High Power Solar Electric Propulsion for Human Space Exploration Architectures

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Roger Myers[1] and Christian Carpenter[2]
Aerojet, Redmond, WA 98072 USA

Abstract: Aerojet has identified an affordable architecture for human exploration of deep space. Following the key tenets of launch and in-space commonality, efficient in-space transportation, and phased capability development drives the overall cost of missions to the Moon, NEOs, Phobos, and the surface of Mars to within NASA's existing Exploration budget while ensuring that risks to the crew and mission are minimized. Using high power solar electric tugs to preposition all non-time critical cargo and using conventional LOX/H_2 high thrust systems for crew transportation enables the use of smaller launch vehicles with great commonality across NASA, DoD, and commercial missions, distributing fixed launch costs across a broad customer base and dramatically exploration costs. Both 300kW and 600kW SEP Tugs are used for pre-placement of habitats, exploration equipment, and return vehicles at the destinations, allowing complete systems verification prior to crew Earth departure, significantly reducing the risks from the crewed portion of the mission. Missions are phased to spread development, production, and mission operations costs and enable incremental expansion of demonstrated exploration capabilities, culminating in a human mission to the surface of Mars in 2033.

I. Introduction

Dramatic reductions in space transportation costs are required to enable human exploration beyond low Earth Orbit (LEO). The need to develop, build, operate and maintain launch vehicles and infrastructure as well as deep space orbit transfer systems, habitats, landers, ascent vehicles, and exploration equipment in our fiscally constrained budget environment requires a major change in the way in which human space exploration is conducted. Previous NASA studies[1,2] have baselined architectures that are unlikely to be affordable. Aerojet recently completed an assessment of launch and in-space architectures that dramatically reduces the cost of human deep-space exploration, including missions to the Moon, Near-Earth Objects, Phobos, and the Martian surface. This paper reviews the overall approach to the in-space architecture study, with particular emphasis on the requirements for solar electric propulsion.

As is well known, the cost of space exploration is dominated by the cost of launch to orbit. The cost of a given launch system includes both fixed and variable costs, where the fixed costs include the ground infrastructure required to manufacture, launch, and operate the vehicle as well as the costs of maintaining the minimum "intellectual capital" required for the launch system. Variable costs include all costs that scale with the number of launchers built and launched. The fixed costs for the Space Shuttle system were approximately \$2.5B/yr, and the variable costs were between \$1.2B and \$0.4B per launch for launch rates between two and four per year. Because the Space Shuttle was used exclusively by NASA, all fixed and variable costs must be fully covered within its budget. The challenge for this study was to identify a human deep space exploration architecture that could accomplish all of the missions listed above within NASA's Exploration budget of \$4 – \$5B/yr, and be flexible enough to accommodate normal annual budget variations. In addition to the budgetary constraints, the mission

[1] Deputy Lead, Space and Launch Systems, Roger.Myers@Aerojet.com
[2] Program Manager, Integrated Systems, Space and Launch Systems, Christian.Carpenter@Aerojet.com

schedule was constrained to an initial launch to the International Space Station in 2016, initial missions beyond LEO before 2020, and a human landing on Mars in the early 2030s.

Aerojet's study quickly identified the following key tenets that must be met to achieve the affordability goals:

1) The launch and in-space architectures must maximize use of common elements across missions, destinations, and customers to minimize development costs and distribute fixed costs across multiple users (NASA, DoD and commercial).
2) Separate crew and cargo missions reduce launch vehicle size and enable use of lighter crewed vehicles, more efficient transportation systems for cargo, and pre-placement/verification of non-time critical architecture elements.
3) The in-space architecture drives launch vehicle requirements – selecting a launch vehicle size or design without first establishing high-level exploration mission requirements and in-space architecture can dramatically increase costs.
4) Modular propulsive stages optimize launched mass, mission flexibility, and commonality across destinations compared to large monolithic stages.
5) The mission order and schedule must enable demonstration of enhanced capabilities in successive missions – similar to the Apollo program, which used Mercury, Gemini, and successive missions to demonstrate increased capabilities before the ultimate landing on the Moon.

Fundamentally, these tenets are driven by simple economics and the rocket equation. First, the need for commonality is clear: unless the development, infrastructure, and industrial base costs are distributed across NASA, DoD, and commercial users there will not be resources to develop the other required elements of a deep space exploration architecture. This will also increase the launch rate, reduce variable production costs, and increase infrastructure and ground crew utilization rates. Additionally, as many architecture elements as possible should be common to multiple destinations – things like engines, stages, power systems, habitats, etc must be useful for missions to the Moon, NEOs, Phobos, and the surface of Mars to reduce the number of separate development and validation efforts. Second, separating crew from cargo, and recognizing that most of the cargo is not time critical, enables use of far more efficient in-space transportation systems while minimizing the crewed vehicle mass, both of which drive down propellant requirements and reduce the launch vehicle size requirements. It is critical to realize that crew radiation protection and health requirements drive one-way trip times to ~6 months, which require a non-Hohmann transfer for missions to the vicinity of Mars, and dramatically increase the sensitivity of launch requirements to vehicle mass for crewed missions. Third, for most missions beyond LEO the total ΔV required for Earth departure, destination arrival, Earth return, and Earth arrival dominates the overall mission ΔV, even when launch is included. The rocket equation thus tells us that most of the launch mass, and therefore most of the cost, will be driven by the propellant required to accomplish these ΔVs. Launch costs, both the size of the vehicle and the number of launches for a given mission, will therefore be controlled by the capability of the in-space propulsion systems. Fourth, modular propulsive stages are required to satisfy stage commonality as well as multiple destinations. The rocket equation shows that breaking the mission into smaller staged segments can actually reduce launched mass, though this must be balanced against the number of separate developments required – there is a balance between optimizing for low launch mass and minimizing the number of separate stages that must be developed. Finally, both budget constraints and risk management dictate that capabilities be developed and validated in a phased manner. For example, demonstrating the transit habitat by having a crew spend six months in lunar orbit can retire the risk of deep space radiation management, environmental controls, psychological factors, etc., at much lower risk than sending them on a mission to visit Phobos.

Major improvements in the performance and capability of solar power and electric propulsion systems have occurred over the past decade. We now have demonstrated solar cells with efficiencies over 35% with credible projections to 40%, and we now have years of experience with a 300kW-class array on the International Space Station (the power generating capability is 264kW BOL though usable power is 84kW to 120kW). Current solar cell technologies are capable of producing >600kW for the same wing area. High performance electric thrusters, whose life limitations are well understood, have been tested at power levels nearing 100kW. In contrast to 10 years ago, it is now possible to credibly discuss development of SEP vehicles with power levels of 300kW to 600kW. This change has enabled reconsideration of space exploration architectures using more realistic projections for available technology. This paper summarizes the approach and results of Aerojet's in-space architecture assessments. Section II provides a summary of the in-space architecture design, including the overall approach, destinations, analysis tools and propulsion systems used to establish the most affordable architecture. Section two presents the results of our analysis, Section III summarizes the architecture elements necessary for affordable human deep-space exploration, and Section IV summarizes the exploration campaign schedule and cost. Finally, Section V provides a summary of the paper.

II. In-Space Architecture Design

The in-space architecture design sets the launch vehicle lift requirements and overall architecture affordability. The space architecture design established the destinations, mission phases, ΔV requirements, propulsion technology trade space and payload elements (delivered mass for each mission phase). The result was an architecture trade space that could be optimized for minimum IMLEO and cost. The resulting in-space architecture was then matched to the launch vehicle options and capabilities to ensure a closed-form solution for human deep-space exploration.

A. Architecture Development Approach

Figure 1 shows the simplified mission phases that were defined in order to identify the ΔV requirements for a set of destinations. This approach reduced the complexity of studying a wide range of propulsion options for each phase and Concepts of Operations (CONOPS), while ensuring that the influence of one phase on all others was properly accounted for. It should be noted that not all phases are required for all missions, and some phases may be performed by non-propulsive elements in certain CONOPS. The approach allowed rapid evaluation of the use of common hardware for different destinations that can dramatically reduce the development and recurring costs of the exploration campaign.

The chosen analysis approach potentially favors propulsion over aerocapture solutions. However, the majority of destinations of interest do not have an opportunity to implement aerocapture as they have no atmosphere. In addition, aerocapture approaches have been studied in detail for Mars and this study was not intended to repeat those efforts. Finally, aerocapture approaches generally result in minimal re-use of space elements favoring single use propulsive stages and the goal of this study was to identify new and innovative approaches that maximize affordability and re-use of space assets. The potential for favoring propulsive approaches over aerocapture was ultimately determined to be an acceptable risk for this study and the breakup of mission phases allows for the easy insertion of aerocapture elements if deemed favorable.

Crew and cargo were separated to identify a minimum launch “bit” that could be used within multiple-launch missions to minimize IMLEO, use low cost launchers that apply to multiple users (NASA/DoD/Commercial), and to maximize launcher capability through dual manifesting. All crewed missions had a crew size of four. Two approaches are evaluated for the Earth insertion and Earth Descent phases: LEO Return and Direct Return. The LEO return approach includes launching crew to LEO and returning them to LEO where they would descend to Earth on a commercial crew or similar ISS crew return vehicle. The LEO return approach enables reuse of certain hardware elements but requires propellant for the LEO insertion. By contrast, the Direct Return approach eliminates the propellant for LEO insertion, resulting in a lower total architecture mass per mission and enables some abort scenarios not available using LEO return, though because LEO hardware is not reusable it may result in a higher mass across multiple missions. Direct Return also leverages the current Orion MPCV capability.

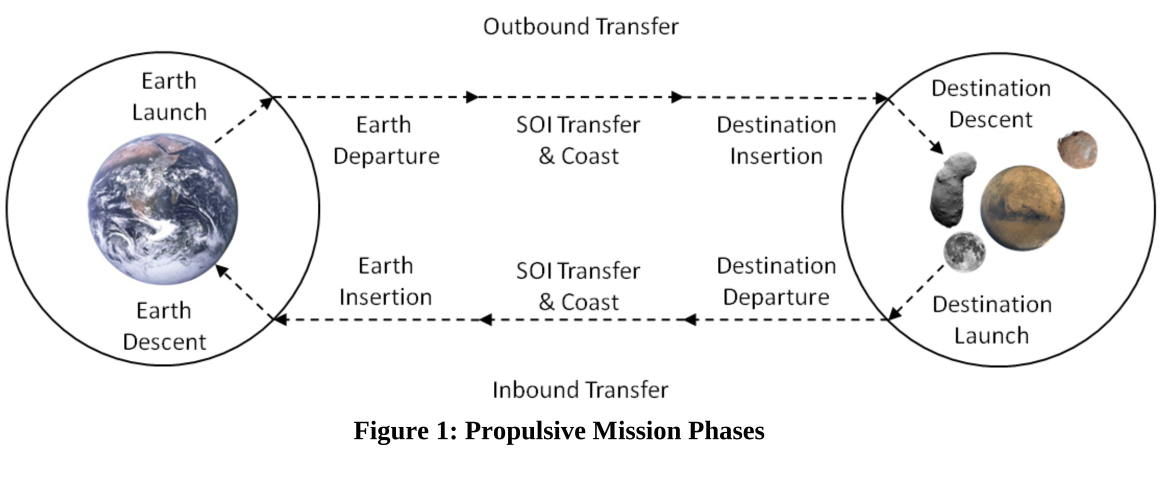


Figure 1: Propulsive Mission Phases

Overall, architecture design requires an iterative approach: for a given mission (delivered mass) at a destination, in-space propulsion system options were evaluated to ensure that the selected performance and power levels fit within the overall exploration campaign timeline. Any inconsistency was resolved prior to moving to the next destination using as many common architecture elements as possible. For example, an initial analysis showed that trip times with a 300kW SEP Tug carrying cargo for the Mars surface mission were too long to permit use of the transportation vehicle in support the Phobos mission (launch dates were too early), so the power level was increased to 600kW and the cases re-run. However, development risk and cost for the 600kW vehicle was too high for the early missions, driving us to phase in a 300kW SEP cargo vehicle for early missions the Moon and NEOs, and introducing a 600kW vehicle in the early 2020s to support the Phobos and Mars surface missions. This iterative approach ensured that the overall exploration campaign to all destinations could be accomplished while maximizing commonality across destinations.

B. Destinations and Trajectory Analysis

The destinations selected for detailed study are as follows: Low Earth Orbit, Lunar Orbit, NEO 2000 SG344, NEO 2008 EV5, NEO 2009 CV, Phobos, and the surface of Mars. These destinations provided bounding cases for mission ΔV and complexity for a flexible in-space architecture. Aerojet collaborated with Dr. Craig Kluever from the University of Missouri to analyze high and low thrust in-space trajectories for missions of interest. The trajectory code we developed was compared to NASA's Mars Exploration Design Reference Architecture (DRA) 5.0[2] results to validate its capability. Table 1 shows that the code results compare well with the results from the NASA DRA 5.0 study of high thrust propulsion. Our non-optimized low thrust trajectory code was used for all low-thrust trajectory analysis in this study. This code has been compared with STK Astrogator and NASA low thrust trajectory codes and the results were generally found to be within 10% for ΔV and trip time. Table 2 presents a comparison of high and low thrust ΔV for two missions. Table 3 presents orbits used for large targets and estimated ascent and descent ΔV. Note that we did not attempt to select optimum Earth departure or target arrival orbits for this simple study – further optimization is quite likely possible. The LEO departure orbit was selected to both maximize initial mass in low Earth orbit (IMLEO) for a given launcher (to minimize the need for large costly vehicles) and enable use of planned Orion and Commercial Crew launch systems. No attempt was made to investigate the use of lunar, Earth, or Venus gravity assist maneuvers to improve mission performance beyond that presented here.

Table 1: Trajectory Model Comparison to NASA's DRA 5.0

	DRA 5.0	**Present Model Results**
Earth Departure Date	8/27/2037	8/27/2037
TMI ΔV, km/s	4.10	3.99
MOI ΔV, km/s	1.75	1.69
Mars Arrival Orbit	250 x 33,793 km	250 x 33,793 km
Mars Arrival Date	2/17/2038	2/18/2038
Stay time	539	548
Mars Departure Date	8/10/2039	8/20/2039
TEI ΔV, km/s	1.52	1.60
Arrival Velocity, km/s	12.0	12.0
Earth Arrival Date	2/27/2040	3/9/2040

Table 2: Comparison of High and Low Thrust ΔV for Earth Space

Maneuver	Impulsive Thrust ΔV, km/s	Continuous Low Thrust ΔV, km/s
LEO to Earth Escape	3.2	7
LEO to LLO	4.8	8

Table 3: Selected Targets and Descent/Ascent ΔV

Target	Rendezvous Orbit	Descent ΔV, km/s	Ascent ΔV, km/s
LEO	407km x 407km @28.5°	Dependent on return strategy	Covered in LV trades
Moon	100km x 100km @0°	2.125 (Ref 3)	1.858 (Ref 3)
Mars	500km x 500km @0°	1.400 (Ref 4)	4.157 (Ref 5)

C. In-Space Propulsion System Performance

Several in-space propulsion options, summarized in Table 4, were considered in this study. All of these performance levels have been demonstrated through ground testing or in space. For the purposes of this study, it was assumed that technology is available for long duration storage of cryogenic propellants in orbit. Without this technology, the cryogenic and Nuclear Thermal Rocket (NTR) stages would not be able to perform beyond LEO. Should NTR propulsion prove practical it will reduce the launch mass for the crewed missions by another ~40% beyond the capabilities of LOX/H_2.

Table 4: Summary of Space Propulsion Performance Capabilities

Element	Propellant	Specific Impulse, s	Thrust Metric
Hall Thruster Propulsion (Ref. 6)	Xenon	2929	40mN/kW
Gridded Ion Thruster Propulsion (Ref. 7)	Xenon	4090	34mN/kW
Gridded Ion Thruster Propulsion	Xenon	6000	25mN/kW
Cryogenic Propulsion (Ref. 1)	LOX/LH_2	452	
Soft Cryogenic Propulsion (Ref. 8)	LOX/CH_4	350	
Semi-Cryogenic Propulsion (Ref. 9)	LOX/RP1	349	
Nuclear Thermal Rocket (NTR, Ref. 2)	LH_2	900	

III. Space Architecture Analysis

The goal of the space architecture analysis was to identify the most affordable and sustainable architecture with a capability to reach multiple deep space destinations, ultimately including the surface of Mars. The analysis used simplified mass models for the architecture element masses and propulsion system performance data provided previously. Following the rocket equation, the architecture analysis was conducted in reverse of the conduct of the mission, starting at the final state using the delivered mass requirements for each mission and working backward to the initial state in LEO. The IMLEO provided the end-state for the launch architecture study and was used to establish launch vehicle payload capability requirements and a launch campaign schedule for different sized launch vehicles that could support a credible human exploration campaign. This approach identified the minimum IMLEO requirements and minimum payload bit sizes for each mission element that meet the trip time constraints. Each mission phase was analyzed to define the propulsive elements of the architecture.

D. CONOPS Trades

The first step in minimizing IMLEO and cost was to trade the concept of operations (CONOPS) for the most challenging mission scenarios. The crew size as fixed at four for all missions considered. As described earlier, crew and cargo missions were separated, though for completeness cases in which the two elements used the same type of propulsion were also analyzed. For example, DRA 5 used NTR propulsion for both cargo and crew. Cargo separation allows prepositioning and check-out of all hardware and equipment not immediately required for the crew. For example, for a human expedition to the surface of Mars, the surface landers and surface payloads, fueled Earth return vehicles, Earth return consumables would all be prepositioned using a cargo vehicle and checked out in Mars orbit and on the surface prior to crew departure from Earth. This timing requirement and the cargo vehicle trip time determine the launch campaign scheduling. As described earlier, we assume either that long-term cryogenic storage technology is available, or that in-situ production of LOX/H_2 is feasible (e.g. storing water, electrolyzing it and liquefying the oxygen and hydrogen after arrival using the SEP Tug power system). The crewed mission consists only of the crew habitat, consumables for the journey, and the propulsion stage for Earth departure and Mars arrival maneuvers. Table 5 presents the CONOPS that were traded.

Table 5: Propulsion CONOPS Trades

Return Method	Crew	Cargo
Direct Return	Cryogenic	Cryogenic
		SEP Hall
		SEP Gridded Ion
	NTR	NTR
		SEP Hall
		SEP Gridded Ion
LEO Return	Cryogenic	Cryogenic
		SEP Hall
		SEP Gridded Ion
	NTR	NTR
		SEP Hall
		SEP Gridded Ion

A mission to Low Mars Orbit (LMO) without a surface landing was analyzed for the 2033 opportunity (the minimum ΔV opportunity in the next 40 years) to establish the relative merit of different in-space architectures and narrow the scope of the study to promising CONOPs and technologies. For this baseline comparison the LEO-LMO crew transfer vehicle was based on a Bigelow BA-330 habitat with a total mass, including consumables, of 25t, and the cargo portion included a preplaced habitat of the same mass, the fueled Earth-return and Earth arrival stages, and consumables for the LMO stay and Earth return journey. The relative performance of the CONOPS with a surface landing will be similar, though the total IMLEO will be increased to account for the landing, ascent, and surface systems. Figure 2 presents results for the LMO mission. The baseline for comparison was the direct return all cryogenic chemical propulsion CONOPS ("cryo crew/cargo") that required an IMLEO of 519t. The analysis showed that the using a 600kW SEP Cargo vehicle with the same cryogenic chemical crew vehicle enabled LEO return (and hardware reuse) for a total mission IMLEO (448t) 14% below that of the all cryo direct return. Switching back to a direct return approach, the use of a cryogenic chemical crew vehicle with an SEP cargo tug reduced the IMLEO to 262t, or 50% below that of the all cryogenic chemical option. Introducing NTR for crew transfer and using SEP cargo results in the lowest mass systems, an IMLEO of 162t for direct return and 175t for LEO return.

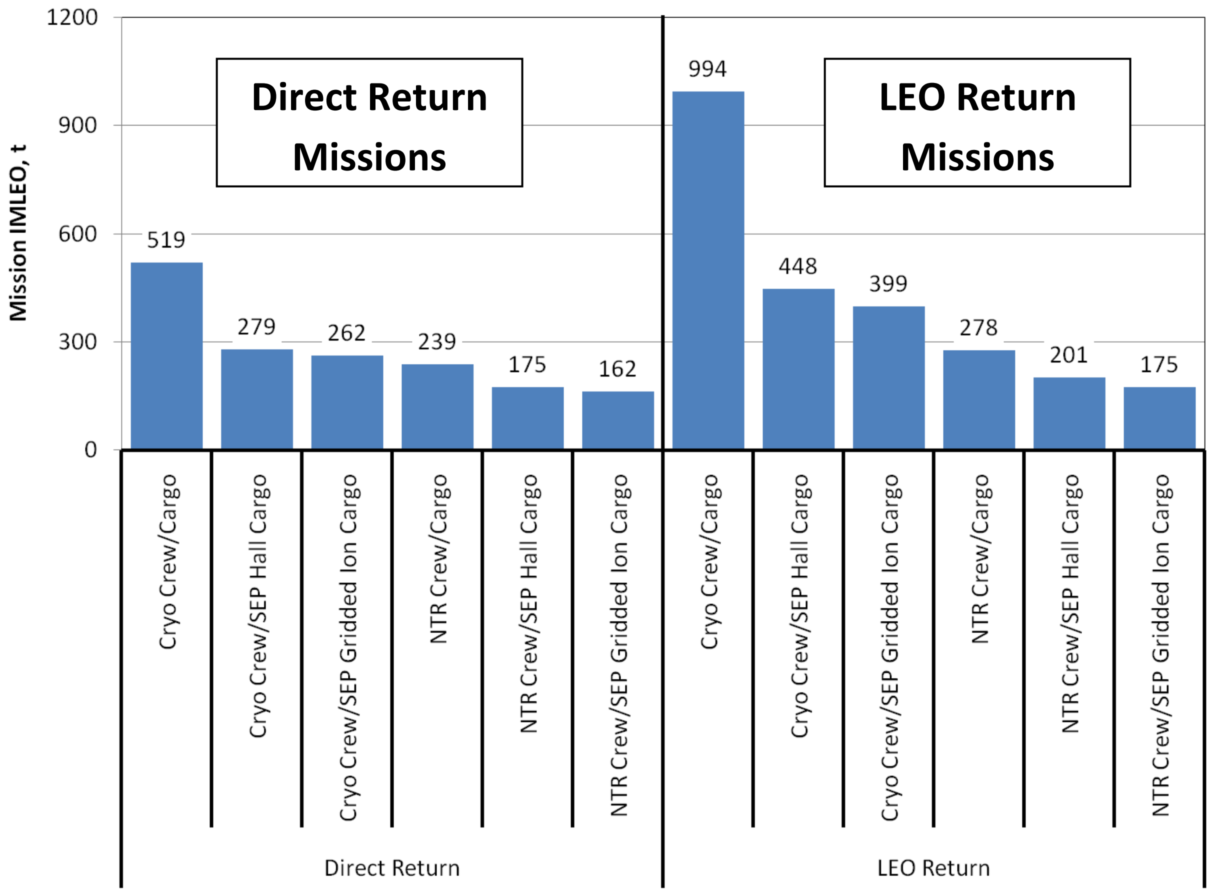


Figure 2: CONOPS Trades for a Mission to Low Mars Orbit with a crew of four.

With the notional Mars CONOPS trades in hand, the Cryo Crew/SEP Cargo architecture was then analyzed for a mission to the NEO 2008 EV5. As shown in Figure 3, the relative results are similar to the LMO mission, showing that the Mars trades apply to other destinations. Based on these analyses, cryogenic chemical (LOX/LH_2) propulsion was selected for near-term crewed missions through the initial Mars surface landing, and SEP was selected for all cargo missions. The development of NTR for crew transfer would provide another large benefit for Mars surface missions, and is enabling for Mars exploration beyond the initial mission due to the 16 year synodic period of Mars' orbit (see Section III.F below). Because the missions analyzed covered only the first steps in a long term sustainable architecture, a direct return architecture was selected as it provided the lowest IMLEO, and therefore lowest cost, for early missions. It should be noted however, that if significant ISRU capabilities were developed at the Moon, Phobos, or Mars, then the IMLEO and cost of the highly reusable LEO return architecture might be greatly reduced to a point where it would be favorable compared to the direct return option. Figure 4 and Figure 5 show examples of the selected CONOPS for the NEO and Mars missions.

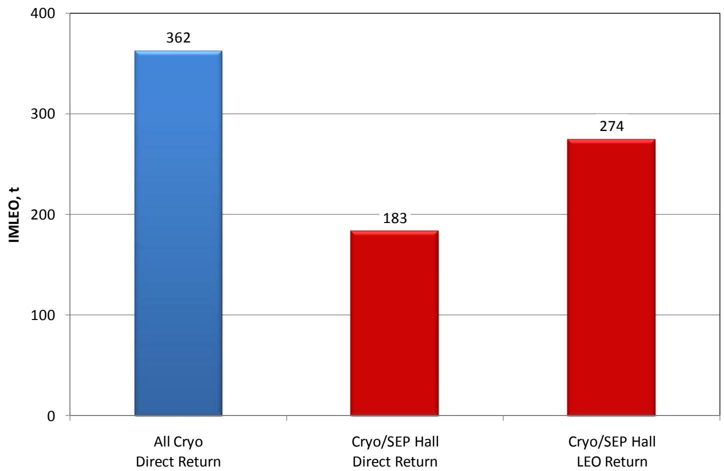


Figure 3: CONOPS Trades for a Mission to 2008 EV5

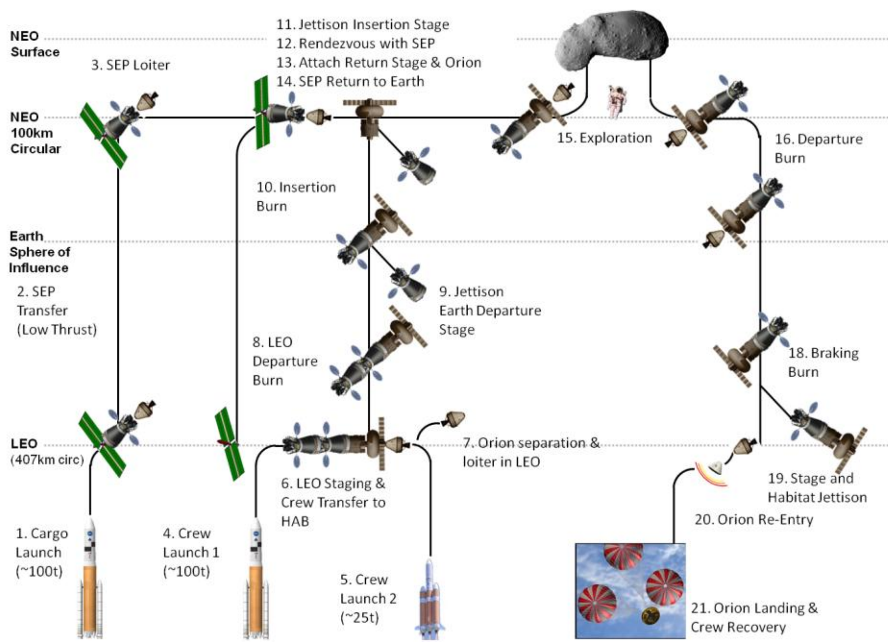


Figure 4: Example NEO Mission Using Cryogenic Propulsion for Crew and an SEP Tug for cargo

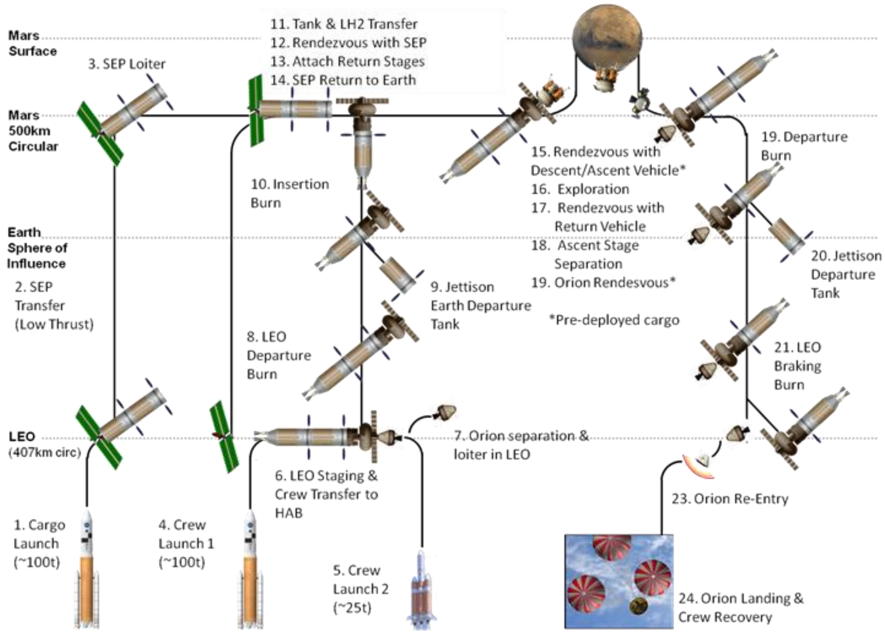


Figure 5: Example Mars Mission CONOPS Using NTR for Crew and SEP for Cargo

E. Sizing of Payload Elements

With the CONOPS and propulsive stage approaches defined, the architecture elements for every destination were sized, leveraging commonality where possible. The Mars surface mission was again taken as the starting point as the bounding case for the overall exploration architecture. Masses for all Mars surface elements supporting a crew of four, including the surface habitat and consumables for 500 days (16,500kg), surface power system (7,300kg), exploration equipment mass (5,000kg), and ISRU plant (1,305kg) were taken directly from NASA's DRA 5.0. For the Mars landers and ascent vehicles, LOX/LH_2, LOX/CH_4, and LOX/RP approaches were traded. While LOX/LH_2 provided the lowest total mass, mass dry mass impact of using a LOX/CH_4 ascent stage was found to be minor. We assumed that all ascent propellant would be entirely generated on the surface of Mars. Our ascent vehicle mass compares well to the Mars DRA 5.0 two-stage ascent vehicle with the dry mass of ~18,433kg for a crew of six[2] (~3,100kg dry mass per person to reach orbit with LOX/CH_4). The descent vehicle was sized based on a notional horizontal descent stage described in Ref. 2 that is consistent with a fairing style body. This has many benefits including the option of landing the aeroshell to provide additional radiation protection, elimination of surface elevator requirements, etc. For the Mars surface mission, this analysis showed the need for three landing vehicles on Mars, two with cargo and one with the crew. Each lander has a wet mass of 50t and a dry mass of 25t, which provided a bounding case for all payloads to all destinations and was used as input to size the SEP cargo vehicles.

F. Sizing of Crewed Propulsion Elements

Crew transport vehicles include the Earth ascent and descent vehicles (Orion[10] or Commercial Crew[11]), the in-space habitat (BA-330[12]), and Mars descent and ascent vehicles[2,5]. In the most affordable CONOPS, using an SEP cargo vehicle to preposition all possible cargo, the most massive Mars payload delivered by high thrust (chemical or nuclear) propulsion is the crew space habitat and consumables at 25t. The space habitat, based on the Bigelow Aerospace BA-330, was required for crew missions up to 6 months for each mission phase. The habitat carries ECLSS systems to support a crew six or less and requires minimal servicing after one year of use. The high thrust crew propulsion system was sized based on a maximum crew transfer time of 6 months for each destination. Options including LOX/LH_2, LOX/CH_4, and LOX/RP were compared on the basis of IMLEO and launch volume. The LOX/H_2 propulsion provided the lowest mass of the cryogenic options considered, while NTR was the lowest mass solution overall. LOX/LH_2 became the baseline for near-term missions with NTR being the baseline for sustained exploration after the first Mars landing. Note that the Earth direct return mission requires a Mars departure payload of 78.9t for the cryogenic approach, which includes both the space habitat and the Earth re-entry vehicle based on the Orion vehicle minimally fuelled for a braking burn at Earth.

A sensitivity analysis was performed to understand the mass and departure window requirements and capabilities of NTR and LOX/LH_2 for crew transport to and from Mars. Because the 2-year minimum ΔV for Mars varies so widely over a 16-year period, a qualitative limit of 650t IMLEO was set in order to stay within the bounds of the DRA 5.0 study – in other words, if the total IMLEO exceeded 650t the opportunity was eliminated. This upper IMLEO limit corresponds to five 130t launches, or ten 70t launches. While the ten seems like a large number of launches, it is critical to remember that most launches will be independent and unmanned – only the final launch will carry crew. This permits extended check-out of the in-space elements prior to crew launch. With the 650t total IMLEO constraint, gaps appeared in the launch opportunities using LOX/H2: humans could no longer be transported to Mars every 2 years. By contrast, use of NTR for crew transport offers a sustainable set of opportunities, enabling sustained exploration. Successful development of ISRU systems on Phobos might relieve this requirement.

Similar analysis was performed for lunar, NEO, and Phobos missions, though for these destinations only LOX/H_2 was considered for crew missions. Overall, this analysis showed that a mix of short term and long term storable cryogenic stages in the range of 70t would meet the requirements for missions to all destinations. In addition, the mission staging showed that a 50t class stage was sufficient for the lower ΔV missions to NEOs and other smaller exploration activities. Therefore common cryogenic stages were sized at 50t and 70t.

G. Sizing of Cargo Propulsion Elements

The sizing of NTR and cryogenic stages and payloads showed that ~70t is the maximum payload a solar electric propulsion vehicle would need to preposition at Mars for crew return to Earth. A sensitivity analysis was performed to understand the impact of SEP performance on the Mars cargo mission. Using SEP mass models, cases were run at varying power levels for a maximum round trip time of ~4y. This round trip time is necessary to permit reusability of the SEP for follow-on cargo missions, enabling a low cost architecture. The analysis showed that an I_{sp} in the range of 3000s was near optimal. This is due largely to the achievable power to mass ratio of the mission the round trip time constraints. Analysis of SEP missions is complicated by the coupling of mass and power to trip time and ΔV. Therefore specific cases were selected for study. The approach for the SEP analysis was to fix a payload mass and then search for an acceptable power level and trip time solution. This approach showed that a SEP stage operating at 3000s I_{sp} and 40mN/kW was the best overall solution and was used to size the vehicles. For Mars payloads at 70t, it was determined that a BOL power level of 600kW was sufficient to meet the round trip time constraint. For NEO and LLO missions, it was found that a 300kW BOL power was sufficient. These data were then used to size the SEP vehicles. SEP vehicles were sized using 300kW and 600kW BOL power levels and a fixed 3000s operating point. Specific mission cases were analyzed to size propellant modules for the SEP vehicle.

The SEP mass model was taken from Ref. 13, which provides more detail as to its derivation. The SEP vehicle is broken into two major components: Xenon (or Krypton) Propellant Modules and a SEP Propulsion Module. The Propulsion Module includes the arrays, structures, mechanisms, Power Management and Distribution (PMAD), avionics, auxiliary propulsion system, and payload interface. The mass model for the Propellant Module was rounded up from 1.6% in the reference to 2%. The result is:

$$M_{\text{propellant Module}} = 0.30*M_{\text{propellant}}^{0.71}$$

It is important to remember that both xenon and krypton are stored at high densities, resulting in the low tank mass fractions. The mass model for the SEP propulsion module is:

$$M_{\text{SEP Propulsion Module}} = 3.8*\text{Power}^{0.57}$$

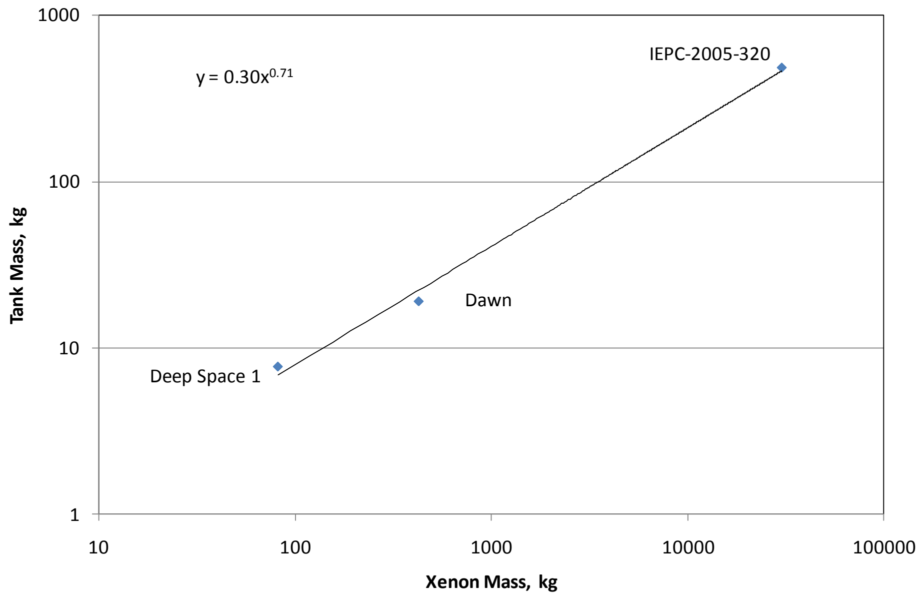


Figure 6: Xenon Propellant Module Mass Model

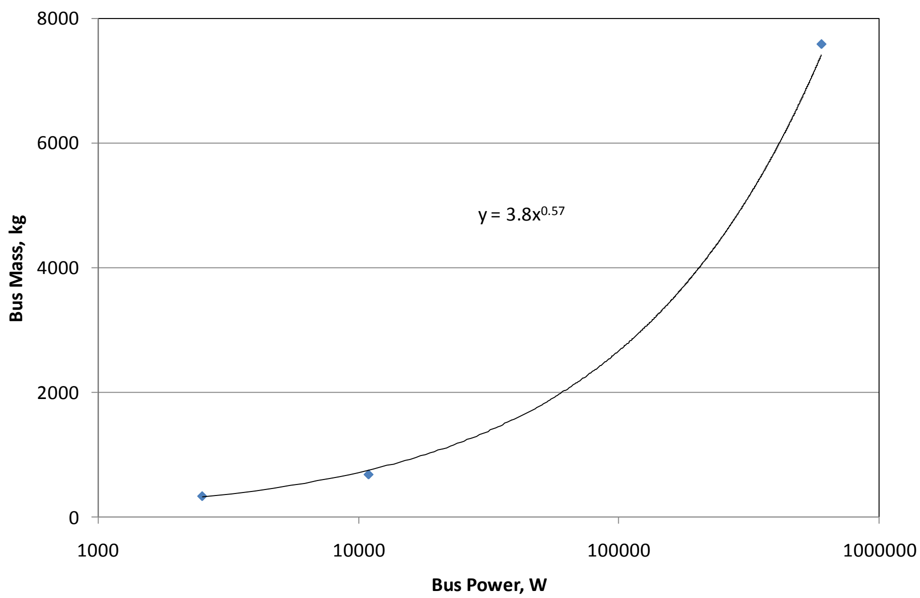
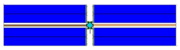
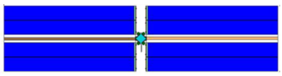


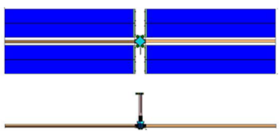
Figure 7: SEP Propulsion Module Mass Model

These mass models were used to size both the 300kW and the 600kW cargo tugs, with the result shown below in Fig. 20.

Figure 20: SEP Cargo Tug Vehicles

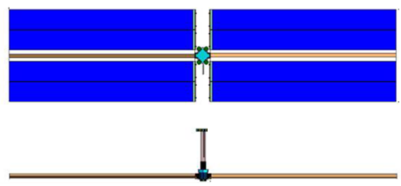
Element Concept	Element Name	Dry Mass, kg	Maximum Wet Mass, kg	Dimensions
	300kW SEP Tug	6,000	36,000	5m dia., 8m overall length (OAL), Stowed
	600kW SEP Tug	8,000	76,000	5m dia., 10m overall length (OAL), Stowed

300kW Cargo Tug



- First flight: 2017
- Dimensions (stowed): 5m dia., 8m overall length (OAL)
- Mass: 36t wet, 6t dry
- Propellants: Xenon or Krypton (30t)
- Propulsion: 3 x 150kW modules @ 3000s
  - Leverages NASA 457M development
  - Carries one spare 150kW module
- Utility: High ΔV maneuvers on long duration cargo missions with 35t class payloads
- Technology: Implements high efficiency radiation hardened solar arrays, 150kW Hall thruster propulsion systems, light weight inert gas propellant tanks

600kW Cargo Tug

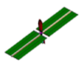
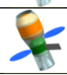
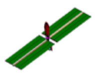
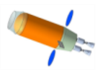
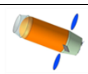


- First flight: 2020
- Dimensions(stowed): 5m dia., 10m OAL
- Mass: 76t wet (via 70t ORU), 8t dry
- Propellants: Xenon or Krypton (65t)
- Propulsion: 5 x 150kW modules @ 3000s
  - Leverages 300kW Cargo Tug propulsion
  - Carries one spare150kw module
- Utility: High ΔV maneuvers on long duration cargo missions with 70t class payloads
- Technology: Implements high efficiency radiation hardened solar arrays, 150kW Hall thruster propulsion systems, light weight inert gas propellant tanks

H. Architecture Summary

With the architecture analysis completed and the architecture elements defined, each mission was planned to determine the launch manifest, IMLEO, and cost. Table 6 presents a summary of the resulting architecture elements. For each destination, the applicable elements are identified with a ✓, and optional elements with an "O". It can be seen from this list that the maximum monolithic element and thus the minimum launch bit required by this architecture is ~70t. This minimum launch bit was used to frame the launch vehicle size and performance requirements.

Table 6: Architecture Elements

Element Concept	Element Name	Dry Mass, mT	Maximum Wet Mass, mT	L2/Moon	NEO	Phobos	Mars Surface
	Orion Vehicle (Direct Return Option)[10]	13	21	✓	✓	✓	✓
	Commercial Crew Vehicle (LEO Option)[11]		9	O	O	O	O
	Space Habitat[12] (Crew of 4)	19.5	25	✓	✓	✓	✓
	300kW Cargo Tug	6	36	✓	✓	O	O
	70t Short Term Cryogenic Stage	6	62	✓	O		
	70t Long Term Cryogenic Stage	14	70	✓	✓	✓	✓
	50t Long Term Cryogenic Stage	9	50	✓	✓	✓	✓
	600kW Cargo Tug	8	76			✓	✓
	NTR Stage	15	40			O	O
	NTR Drop Tank	16	65			O	O
	Mars Descent & Ascent Vehicle	25	50				✓
	Mars Habitat (4 Crew)	25	50				✓
	Mars Exploration Equipment Lander	25	50				✓

IV. Exploration Launch Campaign

The elements above enable an affordable campaign of deep space human exploration. Missions to lunar orbit, NEOs, Phobos, and Mars surface were phased to demonstrate increased capabilities in each succeeding mission to improve the risk management of the overall exploration effort. The overall campaign schedule is summarized in Figure 21. The first mission to lunar orbit or a Lagrange point uses a 300kW Tug to preposition a space habitat, with an Orion vehicle then delivering the crew for a 6-month stay. This mission demonstrates long-duration mission capabilities in deep space while maintaining the option for a fast crew return to Earth if problems arise. This is followed by a NEO mission, the first mission beyond cis-lunar space, which adds prepositioned Earth return stages using a 300kW SEP tug in a less challenging mission than Phobos. The Phobos mission follows with the launch of two 600kW SEP tugs in 2020 – three 600kW tugs are required for this challenging mission. Finally, the Mars surface campaign starts in 2025 and culminates in a crew landing in 2033 for a 500 day stay.

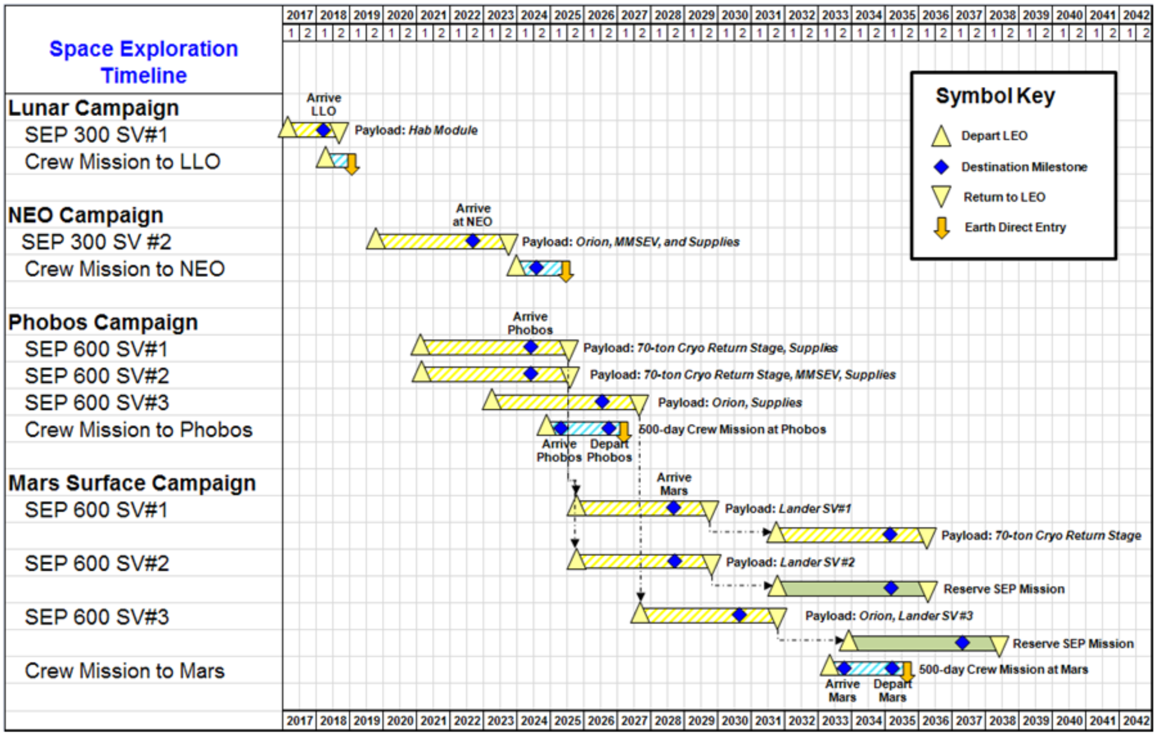


Figure 21: Exploration Architecture Schedule

Establishing the total costs of this exploration campaign requires assessments for both the in-space and launch elements. The details of the launch vehicle trades are presented elsewhere, and included development of both 70t and 130t launch capabilities. While we have shown above that the minimum payload bit size is 70t, enabling use of this size launcher, current policy is driving toward a 130t launcher. With the in-space architecture described above either is suitable for human exploration of deep space, though the 70t vehicle maximizes commonality across NASA, DoD, and commercial users. For this reason it was selected as the baseline for cost estimation. The results of this effort are shown in Figure 22.

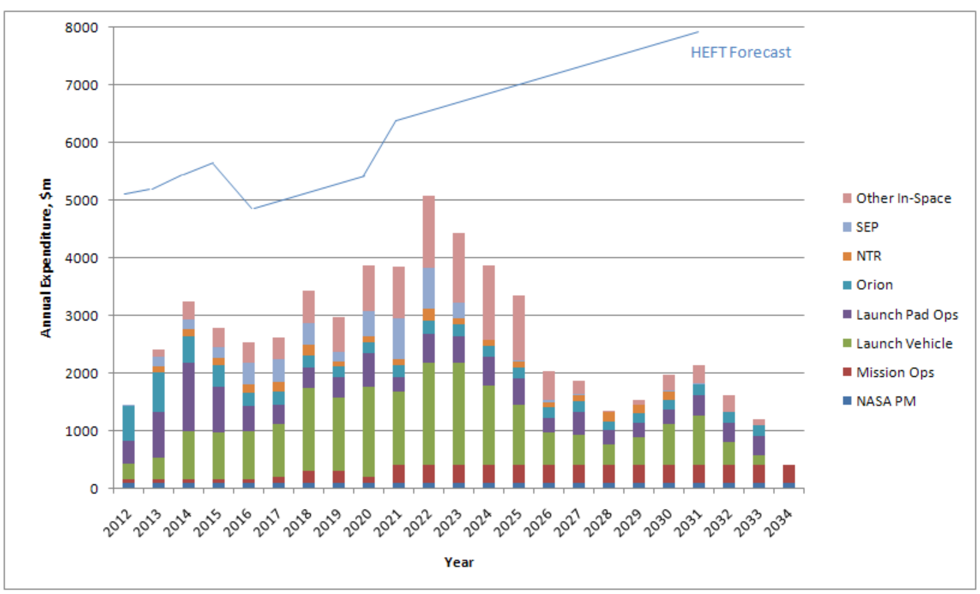


Figure22: Expenditure Profile for the Exploration campaign

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

V. Conclusions

An affordable architecture for human exploration of the Moon, NEOs, Phobos and the Martian surface has been identified by following five key tenets: 1) The launch and in-space architectures must maximize use of common elements missions, and destinations; 2) Separating crew from cargo to reduce launch vehicle size and enable use of smaller crewed vehicles, more efficient SEP transportation systems for cargo, and pre-placement/verification of non-time critical architecture elements; 3) Recognizing that in-space architecture can enable use of smaller launch vehicles with enhanced commonality to multiple users, spreading fixed costs across multiple markets; 4) Modular propulsive stages optimize launched mass and mission flexibility compared to large monolithic stages and enable commonality between missions; and 5) Selecting a mission progression that enables demonstration of enhanced capabilities in successive missions, ensuring early retirement of risks. The SEP cargo vehicles leverage recent advances in solar electric power and propulsion, which make the high performance 300kW and 600kW vehicles possible. Hall thruster based systems with 150kW propulsion modules at a vehicle specific impulse of 3000s and a T/P of 40mN/kW were found to meet the cargo mission requirements, and enabled crew transfer using conventional LOX/H_2 cryogenic propulsion thru the initial Mars surface mission. Additional required technologies are long-term storage of LOX and H_2 and in-situ propellant manufacture for Mars ascent.

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