

Affordable Exploration Architectures Using the Space Launch System and High Power Solar Electric Propulsion

*Presented at the Joint Conference of 30th International Symposium on Space Technology and Science
34th International Electric Propulsion Conference and 6th Nano-Satellite Symposium*

Hyogo – Kobe, Japan

July 4-10, 2015

Roger M. Myers¹, C. Russell Joyner², R. Joseph Cassady³, Steven Overton⁴, Timothy Kokan⁵, James Horton⁶ and W. Andrew Hoskins⁷

Aerojet Rocketdyne
Redmond, West Palm Beach, and Washington DC

Abstract

This study examines the impact of the launch capabilities provided by the planned Space Launch System (SLS) Block 2 vehicle on the solar electric propulsion (SEP) power levels required for large cargo prepositioning missions to Mars orbit. Our analysis shows that the additional launch capability planned for the SLS vehicle enables a dramatic reduction in the required SEP vehicle power level while maintaining reasonable trip times for large cargo pre-positioning (20-40mt). For near-term missions, we show that 50kW SEP modules can be used either singly or in combination, enabling much lower cost human exploration missions. This is a much lower power level, and thus much lower cost, than shown by our previous studies. 50kW SEP modules can also be used for cis-lunar missions and large science missions such as the Asteroid Redirect mission with smaller launchers such as the SLS Block 1, Falcon 9 and 9H, and EELV launchers, ensuring that the SEP modules have multiple applications. We show that this modular approach enables a significant reduction in total development and production costs for high power SEP and provides for a more gradual phasing of system development to fit within available annual budgets.

1. Introduction

Major reductions in space transportation costs are required to enable human exploration beyond low Earth orbit. 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 human space exploration is conducted. Multiple studies have examined a wide range of exploration architectures¹⁻³, and over the past four years the focus of these efforts has shifted from mission performance to affordability. Aerojet Rocketdyne completed an assessment in 2010 of launch and in-space architectures that showed the potential to dramatically reduce the cost of human deep-space exploration, including missions to the

¹ Executive Director, Advanced In-Space Programs, Roger.Myers@rocket.com

² Fellow, Systems Analysis, Claude.Joyner@rocket.com

³ Executive Director, Space, Joseph.Cassady@rocket.com

⁴ Program Manager, Advanced In-Space Integrated Systems, Steven.Overton@rocket.com

⁵ Specialist Engineer, Mission Architecture, Timothy.Kokan@rocket.com

⁶ Senior Engineer, Mission Architecture, James.Horton@rocket.com

⁷ Manager, Advanced In-Space Business Development, Andrew.Hoskins@rocket.com

Moon, Phobos, and the Martian surface⁴. Since that time several groups have extended those results⁵⁻¹⁰ to incorporate new developments in both launch and in-space transportation systems. This paper presents the results of continued efforts at Aerojet Rocketdyne to identify affordable approaches to human deep space exploration, with particular emphasis on the requirements for solar electric propulsion.

After completion of the 2010 study NASA made several key decisions and put programs in place that impact the specific exploration architecture options:

- 1) Build the Space Launch System and produce these launchers at a rate of 1 or 2 per year.
- 2) Started development of a 13kW Hall thruster system (2013)
- 3) Completed the first phase of new high performance solar array programs (2014)

The purpose of this study is to assess the impacts of these decisions on the SEP approaches presented in our 2011 paper, in which we established an architecture using 300-600kW SEP cargo vehicles with individual thruster power levels of 100kW. Those cargo vehicles required a significant extension in the state of art for electric propulsion with resulting technical, schedule and cost risks. In section 2 we review the key tenets of our exploration architecture design. In sections 3 and 4 the SLS launch vehicle capabilities are summarized, along with assumed performance of the SEP subsystem for this study. Finally in Sections 5 and 6 we summarize the approach used for the mission analysis, present the key results and present two approaches to scaling the core 50kW SEP Transportation Module. A summary is provided in section 8.

2. Exploration Architecture Framework

Our prior study⁴ found that exploration architecture affordability is driven by 5 key tenets, which we have updated to reflect the progress made since that paper:

1. Separate cargo and crew missions to reduce required launch mass and enable the use of lighter crewed vehicles, more efficient in-space transportation systems for cargo, and pre-placement verification of non-time critical mission elements.
2. Maximize commonality for all elements of the in-space architecture across missions, destinations, and customers to minimize development costs and distribute fixed costs across NASA, DoD, and commercial users.
3. Use modular propulsive stages to optimize launched mass, mission flexibility, and commonality across customers and destinations
4. Maximize leverage from existing and planned upgrades to the Space Launch System and Orion.
5. Phase the mission order to enable critical demonstrations when needed while staying within annual budget appropriations.

The benefit of the first of these tenets has been demonstrated multiple times: efficient pre-placement of non-time critical exploration systems results in a factor of two reduction in launch mass, enables shorter crewed trip times by reducing crew vehicle mass, and reduces overall mission risk by enabling on-site check-out of critical systems before the crew departs Earth for their destination. The second tenet has also been amply demonstrated in the use of common (or nearly common) EP systems for DoD and commercial missions¹¹⁻¹⁴ but to date it has presented a bigger challenge when extended to human exploration. For example, the high power SEP vehicles (300-600kW) in our 2010 study were to be used for all NASA human exploration destinations, but maintaining commonality for the defense and commercial mission was extremely difficult. Not only would these vehicles be very costly, but all these costs would have to be fully covered by a single user as only NASA human exploration missions have an identified need for large cargo transportation to the Moon and Mars. Projections for NASA science,

DoD, and commercial space vehicles indicate that for the next 2-3 decades their power requirements are unlikely to exceed 50kW power system requirements, providing a good upper limit to the vehicle power level to maximize commonality. The system level at which commonality is maintained is addressed by the third tenet. To date it has been most successfully maintained at the subsystem level: subsystems and components are used today across many missions and markets. The use of modular propulsion systems with multiple in-space transportation users represents the next level of commonality, and is driven by the shift from the payload-focused spacecraft of today to the higher power transportation-focused spacecraft needed for the future.

Only since the decision to build SLS and continue Orion has the fourth tenet started to be examined. The impact of launch vehicle capability on SEP systems and commercial mission capabilities has been studied since the late 1990s¹⁵, and it is clear that the launch vehicle drop-off orbit has a first-order impact on in-space propulsion system requirements. However, the opportunities enabled by SLS were not integrated into our prior architecture analyses. Our 2010 study assumed launch drop-off in LEO, and only a few studies since then have studied higher energy options. Additionally, the high speed reentry capabilities of the Orion capsule and the resulting return orbit flexibility have not been fully incorporated. This study focuses on the first of these opportunities.

3. Launch Vehicle Capability Comparisons

NASA development of the SLS vehicle is planned in phases, with the Block 1 SLS first flight scheduled for early 2018, Block 1B planned for the Exploration Mission 2 in the early 2020s, and the final Block 2B vehicle in the late 2020s. The launch vehicle configurations are summarized in Figure 1.

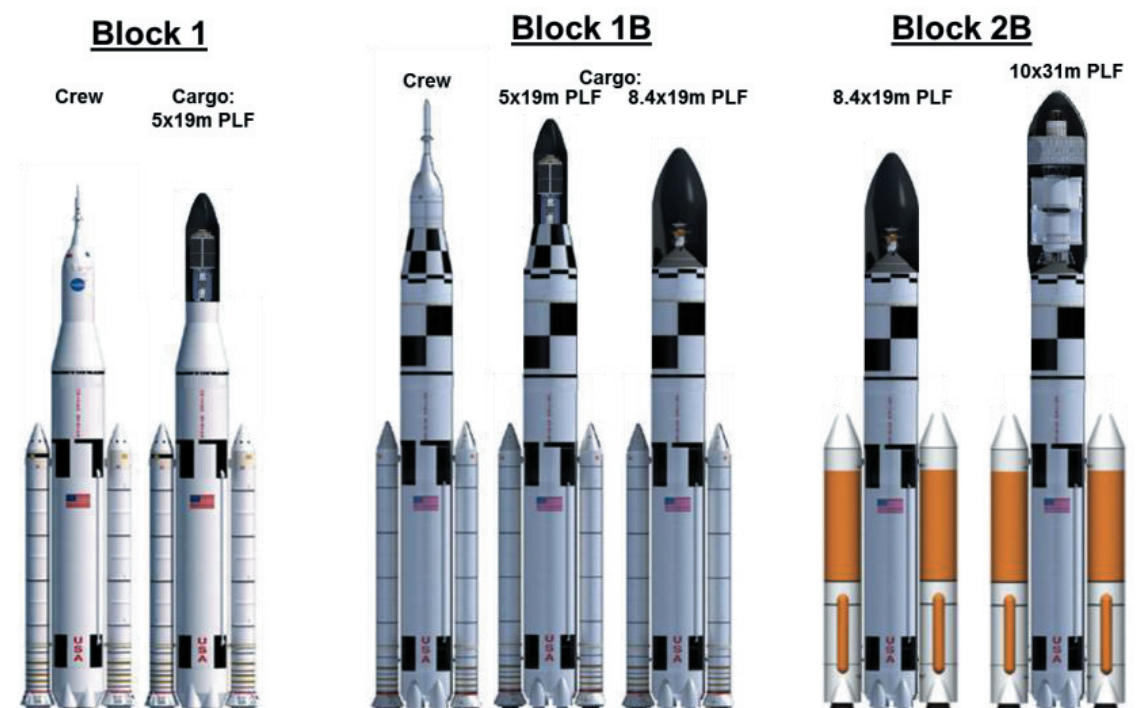


Fig. 1: SLS Block 1, Block 1B, and Block 2B configurations.

Note that there is a high level of commonality across the different Blocks, and both Block 1 and Block 1B have both crew and cargo configurations. Block 2B is focused on large cargo capability beyond LEO. The key difference between Block 1 and Block 1B is the addition of the LOX/H₂ Exploration Upper Stage, and between Block 1B and 2B is the addition of advanced boosters, each of which increases the launcher capability. Net payload mass as a function of injected velocity is shown in Figure 2 for Block 1B and 2B. Our study considered launch vehicle capabilities ranging from a high of 130mT to low Earth Orbit for the Block 2B down to 38mT to a C3 of 0 (Earth escape) for the Block 1B.

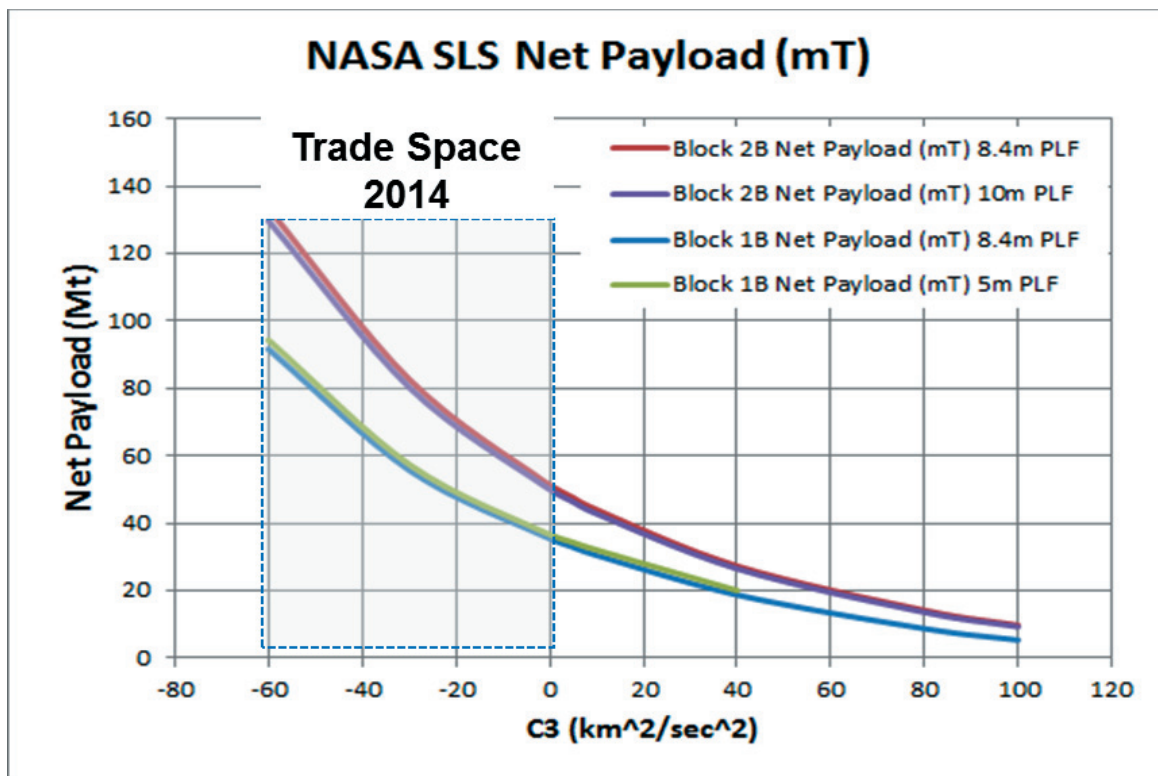


Figure 2: SLS throw weight capabilities used for this trade study.

Current plans call for one to two SLS launches per year starting in the mid-2020s, which supports the human exploration mission objectives of reaching the vicinity of Mars in the early 2030s.

4. SEP System Capability Assumptions

Significant advancements have been made in the last 5 years in the development of higher power SEP systems. First, Aerojet Rocketdyne has successfully supported three AEHF missions and several more are planned with our Zero Erosion long life Hall thruster technology. Second, continued work by NASA has further illuminated the physics of achieving very long Hall thruster life, along with development of a 13kW long-life xenon Hall thruster. Third, high power solar array technology has advanced considerably with investments in industry and NASA. Notably the work on the Mega-ROSA and the Mega-Flex arrays^{16, 17} has advanced the specific power to between 200 and 400W/kg and the reduced the packaging volume to over 50kW/m³. For this study we used a conservative array specific mass of 132 W/kg as we did not explicitly account for array degradation. Note that for these missions there is no powered transfer thru the van Allen radiation belts in any of this analysis as the vehicle is always launched to a very high

orbit or escape. Additionally, for the 100kW vehicle we included a total of 10 Hall thrusters (8 active and 2 spares). Masses for all 100kW SEP Transportation Stage elements are summarized in Table 1 below.

SEP Cargo Vehicle Dry Mass	Mass, kg
Structures and Mechanisms	634
Main Propulsion System (thrusters, gimbal, PPU, cabling, tanks, feed system)	1234
ACS/RCS (dry)	86
Power System (solar array, PMAD, cabling)	1493
Avionics (GN&C, C&DH, Communications)	75
Thermal Control (radiators, insulation, etc)	706
Dry Mass w/o growth	4228
Reserve (30%)	1268
Total Mass for 100kW SEP Transportation Module	5496

Table 1: Masses for main elements of the SEP Transportation Module

For the purposes of this initial study, the power scaling between 100 and 400kW was done by holding constant the specific mass of the subsystems that scale with power. The number of tanks was adjusted to accommodate the total propellant mass, in 2000 kg increments.

5. Study Approach

With the SLS launch vehicle and SEP vehicle capabilities defined above, the remaining key constraints for the study are the destination orbit and the maximum allowable trip time for the cargo. For this assessment we chose a 1 Sol circular orbit at Mars (~17,000km) and a maximum allowable cargo trip time of 3.5 years. The analysis included 5% ΔV margin. Note that the allowable trip time has been extended significantly since our 2010 study, in which the trip time was constrained by the need to reuse the 600kW SEP Cargo transports due to their very high cost. For this study we varied the SEP vehicle power level between 100kW and 400kW and calculated the delivered payload as a function of power for specific impulses between 2500s and 3000s. Initial low thrust trajectories were calculated using the NASA developed Varitop¹⁸ and SECKSPOT¹⁹ codes, and the results for the final “optimized” cases were verified using Copernicus^{20,21} to ensure accurate low-thrust finite-burn propellant sizing trends.

6. Results

Starting with a straight comparison of the SLS Block 1B and 2B capabilities we calculated the delivered useful payload to Mars for a 100kW SEP Transportation stage for different drop off orbits, where the mass of xenon propellant decrease for higher initial orbits as the mission ΔV and trip time decrease. The 100kW power level is three or six times less than our 2010 study. Results are summarized in Figure 3. For this initial comparison and trip times of interest (~3.5 years or less), it is clear that the drop off orbit energy states of 35,786km (GEO) or higher orbits are necessary, and for the SLS Block 2B there is little payload penalty but substantial decrease in trip time by increasing that to ~440,000 km (LDHEO energy levels) to get to a trip time under 3 years. For comparison, if the same mission trades were run with an all cryogenic LOX/H₂ stage for cargo delivery the payload drops by more than 50% and the one-way trip 300 days. Changing the Mars orbit insertion stage to a storable chemical (NTO/MMH) with a LOX/H₂ Earth-departure stage, the payload drops to ~25% of the SEP capability. These comparisons make the benefits of SEP for cargo and logistics clear: either two or four times the non-time sensitive cargo can be delivered with SEP than with a chemical propulsion system, albeit with a trip time that is three times as long.

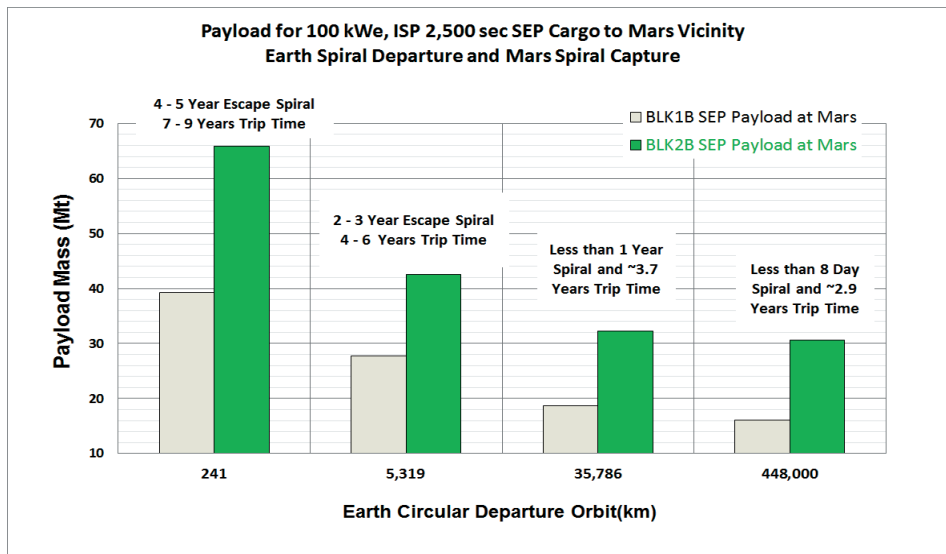


Figure 3: Impact of circular drop-off orbit on delivered payload mass and trip time to the Mars destination orbit for a 100kW SEP Cargo carrier using either the SLS Block 1B and 2B vehicles

Further comparing direct injection orbits to those requiring spiral Earth escape, we find that direct injection using either the Block 1B or 2B with a 100kW SEP system provides both higher payload and shorter trip time, as shown in Figure 4. This is driven by the trade between propellant mass, payload, and trip time for the performance of the EUS upper stage. As shown in Figure 5, the Block 2B vehicle with a 3000s SEP transportation system can deliver 41.8mT to the final 1 SOL Mars orbit in just under 3.5years with direct injection. Given this result, we focused on $C3 = -1.8 \text{ km}^2/\text{s}^2$ for all subsequent evaluations.

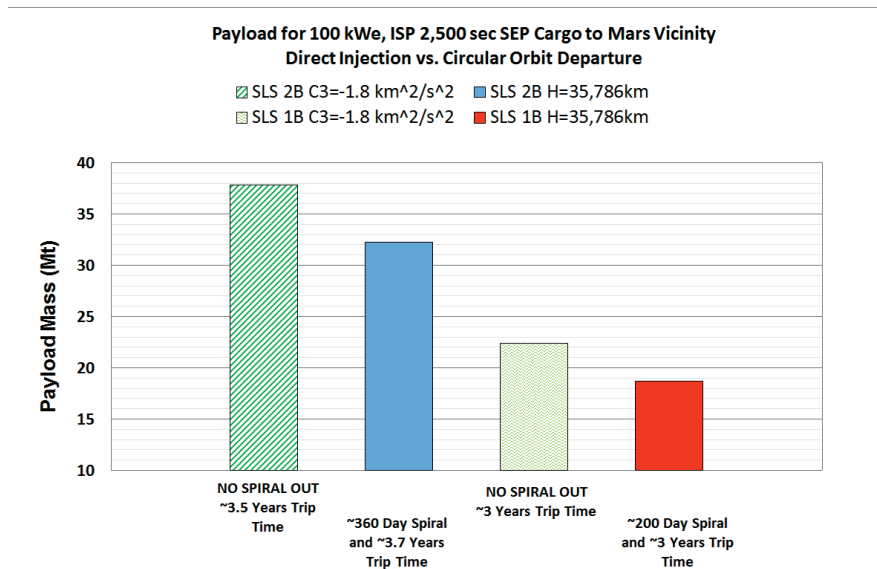


Figure 4: Comparison of delivered payload mass for SLS Block 1B and 2B for drop off orbits at GEO with spiral out and for drop off at $C3 = -1.8 \text{ km}^2/\text{s}^2$ (no spiral).

To examine the impact of power level and specific impulse on the delivered payload and trip time for the SLS Block 2B we varied the power between 100 and 400kW and the I_{sp} between 2,000s and 3,000s. The launcher drop-off energy was fixed at a $C3$ of $-1.8 \text{ km}^2/\text{s}^2$. The result is shown in Figure 5, where it is

seen that the maximum delivered payload of 41.8mT is obtained with a 100kW SEP vehicle at 3000s I_{sp} , and the minimum of 4.5mT payload is delivered at a power level of 400kW at an I_{sp} of 2000s. While these trends are not surprising given the scaling of power system and propellant mass for a given launch vehicle throw weight, note that the impact of power level is much larger than the impact of I_{sp} . For a given I_{sp} , the delivered payload mass increases essentially linearly with decreasing power level, with the slope being much higher for the lower specific impulse cases.

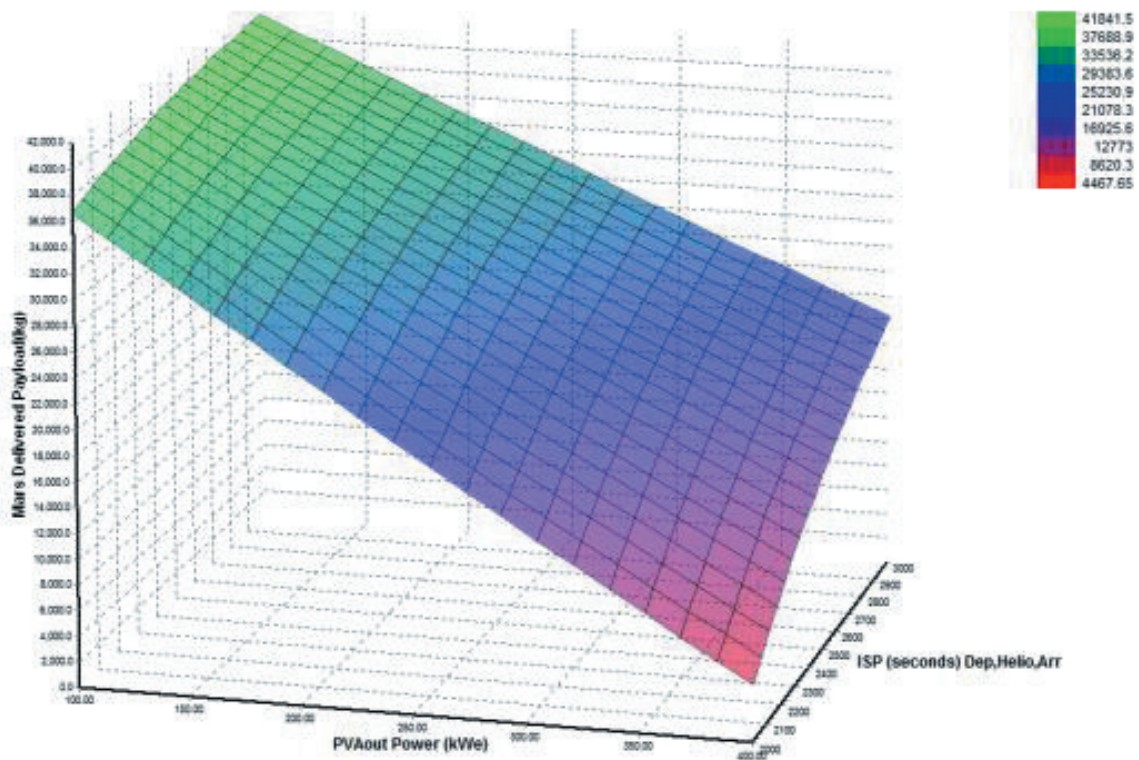


Figure 5: Impact of power and Isp variations on delivered payload mass for SLS Block 2B with a $C3=-1.8\text{km}^2\text{s}^2$.

For these cases, the trip time to the final 1 Sol Mars orbit for the lowest power, highest I_{sp} case (100kW, 3000s I_{sp}) with a payload of 41.8mT, is 3.4 years,, and the fastest trip time with a payload of 4.5mT is 2.4 years. These results are shown in Figure 6.

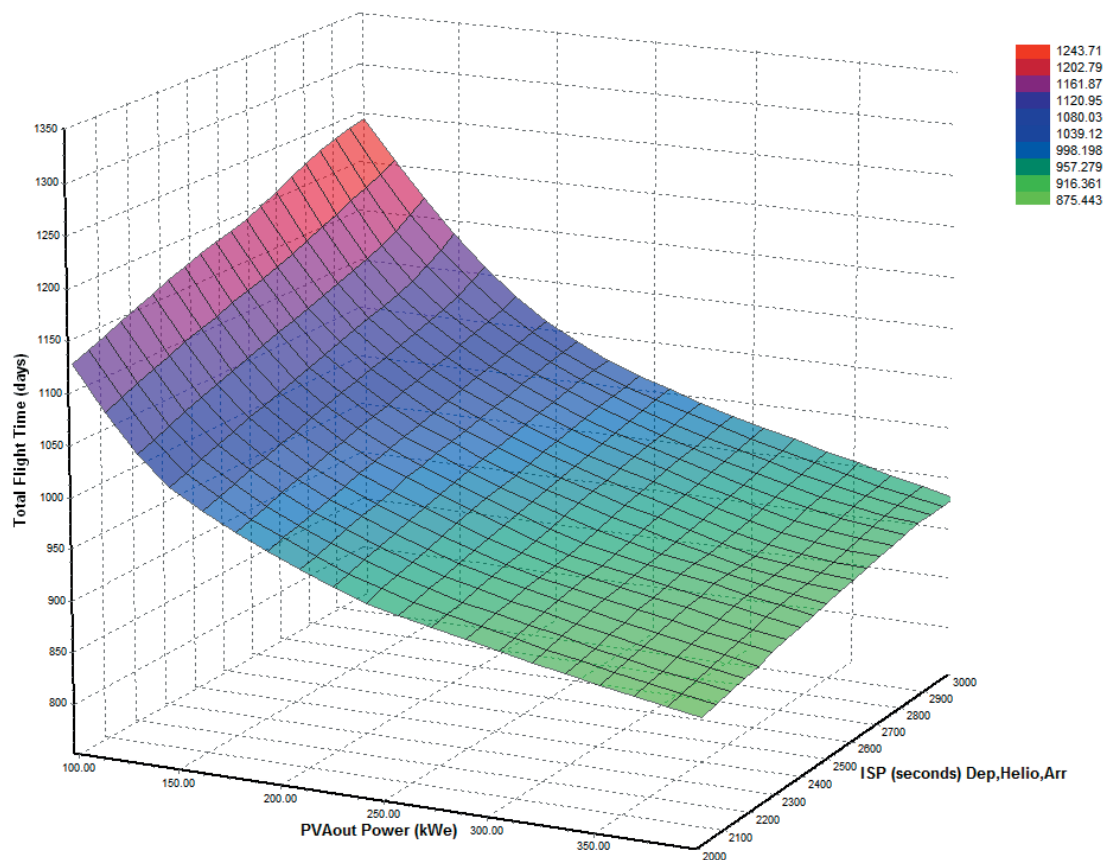


Figure 6: Total flight time from drop off by SLS Block 2B at $C3=-1.8\text{km}^2/\text{s}^2$ to arrival at destination orbit as a function of SEP power level and I_{sp} .

To summarize these results, the SLS Block 2B enables the SEP Cargo Vehicle to deliver 41.8mT to a 1 Sol orbit at Mars in 3.4 years. Increasing the power level decreases the flight time at the expense of payload mass.

7. SEP Transportation Module Configuration Options

In order to minimize the cost of the SEP Transportation Module it should be applicable to multiple missions for NASA, Defense, and commercial missions. We must avoid unique mission solutions requiring major development programs for each mission. This common SEP Transportation Module will result in distributing the fixed costs over multiple users and provides a production base to maximize manufacturing efficiencies. To date most spacecraft developments have used common subsystems at the component level, with thrusters, valves, tanks etc. being used for multiple platforms, and each space vehicle being uniquely optimized for its particular mission and destination. Technology maturation and capability margin demonstrations in both the power and propulsion elements are allowing us to consider shifting the “common elements” to a higher level of integration, namely the development of an integrated common SEP power and propulsion module that is applicable to multiple missions without separate, major development programs for each mission. This approach has the advantage of enabling a single

major module development program to serve the transportation needs for multiple missions and customers.

The transportation module envisioned is sized for 50kW, but is throttleable down to 30kW and accommodates either size arrays: either two 15kW wings or two 25kW wings. Each of the thruster strings consists of a power processing unit and thruster and consumes up to 14kW of input power, so there are either three or five thruster strings on the module, allowing for one spare thruster string. The common structure is designed to accommodate two, three, four or five xenon tanks, each of which holds up to 2,000 kg of xenon. The 30-50kW SEP Transportation Module is sized to serve NASA science missions, exploration robotic precursor and logistics missions, Defense, and emerging commercial transportation applications.

Growth to the 100kW – 200kW power range required for large scale human exploration of the vicinity of Mars can be accomplished in either of two ways as shown in Figure 7. First, multiple self-contained modules can be joined together with a lightweight structure with the payload mounted in the center. The low-thrust of the SEP vehicle enables a simple approach to this scaling: the structure can be very lightweight and the primary development will be GN&C software to ensure thrust vector alignment and

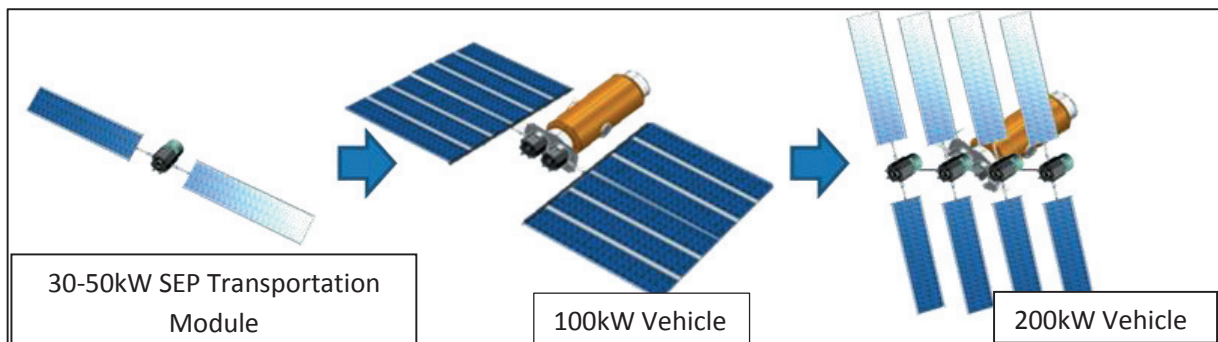


Figure 7: Two options for extending the 50kW SEP Transportation Module to 100kW or 200kW.

vehicle control. The primary issues will be thermal: accommodation of radiators on each module and the view-factors to adjacent modules will need to be addressed. The second scaling approach to a 100kW vehicle is to increase the array wing size to 50kW each while maintaining the module power level at 50kW: each array wing provides the entire power for each 50kW module.

8. Summary

An initial evaluation of using the SLS Block 2B launch vehicle to reduce the power level for SEP cargo and logistics has been completed. In contrast to our 2010 study, we show that this approach enables the reduction of SEP power levels to as low as 100kW for Mars payloads replacement of nearly 42mT in a 1 Sol orbit at Mars. Using these lower power levels significantly reduces both the development and production cost and risk of SEP transportation systems while providing the opportunity to create a common SEP Transportation Module with much broader applications across the existing and emerging space market.

References

1. "Exploration Systems Architecture Study," NASA TM-2005-214062, November 2005.
2. "Human Exploration of Mars Design Reference Mission 5.0," NASA-SP-2009-566, July 2009.
3. Spores, R., et al, "A Solar Electric Propulsion Cargo Vehicle to Support NASA Lunar Exploration Vehicles;" IEPC-2005-320; 29th International Electric Propulsion Conference, Princeton NJ, Oct. 2005. Download from ERPS website.
4. Myers, R.M. and Carpenter, C., "High Power Solar Electric Propulsion for Human Space Exploration Architectures," IEPC 2011-261; 32nd International Electric Propulsion Conference, Wiesbaden, Germany, Sept 11-15 2011. Download from ERPS Website.
5. Carberry, et.al., "Second Mars Affordability and Sustainability Workshop," Organized by Explore Mars, Inc., Keck Institute for Space Sciences, Oct. 14-16, 2014.
6. Cassady, J., et al, "A Sustainable Concept for Human Exploration Spacecraft," Space Exploration International Conference, Strasbourg, France, Oct 29-31, 2014.
7. Joyner, C.R., Levack, J.H, and Crowley, J. "Determining an Affordable Mars Mission Capable Thrust Size," Nuclear and Emerging Technologies for Space, Albuquerque, NM, February 2015.
8. Merrill, R., Qu, M., Nobel, H., and Strange, N., "Mars Conjunction Crewed Missions with a Reusable Hybrid Architecture", 8.0104 IEEE Aerospace Conference, March 8, 2015.
9. Price, H. W., Baker, J. D, Strange, N.J, and Woolley, R.C., "Human Missions to Mars Orbit, Phobos, and Mars Surface Using 100-kWe-Class Solar Electric Propulsion," AIAA 2014-4436; Space 2014 Conference and Exposition, San Diego, CA, August 2014.
10. Price, H., Baker, J.P., and Naderi, F., "Humans to Mars: Thoughts Toward an Executable Program," Humans to Mars Conference, Washington DC, April 2015.
11. Mathers, A., de Grys, K., and Paisely, J. "Performance Variation in BPT-4000 Hall Thrusters," IEPC-2009-144, 31st International Electric Propulsion Conference, Ann Arbor, Michigan. Download from ERPS Website.
12. "Aerojet Rocketdyne to Provide Electric Propulsion for Orbital Sciences Corporation's GEOStar-3 Satellites," Press release, February 3, 2015.
13. Patton, C., "All Electric Satellites: Revolution or Evolution," Via Satellite, May 1, 2013.
14. Global News Wire, "European Space Propulsion Receives European Space Agency Contract for Qualification of 5-Kilowatt Hall Thruster," May 17, 2015.
15. Oleson, S.R., Myers, R.M., Kluever, C.A., Riehl, J.P., and Curran F.M, "Advanced Propulsion for Geostationary Orbit Insertion and North-South Station Keeping; Journal of Spacecraft and Rockets, Vol. 34, No. 1, January-February 1997, pp. 22-28.
16. NASA Space Technology Mission Directorate, "Advanced Solar Array Systems," Oct 31, 2012; www.nasa.gov.
17. NASA Tech Briefs On-Line Edition; "Fact: Mega-ROSA, SOLAROSA", Oct. 26, 2012.
18. Sackett, L.L, Malchow, H.L., and Edelbaum, T.N., "(SECKSPOT) Solar Electric Geocentric Transfer With Attitude Constraints: Analysis," NASA CR-134927, 1975.
19. Sauer, Jr., C.G., "Users Guide To The Operation Of Varitop: A General Purpose, Low-Thrust, Trajectory Optimization Program," NASA Jet Propulsion Laboratory, Pasadena, California, 1998.
20. Ocampo, C., Senent, J. S., and Williams, J., "Theoretical Foundation of Copernicus: A Unified Systems for Trajectory Design and Optimization," NASA Technical Reports Server, Document ID: 20100017707; Report Number: JSC-CN-20552, May 21, 2010.
21. Williams, J., Senent, J. S., Ocampo, C., Mathur, R., and David, E. C., "Overview and Software Architecture of Copernicus Trajectory Design and Optimization System," NASA Technical Reports Server, Document ID: 20100017807; Report Number: JSC-CN-20553, May 21, 2010.