Power Level
PLM	Project Lift Cycle Management
POR	Program of Record
POV	Point of View
PPE	Personal Protective Equipment
PPM	Program Performance Metric
PRF	Parachute Refurbishment Facility
PWR	Pratt-Whitney Rocketdyne

QD	Quick Disconnect
QFD	Quality Function Deployment

RAA	Responsibility, Accountability, and Authority
RAC	Requirements Analysis Cycle
RF	Radio Frequency
RLV	Reusable Launch Vehicle
RP	Rocket Propellant
RSRM	Reusable Solid Rocket Motor
RVD	Reference Vehicle Design
RVLM	Removable Vehicle Launch Mounts

SAFIR	Single Aperture Far Infrared Telescope
SATS	Systems Analysis and Trade Study
SBIRS	Space Based Infrared System
SBKF	Structural Buckling Knockdown Factor
SCAPE	Self Contained Atmosphere Protective Ensemble
SDHLLV	Shuttle Derived Heavy Lift Launch Vehicle
SE&I	Systems Engineering and Integration
SERT	Space Solar Power Exploratory Research and Technology
SHAB	Surface Habitation
SI	Stellar Imager
SL	Sea Level

SLS	Space Launch System
SMD	Science Mission Directorate
SPECS	Submllimeter Probe of the Evolution of Cosmic Structure
SPOC	Space Program Operations Contract
SRB	Solid Rocket Booster
SRM	Solid Rocket Motor
SSC	Stennis Space Center
SSME	Space Shuttle Main Engine
SSP	Space Shuttle Program
STA	Static Test Article
STS	Space Transportation System
SW	Software
TA	Technology Area
TBD	To Be Determined
TBR	To Be Revised
TLI	Trans-Lunar Injection
TMI	Trans-Mars Injection
TPS	Thermal Protection System
TransHab	Transit Habitat
TRL	Technology Readiness Level
TVC	Thrust Vector Control
U.S.	United States
UGCS	Universal Ground Control System
ULA	United Launch Alliance
ULC	Universal Launch Complex
UML	Universal Mobile Launcher
USA	United Space Alliance, LLC.
USAF	United States Air Force
VAB	Vehicle Assembly Building
Vac	Vacuum
VASIMR	Variable Specific Impulse Magnetoplasma Rocket
VOC	Voice of the Customer
W&S	Weight and Sizing
Wt	Weight

APPENDIX B BAA TECHNICAL OBJECTIVES AND USA SOW TRACEABILITY MATRIX

Report Paragraph	Report Paragraph Title	NASA BAA Technical Objectives	USA Statement of Work (SOW)
1.0	**Background, Experience, and Summary**	(b) (4)	[redacted]
1.1	Introduction		
1.2	Background		
1.3	Experience		
1.4	Executive Summary		
2.0	**Study Approach & Methodology**	(b) (4)	4.0
2.1	Trade Approach		
2.2	Trade Process		
2.3	AHP, QFD, and Pugh Matrix Tools		
3.0	**Trade Space Boundaries and Constraints**	• Identify how alternative GR&As impact the identified alternative system solutions	4.1
3.1	Systems Analysis GR&A		
3.2	Trade Study GR&A		
4.0	**Trade Input Considerations**	(b) (4)	(b) (4)
4.1	Missions & Objectives		4.2
4.2	Engine & Propulsion Systems		4.0
4.3	Reference Vehicle Designs		4.0
5.0	**Figures Of Merit**	• Provide a recommended list of key decision attributes and rationale associated with each • Provide a recommendation for the weighting of the recommended key decision attributes • Identify how changes to the weighting of key decision attributes affect the architectures	4.3
5.1	Voice of the Customer		
5.2	Customer Critical Requirements		
5.3	Characteristics and Attributes		

Report Paragraph	Report Paragraph Title	NASA BAA Technical Objectives	USA Statement of Work (SOW)
6.0	**Trade Results**	• Identify how innovative or nontraditional processes or technologies can be applied to the heavy lift systems to dramatically improve its affordability and sustainability • 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 • Identify how incremental development testing, including ground and flight testing, of heavy lift system elements can enhance the heavy lift system development • Identify capability gaps associated with the heavy lift system, and for each capability gap, identify specific areas where technology development may be needed • Identify capability gaps associated with the First Stage main engine functional performance and programmatic characteristics required to support each heavy lift system studied • Identify capability gaps associated with the Upper Stage main engine functional performance and programmatic characteristics required to support each heavy lift system studied • Identify capability gaps associated with all other technical aspects of heavy lift system • Identify capability gaps associated with the In-Space space propulsion elements functional performance and programmatic characteristics required to support each heavy lift system studied • Identify capability gaps associated with all other technical elements of the In-Space space propulsion element • Identify which In-Space space propulsion elements, if any, should be demonstrated via space flight experiments	4.0, 4.5, and 4.6
6.1	Architecture		
6.2	Launch Vehicle Stages		
6.3	In-Space Stage		
6.4	System		
6.5	Implementation		
7.0	**HLLV Systems Analysis Tool**	(b) (4) [redacted]	4.4
8.0	**Systems Analysis and Trade Study Conclusions and Recommendations**	[redacted]	4.0 and 4.6
8.1	Conclusions		
8.2	Recommendations		

Report Paragraph	Report Paragraph Title	NASA BAA Technical Objectives	USA Statement of Work (SOW)
Appendix A	Acronyms	(b) (4) [redacted]	[redacted]
Appendix B	BAA Technical Objectives and USA SOW Traceability Matrix	[redacted]	[redacted]
Appendix C	Cost Evaluations	[redacted]	[redacted]
Appendix D	DTI Independent Assessment Study Report	[redacted]	[redacted]

APPENDIX C COST EVALUATIONS

When addressing the cost component of the trade study, USA used a parametric evaluation approach due to the numerous trade study cases that were analyzed. The process for the parametric evaluation is to develop a baseline architecture cost and then leverage the FOMs, as developed and discussed in Section 5.0, on the other configurations to develop the discrete cost values.

USA selected the Trade 6 - Trial Case 6 scenario, Alternative Configuration 1 (Baseline) as the cost "baseline" architecture. The selected baseline architecture was cost estimated using parametric estimating techniques by cost element, inputting Industry data from previous SDHLLV efforts, and escalating the inputs for CY 2011 dollars. On developing the cost estimates, a cost estimate validation was accomplished using recent Industry and NASA cost estimating values (external) along with recent USA proposal estimates (internal).

With respect to the alternative configurations, because cost is both directly and indirectly affected by the FOMs, it is important to assess the options against all FOMs that impact cost. USA established and defined an Affordability FOM (VOC metric), then prioritized the FOMs via AHP pairs-wise comparison, which resulted in a relative importance of 47 percent at the program level. To determine the distribution of the FOM, USA established Affordability FOM allocations at the architecture and vehicle and element system (ground and mission) levels with associated target levels. Figure C-1 depicts the Affordability FOM decomposition and allocation of the FOMs that directly affect LCCs. For each Trade Trial Case, USA ranked each alternative configuration architecture with corresponding vehicle and element system against the baseline configuration.

The other four VOC components, Schedule, Performance, Operability, and Reliability FOMs, also affect affordability indirectly. These were identified, and a smaller scaling factor against the baseline configuration costs was applied.

C.1 Baseline Architecture Cost GR&A

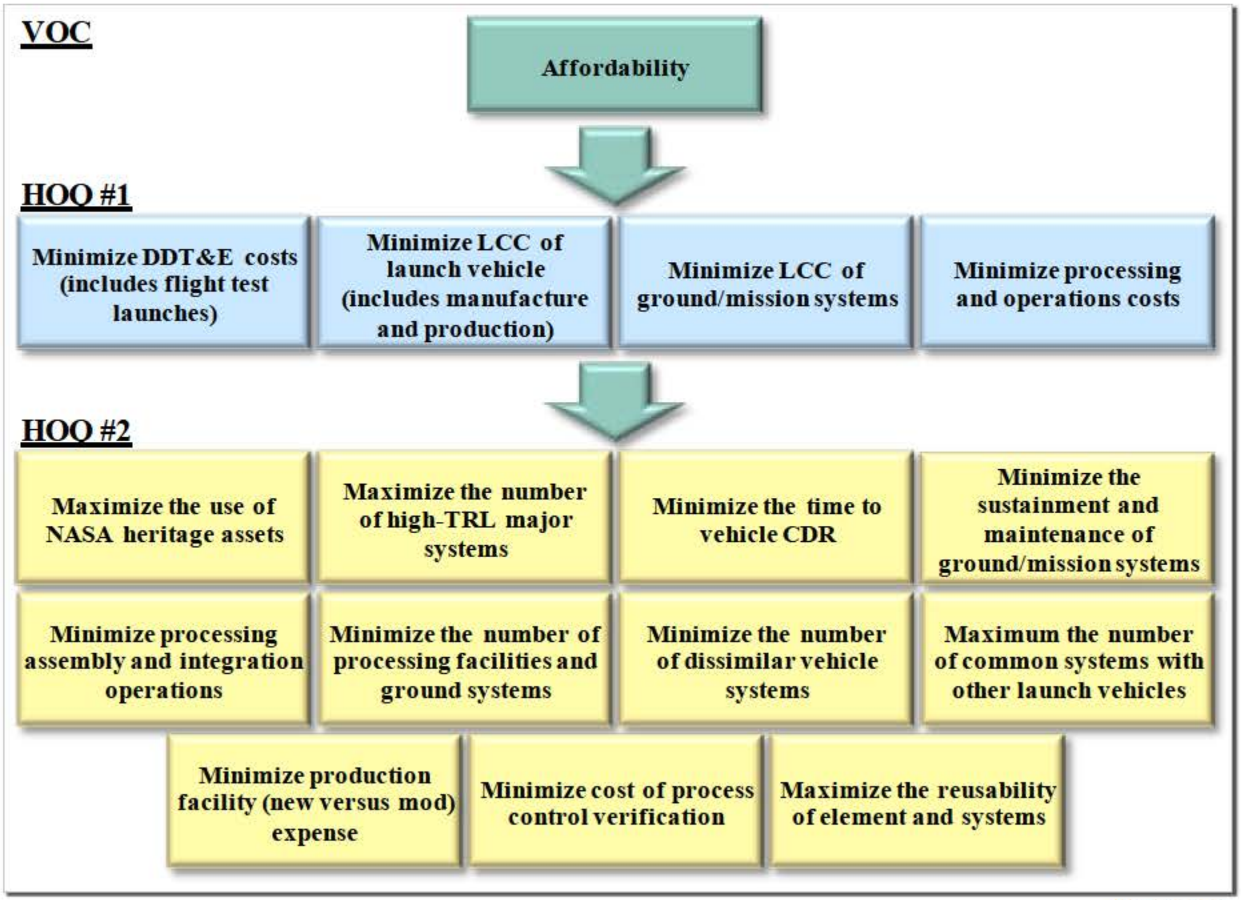
To follow are the costing GR&A for the baseline architecture:

a. Partial STA and full-up MPTA (DDT&E costs) and two flight tests (flight Hardware (HW) costs)

b. Block 1 configuration flight test article in 2015; Block 2 configuration flight test article in 2018

c. IOC for Phase 1 (Block 0 vehicle configuration) capability in CY 2016 with Phase 2 (Block 1) IOC 3 years later; Phase 3 (Block 2) IOC 3 years later

d. Contractors assume element design authority



HLPT-SATS_036

Figure C-1. Affordable FOMs that affect LCC

e. NASA assumes system design and integration management - NASA costs not included in USA parametric estimate

f. NASA implements partial implementation affordability initiatives, which reduce both NASA CS and Contractor costs

g. Cost estimated at FY 2011 costs with 4-percent escalation thereafter

h. Existing SSP and Cx contracts modified to support IOC milestones

i. Use SSP and CxP sunken costs on DDT&E and facilities and systems

j. First three Core Stage tanks are partially complete (using ET elements)

k. 21st Century Launch Complex sustains and improves KSC infrastructure (i.e., not an SLS Program cost burden - SLS Program will fund SLS-specific requirements only)

l. Excludes SLS missions operations, payload assets, payload integration, payload carrier, and payload mission support

C.2 Configuration Processing Comparison

To determine the operations cost associated with the baseline configuration and options, USA developed a ground processing and operations concept of operations, Figure C-2, to determine the flow approach.

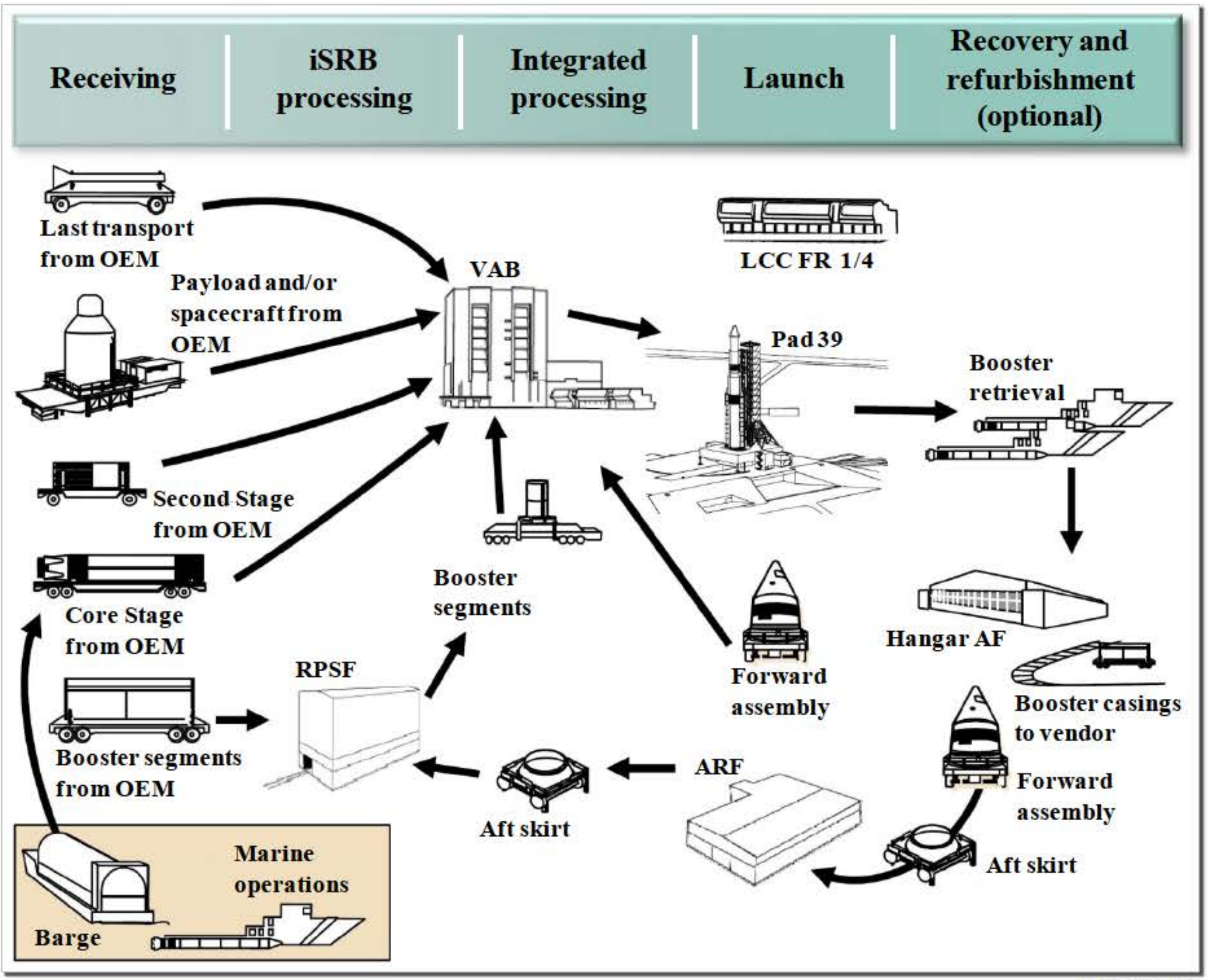
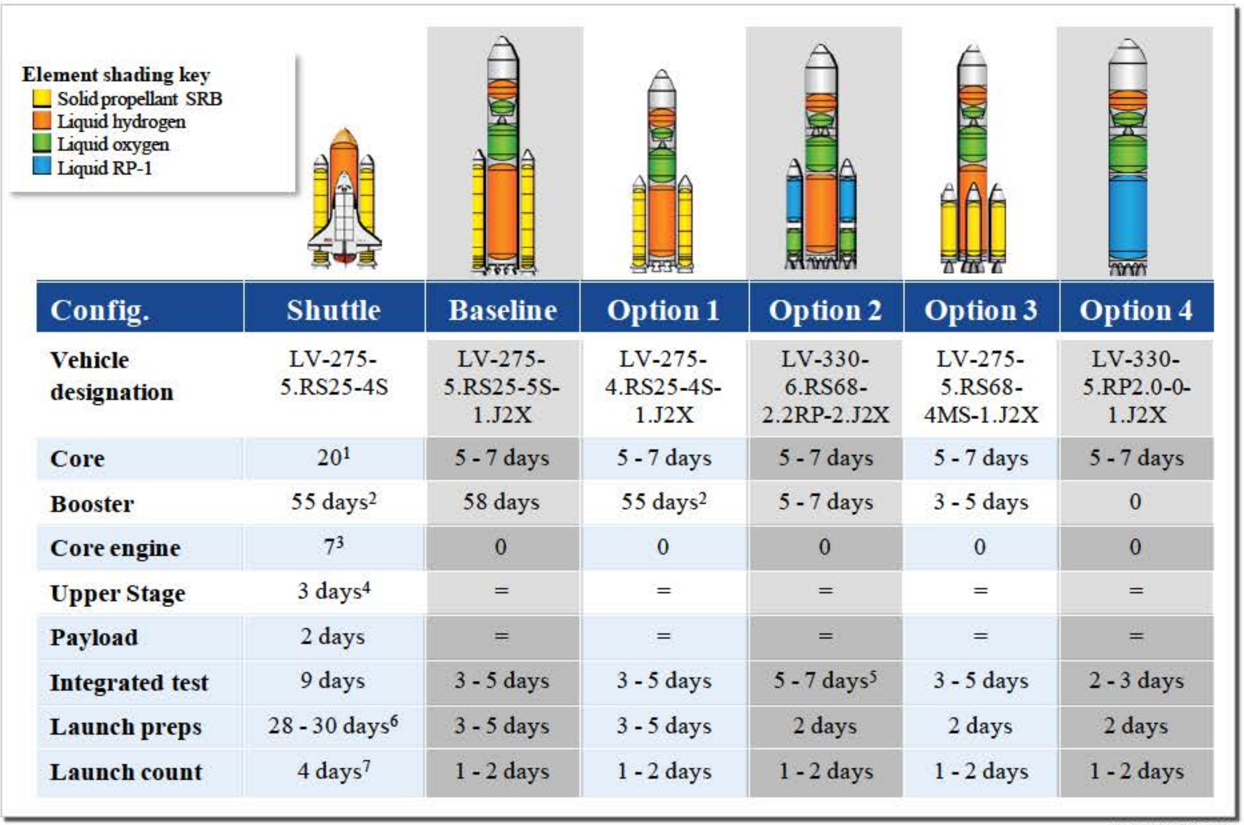


Figure C-2. Processing and launch flow concept of operations

Using the SSP processing and launch flow as a benchmark, USA then compared the baseline and options against the benchmark, as shown in Figure C-3, with comparison notes in Table C-1, to determine the recurring costs for each configuration.



Config.	Shuttle	Baseline	Option 1	Option 2	Option 3	Option 4
Vehicle designation	LV-275-5.RS25-4S	LV-275-5.RS25-5S-1.J2X	LV-275-4.RS25-4S-1.J2X	LV-330-6.RS68-2.2RP-2.J2X	LV-275-5.RS68-4MS-1.J2X	LV-330-5.RP2.0-0-1.J2X
Core	20[1]	5 - 7 days	5 - 7 days	5 - 7 days	5 - 7 days	5 - 7 days
Booster	55 days[2]	58 days	55 days[2]	5 - 7 days	3 - 5 days	0
Core engine	7[3]	0	0	0	0	0
Upper Stage	3 days[4]	=	=	=	=	=
Payload	2 days	=	=	=	=	=
Integrated test	9 days	3 - 5 days	3 - 5 days	5 - 7 days[5]	3 - 5 days	2 - 3 days
Launch preps	28 - 30 days[6]	3 - 5 days	3 - 5 days	2 days	2 days	2 days
Launch count	4 days[7]	1 - 2 days	1 - 2 days	1 - 2 days	1 - 2 days	1 - 2 days

HLPT-SATS_038

Figure C-3. Processing and launch flow comparisons

Table C-1. Comparison notes

Element	Notes
Core	[1] SSP includes ET receipt to SRB mate plus umbilical mates and leak tests (Orbiter and ET)
Booster	[2] Include aft booster buildup and stacking – (SSP stacking operations runs three shifts) – Liquid booster configuration assumes receipt with engines integrated; includes umbilical mate and checkout
Core engine	[3] SSP values include orbiter engine integration and leak tests timelines – All HLLV configurations assume engines received integrated with core
Upper stage	[4] Orbiter mate as baseline
Payload	– All HLLV configurations assume payload integration and checkout prior to rollout
Integrated test	All vehicles include integrated testing and system closeouts operations prior to rollout [5] Assumes liquid booster involve more systems verifications than solids (valves, monitoring, purges, instrumentation, etc.)
Launch preps	[6] SSP includes ground interface mates and validations, battery installations, vehicle closeouts, and testing prior to fueling – All HLLV configurations assume vehicle umbilical mates, validations, and closeouts made prior to rollout. Ground interfaces are made and validated by autocoupler – SRB configurations assume SSP-like TVC fuel loading, aft skirt closeouts, ignition S&A pin removal, and forward skirt closeouts
Launch count	[7] SSP - 3 days driven by payload and crew requirements – All configurations include pressurizations, fuel loading, and terminal count to launch

C.3 Baseline Architecture Parametric Cost Estimate

In reviewing the cost modeling tools available, PRICE-H and NAFCOM, it was determined that for a complex system like SLS with unique conditions (e.g., assets already available) and implementing a revised business operating model different than NASA has used previously, the model-based cost estimating would not be appropriate. In reviewing the historical cost data from previous bottoms-up estimating, the baseline cost was based on the Industry Team's STS Derived Launch Vehicle Trades (case F) budgetary estimate. The budgetary estimate was adjusted for changes in architecture, escalated to FY 2011 costs, and then escalated 4 percent for out year projected costs. The total SLS Program costs for the baseline configuration are show in Table C-2, parametric costs estimates by FY, and Figure C-4, parametric cost funding profiles.

A key assumption to SLS cost estimating is allocating ground infrastructure and modification costs necessary for KSC to be the launch complex for numerous programs. There are two distinct categories for KSC capability and capacity: (a) universal infrastructure, which means that all programs use the same facilities and equipment, and (b) special infrastructure, which is unique to a single program only. In the cost estimating, it is assumed that the universal infrastructure, typically institutional capability and capacity, will be funded with KSC funding under 21st Century Launch Complex or equivalent budget source and not SLS Program funding. For the SLS Program to be successful, MSFC needs to take advantage of the synergy at KSC between SLS and numerous commercial crew launch vehicle requirements.

(b) (4)

[redacted]

(b) (4)

[redacted]

Using a cost-sharing methodology, the areas of potential synergy in the architectures are in the (a) launch vehicles, (b) ground systems, and (c) maybe missions systems. All three areas should be thoroughly examined for common requirements that lean toward a universal solution and provide the cost-sharing opportunities:

a. In the launch vehicles area, the selection of SLS vehicle systems that share systems with commercial vehicles; e.g., engines, avionics, and other components, that share an Industry base, supply chain, etc.

b. In the ground systems area, the selection of SLS-required infrastructure (facilities, GSE/STE, etc.) that share with the Commercial Crew-required infrastructure: universal mobile launcher concept that can enable multiple vehicles without incurring full cost of dedicated launch facility by all operators, even if the architectures are different

c. In the mission systems area, the selection of SLS mission systems and processes that share with the Commercial Crew (human-rated mission operations)-required systems

In summary, capitalize on anything that is not a point solution and consider a design solution that helps alignment with facilities, find synergies with common direction. This type of analyses and decisions is a NASA HQ programmatic decision, not necessarily a commercial crew or SLS initiative, but the potential benefits to both programs could be significant.

C.4 Contractor DDT&E Parametric Cost Breakout

For DDT&E costs from Industry for the Block 0 configuration, BOE/BOM verification from SDHLLV Design to Cost, Cycle 3, In-Line Heavy (Case F), 13 May 2005, was used to develop the estimates. It should be noted that the Block 0 System Testing and Flight Test Vehicle HW are accounted for in separate line items on the Total Program Cost chart, Figure C-2, and not in the Block 0 DDT&E rollup costs depicted in Table C-3.

(b) (4)

[redacted]

For DDT&E costs from Industry for the Block 1 configuration, BOE/BOM verification from SDHLLV Design to Cost, Cycle 3, In-Line Heavy (Case F), 13 May 2005, was used to develop the estimates. It should be noted that the Block 1 System Testing and Flight Test Vehicle HW are accounted for in separate line items on the Total Program Cost chart, Table C-2, and not in the Block 1 DDT&E rollup costs depicted in Table C-4.

(b) (4)

[redacted]

(b) (4)

[redacted]

C.5 Contractor Production Parametric Cost Breakout

For Production costs from Industry for the SLS Block 0 configuration (SDHLLV, Stretched ET, 3 RS-25D, Shuttle-like MPS, 4-segment PBAN plus spacer SRB, no U.S. element), the BOE/BOM data from SDHLLV Evolution Path 1 Recurring Costs Estimates, 13 May 2005, was used to develop the estimates. The element costs have PM and SE&I embedded, and the first Block 0 Production article is a single IOC flight, one flight per year thereafter until 2019 (three flights total). It should be noted that the Block 0 system testing and flight test vehicle HW are accounted for in separate line items on the Total Program Cost chart, Table C-2, and not in the Block 0 Production rollup costs depicted in Table C-5.

(b) (4)

[redacted]

For production costs from Industry for the SLS Block 1 configuration (SDHLLV, Stretched ET, 5 RS-25D migrating to RS-25E, Shuttle-derived MPS, 5-segment PBAN SRB, no U.S. element), the BOE/BOM data from SDHLLV Evolution Path 1 Recurring Costs Estimates, 13 May 2005, was used to develop the estimates. The element costs have PM and SE&I embedded, and the first Block 1 production article is a single IOC flight. Long-lead parts costs are included in FY 2019 for two flights per year thereafter, projected to start in 2020 (costs not estimated or projected). It should be noted that the Block 1 system testing and flight test vehicle HW are accounted for in separate line items on the Total Program Cost chart, Table C-2, and not in the Block 1 production rollup costs depicted in Table C-6.

(b) (4)

[redacted]

(b) (4)

[redacted]

C.6 Parametric Cost Estimate Comparison using FOMs

Once the baseline costs were determined, using the Affordability FOMs as shown in Figure C-1 and the other FOMs that indirectly affect cost, cost elements were adjusted by scaling cost in the direction based on Trial Case 6 results and their influence magnitude, yielding cost estimate comparisons as shown in Figure C-7.

(b) (4)

[redacted]

C.7 Parametric Cost External Validation

To validate the parametric cost estimates, with the recognition that some estimating GR&A could be slightly different, USA looked at four sets of data for a comparable product cost analogy. The Industry Heavy Lift Vehicle (HLV) Trade Study In-Line Cost Estimate, dated October 2010, and the NASA Shuttle-Derived HLLV Assessment, dated March 2010, were used as reasonable Rough Order Magnitude (ROM) budgetary estimates for benchmarking. The Industry estimate broke cost into DDT&E, production, and sustaining cost elements, and the estimates were segregated by function and WBS. The estimates were developed using historical data and modeling (PRICE-H and NAFCOM). The GR&A were very similar to USA's Trade Study GR&A. The total program as defined, no U.S. element and 50 mT to LEO, including three flight tests, was estimated at design and development = $2.2B, system test HW = $2.7B, and first flight article = $650M.

The NASA estimate broke cost into DDT&E and total cost (DDT&E and 18 operational flights). The in-line, three SSME, four-seg SRB, no U.S. element, 70 to100 mT to LEO was estimated at DDT&E = $10.4B and total = $35.7B. The in-line (HEFT), five SSME, five-seg SRB, no U.S. element, 100 mT to LEO was estimated at DDT&E = $14.9B and total = $42.0B. The in-line (HEFT), five SSME, five-seg SRB, U.S. element, 130 mT to LEO was estimated at DDT&E = $16.9B and total = $45.5B.

Adjusting for milestones and performance capability, the USA parametric cost estimates for the different Block vehicles are within range of Industry and NASA's benchmark estimates.

C.8 Parametric Cost Internal Validation

As a cost element data point, USA's trade study parametric costs were reviewed against the USA SLS iSRB proposal submitted to MSFC Procurement Office on May 24, 2011, for both reusable and nonreusable SRB options to determine if the Booster Stage cost element line item was reasonable. The booster includes all SRB components necessary to fully integrate the SLS booster and RSRM (segments) that would be provided as GFE to USA for integration. In comparing the costs, as show in Table C-7, then adjusting for milestones and accelerating the performance capability of the iSRB, the USA parametric costs for the SLS Booster Stage are within range of bottoms-up estimates.

(b) (4)

[redacted]

APPENDIX D DTI INDEPENDENT ASSESSMENT STUDY REPORT

(b) (4)

[redacted]

DTI Proprietary Information

Davidson Technologies Inc.

(b) (4)

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

(b) (4)

[redacted]

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

Davidson Technologies Inc.

(b) (4)

[redacted]

(b) (4)

[redacted]

(b) (4)

[redacted]

DTI Proprietary Information

Davidson Technologies Inc.

(b) (4)

[redacted]

DTI Proprietary Information

Davidson Technologies Inc.

(b) (4)

[redacted]

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

Davidson Technologies Inc.

(b) (4)

[redacted]

DTI Proprietary Information

Davidson Technologies Inc.

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

Davidson Technologies Inc.

(b) (4)

[redacted]

(b) (4)

[redacted]

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

(b) (4)

[redacted]

(b) (4)

[redacted]

(b) (4)

[redacted]

DTI Proprietary Information

Davidson Technologies Inc.

(b) (4)

[redacted]

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

(b) (4)

[redacted]

(b) (4)

[redacted]

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

(b) (4)

[redacted]

DTI Proprietary Information

Davidson Technologies Inc.

(b) (4)

[redacted]

DTI Proprietary Information

Davidson Technologies Inc.

(b) (4)

[redacted]

Davidson Technologies Inc.

DTI Proprietary Information

Broad Agency Announcement NNM10ZDA001K

Heavy Lift & Propulsion Technology Systems Analysis and Trade Study

June 3, 2011

POC: Steven Davis
(b) (4), (b) (6)

Contents

1 PROBLEM DEFINITION AND BACKGROUND

This report was prepared under a NASA contract for a Heavy Lift and Propulsion Technology Systems Analysis and Trade Study awarded in response to 2010 NASA Broad Agency Announcement NNM10ZDA001K) NASA's overarching motivation for soliciting this input is described in the BAA Announcement:

> *NASA is seeking an innovative path for human space exploration that strengthens its capability to extend human and robotic presence throughout the solar system. The information also may help lay the groundwork for humans to safely reach multiple potential destinations, including asteroids, Lagrange points, the moon and Mars.*

The BAA also implies that, beyond surmounting technical challenges, the affordability of developing domestic heavy lift capability will strongly affect the viability of the program:

> *The focus will be on developing affordable system concepts that may be used by multiple entities, such as the Department of Defense, commercial corporations and international space agencies.*

This document is SpaceX's final report in the Heavy Lift & Propulsion Technology Systems Analysis and Trade Study. The report's main finding is the presentation of an affordable path forward for a super heavy lift system that has applicability to a much wider market than the current plan and is thus sustainable on a recurring basis. The document is structured as follows:

Section 2 contains a market analysis of the commercial, civil, national security, and super heavy lift markets. Section 3 discusses the top-level trade study framework, including the trade study methodology, the ground rules and assumptions, and the key decision attributes used in the study. Section 4 presents a trade tree and performs a top-level juncture analysis using the pre-described decision attributes as the drivers for making recommendations at each juncture. This analysis is designed to be very high-level with recommendations, when possible, made based on mostly qualitative rationale. Section 5 provides a further architecture downselect. The level of these trades is lower than that of Section 4, but not yet low-level enough to be considered a comprehensive downselect. The trades are performed in a similar method to that of Section 4. Section 6 presents a downselected configuration in detail, including trajectory profiles and potential options upgrade options. Section 7 presents a sensitivity analysis on the configuration presented in Section 6. Section 8 describes the ground processing concept of operations for the configuration presented in Section 6. Section 9 switches gears and performs a top-level trade study using weighted metrics on various types of in-space propulsion; the decision attributes qualitatively described in Section 3 are quantitative inputs to the top-level study. Section 10 presents conclusions, including a discussion of cost data and capability gaps. Appendix A shows a compliance matrix between the report and the Statement of Work. Appendix B shows a compliance matrix between the report and the original Broad Agency Announcement.

2 MARKET ANALYSIS

The market consists of commercial, civil, and national security space segments. The following analysis describes the current and projected demand for each segment through 2020, which is summarized in Figure 1. Using Evolved Expendable launch Vehicle (EELV)–class definitions for mass and adjusting NASA and FAA commercial classes to the EELV standard, on average, there are 11 small/medium class launches per year, 21 intermediate class launches per year, and approximately 1 heavy class launch every other year.

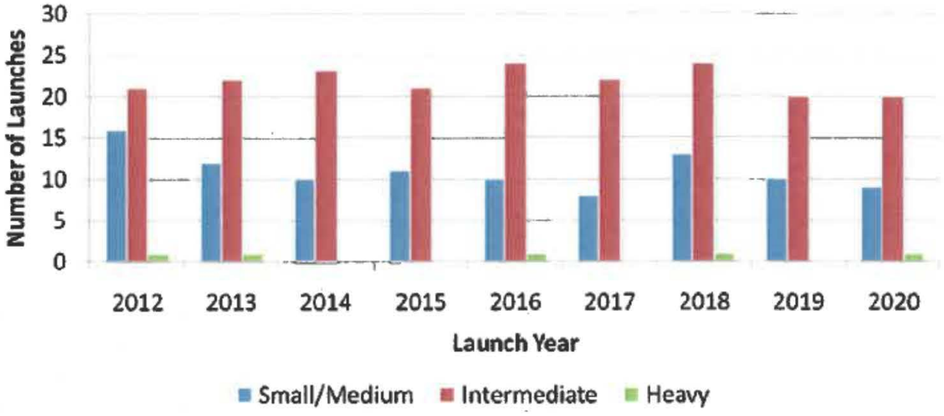


Figure 1: Projected number of launches by mass class

The mass classes are defined as follows:

- Small/Medium: Less than 3860 kg (total payload weight)
- Intermediate : Between 3860 and 8620 kg
- Heavy: Greater than 8620 kg

2.1 Commercial Market Outlook

(b) (3)

[redacted]

(b) (3)

[redacted]

2.2 Civil Market Outlook

(b) (3)

[redacted]

2.3 National Security Space Market Outlook

(b) (3)

[redacted]

(b)(3)

[redacted]

2.4 Super Heavy-Lift Market Outlook

(b)(3)

[redacted]

(b) (3)

[redacted]

(b) (3)

[redacted]

3 TRADE STUDY FRAMEWORK

3.1 Trade Study Methodology

The steps used in performing the trade study were as follows:

1. Establish and understand the study ground rules and assumptions.
2. Clearly state the key decision attributes and rationale behind the attributes.
3. Set up a trade tree and perform a high-level juncture analysis using the decision attributes as the inputs to the decisions at each juncture. The analysis is mostly qualitative with a clear focus on the decision attributes set up in Step 2.
4. Perform lower level trades with similar methodology to those used in Step 3.
5. Use the results from Steps 3 and 4 to create a down-selected baseline configuration that is used as a point solution for additional analysis.
6. Demonstrate the evolvability and configuration options with regard to the point solution.

Steps 1 and 2 are examined below within Section 3 of the report. The remaining steps are examined in subsequent sections.

3.2 Study Ground Rules and Assumptions

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

[redacted]

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

[redacted]

(b) (3)

[redacted]

(b) (3)

[redacted]

4 TRADE TREE AND JUNCTURE ANALYSIS

(b) (3)

[redacted]

(b)(3)

[redacted]

(b)(3)

[redacted]

(b) (3)

[redacted]

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

[redacted]

(b) (3)

[redacted]

5 ADDITIONAL ARCHITECTURE TRADES

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

[redacted]

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

[redacted]

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

[redacted]

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

[redacted]

6 DOWNSELECTED CONFIGURATION

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

[redacted]

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

[redacted]

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

[redacted]

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

[redacted]

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

[redacted]

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

[redacted]

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

[redacted]

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

[redacted]

7 SENSITIVITY ANALYSIS

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

[redacted]

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

[redacted]

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

[redacted]

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

[redacted]

8 CONCEPT OF OPERATIONS

The operations concept for a launch vehicle from manufacturing through test and launch is among the most important drivers for recurring cost. For Falcon 1 and Falcon 9, SpaceX has pursued an organized, fast-paced concept of operations that has figured prominently in lowering launch costs. A similar series of operations is proposed for the heavy and super-heavy lift launch vehicles.

8.1 Manufacturing and Test

Figure 22 shows the major steps in the process flow from manufacturing through transportation of all flight hardware to the launch site. The first-stage tanks will be constructed very close to the launch and integration facilities, at Kennedy Space Center (KSC) or Cape Canaveral Air Force Station (CCAFS). Manufacturing the first stage in Hawthorne was considered but ultimately rejected due to dramatically higher transportation costs for assemblies of this size. The most practical option for transport from the West coast, barge via the Panama Canal, would lead not only to high transportation costs, but also an elongated schedule and the potential for serious weather-related delays. Processing flight assemblies rapidly (while attending to quality control) is a foundational principle at SpaceX because it minimizes the time that capital is tied up in hardware on the ground. For these reasons, building very large components near the launch site presents a substantial advantage.

(b)(4)

Figure 22: Manufacturing & Test Process Flow

A leading candidate for tank manufacturing is the Assembly and Refurbishment Facility (ARF), which could comfortably hold (b)(4) outer diameter first-stage cores simultaneously in the high bay. Additionally, the ARF main doors have ample capacity for first-stage cores. The ARF is conveniently located and would cost significantly less than a new facility. Some alternative facilities are marked in green on Figure 24.

The [redacted] engines for the first stage and strap-on boosters will be shipped from the factory in Hawthorne to a test facility. The preferred choice for [redacted] testing is the SpaceX Rocket Development Facility in McGregor, TX; backup options include a Stennis Space Center test stand or a converted Cape/KSC launch pad.

First-stage tank acceptance testing will occur at a new structural test stand to be built at KSC based on a scaled version of the Falcon 9 first-stage structural test stand. Once tank and subassembly acceptance testing is complete, the [redacted] engines will be delivered to a horizontal integration facility at KSC or CCAFS for integration with the first-stage core thrust structure. The preferred choice for this facility is a new hanger, scaled up from the Falcon 9 integration facility at LC-40. Some backup integration facilities are shown in green in Figure 24, including the Vertical Assembly Building (VAB). Acceptance testing of the first-stage integrated thrust assembly (engines + tank) in the single core configuration will occur at a test stand to be built at KSC or CCAFS, preferably a converted CCAFS/KSC launch pad. Some prospective facilities for testing the first stage are shown in orange in Figure 24. The baseline approach for the triple core vehicle would be to conduct a static-fire test on each core individually, then assemble the three cores to form the first stage

The entire second stage, including engines and all other assemblies easily transported by truck, will be manufactured at SpaceX headquarters in Hawthorne, where Falcon 9 components are currently manufactured. Following the same processes already in place for Falcon 9, the second-stage engine and tanks would be shipped to SpaceX's McGregor, TX, test facility for individual acceptance testing. Assembly of the second-stage engine and tanks would occur in parallel in McGregor. Static-fire testing of booster stages is planned prior to final integration of the integrated launch vehicle. Second-stage testing would occur in the McGregor test stand which has processed two Falcon 9 upper stages to date.

As soon as acceptance testing is complete, each stage will be transported to the same horizontal integration facility used for mating the first stage thrust structure to the tank. Other assemblies such as the interstage and flight termination system would also be received and installed at this facility. Most flight assemblies will be transported using trucks, but the first-stage cores are a special case. Each core will be transported either by rail or road through all stages of processing (manufacturing to final vehicle integration). For the preferred rail option, a scaled-up variant of the Falcon 9 transporter-erector would carry the core, while the road option would use a wheeled transporter such as a KAMAG.

8.2 Integration and Launch

As hardware arrives at the horizontal integration facility, the final integration and launch process flow begins (Figure 23). All flight hardware is planned to arrive by [redacted] The payload will arrive in Florida by [redacted] and be processed in a separate facility. Checkouts of the stage propulsion and avionics systems will be proceed between [redacted] and [redacted] Note that the timeline presented here has substantial margin built in. Following checkouts, the upper stage will be mated to the first stage at [redacted] The next step is to transfer the payload from its processing facility and horizontally mate to the launch vehicle at [redacted]

[redacted]

The fully integrated vehicle is lifted via crane and secured to the transporter-erector 8 days prior to launch. [redacted] At [redacted] the vehicle will be transported to the pad and raised to the vertical position. A brief static-fire test is planned in order to check out the first-stage systems and automated ground systems which load propellant and helium. The static fire could also verify cross-feed functionality, if necessary. Immediately following static fire, the vehicle will return to the integration facility for [redacted] about [redacted] days before launch. After several successful flights, the static-fire process would be phased out, accelerating the launch schedule by about [redacted] days. After [redacted] install, the vehicle will roll out to the pad and go vertical once more, then launch on the following day. At ignition, hold-down pylons restrain the vehicle to confirm that all telemetry parameters are within expected ranges before releasing the vehicle. This check, analogous to an airliner's end-of-runway check, assures the vehicle's reliability before liftoff.

In order to minimize the cost of ground equipment and maintenance, both single- and 3-core vehicles would launch from the same pad. Using one pad for both vehicles will not create a bottleneck because only a few super heavy launches are expected per year. While other launch pads could perhaps be upgraded to accommodate the single-core vehicle, only the LC-39 launch pads are large enough for the super-heavy variant and even then would require modifications. As the designated SLS launch pad, LC-39B is preferred.

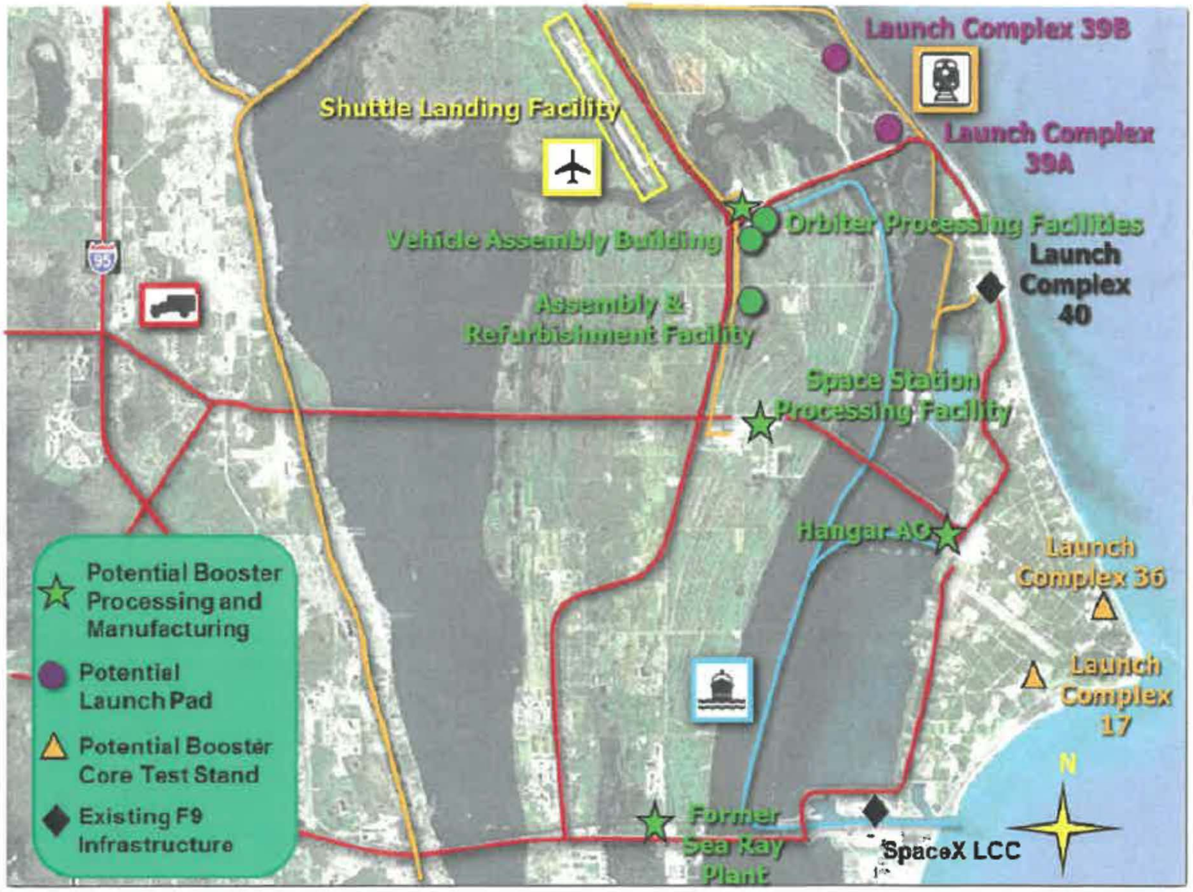


Figure 24: Potential Manufacturing & Launch Facilities

The SpaceX integration and launch flow enables use of a single pad across a robust customer base. With reduced time on the pad for the launch vehicle, multiple users are able to leverage the sharing of large infrastructure sustainment costs. SpaceX has a diverse launch manifest that in the future will include Falcon 9 cargo/crew/commercial, Falcon Heavy commercial/cargo, and (b) (4) (b) (4) launches. In addition, SpaceX vehicles are designed for rapid countdown and launch operations using highly automated countdown sequences as well as quick turnaround in case of a launch scrub.

8.3 Launch Facility Upgrades

With a focus on reliability and low cost, SpaceX will use its experience with three different launch site activation efforts to reduce cost, increase operational availability, and reduce on-pad and pad turnaround times. To the extent that it makes sense, we will leverage existing facilities while not tying ourselves to legacy concepts and expensive heritage operations and maintenance costs. The SpaceX horizontal processing and integration concept streamlines pad infrastructure requirements while enabling robust integration with existing pad systems.

A significant element of the launch operations and vehicle integration effort is a review of the required launch pad, ground system, and facilities and infrastructure required to support a heavy-lift booster. Leveraging the concept of operations described above, the SpaceX team developed a comprehensive set of modifications required to support the launch base processing of the (b) (4) (b) (4) vehicle. The SpaceX-estimated price of launch pad, ground systems, and KSC infrastructure totaled (b) (4) (Table 1), which includes a large amount of margin (b) (4) over the current internal cost estimates. This large margin is included due to general uncertainty, the assumed financial risk in a firm fixed-price contract, and the uncertainty in specific assets on the Range. A roll-up of the necessary upgrades, along with descriptions of those upgrades and any assumptions on existing or future capability that can or will be provided by NASA, is discussed below.

Table 1: Estimated Launch Pad, Ground System, Facilities and Infrastructure Costs

Launch Pad Upgrades	(b) (4)
(b) (4)	
Ground System Upgrades	(b) (4)
(b) (4)	
Infrastructure	(b) (4)
(b) (4)	
Total	(b) (4)

8.3.1 Launch Pad Upgrades

In assessing the required upgrades to LC-39 to support the [redacted], we made a number of assumptions on what work would be accomplished as part of the NASA 21st Century Launch Complex effort or as part of the baseline NASA Heavy Lift program. This enabled SpaceX to bound the work to that specifically required to support the [redacted] concept.

Ground Rules and Assumptions:

SpaceX assumed the primary LC-39 commodity, pneumatics, power, and communications/data systems would be maintained by NASA in the post-shuttle period before a heavy-lift variant is ready to fly. Therefore, the cost estimate for launch pad, ground, and infrastructure improvements is focused on those items unique to the [redacted] vehicle. The estimate includes all costs to achieve initial launch capability but does not include the recurring costs of maintaining the basic pad and supporting systems in ensuing years. It also assumes that NASA will maintain any infrastructure outside the fence line of the pad such as gas and commodity supply lines, communications and data circuits, etc. NASA insight was assumed to be similar to what SpaceX has experienced on the COTS and CRS programs as commercially procured services.

Liquid Oxygen System—LC-39 capacity of approximately 970,000 gallons would support [redacted] The [redacted] core is estimated to require [redacted] The SpaceX estimate includes minor mission-unique modifications to the LOX system storage, pump, and pad interfaces to support [redacted]

Fuel (RP-1) System—SpaceX assumed the existing cross-country lines from the previous RP-1 tank area would be left in place. The SpaceX estimate includes the costs of adding RP-1 storage tanks as well as a fuel pump system and pad interface to support Falcon Heavy and Falcon Super Heavy.

Gaseous Nitrogen and Helium—SpaceX assumed LC-39 would remain connected to the existing gaseous nitrogen and helium pipelines. In order to support SpaceX unique engine spin start and vehicle pressurization requirements, SpaceX included updates to the helium and nitrogen systems to modify the storage, pressurization, and supply system to support both [redacted]

Lightning Protection—SpaceX assumed the existing lightning protection system would be maintained to support any future heavy-lift concept. We did include estimated costs to upgrade the system to meet vehicle-unique electromagnetic interference and lightning susceptibility requirements.

Water Deluge/Acoustic Suppression—SpaceX assumed the existing water system at LC-39 would be maintained and that the capacity available would be ~300,000 gallons. The high-end estimate of water flow rate for the [redacted]. The SpaceX estimate includes upgrades to the existing infrastructure to support the required flow rates as well as modifications to plumbing and orifices to support the necessary acoustic suppression requirements for both variants of the [redacted]

Flame Duct—SpaceX assumed significant modifications to the flame duct would be needed to expand it beyond the current 58-ft width. [redacted] In addition to costs for expanding the flame duct, the estimate includes the cost of potential structural upgrades and flame resistance modifications to accommodate both [redacted]

Launch Pad Electrical Power, Communications, and Data—SpaceX assumed the existing LC-39 infrastructure in these three areas would be maintained to support the future heavy-lift vehicle. Our estimate includes costs to implement the necessary interfaces with our unique transporter-erector and ground support equipment systems. We assume ground and vehicle power will be available and that critical circuits will be supported by uninterruptable power supplies or other backup power means. We assumed the existing communications architecture would support our communication and network data requirements and we would be permitted to use our SpaceX command and control suite to execute test and countdown operations from either a SpaceX unique control center or a shared-use launch control center. We included a cost estimate to complete the network and data interfaces to tie our system into the NASA backbone as well as estimated costs associated with developing any unique payload customer pad interface requirements.

Crew Ingress/Egress--The launch pad upgrades scoped in this study do not include astronaut ingress and egress. It is assumed that any ingress/egress system will be developed separately and may be provided by NASA as part of its 21st Century Launch Complex effort.

8.3.2 Ground Systems

In addition to launch pad upgrades, there are a number of significant ground system upgrades required to support booster and engine test operations. The ground system upgrades included in the cost estimate are a booster tank load testing capability, a large engine test stand, and the transporter-erector used to move the integrated booster from the integration facility to the pad.

Booster Structural Load Test Capability--The SpaceX concept locates manufacturing of the large 20-ft-diameter tanks at or near KSC. As noted above, a primary option for this work is the existing Assembly and Refurbishment Facility at KSC. Once manufacturing is complete, the tanks will need to complete design qualification and acceptance testing, which includes structural strength as well as burst pressure tests. These tests will be performed at a SpaceX-developed structural test stand at or near the stage manufacturing site.

Large Engine Test Stand--In addition to booster tank testing, large-scale engine testing of multi-engine thrust assemblies is required to validate design and to sufficiently check out flight thrust assemblies. This test stand will be capable of longer duration full-stage firings. Primary options for this capability are vacant launch sites on CCAFS such as LC-36 or LC-17. The development of such a test capability is included in the estimated launch operations and integration estimate. It assumes a site would be available on KSC or CCAFS and that a real estate agreement (similar to the Air Force agreement for SpaceX license use of LC-40) can be reached to allow SpaceX use.

Transporter-Erector--The transporter-erector is a major element of ground support equipment and is used to transport the integrated booster stack from its horizontal processing facility to the launch pad and to raise the entire assembly to the vertical launch position once at the pad. (b)(4) (b)(4) (b)(4) The transporter-erector consists of a strong back used to carry the loads of the integrated horizontal stack, coupled with a launch mount that acts as the interface between the rocket and the ground propellant and pneumatics systems. Hold-downs secure the booster to the launch mount prior to going vertical. Upon arrival at the pad, hydraulics is used to raise the entire integrated booster/transporter/erector into the vertical launch configuration (see Figure 23). It is assumed that SpaceX will manufacture the transporter-erector at or near KSC. Costs for upgrades to the hydraulics system at LC-39 are included in the launch pad upgrades estimate.

8.3.3 Facilities and Infrastructure

The final area requiring improvements to support the (b)(4) concept is supporting facilities and infrastructure. The main elements included in this section are the booster tank manufacturing facility, the horizontal integration facility, and any upgrades required to the transportation infrastructure to handle the movement of the booster components.

Booster Stage Manufacturing--Due to the logistics associated with the large first-stage and strap-on boosters, SpaceX plans to manufacture these elements on or near KSC. This location allows us to avoid the large recurring costs of rail or ship transport of these major assemblies to Florida. Our baseline is the Assembly and Refurbishment Facility on KSC as it is sized for this type of work and resides in close proximity to the Vehicle Assembly Building and launch pad. The facilities and infrastructure cost estimate includes the cost of upgrading an existing facility on or off KSC to accommodate booster stage production requirements.

Horizontal Integration Facility--The baseline horizontal integration facility will be a SpaceX facility on KSC where the (b)(4) will be assembled and tested before rolling to the launch pad. As mentioned, an alternate integration capability is available using the transfer aisle of the Vehicle Assembly Building. The integration facility will be based on existing facilities designed and built at our Cape LC-40 and Vandenberg SLC-4East launch sites, which allow for testing of individual stages, mating of the first stage and strap-on cores, and lift of the integrated booster assembly into the transporter/erector.

Transportation Upgrades—Modernization of the basic transportation infrastructure may be required to support the movement of large launch vehicle assemblies, and related costs are included in the launch and integration cost estimate. In particular, modifications may be required to the pad access roads to accommodate the SpaceX horizontal transporter-erector. This could be the addition of rails from the horizontal integration facility to the pad or a modification to the road infrastructure to handle a wheeled transporter. In addition, specialized ground support equipment will be required to move booster assemblies from the manufacturing facility to the structural and engine test stands; a cost estimate is included in the scope of this effort.

In summary, the [redacted] launch operations and integration concept described here is aimed at achieving a reliable, low-cost heavy-lift capability while achieving increased operational availability and flexibility.

9 IN-SPACE PROPULSION

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

[redacted]

(b) (3)

[redacted]

(b) (3)

[redacted]

(b)(3)

[redacted]

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

[redacted]

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

[redacted]

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

[redacted]

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

[redacted]

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

[redacted]

10 CONCLUSIONS

10.1 Cost Discussion

10.1.1 Development Costs

(b)(4) This would be the price in a firm fixed-price contract; thus, SpaceX's internal cost estimates are actually much lower, with the final price reflecting an increase over cost to account for uncertainty and inherited risk.

The (b)(4) price includes all design, development, test, and evaluation; systems engineering; launch site, test facility, and manufacturing facility upgrades; and an HLV test flight. The breakdown (with systems engineering, systems integration, and management costs built in) is approximately:

(b)(4)

The expected development costs as a function of time are shown below in Figure 30. These cost values have been broken down into much more detail and shared with NASA personnel, but that proprietary data is not included here.

(b)(4)

Figure 30: Estimated SLV development costs as a function of time

To support the cost estimates above, Table 5 shows empirical data points of SpaceX's previous development costs ("blank-sheet to operational"):

Table 5: SpaceX Empirical Cost Data Points

(b)(4) [redacted]

SpaceX's total company expenditures since inception in 2002 are approximately (b)(4) [redacted]. This value includes all of the developments above, as well as the cost of SpaceX's seven launches as of May 2011. SpaceX is aware that the (b)(4) [redacted] is likely not consistent with the standard costing models. However, these costing tools are not yet calibrated to SpaceX's demonstrated capabilities and culture. As an example, one standard tool estimated that Falcon 9 development would cost (b)(4) [redacted] when in fact it cost (b)(4) [redacted].

10.1.2 Recurring Costs

As discussed in this study, recurring costs are minimized through simple design and maximum commonality across vehicles. This can be seen by examining three launch vehicles: Falcon 9, (b)(4) [redacted] on the assumption that the (b)(4) [redacted]

(b)(4) [redacted]

The production rate, given the development and facilities costs quoted in this study, would be on the order of (b)(4) [redacted]

10.2 Capability Gaps

In performing the trade study, several capability gaps were identified, most of which were mentioned in the analyses in previous sections. However, with regards to the recommended configurations, it is useful to consolidate the key capability gaps in one section. Note that these are relevant to the flight vehicle with ground and testing needs discussed in Section 8. With regards to technical readiness, all items listed in the table below are based on heritage and/or standard technologies such that SpaceX believes the underlying technical risk of said developments are low. However, with the exception of ID 5, the actual physical demonstration of the heavy-lift versions of these items has not yet begun.

Table 6: Key Capability Gaps Identified During Study

(b) (4) [redacted]

Due to the suggested use of the current (b) (4) [redacted] along with common Avionics and software, the number of capability gaps in the baseline, non-upgraded vehicle is much smaller than was initially expected.. This results in a configuration which minimizes development time.

10.3 Concluding Remarks

The all (b) (4) [redacted] configuration provides a payload capability of (b) (4) [redacted] to LEO, with potential performance upgrades to (b) (4) [redacted] The configuration minimizes cost and development time while maximizing reliability and sustainability through:

- Common propellants between stages.
- (b) (4) [redacted]
-
-

APPENDIX A: SOW COMPLIANCE MATRIX

ID	SOW Text (sub-bullets identified)	SpaceX Compliance
1	Identify and define the market.	Section 2 of Final Report
2	[sub-bullet] Identify National Aeronautics and Space Administration (NASA), other United States (US) Government, US Commercial, and international launch opportunities.	Section 2 of Final Report
3	Identify, review and document ground rules and assumptions, as well as identify how alternative ground rules impact the identified alternative system.	Section 3 of Final Report
4	Define the top-level trade space.	Sections 3 and 4 of Final Report
5	[sub-bullet] Identify and analyze alternative launch vehicles meeting the objectives of a heavy lift system.	Sections 6 and 7 of Final Report
6	[sub-bullet] Define and weigh a series of decision attributes (with rationale) and establish metrics for quantitative assessment.	Sections 3 and 4 of Final Report
7	[sub-bullet] Update trade study models.	Sections 3 through 7 of Final Report
8	Perform system analysis and trade studies identifying how alternative heavy lift system solutions address key decision attributes.	Sections 4 through 7 of Final Report
9	Downselect designs for further study.	Section 6 of Final Report
10	Analyze how changes to the weighting affect the outcome of the architecture assessment.	Section 5.4 of Final Report
11	Conduct trade studies designed to optimize first and upper stages (including in-space stages and components) of heavy lift systems.	Sections 4 through 6 and Section 9 of Final Report
12	Perform sensitivity analyses.	Section 7 of Final Report
13	Identify capability gaps discovered during the trade study and systems analyses on the first and upper stages (including in space stages and components) and during any other trade studies conducted as a result of examining the heavy lift trade space.	Sections 4, 5, 6, 8, and 10 of Final Report
14	Determine an estimated overall lifecycle cost, development schedule and production rate.	Section 10 of Final Report
15	Where suitable, identify opportunities for space flight experiments of in-space propulsion elements	Section 9 of Final Report
16	Support weekly telephone status meetings as required.	Supported over course of BAA Process
17	Prepare briefing packages describing the status of the study and results to date for each of the Technical Interchange Meetings (TIMs) in accordance with DRD 1382MA-002.	TIM-1 and TIM-2 briefings
18	Submit a Final Study Report within six months from authorization to proceed effective date in accordance with DRD 1382MA-003.	Final Report
19	[sub-bullet] Analyze impact of ground rules and assumptions.	Sections 3 through 7 of Final Report

APPENDIX B: BAA COMPLIANCE MATRIX

ID	BAA Text	SpaceX Compliance
1	Provide a recommended list of key decision attributes and rationale associated with each. As a point of reference, NASA's Heavy Lift Launch Vehicle study utilized the following list of system attributes: Life-cycle cost; operability-support manifest launch rate; safety and reliability; performance (mass, delivery orbit); schedule – initial human flight; extensibility to support Modified Flex Path Missions.	SOW ID's 3 and 4
2	Provide a recommendation for the weighting of the recommended key decision attributes.	SOW ID's 3 and 4
3	Identify how changes to the weighting of key decision attributes affect the architectures.	SOW ID 10
4	Identify how alternative ground rules and assumptions (Reference NASA HLLV Study) impact the identified alternative system solutions.	Sections 3 through 7 of Final Report (no 1-to-1 correspondence with SOW)
5	Identify how innovative or non-traditional processes or technologies can be applied to the Heavy Lift Systems to dramatically improve its affordability and sustainability.	Sections 6, 8, and 10 of Final Report (no 1-to-1 correspondence with SOW)
6	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.	SOW ID's 1 and 8
7	Identify how incremental development testing, including ground and flight testing, of Heavy Lift System elements can enhance the heavy lift system development.	Sections 3 through 8 of Final Report (no 1-to-1 correspondence with SOW)
8	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).	SOW ID 13
9	Identify capability gaps associated with the first-stage main engine functional performance and programmatic characteristics required to support each Heavy Lift System studied.	SOW ID 13
10	Identify capability gaps associated with the upper-stage main engine functional performance and programmatic characteristics required to support each heavy lift system studied.	SOW ID 13
11	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.	SOW ID 13
12	Identify capability gaps associated with the in-space space propulsion elements functional performance and programmatic characteristics required to support each Heavy Lift System studied.	SOW ID 15
13	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.	SOW ID 15
14	Identify what in-space space propulsion elements, if any, which should be demonstrated via space flight experiments.	SOW ID 15