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Systems Architecting Methodology for Space Transportation Infrastructure

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This paper presents a systems architecting methodology for comprehensive but transparent exploration of available options for future space transportation infrastructure. Evaluation of launch architectures requires the assessment of options over a range of dimensions, which can be broadly grouped into technical performance, time to initial capability, cost, and satisfaction of stakeholders' various needs. The challenge is to fairly compare a broad range of architectures across these dimensions. This paper describes the methodology as applied to a super heavy lift launch infrastructure. The paper investigates the trade-offs associated with stage propellant selection, launch vehicle configuration and other relevant design parameters. The study considers potential LEO-class vehicles derived from the baseline vehicle to deliver early benefit from the heavy lift vehicle and provide an ongoing affordable LEO service. The technical assessment methodology is validated against existing launch vehicles. The paper demonstrates how a field of 192 possible launch vehicles can be transparently reduced to seven possible designs on technical considerations, and how further narrowing the design space requires weighting competing stakeholder priorities. The paper further shows how coarse

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tradespace exploration early in the process can inform decision-making on future launch developments. The paper closes with suggestions for future work in the area.

Nomenclature

ΔV_{1a}	=	Boosters Delta V during Vertical Ascent (m/s)
ΔV_{1b}	=	Boosters Delta V after Vertical Ascent (m/s)
ΔV_{2}	=	Upper Stage Delta V (m/s)
ΔV_{ca}	=	Core Delta V during Vertical Ascent (m/s)
ΔV_{cb}	=	Core Delta V after Vertical Ascent until Boosters Burnout (m/s)
ΔV_{cc}	=	Core Delta V after Boosters Burnout (m/s)
ΔV_{pen}	=	Aerodynamic and Gravity Delta V Penalty (m/s)
ΔV_{req}	=	Total Required Delta V (m/s)
$GLOM$	=	Gross Lift-Off Mass (mt)
GR	=	Payload Gear Ratio
m_c	=	Total Core Stage Mass (mt)
m_{d_i}	=	i-th Stage Total Inert Mass (mt)
m_{d2}	=	Upper Stage Dry Mass (mt)
m_{p2}	=	Upper Stage Propellant Mass (mt)
m_{db}	=	Boosters Dry Mass (mt)
m_{pb}	=	Boosters Propellant Mass (mt)
m_{dc}	=	Core Stage Dry Mass (mt)
m_{pc}	=	Core Stage Propellant Mass (mt)
m_{pi}	=	i-th Stage Propellant Mass (mt)
m_{pl}	=	Payload Mass (mt)
μ_i	=	i-th Stage Inert Mass Fraction
η	=	Design Thrust/Weight Ratio

$\vec{p}$ = Parameters Vector

T_b = Booster Thrust (kN)

T_c = Core Stage Thrust (kN)

t_b = Boosters Burn Time (s)

t_c = Core Stage Burn Time (s)

VAB = Vehicle Assembly Building

$V_\oplus$ = Earth Equatorial Rotational Velocity (m/s)

V_{esc} = Escape Velocity (m/s)

$V_{staging}$ = Design Boosters Staging Velocity (m/s)

$\vec{x}$ = Design Vector

I. Introduction

As the Space Shuttle Program ended, NASA faced the challenge of developing a new system for affordable access to space. In February 2010, the FY11 NASA budget proposal drew a new direction for the American human spaceflight program. The proposal outlined the cancellation of the Constellation program, and the launch vehicles planned therein, namely the Ares I and Ares V vehicles. A new launch vehicle will need to be developed to support any substantial human exploration beyond LEO, whether to an asteroid, the Moon or Mars [1]. Ideally, the launch vehicle would also bring astronauts to the International Space Station and perform heavy lift to LEO.

Notwithstanding the missions that will constitute the new plans, the future human exploration program will require payload capabilities significantly greater than the ones offered by current transportation capabilities. The human mission to a Near Earth Object mentioned by President Obama as a milestone of the future program [2] will be no exception to this. Furthermore, in 2009, the Augustine Committee recognized the need for heavy lift capability, and the existing uncertainty on what will be the "smallest largest" payload in terms of mass and volume to be carried by the future vehicle – that is, the largest monolithic payload to be carried by a single flight.

This paper describes a systems architecting methodology for space transportation infrastructure, with an application to heavy launch vehicles for future human exploration beyond Low Earth Orbit.

Making an informed decision on launch vehicle development requires the understanding of the interconnection between each architectural decision. This requires insights on the global influence of each architectural decision on the launch system and its supporting infrastructure. Research in academic literature focused mainly on launch vehicle optimization, i.e. the optimization of a launch vehicle point design at a system and sub-system level to perform a set of pre-defined trajectories [3,4,5,6,7]. Such literature discusses coupling between design decisions (such as propellant allocation per launch vehicle stage) and operations (such as trajectory design). A further traditional focus of research in the last 40 years has been related to trajectory optimization in guidance, navigation and control, where trajectories are optimized for a given launch vehicle system [8,9,10,11,12,13].

Traditional optimization studies, as such, require an a priori definition of the launch vehicle architecture. Decisions such as booster diameter, hardware commonality, propellant type and engine selection are usually not considered as part of the optimization routines. Those decisions, however, are at the heart of the current debate on future launch vehicles. Hardware commonality has been a key element in the development of modern launch vehicle infrastructure, such as seen in the Delta [14] and Atlas [15] launch vehicle families. Product platforming and hardware commonality are cost-effective means to reduce development costs and risks in complex programs [16, 17]; researchers developed methods to identify and evaluate opportunities for hardware commonality in aerospace portfolios [18], while recognizing limitations of platforming in divergence and lifecycle offsets [19] and analyzing incentive structures to implement commonality within large programs [20]. Examples of hardware commonality assessment studies can be found, for instance, in the analysis of space exploration infrastructure [21] and families of new aircraft systems [22]. Commonality is also included as explicit consideration in pre-phase A systems architecting studies, during the identification of concepts of interest for further decision-making and design efforts. Comprehensive system architecting requires tools to support decision makers by conducting comprehensive analysis of the design space. In recent years, researchers at MIT developed a set of tools for quantitative analysis of system architectures, such as the Algebra of Systems (AoS) [23] and the Architectural Decision Graph (ADG) Framework [24]. AoS and ADG are formal meta-languages to generate and evaluate system architectures and identify architectural insights of interest to decision makers.

This paper builds on this line of research and complements the ADG Framework with a hybrid optimization approach applied to the systems architecting problem of heavy lift transportation systems. The work focused in understanding the architectural implications of high-level decisions in architecting a launch vehicle from a technical performance, program management and stakeholder satisfaction perspectives, and developed a model for its design space exploration.

The paper analyzes the trade-offs associated with launch vehicle sizing, propellant type, engine selection and ground operations infrastructure. It further considers how early payload capabilities can be derived by considering down-ward derivative vehicles sharing components with the main baseline of the heavy lift vehicle. The paper is structured as follows. Section II presents the study formulation, assumptions and evaluation metrics for performance, cost, policy risk and hardware commonality. Section III presents the hybrid optimization approach that has been developed for tradespace exploration of launch vehicles and its validation against existing systems with known performance. Section IV presents the results obtained by applying the framework to the human space exploration study and discussing tradeoffs of interest for consideration from decision-makers. Section V draws conclusions on the study and outlines avenues of future research.

II. Study Formulation

We consider a scenario where a payload capability of 30 metric tons to Earth Escape orbit is required for future exploration missions. As discussed in Section V of this report, further research will be required to assess the impact of this parameter in the selection of an architecture. A transportation system with a 30mt payload capability to Escape qualifies as "super" heavy, since existing heavy lift expendable launch vehicles are able to lift at most 9.3mt to Earth Escape orbit (such as the Delta IV Heavy [14]). Being 30mt about 3 times the maximum capacity allowed by existing assets, a new launch vehicle will most likely be required to fulfill the desired payload capacity requirement.

A. Architectural Variables

The following paragraphs describe the architectural variables investigated in this study. A structural morphological matrix [24] summarizing the architectural decisions for the expendable super heavy lift architecture is presented in Table 1. An architecture is defined by selecting one option on each row of the matrix. Compatibility constraints prune infeasible architectures from the design space are shown in Table 2. The feasible design space resulting from the application of architectural constraints consists in 43 possible launch vehicle architectures per vehicle diameter (129 total).

1. Configuration

Three configurations are being considered in this study. The first configuration is a 2.5 stage to orbit expendable vehicle, composed of a first stage (a core and strap-on boosters) and an upper stage. The main elements of the launch vehicle configuration are shown in Figure 1. The core and the strap-on boosters are operated in parallel staging mode to reduce the size of the engines required for lift-off and to allow a check-out of all the first stage engines before committing the vehicle to lift-off. The second configuration is a three-stage launch vehicle operating in serial staging mode (Figure 2), analogous to the Saturn V of the Apollo era. The third configuration is a clustered launch vehicle (Figure 3), where multiple identical stages are stacked horizontally. Stage modules are serially staged three at a time to maintain thrust symmetry. This configuration implies the need for air starts of engines in successive stages of the burn sequence.

2. Booster Diameter

Three possible stage diameter options are considered: 5m (such as Delta IV), 7m (an option considered in a Boeing study for growth evolution of the Delta launch vehicle family11)) and 8.4m (such as the External Tank of the Space Shuttle). This study considers stages with equal diameter within each configuration. Hammerhead fairings are considered with a diameter that is 1.5 times the stage diameter (this limit being dictated by structural constraints). A reference 25m payload fairing length is assumed. A thrust/weight ratio of 1.2 at launch is assumed for each baseline vehicle. A minimum thrust/weight ratio of 1.0 is assumed at Booster Engine Cut-Off (BECO) for each parallel-staged vehicle.

3. Hardware Commonality

Platforming options include hardware commonality in first stage booster engines (referred to as Common Engines - CE), and commonality in booster stages (referred to as Common Boosters - CB) or both.

4. Propellant Types

Four propellant combinations have been considered for the boosters, both liquid and solid. Liquid propellants include LOX/LH2 and LOX/RP1; solid propellants include SRB 4 segments (PBAN) and SRB 5 segments (HTPB). Propellant options for the core stage include LOX/LH2 and LOX/RP1; LOX/LH2 and LOX/CH4 have been considered for the upper stage.

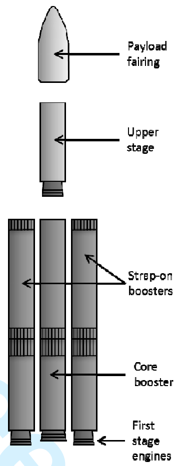

Figure 1 Parallel Staging Configuration

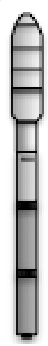

Figure 2 Serial Staging Configuration

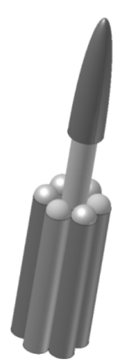

Figure 3 Clustered Staging Configuration

Table 1 Structural morphological matrix for expendable super heavy lift launch vehicles

Architectural Decisions	Alternatives				
Configuration	Serial	Parallel	Clustered		
Booster diameter	5m	7m	8.4m		
Booster type	Liquid (LOX RP1)	Liquid (same as core)	SRB 4 seg.	SRB 5 seg.	No booster (serial st.)
Common Engines	Yes	No			
Common Boosters	Yes	No			
Core stage prop. type	LOX RP1	LOX LH2			
Upper stage prop. type	LOX LH2	LOX CH4			
Third stage prop. type	LOX LH2	LOX CH4	No 3rd s. (parallel st.)		

Table 2 Constraints for architectural enumeration

Constraint	Motivation
Third stage prop. type = "No 3rd s. (parallel st.)" if Staging mode = "Parallel"	No 3rd stage in parallel staging configurations.
Third stage prop. type != "No 3rd s. (parallel st.)" if Staging mode = "Serial"	Need for 3rd stage in serial staging configurations.
Third stage prop. type = "No 3rd s. (parallel st.)" if Staging mode = "Clustered"	No 3rd stage in clustered staging configurations.
Common Engines = "No" if Booster Type = "SRB 4 Seg."	No common engines if booster is SRB 4 segments.
Common Engines = "No" if Booster Type = "SRB 5 Seg."	No common engines if booster is SRB 5 segments.
Common Boosters = "No" if Booster Type = "SRB 4 Seg."	No common core/boosters if booster is SRB 4 segments.
Common Boosters = "No" if Booster Type = "SRB 5 Seg."	No common core/boosters if booster is SRB 5 segments.

B. Evaluation Metrics

The evaluation metrics considered for this study are Payload Gear Ratio, Gross Lift-off Mass, Vehicle Height, Vehicle Width, Vehicle Aspect Ratio, Core/Booster Thrust, Total Lifecycle Cost, Feasible Downward Derivative and Stakeholder Compliance.

Payload Gear Ratio is an efficiency metric for mass performance and is defined as the ratio between the Gross Lift-off Mass (GLOM) and payload mass:

$$GR = \frac{GLOM}{m_{pl}} = \frac{m_b + m_{db} + m_c + m_{dc} + m_2 + m_{d2} + m_f + m_{pl}}{m_{pl}} \quad (1)$$

GLOM is also used as a first-order proxy for total lifecycle cost of the launch vehicle. Dry masses m_{d_i} are assumed proportional to total stage mass:

$$m_{d_i} = \mu_i \left(m_{d_i} + m_{p_i} \right) \quad (2)$$

Where μ_i depends on the propellant type employed on the stage.

Vehicle height is defined as the sum of the stages height and the fairing. *Stage length* is estimated by calculating the oxidizer and fuel volume, using an average fuel and oxidizer densities, and an overhead to take in account for the tankage and interstages. Oxidizer and fuel mass for each stage are estimated assuming an optimal mixture ratio for the propellant combination assumed in the vehicle architecture.

Vehicle width is defined as the horizontal envelope of the vehicle. Height and width are proxies for ground support infrastructure cost (driving the sizing of the Vehicle Assembly Building required for ground processing, while assuming vertical integration of the launch stack). Height and width are also used to assess whether existing Vehicle Assembly facilities could be used for the architectures being analyzed. Vehicles exceeding the dimensions of available infrastructure (such as the VAB at Kennedy Space Center) require investment in new assembly buildings, therefore driving up the overall cost of the architecture.

Vehicle Aspect Ratio is defined as the height/width ratio of the vehicle and is used to assess whether the architecture results in a feasible vehicle from an aerodynamic and structural perspective. A high aspect ratio results in a "pencil-shaped" vehicle, which is likely to be infeasible from a structural and dynamic stability perspective, whereas a low aspect ratio results in an aerodynamically inefficient vehicle.

Core Thrust and Booster Thrust are used as proxies for Design, Development, Test & Evaluation (DDT&E) and recurring costs associated with the engines. This evaluation metric is also a measure of feasibility of the proposed launch vehicle architectures. It is assumed that engine layouts are feasible only if their envelope diameter does not exceed stage diameter by more than 33% (assuming a stage skirt with the same aspect ratio of the one employed in the Saturn V launch vehicle). Engine layouts considered are shown in Figure 4. An additional metric considered in the assessment of derivative vehicle is Booster

Engine Cut-off (BECO) Core Thrust/Weight ratio, defined as vehicle thrust at booster cut-off, and used to estimate whether core engines alone comply with thrust/weight constraints when booster thrust is depleted.

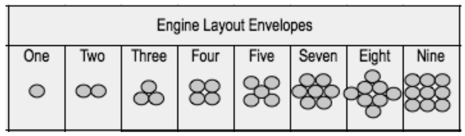

Figure 4 Engine Layouts considered

Thrust per stage is distributed among one or more engines, which layout is driven by thrust symmetry considerations: engines must fit within the stage diameter, while providing a symmetrical thrust. These constraints allow a maximum thrust level to be determined for each stage diameter, limited by current technology constraints and propellant selection decisions.

For parallel staging architectures we assume that booster/core engines have a throttling profile such as the one shown in Figure 5. In this scenario, boosters provide 100% thrust until burnout, whereas core engines are throttled back to 60% in the 10%-75% portion of the burning time profile (time in the Figure is normalized to Main Engine Cut-Off time, MECO). This throttling profile resembles that in existing launch vehicles [14,15]. It optimizes propellant expenditure since core propellant is not used to accelerate inert mass associated with the boosters. This level of fidelity is appropriate as the framework is concerned with the identification of architectures of interest for more detailed study. However, designers need to refine vehicle sizing on selected architectures obtained by the analysis in this framework with more detailed trajectory analysis and consideration of trajectory constraints (such as dynamic pressure and staging velocity).

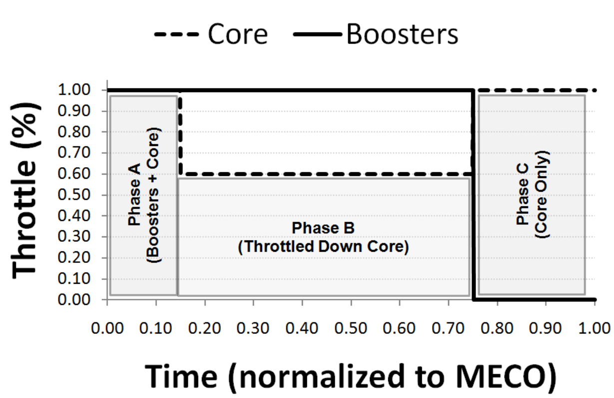

Figure 5 Assumed Throttling Profile

Total Lifecycle Cost is estimated using a proxy variable. Lacking the data resources required to create detailed Cost Estimating Relationships (CER) we rely on the engineering adage "parts aggregate cost". The proxy for development cost is the unique number of development projects required, and the operating cost proxy is unique number of elements to operate. For development costing architectural elements are appropriately discounted if they already exist (e.g. SRBs), exist but require modification (e.g. increased thrust rating on an existing engine), or once existed but the manufacturing lines are now closed (e.g. the F-1 engine). The discounted values are shown in Table 3. A discount is also awarded for common elements in both development and operating costs. To create these cost proxies the following elements of each architecture are considered: upper stage and engine, second stage and engine (for serial rockets only), first or core stage and engine, booster stage and engine (for parallel rockets only), and launch infrastructure. Figure 6 shows an example of an operation project accounting for parallel staged rockets. Similar schemes were applied for serial rockets, and development cost accounting.

Table 3 Cost Proxy Discounts

	Open Line	Closed Line	No Previous Line
Existing hardware	0.2	0.4	-
Modified hardware	0.6	0.8	-
New Hardware	-	-	1.0

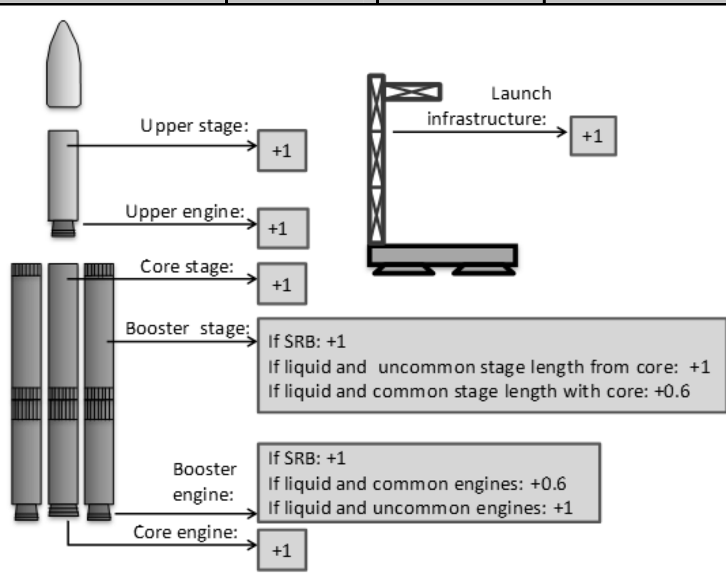

Figure 6 Sample Cost Proxy: Parallel Staging Vehicle, Operation Project Accounting

Stakeholder Compliance is a proxy metric for Policy Risk and it is measured by the value of the launch vehicle program to 3 abstracted key stakeholders: partners that have direct interest in launch capability including the National Security Space community, Commercial space companies, and International Space Partners (hereafter "Partners"), the U.S. congress that allocates funding (hereafter "Congress"), and the executive branch of government consisting of both the White House and NASA Headquarters that set goals for NASA (hereafter "Executive"). We identify each stakeholder's needs from a launch vehicle family and evaluate the expected satisfaction of these needs for each individual proposed launch vehicle architecture. The Partners are satisfied by a vehicle whose derivative (core and upper stages) provides a useful payload range. They also benefit from the opportunity to provide the propulsion system with national industrial base assets. Congress benefits from an architecture that meets the payload and industrial base requirements specified in the FY2011 Authorization Act. Executive benefits from using specific engine technology and a payload size that is useful to Partners to fulfill strategy articulated in the NASA FY2011 Budget Request. A "stoplight" categorization is made to categorize satisfaction of each stakeholder. Dark gray indicates no satisfaction, light gray is partial, and white is full satisfaction of key identified needs for that stakeholder. Details are shown in Table 4.

Mission Requirements are defined by the change in velocity required to achieve Earth Escape orbit. The required change in velocity is estimated as:

$$\Delta V_{req} = V_{esc} + V_{pen} - V_{\oplus} = \sqrt{\frac{2\mu_{\oplus}}{R_{\oplus}^2}} + \Delta V_{pen} - V_{\oplus} \quad (3)$$

The penalty term ΔV_{pen} in Eq. 2 is the sum of the additional change in velocity required to overcome gravity and aerodynamic losses. This study considers a constant penalty term of ΔV_{pen} =2,000 m/s, assuming that the vehicles perform a gravity-turn direct ascent to orbit. This represents a worst-case assumption, since a typical value for expendable two stages to orbit configurations is ~1,500±500 m/s.

Architecture feasibility is defined by setting three level tolerance thresholds across the metrics considered (Table 5). Criteria are defined with respect to existing capabilities (such as existing vehicles and existing engines). They define first-order screening rules to prune out infeasible vehicles from the design space.

Each criteria may be violated (and thus assigned red) when the resulting launch vehicle is infeasible; the launch vehicle is taken out of consideration. It may be nearly violated (light gray flag) when the design operates near the constraint of the criterion – this counts against the architecture but it remains a possible option left in the design space for further analysis. Otherwise, the architecture can pass the criteria, being well within the corresponding constraint, and therefore is assigned a white flag. A color-coded "stoplight chart" summarizes the results of the analysis and allows the identification of architectural tradeoffs.

Table 4 Evaluation Framework for Stakeholder Satisfaction

	Partnerships (NSS, Commercial, International)		
S/H Need	**Fully Satisfied**	**Partially Satisfied**	**Not Satisfied**
Use of Vehicle (Payload to LEO) (C+U)	25 – 35 mt	20 – 25 mt or 35 – 45 mt	< 20 mt or > 45 mt
Use of Engine for RP1 & Size of RD-180 for retrofit (NSS only)	RD-180 engines (option for dev of new engine)	Uses RS-68 engines	Other
	Congressional Guidance (NASA FY2011 Authorization Act)		
S/H Need	**Fully Satisfied**	**Partially Satisfied**	**Not Satisfied**
Payload Capacity	115 – 130+ mt to LEO_baseline	100 – 115 mt to LEO baseline	< 100 mt to LEO baseline
Engine Industrial Base	SSME or RS-68 AND SRB	SSME or RS-68, OR SRB	No US engine
	Executive Guidance (NASA FY2011 Budget)		
S/H Need	**Fully Satisfied**	**Partially Satisfied**	**Not Satisfied**
International and/or Commercial Opportunities	25 – 35 mt	20 – 25 mt	< 20 mt or > 35 mt
Engine Technology	Hydrocarbon size of RD-180	Hydrocarbon Core	Hydrogen Core

Table 5 Criteria for Launch Vehicle architecture feasibility

			Architecture feasibility criteria	
Metrics	**Envelope Basis**	**Fully Satisfied**	**Partially Satisfied**	**Not Satisfied**
Height	Height of the Vehicle Assembly Building (VAB) at KSC.	< 0.9 VAB (125.1)	125.1m – 152.9m	> 1.1 VAB (152.9m)
Width	Space Shuttle Orbiter Wing Span.	< 0.9 Shuttle Wing Span	21.6m – 26.4m	> 1.1 Shuttle Wing Span

		(21.6m)		(26.4m)
Aspect Ratio (AR)	Delta IV Heavy	< 0.8 AR Delta IV (14.4)	14.4 – 17.3	> 1.2 AR Delta IV (17.3)
GLOM	Saturn V	< 0.6 Saturn V GLOM (1800mt)	1800mt – 2400mt	> 0.8 Saturn V GLOM (2400mt)
Core Thrust	Existing Engines (RD-180, SSME, RS-68, J2-X)	<= Existing Engine Thrust	Existing Engine Thrust – 1.2 Existing Engine Thrust	> 1.2 Existing Engine Thrust
BECO Core Thrust	T/W > 1	> 1.0 T/W at BECO	0.8 T/W at BECO – 1.0 T/W at BECO	< 0.8 T/W at BECO

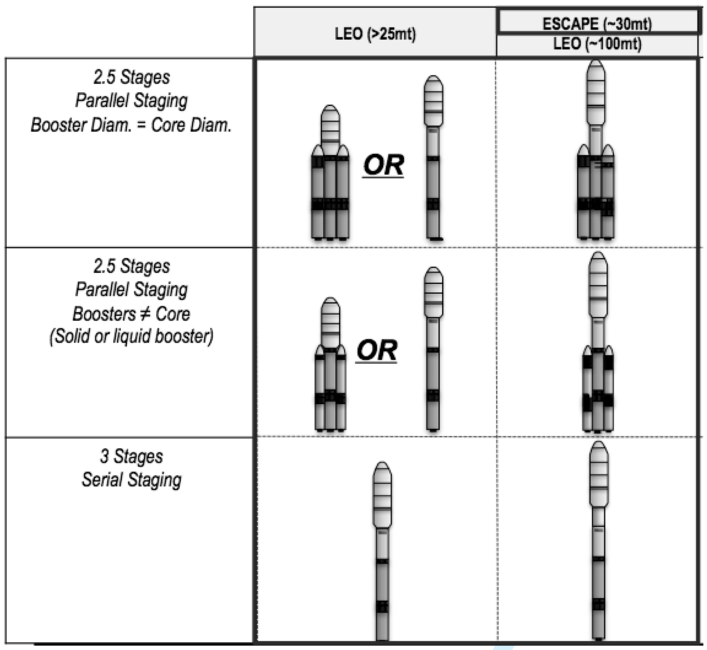

Figure 7 Downward derivative strategies

Figure 7 shows the downward derivative strategies that have been included in this study. Derivatives of parallel-staged vehicles (with either liquid or solid boosters) include a core/upper stage variant and a core/boosters variant. Similarly, a first/second stage variant can be derived from a three-stage serial vehicle. Downward derivatives are of interest to assess evolutionary paths for the development of the heavy lifter, assuming that early payload capability is desired (for instance, for ~25mt-class payloads to LEO). Table 6 shows the payload to LEO criteria used to evaluate downward derivatives. Payload gear ratio efficiency of the vehicle optimized for an Escape mission is compared to the efficiency of the same vehicle configuration optimized to LEO as an additional evaluation metric.

Table 6 Criteria for derivative evaluation

		Derivatives feasibility criteria	
Metrics	**Fully Satisfied**	**Partially Satisfied**	**Not Satisfied**
Payload to LEO Booster-Core	25-35mt to LEO	20-25mt or > 35mt to LEO	<20mt to LEO
Payload to LEO Core-Upper	25-35mt to LEO	20-25mt or 35-45mt to LEO	<20mt or >45mt to LEO
Baseline Vehicle to LEO	<10% inefficient	10-20% inefficient	>20% inefficient compared to similar optimized vehicle

III. Hybrid Optimization Approach for Preliminary Sizing of Launch Vehicles

We propose a hybrid optimization approach for optimal sizing of parallel-staging vehicle architectures based on assumptions on the maximum staging velocity and the throttling profile.

While optimal sizing of a serial-staging vehicle is straightforward and can be solved with gradient-based optimization methods or by the use of a Lagrangian formulation [26], parallel-staging vehicles require more complex trajectory simulations, consideration of booster/core burn times and optimization of throttling profiles (while respecting dynamic pressure, staging velocity and other types of constraints).

The proposed hybrid optimization framework couples a heuristic algorithm (i.e. genetic algorithms [27]) and a gradient-based algorithm (i.e. Sequential Quadratic Programming [28]) to solve the optimization problem for a parallel-staging launch vehicle architecture. The heuristic algorithm provides breadth in the exploration by identifying “sweet spots” in the objectives function vector. The gradient-based algorithm provides depth by refining the solution provided by the heuristic algorithm. A hybrid formulation is required in presence of non-convex objective functions, where local minima lead gradient-based algorithms to identify suboptimal solutions. In modeling parallel staging launch vehicles, the objective function featured local minima and instability of solutions (i.e. different solutions obtained for different initial conditions) with pure gradient-based optimization attempts. An optimization problem is generally defined as:

$$\begin{aligned} \min \quad & \vec{J}(\vec{x},\vec{p}) \\ \text{s.t.} \quad & h(\vec{x},\vec{p})=0 \qquad (4) \\ & g(\vec{x},\vec{p})=0 \end{aligned}$$

Where $\vec{x}$ is the input design vector the components of which represent propellant allocations for each launch vehicle stage, and $\vec{p}$ is the parameters vector describing the architecture. The parameters vector represents a possible architecture as defined in the structural morphological matrix. $\vec{J}(\vec{x})$ is the vector of desired objectives (i.e. functions to be minimized, such as Payload Gear Ratio and/or Vehicle Height), $\vec{h}(\vec{x})$ is a vector of equality constraints and $\vec{g}(\vec{x})$ is a vector of inequality constraints. N optimization problems such as the one presented in Eqn. (4) are set up, one for each of the N architectures in the design space that are mapped to a set of N parameters vectors. This paper presents the results of the framework where the single-objective function to be minimized is Payload Gear Ratio. Other evaluation metrics are used for successive identification of architectural tradeoffs.

The optimization problem is defined as:

$$\begin{aligned} \min \quad & \mathrm{J}\left(\vec{\mathrm{x}}=[m_b, m_c, m_2, t_b, t_c], \vec{p}=[m_{pl}, m_f]\right)=\frac{m_b+m_c+m_2+m_f+m_{pl}}{m_{pl}} \\ \text{s.t.} \quad & C_1: \quad \Delta V_{1a}+\Delta V_{ca}+\Delta V_{1b}+\Delta V_{cb}+\Delta V_{cc}+\Delta V_2=\Delta V_{req}+\Delta V_{pen}-V_{\oplus} \\ & C_2: \quad \Delta V_{1a}+\Delta V_{ca}+\Delta V_{1b}+\Delta V_{cb}\le V_{staging} \qquad (5) \\ & C_3: \quad T_b+T_c\ge \eta GLOM \\ & C_4: \quad m_b-2m_c=0 \\ & C5: \quad T_b-2T_c=0 \end{aligned}$$

The design vector $\vec{x}$ is composed by boosters (m_b), core (m_c), upper stage (m_2) propellant mass and booster (t_b) and core (t_c) burn time. The parameters vector $\vec{p}$ is composed of payload (m_{pl}) and fairing (m_f) mass. The objective function J is the Payload Gear Ratio. C_1 is an equality constraint representing the launch vehicle delta V requirement. Core and booster stage propellants are partitioned to account for the three-segment throttling profile described in Section II. C_2 is an inequality constraint representing the upper boundary on staging velocity. C_3 ensures vehicle architectures a thrust/weight ratio greater than η. C_4 and C_5 are equality constraints for common booster and common engine hardware commonality options respectively.

The approach has been validated by evaluating its ability to reproduce existing launch vehicle architectures (Figure 8 and Figure 9). In both cases, upper stage length is under-estimated. This result was expected as the Delta IV Heavy and Atlas V Heavy have a ~3m diameter Centaur upper stage [29], despite their larger stage diameter [14, 15], whereas our formulation assumes propellant tanks of the same diameter size of associated stages.

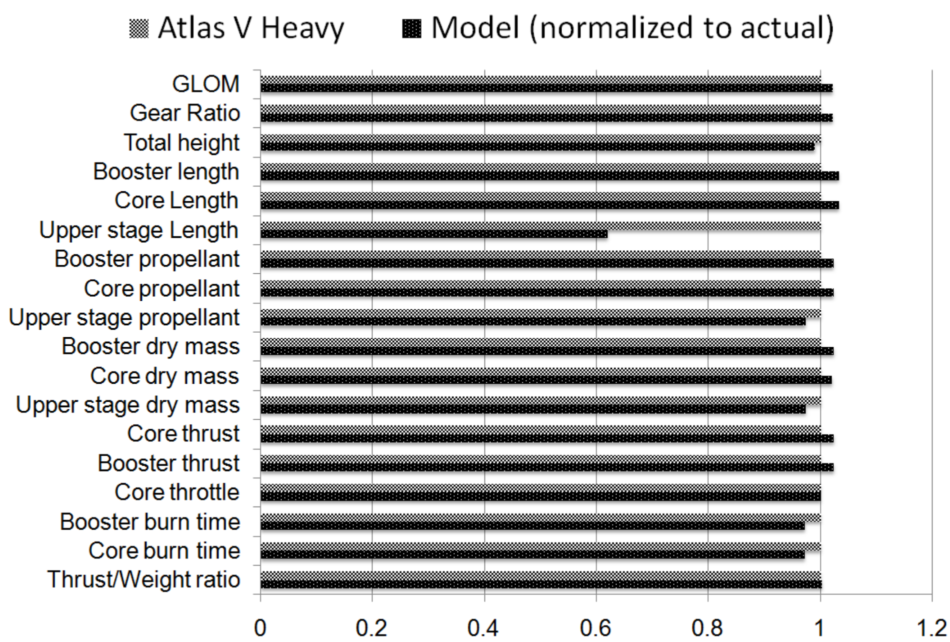

Figure 8 Model Validation with Atlas V Heavy

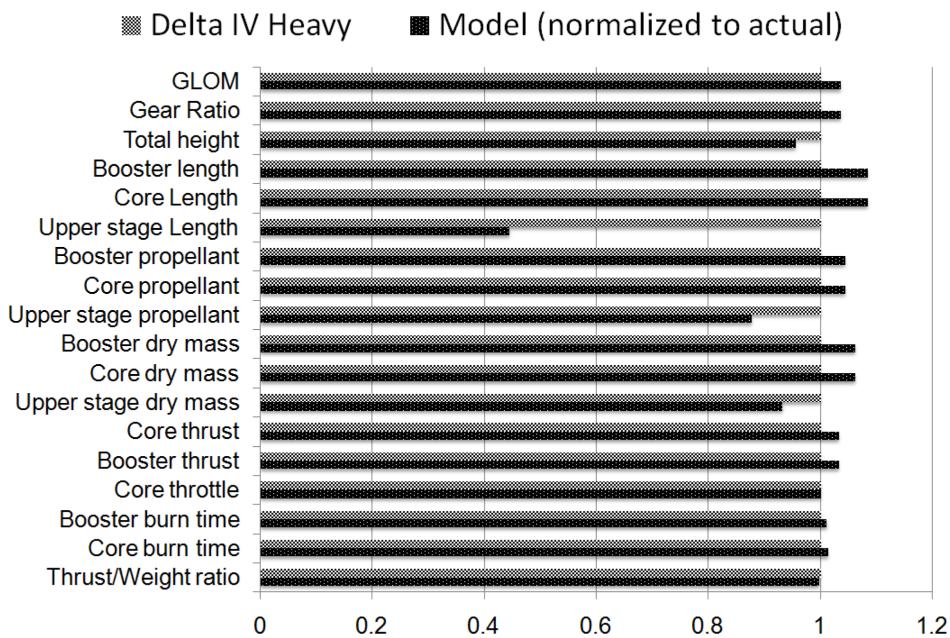

Figure 9 Model Validation with Delta IV Heavy

IV. Discussion of Results

The model presented in Section III is applied to explore options for a baseline heavy lift vehicle for 30mt to Escape, with the assumptions discussed in Section II. The results of the analysis are summarized in the stoplight charts shown in Table 11 (5m diameter core), Table 12 (7m diameter core) and Table 13 (8.4 diameter core) in the Appendix (Section VI). The main tradeoff resulting from this investigation is the one observed between the selection of propellant types, vehicle height, core stage diameter and payload gear ratio. While high performance propellants are desired to maximize performance (i.e. minimize payload gear ratio), they result in tall vehicles as propellant density is inversely proportional to specific impulse. Stages with lower performance propellants result in a vehicle of smaller dimensions, but of increased payload gear ratio (therefore increased GLOM). This reflects the known result in the literature of the inverse relationship between propellant molecular mass and specific impulse. Height limitations can be alleviated by increasing stage diameter. However, large diameters require investment in manufacturing capability and pose challenges in terms of ground processing and integration due to large vehicle width (i.e. having a first stage with two 8.4m boosters and a 8.4m core results in a 25.2m vehicle horizontal span. For comparison, the Space Shuttle Orbiter wing span was 24m). Small diameters pose aspect ratio issues leading to “pencil-shaped” vehicles.

Only seven out of the 192 baselines investigated were found to be viable according to the feasibility criteria that have been specified in the formulation of the study. These viable baselines are summarized in Table 7. LEO performance of the associated downward derivatives is shown in Table 8. These baselines provide a set of candidate designs for further study.

None of the 43 launch vehicle architectures with a 5m core diameter were feasible for a 30mt payload capability to Escape – the main constraint being the diameter resulting in unrealistic vehicle height and high aspect ratios.

In addition, "pencil-shaped" vehicles result from 5m core diameters, even considering high-density propellants such as LOX/RP1 and LOX/CH4; clustered stages were considered to mitigate these problems and allocate first stage propellant in multiple 5m-diameter modules. While clustering configurations succeed in meeting height criteria, they fail in meeting aspect ratio requirements. It can be concluded therefore that "super" heavy lift vehicles delivering 30mt to Escape will require larger diameters, thereby exceeding the existing US tooling capability (5m at the Decatur manufacturing plant).

Four parallel-staging configurations were viable with a 7m core diameter. The RP1/RP1/LH2 and RP1/LH2/LH2 architectures (i.e. LOX/RP1 boosters, LOX/RP1 core stage, LOX/LH2 upper stage) enable the use of existing first-stage engine technology (RD-180 class), but provide the most inefficient LEO derivatives across the set of feasible options. The LH2/LH2/LH2 architecture has the best gear ratio performance across the viable parallel-staged baselines – being the launch vehicle stack with the highest available specific impulses. It further provides the highest number of opportunities for hardware commonality (i.e. in engines). On the other hand, this architecture results in the highest vehicle height across the set of viable parallel-staged architectures (still within the VAB envelope) and requires the development of high-thrust LOX/LH2 engine technology for first-stage use. The LH2/LH2/CH4 architecture allows upper stage use of methane, thus mitigating cryogenic requirements of in-orbit refueling infrastructure (i.e. orbiting propellant depots). However, it results in over-designed booster-core downward derivatives– as optimal launch vehicles will feature a delta-V split biased toward higher specific impulse stages (thus resulting in large core and boosters). Core-upper stage derivatives are under-designed (with a 13mt payload capability to LEO) due to the relatively low specific impulse provided by the LOX/CH4 upper stage.

8.4m core stage diameter vehicles in a parallel configuration hit width constraints as specified previously, lying in the “light gray” zone of the stoplight chart. Solid propellant boosters mitigate dimension issues, due to higher propellant density resulting in smaller booster diameter (SRBs being 3.71m in diameter).

A. Affordability and Stakeholder Compliance

To measure the relative affordability of the proposed launch vehicle family architectures, we consider the number of operations projects and number of development projects. We also consider the number of development projects of the derivative vehicle. This represents the effort required to gain initial capability and is important in understanding the sustainability of the architecture. A high number of development projects to initial capability represents a program with large costs and schedule risk before any benefit is received from initial expenditure of resources. For the 7 architectures that passed screening on technical metrics, the cost proxy results are shown in Table 7, Table 8 and Table 9.

For each proxy category we divide the range of projects by three thresholds and label the fewest number of projects white, middle range light gray, and highest number of projects dark gray.

Three specific architectures stand out with the fewest number of projects across all categories: the 2.5 stage vehicle with common RP1 booster/core and LH2 upper stages, the 2.5 stage vehicle with all LH2 and common booster/core stages, and the 2.5 stage LH2 core/upper stage with 5 segment SRBs. Architectures with methane do poorly because of the lack of industrial experience in using that propellant in rocket engines (creating the cost of another development project). Losing the opportunity for commonality within the architecture penalizes the 2.5 stage RP1 booster, LH2 core, and all 3 stage vehicles.

Our evaluation of stakeholder compliance (policy risk) is not intended to describe the “right” architecture to select, however it allows a reasonable measure of the satisfaction of major stakeholder groups. The caveat that accompanies this analysis is that stakeholder needs are subject to change as has been the case in particular with congressional and executive guidance over the last decade. The satisfaction level measured for stakeholder compliance is shown in Table 10.

Immediately visible is that no architecture satisfies all three stakeholders completely. In addition, no single architecture completely satisfies Congress. There are 2 vehicles that satisfy Congress more than others- the 2.5 stage SRB and 2.5 stage with methane upper stage. These vehicles satisfy Congress because of the large payload range and the support of an American rocket engine industrial base. However this large payload capacity leads to an oversized derivative vehicle that is less useful for the Partners. The vehicles that satisfy Partners and Executive the most are the 2.5 stage rockets with common booster and core (of either RP1 or LH2) and do more poorly with Congressional needs.

B. Preferred Launch Vehicle Baselines

The preferred launch vehicle baselines resulting from this analysis are the SRBs/LH2/LH2 and LH2/LH2/LH2 vehicle architectures. Core and booster diameters of 7 or 8.4m are required to allow the desired capability of 30mt to escape. As highlighted in previous results, there is no “dominant” architecture between the two as the optimal outcome is determined by the weights of decision-makers to stakeholder compliance and lifecycle cost. If lifecycle cost has preferential weighting, a LH2/LH2/LH2 strategy is preferred as it maximizes opportunities for hardware commonality and allows NASA and commercial markets to share fixed and recurring costs of core and upper stages. If key political stakeholders are more heavily weighted, then a SRB/LH2/LH2 vehicle is preferred as it maximizes compliance with stakeholder goals.

C. Launch Vehicle Family Strategy

A family strategy of interest emerging from the results of this analysis is that of investing in early development of a LOX/LH2 first stage technology with sufficient thrust to be used either with a solid or liquid booster. In this strategy, the decision between liquid and solid technology for the boosters can be deferred to later phases of the exploration program (Figure 10).

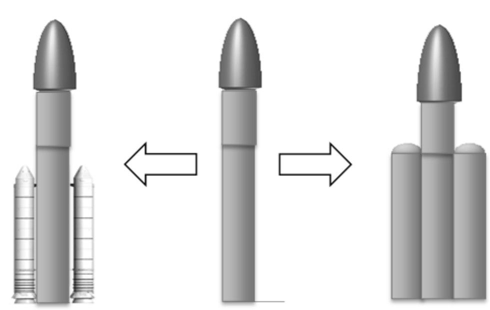

Figure 10 Launch Vehicle Family Development Strategy

A hydrogen-based core-upper derivative results in market-competitive LEO capability for ISS re-supply needs and other payloads in the 25mt class, while providing the required building blocks for successive evolution to "super" heavy lift capability. From stakeholders' perspectives, it complies with the Appropriation Act and keeps competitive forces on liquid engine and solid manufacturers. In conclusion, it provides several pathways for the evolution of the final system.

Table 7 Super Heavy Lift Vehicle - Viable Baselines

				BASELINE VEHICLE					
Vehicle Architecture	Dia. (m)	Common Booster Length	Common Engines	Height (m)	Width (m)	Aspect Ratio	GLOM (mt)	Engines Diam. Fit Check	Core T/W @ BECO Check
2.5 stage RP1, RP1, LH2	7	Yes	Yes	69.4	21	9.9	2196	0.53	1.77
2.5 stage LH2, LH2, LH2	7	Yes	Yes	92.4	21	13.2	1699	0.29	1.74
2.5 stage LH2, LH2, CH4	7	Yes	No	98.2	21	14.0	2196	0.43	1.05
2.5 stage RP1, LH2, LH2	7	No	No	111.5	21	15.9	1840	0.49	0.83
2.5 stage SRB 5seg, LH2, LH2	8.4	No	No	79.5	16	9.5	2287	0.00	N/A
3 stage RP1, LH2, LH2	8.4	n/a	n/a	99.8	8.4	11.9	1232	0.43	N/A
3 stage LH2, LH2, LH2	8.4	n/a	n/a	116.5	8.4	13.9	1099	0.49	N/A

Table 8 Downward Derivatives Performance for Viable Baselines

				LEO PERFORMANCE		
Vehicle Architecture	Dia. (m)	Common Booster Length	Common Engines	Vehicle Inefficiency to LEO	B-C Deriv Payload to LEO	C-U Deriv Payload to LEO
2.5 stage RP1, RP1, LH2	7	Yes	Yes	14%	33	35
2.5 stage LH2, LH2, LH2	7	Yes	Yes	6%	53	29
2.5 stage LH2, LH2, CH4	7	Yes	No	2%	81	13
2.5 stage RP1, LH2, LH2	7	No	No	8%	44	55
2.5 stage SRB 5seg, LH2, LH2	8.4	No	No	7%	48	Infeasible
3 stage RP1, LH2, LH2	8.4	N/A	N/A	7%	N/A	60
3 stage LH2, LH2, LH2	8.4	N/A	N/A	3%	N/A	71

Table 9 Cost Proxy Evaluations

				COST PROXY			
Vehicle Architecture	Dia. (m)	Common Booster Length	Common Engines	# Projects to first flight	# Dev Projects	# Ops Projects	Total # Projects
2.5 stage RP1, RP1, LH2	7	Yes	Yes	1.6	3.4	6.2	9.6
2.5 stage LH2, LH2, LH2	7	Yes	Yes	1.6	3.4	6.2	9.6
2.5 stage LH2, LH2, CH4	7	Yes	No	2.4	5	6.6	11.6
2.5 stage RP1, LH2, LH2	7	No	No	2.8	4.6	7	11.6
2.5 stage SRB 5seg, LH2, LH2	8.4	No	No	1.8	3.6	6	9.6
3 stage RP1, LH2, LH2	8.4	n/a	n/a	3.2	4.2	7	11.2
3 stage LH2, LH2, LH2	8.4	n/a	n/a	3.2	4.2	7	11.2

Table 10 Stakeholder Compliance (Policy Risk) Evaluations

				PARTNERSHIPS		CONGRESSIONAL GUIDANCE		EXECUTIVE GUIDANCE	
Vehicle Architecture	Dia. (m)	Common Booster Length	Common Engines	Use of Vehicle-Derivative	Use of Engine	Payload Capacity	Engine Industrial Base	Int'l or Comm'l Opportunity	Engine Tech

2.5 stage RP1, RP1, LH2	7	Yes	Yes	25-35mt	RD-180	112 mt	None	25-35 mt	RP
2.5 stage LH2, LH2, LH2	7	Yes	Yes	25-35mt	RS-68	109 mt	RS-68	25-35 mt	-
2.5 stage LH2, LH2, CH4	7	Yes	No	-	RS-68	119 mt	RS-68	-	-
2.5 stage RP1, LH2, LH2	7	No	No	-	RS-68	99 mt	RS-68	-	-
2.5 stage SRB 5seg, LH2, LH2	8.4	No	No	-	RS-68	108 mt	RS-68 + SRB	-	-
3 stage RP1, LH2, LH2	8.4	n/a	n/a	-	RD-180	92 mt	RS-68	-	RP
3 stage LH2, LH2, LH2	8.4	n/a	n/a	-	RS-68	88 mt	RS-68	-	-

V. Conclusions

The paper demonstrates how a field of 192 possible launch vehicles can be transparently reduced to seven possible designs on technical considerations, almost all of which are represented by particular viewpoints in the current debate, and how stakeholder screening can further narrow the design space. The paper illustrates two preferred architectures based on the analysis and proposes a corresponding strategy for the development of a family of launch vehicles to meet contingent and future needs for space exploration.

This paper presented a systems architecting framework for launch vehicle design and its application to the analysis of “super” heavy lift launch vehicles for future manned space exploration. The framework consists of a hybrid optimization method for preliminary sizing of parallel-staging and serial-staging launch vehicle configurations and assessment of proxy evaluation metrics for performance, cost and stakeholder compliance.

Several avenues of future research are identified by this study. The two main challenges highlighted by this paper are the analysis of requirements of future human space exploration, in order to improve the understanding of the required payload mass and volume requirements. While the study in this paper assumes a 30mt payload capability to escape, it does not consider a volume analysis on exploration payloads nor does it consider the impact of a varying payload requirement on the performance of the launch vehicles. Changing requirements result from the uncertainty surrounding the selection and design of future exploration payloads. Another evolution of the present study will analyze the additional infrastructure required in future exploration missions, including in-space transportation assets. In-space elements for in-orbit refueling and in-orbit assembly operations need to be considered concurrently to the selection of the launch vehicle. Similarly to the launch problem, the analysis of in-space elements will also require assumptions on desired payload and delta-V capabilities. Therefore, an uncertainty analysis will be required for elicitation of transportation needs, therefore providing an accurate specification of performance requirements.

VI. Appendix

Table 11 Stoplight Chart, 5m diameter core

5m Diameter Vehicles	Common Booster Length	Common Engines	Height Check	Width Check	Aspect Ratio Check	GLOM Check	Eng Thrust Check	Core T/W @ BECO Check	Baseline Vehicle Inefficiency to LEO	B-C Deriv Payload to LEO	C-U Deriv Payload to LEO
LiquidRP1LH2	Yes	Yes	110.5	15	22.1	2196.2	1.04315	1.77	14%	33.3	34.9
LiquidRP1LH2	Yes	No	103.8	15	20.8	2061.4	1.19635	0.71	10%	41.7	11.1
LiquidRP1LH2	No	Yes	121.2	15	24.2	2127.2	1.01085	1.13	11%	29.9	31.2
LiquidRP1LH2	No	No	99.2	15	19.8	2059.0	1.22191	0.69	11%	44.3	8.8
LiquidRP1CH4	Yes	Yes	109.3	15	21.9	3435.4	1.62261	2.06	7%	54.3	37.5
LiquidRP1CH4	Yes	No	101.5	15	20.3	3133.2	1.78455	0.83	5%	67.9	12.4
LiquidRP1CH4	No	Yes	123.1	15	24.6	3312.7	1.56527	1.34	6%	54.1	32.8
LiquidRP1CH4	No	No	85.7	15	17.1	3099.4	1.86423	0.82	6%	75.7	7.3
LiquidLH2LH2	Yes	Yes	155.7	15	31.1	1699.4	0.68453	1.74	6%	53.4	29
LiquidLH2LH2	Yes	No	145.8	15	29.2	1580.6	0.74099	0.91	5%	55.8	11.8
LiquidLH2LH2	No	Yes	182.4	15	36.5	1624.2	0.65484	1.10	5%	46.6	24.2
LiquidLH2LH2	No	No	140.5	15	28.1	1579.9	0.75197	0.91	6%	56.9	10.7
LiquidLH2CH4	Yes	Yes	181.0	15	36.2	2413.4	0.96639	1.95	3%	76.3	30.9
LiquidLH2CH4	Yes	No	167.0	15	33.4	2195.9	1.00867	1.05	2%	80.8	13.4
LiquidLH2CH4	No	Yes	212.9	15	42.6	2291.7	0.91837	1.24	2%	71.2	25
LiquidLH2CH4	No	No	145.8	15	29.2	2185.2	1.05011	1.04	2%	84.1	10
SRB_4segRP1LH2	No	No	124.5	12.6	24.9	2703.6	1.03107	0.65	9%	10.9	25.3
SRB_4segRP1CH4	No	No	221.7	12.6	44.3	4264.7	3.22121	1.10	6%	Infeasible	69.9
SRB_4segLH2LH2	No	No	217.4	12.6	43.5	2163.3	0.23055	0.24	6%	36.3	Infeasible
SRB_4segLH2CH4	No	No	375.6	12.6	75.1	3148.8	1.39774	0.81	2%	26.6	34.5
SRB_5segRP1LH2	No	No	103.9	12.6	20.8	2715.3	0.16799	0.12	8%	27.3	Infeasible
SRB_5segRP1CH4	No	No	196.4	12.6	39.3	4199.4	2.25008	0.84	5%	3.4	43.3
SRB_5segLH2LH2	No	No	176.1	12.6	35.2	2287.3	0	0.00	7%	48.0	Infeasible
SRB_5segLH2CH4	No	No	310.4	12.6	62.1	3129.3	0.63211	0.42	1%	46.0	3.1
3Stage_RP_RP_LH			180.9	5	36.2	1825.9	2.56154	N/A	9%	N/A	54.5
3Stage_RP_RP_CH			207.5	5	41.5	2540.7	3.56438	N/A	5%	N/A	84.5
3Stage_RP_CH_CH			194.5	5	38.9	2202.3	3.08962	N/A	4%	N/A	88.5
3Stage_LH_LH_LH			254.0	5	50.8	1098.7	1.3013	N/A	3%	N/A	71.2
3Stage_LH_LH_CH			301.4	5	60.3	1386.1	1.64168	N/A	1%	N/A	97.2
3Stage_LH_CH_CH			342.2	5	68.4	1826.1	2.16289	N/A	1%	N/A	93.2
3Stage_RP_LH_LH			207.0	5	41.4	1231.9	1.7283	N/A	7%	N/A	60.4
3Stage_RP (F1)_LH_LH			247.9	5	49.6	1295.0	1.81681	N/A	7%	N/A	43.3
2Stage_RP_LH			232.2	5	46.4	2395.6	3.3609	N/A	13%	N/A	N/A
2Stage_RP_CH			295.6	5	59.1	3926.6	5.50871	N/A	11%	N/A	N/A
2Stage_LH_LH			386.9	5	77.4	1800.7	2.13277	N/A	8%	N/A	N/A
2Stage_LH_CH			548.4	5	109.7	2824.7	3.34564	N/A	6%	N/A	N/A
2Stage_RP (F1)_LH			303.0	5	60.6	2284.3	3.2047	N/A	20%	N/A	N/A
Clustered_RP_LH			87.6	15	17.5	2828.0	0.99187	N/A	8%	63.1	Infeasible
Clustered_RP_CH			71.9	15	14.4	4558.0	1.59864	N/A	5%	105.9	Infeasible
Clustered_LH_LH			100.7	15	20.1	2260.0	0.6692	N/A	5%	116.6	Infeasible
Clustered_LH_CH			100.4	15	20.1	3215.0	0.95198	N/A	1%	173.0	Infeasible
LiquidRP1LH2LH2			193.0	15	12.9	1839.9	0.93016	0.83	8%	43.7	55.0
LiquidRP1LH2CH4			193.0	15	12.9	2597.2	1.39387	0.94	5%	75.3	44.8

Table 12 Stoplight Chart, 7m diameter core

7-m diameter Vehicles	Common Booster Length	Common Engines	Height Check	Width Check	Aspect Ratio Check	GLOM Check	Eng Thrust Check	Core T/W @ BECO Check	Baseline Vehicle Inefficiency to LEO	B-C Deriv Payload to LEO	C-U Deriv Payload to LEO
LiquidRP1LH2	Yes	Yes	69.4	21	9.9	2196.2	0.52729	1.77	14%	33.3	34.9
LiquidRP1LH2	Yes	No	65.9	21	9.4	2061.4	0.60473	0.71	10%	41.7	11.1
LiquidRP1LH2	No	Yes	74.8	21	10.7	2127.2	0.51097	1.13	11%	29.9	31.2
LiquidRP1LH2	No	No	63.6	21	9.1	2059.0	0.61765	0.69	11%	44.3	8.8
LiquidRP1CH4	Yes	Yes	68.7	21	9.8	3435.4	0.8202	2.06	7%	54.3	37.5
LiquidRP1CH4	Yes	No	64.8	21	9.3	3133.2	0.90206	0.83	5%	67.9	12.4
LiquidRP1CH4	No	Yes	75.8	21	10.8	3312.7	0.79121	1.34	6%	54.1	32.8
LiquidRP1CH4	No	No	56.7	21	8.1	3099.4	0.94234	0.82	6%	75.7	7.3
LiquidLH2LH2	Yes	Yes	92.4	21	13.2	1699.4	0.29337	1.74	6%	53.4	29
LiquidLH2LH2	Yes	No	87.4	21	12.5	1580.6	0.31757	0.91	5%	55.8	11.8
LiquidLH2LH2	No	Yes	94.3	21	13.5	1668.4	0.28812	1.58	5%	46.6	24.2
LiquidLH2LH2	No	No	84.7	21	12.1	1579.9	0.32227	0.91	6%	56.9	10.7
LiquidLH2CH4	Yes	Yes	105.3	21	15.0	2413.4	0.41417	1.95	3%	76.3	30.9
LiquidLH2CH4	Yes	No	98.2	21	14.0	2195.9	0.43229	1.05	2%	80.8	13.4
LiquidLH2CH4	No	Yes	103.9	21	14.8	2465.8	0.42305	2.11	2%	71.2	25
LiquidLH2CH4	No	No	87.4	21	12.5	2185.2	0.45005	1.04	2%	84.1	10
SRB_4segRP1LH2	No	No	76.5	14.6	10.9	2703.6	0.52119	0.65	9%	10.9	25.3
SRB_4segRP1CH4	No	No	126.1	14.6	18.0	4264.7	1.62827	1.10	6%	Infeasible	69.9
SRB_4segLH2LH2	No	No	123.9	14.6	17.7	2163.3	0.09881	0.24	6%	36.3	Infeasible
SRB_4segLH2CH4	No	No	204.6	14.6	29.2	3148.8	0.59903	0.81	2%	26.6	34.5
SRB_5segRP1LH2	No	No	66.0	14.6	9.4	2715.3	0.08492	0.12	8%	27.3	Infeasible
SRB_5segRP1CH4	No	No	113.2	14.6	16.2	4199.4	1.13737	0.84	5%	3.4	43.3
SRB_5segLH2LH2	No	No	102.8	14.6	14.7	2287.3	0	N/A	7%	48	Infeasible
SRB_5segLH2CH4	No	No	171.4	14.6	24.5	3129.3	0.2709	0.42	1%	46	3.1
3Stage_RP_RP_LH			92.3	7	13.2	1825.9	1.28077	N/A	9%	N/A	54.5
3Stage_RP_RP_CH			105.9	7	15.1	2540.7	1.78219	N/A	5%	N/A	84.5
3Stage_RP_CH_CH			99.2	7	14.2	2202.3	1.54481	N/A	4%	N/A	88.5
3Stage_LH_LH_LH			129.6	7	18.5	1098.7	0.5577	N/A	3%	N/A	71.2
3Stage_LH_LH_CH			153.8	7	22.0	1386.1	0.70358	N/A	1%	N/A	97.2
3Stage_LH_CH_CH			174.6	7	24.9	1826.1	0.92695	N/A	1%	N/A	93.2
3Stage_RP_LH_LH			105.6	7	15.1	1231.9	0.86415	N/A	7%	N/A	60.4
3Stage_RP (F1)_LH_LH			126.5	7	18.1	1295.0	0.90841	N/A	7%	N/A	43.3
2Stage_RP_LH			118.5	7	16.9	2395.6	1.68045	N/A	13%	N/A	N/A
2Stage_RP_CH			150.8	7	21.5	3926.6	2.75436	N/A	11%	N/A	N/A
2Stage_LH_LH			197.4	7	28.2	1800.7	0.91405	N/A	8%	N/A	N/A
2Stage_LH_CH			279.8	7	40.0	2824.7	1.43385	N/A	6%	N/A	N/A
2Stage_RP (F1)_LH			154.6	7	22.1	2284.3	1.60235	N/A	20%	N/A	N/A
Clustered_RP_LH			57.7	21	8.2	2828.0	0.49594	N/A	8%	63.1	Infeasible
Clustered_RP_CH			49.7	21	7.1	4558.0	0.79932	N/A	5%	105.9	Infeasible
Clustered_LH_LH			64.3	21	9.2	2260.0	0.2868	N/A	5%	116.6	Infeasible
Clustered_LH_CH			64.2	21	9.2	3215.0	0.40799	N/A	1%	173.0	Infeasible
LiquidRP1LH2LH2			111.5	21	15.9	1839.9	0.46508	0.83	8%	43.7	55.0
LiquidRP1LH2CH4			111.8	21	15.9	2597.2	0.69694	0.94	5%	75.3	44.8

Table 13 Stoplight Charts, 8.4m diameter core

8.4-m diameter Vehicles	Common Booster Length	Common Engines	Height Check	Width Check	Aspect Ratio Check	GLOM Check	Eng Thrust Check	Core T/W @ BECO Check	Baseline Vehicle Inefficiency to LEO	B-C Deriv Payload to LEO	C-U Deriv Payload to LEO
LiquidRP1LH2	Yes	Yes	56.3	25.2	6.7	2196.2	0.30131	1.77	14%	33.3	34.9
LiquidRP1LH2	Yes	No	53.9	25.2	6.4	2061.4	0.34556	0.71	10%	41.7	11.1
LiquidRP1LH2	No	Yes	60.1	25.2	7.2	2127.2	0.29198	1.13	11%	29.9	31.2
LiquidRP1LH2	No	No	52.3	25.2	6.2	2059.0	0.35295	0.69	11%	44.3	8.8
LiquidRP1CH4	Yes	Yes	55.8	25.2	6.6	3435.4	0.46869	2.06	7%	54.3	37.5
LiquidRP1CH4	Yes	No	53.1	25.2	6.3	3133.2	0.51546	0.83	5%	67.9	12.4
LiquidRP1CH4	No	Yes	60.7	25.2	7.2	3312.7	0.45212	1.34	6%	54.1	32.8
LiquidRP1CH4	No	No	47.5	25.2	5.7	3099.4	0.53848	0.82	6%	75.7	7.3
LiquidLH2LH2	Yes	Yes	72.3	25.2	8.6	1699.4	0.2567	1.74	6%	53.4	29
LiquidLH2LH2	Yes	No	68.8	25.2	8.2	1580.6	0.27787	0.91	5%	55.8	11.8
LiquidLH2LH2	No	Yes	73.6	25.2	8.8	1668.4	0.2521	1.58	5%	46.6	24.2
LiquidLH2LH2	No	No	66.9	25.2	8.0	1579.9	0.28199	0.91	6%	56.9	10.7
LiquidLH2CH4	Yes	Yes	81.2	25.2	9.7	2413.4	0.3624	1.95	3%	76.3	30.9
LiquidLH2CH4	Yes	No	76.3	25.2	9.1	2195.9	0.37825	1.05	2%	80.8	13.4
LiquidLH2CH4	No	Yes	80.2	25.2	9.6	2465.8	0.37017	2.11	2%	71.2	25
LiquidLH2CH4	No	No	68.8	25.2	8.2	2185.2	0.39379	1.04	2%	84.1	10
SRB_4segRP1LH2	No	No	61.2	16	7.3	2703.6	0.29782	0.65	9%	10.9	25.3
SRB_4segRP1CH4	No	No	95.7	16	11.4	4264.7	0.93044	1.10	6%	Infeasible	69.9
SRB_4segLH2LH2	No	No	94.1	16	11.2	2163.3	0.08646	0.24	6%	36.3	Infeasible
SRB_4segLH2CH4	No	No	150.2	16	17.9	3148.8	0.52415	0.81	2%	26.6	34.5
SRB_5segRP1LH2	No	No	53.9	16	6.4	2715.3	0.04852	0.12	8%	27.3	Infeasible
SRB_5segRP1CH4	No	No	86.7	16	10.3	4199.4	0.64993	0.84	5%	3.4	43.3
SRB_5segLH2LH2	No	No	79.5	16	9.5	2287.3	0	N/A	7%	48	Infeasible
SRB_5segLH2CH4	No	No	127.1	16	15.1	3129.3	0.23704	0.42	1%	46	3.1
3Stage_RP_RP_LH			90.6	8.4	10.8	1825.9	0.64039	N/A	9%	N/A	54.5
3Stage_RP_RP_CH			100.0	8.4	11.9	2540.7	0.8911	N/A	5%	N/A	84.5
3Stage_RP_CH_CH			95.4	8.4	11.4	2202.3	0.7724	N/A	4%	N/A	88.5
3Stage_LH_LH_LH			116.5	8.4	13.9	1098.7	0.48799	N/A	3%	N/A	71.2
3Stage_LH_LH_CH			133.3	8.4	15.9	1386.1	0.61563	N/A	1%	N/A	97.2
3Stage_LH_CH_CH			147.7	8.4	17.6	1826.1	0.81108	N/A	1%	N/A	93.2
3Stage_RP (RD180)_LH_LH			99.8	8.4	11.9	1231.9	0.43208	N/A	7%	N/A	60.4
3Stage_RP (F1)_LH_LH			114.3	8.4	13.6	1295.0	0.4542	N/A	7%	N/A	43.3
2Stage_RP_LH			108.8	7	15.5	2395.6	0.84022	N/A	13%	N/A	N/A
2Stage_RP_CH			131.2	7	18.7	3926.6	1.37718	N/A	11%	N/A	N/A
2Stage_LH_LH			163.6	7	23.4	1800.7	0.79979	N/A	8%	N/A	N/A
2Stage_LH_CH			220.8	7	31.5	2824.7	1.25461	N/A	6%	N/A	N/A
2Stage_RP (F1)_LH			133.9	7	19.1	2284.3	0.80118	N/A	20%	N/A	N/A
Clustered_RP_LH			48.1	25.2	5.7	2828.0	0.24797	N/A	8%	63.1	Infeasible
Clustered_RP_CH			42.6	25.2	5.1	4558.0	0.39966	N/A	5%	105.9	Infeasible
Clustered_LH_LH			52.8	25.2	6.3	2260.0	0.25095	N/A	5%	116.6	Infeasible
Clustered_LH_CH			52.7	25.2	6.3	3215.0	0.35699	N/A	1%	173.0	Infeasible
LiquidRP1LH2LH2			85.5	25.2	10.2	1839.9	0.23254	0.83	8%	43.7	55.0
LiquidRP1LH2CH4			85.7	25.2	10.2	2597.2	0.34847	0.94	5%	75.3	44.8

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