High Power Solar Electric Propulsion Impact on Human Mars Mission Architecture

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Abstract: Future human exploration missions to Mars are being studied by NASA, industry and academia. Many approaches to the Mars mission are being examined that use various types of propulsion for the different phases of the mission. The choice and implementation of propulsion system options can be optimally determined based on specific mission criteria such as: launch system, trip time, cargo or payload requirement at Mars, departure and arrival orbits, and mission campaign schedule. One way to reduce the size for a particular Mars mission architecture and possible cost of the mission is to split the cargo from the crew and pre-position the cargo or payloads required over several trajectories or missions. This can reduce the size of the most expensive item, the Mars crew vehicle. This paper will discuss the current Solar Electric Propulsion (SEP) cargo stage sizing for Aerojet Rocketdyne (AR) Mars architecture work performed in 2017 that is examining the impact of SEP system design choices. Trades are presented varying SEP power and voltage, spanning various Space Launch System (SLS) departure orbits while taking into account the mission requirements across several Mars mission opportunities beginning in the late 2030's.

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Nomenclature

AEHF	=	Advanced Extremely High Frequency
AR	=	Aerojet Rocketdyne
C3	=	Characteristic Energy, km^2/sec^2
EMC	=	Evolvable Mars Campaign
HEO	=	High Earth Orbit (e.g., 1,000 km circular or higher)
HEEO	=	Highly Elliptical Earth Orbit (e.g., 1,000 km x 21,000km)
I_{SP}	=	Specific Impulse, a measure of efficiency (Thrust per pound of fuel burned), higher is better
LDHEO	=	Lunar Distant High Earth Orbit (e.g., 241 km or higher perigee x 385,000 to 440,000km apogee)
LDRO	=	Lunar Distant Retrograde Orbit (e.g., approximately 70,000km semi-major axis around the moon)
LEO	=	Low Earth Orbit (e.g., 185 km circular)
LLO	=	Low Lunar Orbit
LGA	=	Lunar Gravity Assist
MOC	=	Mars Orbit Capture
MSC	=	Mars Study Capability (Team)
PPU	=	Power Processing Unit
PSM	=	Payload System Mass
SEP	=	Solar Electric Propulsion
SLS	=	Space Launch System
TEI	=	Trans-Earth Injection
TMI	=	Trans-Mars Injection
TRL	=	Technology Readiness Level
V_∞	=	Hyperbolic Excess Velocity, km/s

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 have baselined architectures that may present challenges to the technical and affordability aspects of a future Mars architecture, depending on the transportation system designs.[1, 2]

Aerojet Rocketdyne has been analyzing many of the elements that make up the technology portfolio for accomplishing any Mars architecture (e.g., Solar Electric Propulsion (SEP), power processing, chemical propulsion for landers, and nuclear thermal propulsion for Mars crew vehicles). The AR Mars architecture studies are examining SEP systems across power ranges from 100 to 250 kWe as part of a human Mars mission architecture where the SEP vehicle takes the cargo separately from the crew. The power level along with the design specific impulse (ISP) and specific mass for a SEP system play a significant role in determining how effective the propulsion system may be for delivering cargo during each mission opportunity for the Mars architecture.[3]

Understanding the key impacts of the propulsion system requirements for both cargo transfer and crew transfer to Mars is necessary to credibly assess the capability to successfully perform a Mars mission. Power levels can trade with ISP and SEP system mass to permit higher payload masses to Mars and will impact trajectory transfer time or the maximum payload capability of the launch system for a given departure orbit.

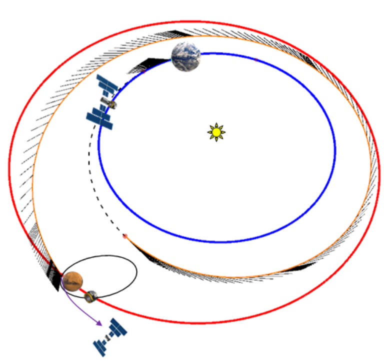


Figure 1: Example of the Earth to Mars SEP System for Cargo Delivery or Pre-positioning Mars Mission Elements.

Besides the impact of propulsion, other elements are important and can be strong drivers. Particularly important is the type and size of cargo (e.g., surface landers, pre-positioned propellant stages, Mars orbital taxi, etc.). The type and size of the payload can also impact the SEP sizing parameters because of when they are needed at Mars within a mission campaign. The SEP cargo vehicle size will be driven by the payload size and the ISP and power level for the thrusters. It will be impacted by how many cargo vehicles are used to get the total mission campaign required payload delivered and how often the vehicles are flown. The optimum SEP power level can be affected by whether the SEP is launched via a commercial launch system or the NASA SLS Block 1B or Block 2B vehicle for a given optimum departure orbit. Figure 1 illustrates a mission type AR is examining for Mars SEP cargo missions. The SEP cargo flies a trajectory that delivers surface payload or pre-positions return stages ahead of a crew mission. Figure 1 also illustrates a unique optimization for the SEP trajectory whereas using thrusting, then coast, and final thrusting for Mars fly-by or arrival is more fuel efficient and can reduce time-of-flight.

The Mars SEP cargo mission will need to use trades to determine the best power level for optimizing the timing of the cargo mission relative to the crew mission trip time and departure year. Thus, any Mars architecture study needs to collaboratively analyze the Mars crew vehicle propulsion, the Mars mission payloads, SEP power and propulsion capability, the SEP earth to orbit launch system method and departure orbit, the year of launch, and how long the mission will last to understand the impact of the choices. AR studied many of these design attributes based on early NASA SLS ground rules and system concepts in 2015.[4]

AR in 2017 updated the ground rules used in the Mars SEP cargo mission studies to align with results from the Evolving Mars Campaign (EMC) and the on-going Mars Study Capabilities (MSC) Team activity. AR is evaluating many of the architecture elements, designs and mission trajectory attributes in order to determine a path that provides for the most affordable Mars mission systems and propulsion technologies.[1, 2]

II. Ground Rules and Assumptions

The 2017 analysis ground rules and assumptions forming the basis of the update to the 2015 Mars SEP cargo studies were drawn from Percy[1] and Merril[2]. The mission opportunities assessed spanned from the early 2030s to late 2040s. A SLS Block 2B launch vehicle was assumed to be available to launch Mars SEP cargo missions to pre-position surface landers for the entire mission campaign. A typical Mars cargo mission would begin with an SLS launch into a highly elliptical orbit. A SEP tug would then begin a slow spiral trajectory from the drop-off orbit to an orbit corresponding to C3 = -2 km^2/sec^2. Next, the SEP tug would use a Lunar Gravity Assist (LGA) to achieve interplanetary trajectory starting at a C3 = 2 km^2/sec^2. The payload would then thrust along a heliocentric trajectory, with optimized periods for coasting, until Mars sphere of influence is reached, where at an orbit arrival velocity at or under 5 km/sec the payload (surface lander) is aero-captured into a 1 Sol Mars orbit and the SEP stage does a Mars flyby continuing in heliocentric space. Figure 2 depicts a Mars architecture campaign chart showing how the pre-positioning of the key mission elements is performed in conjunction with later Mars crew missions.

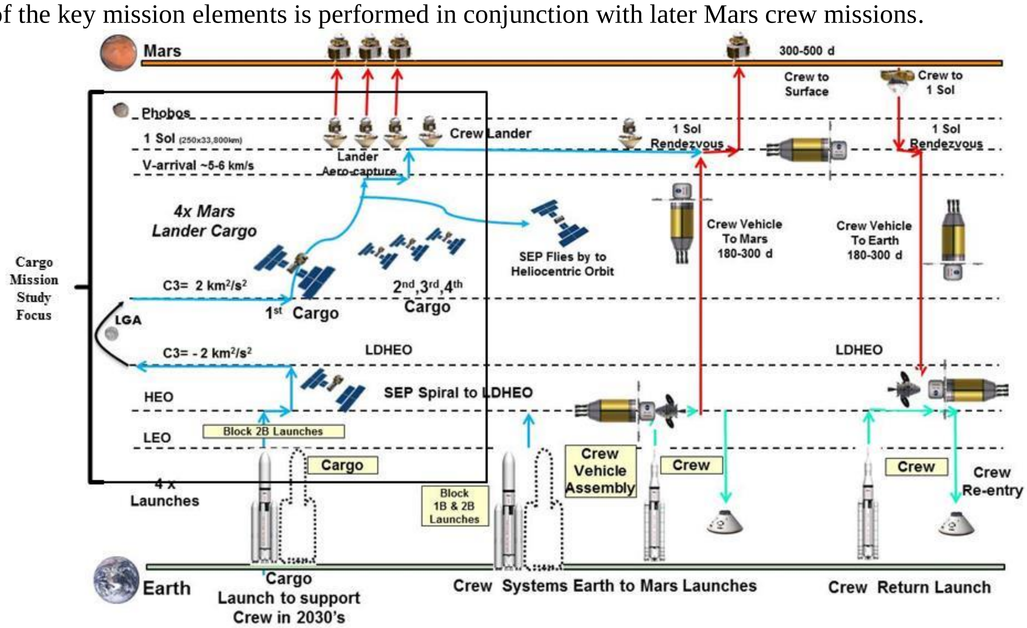


Figure 2: Mars Exploration Architecture and Mission Elements.

III. Launch Vehicle Capability

NASA development of the SLS vehicle is planned in phases, with the Block 1 SLS first flight scheduled for the late 2019 timeframe, Block 1B planned for later Exploration Mission sets in the early 2020's, and the final Block 2B vehicle in the late 2020's. The launch vehicle configurations are summarized in Figure 3.[5]

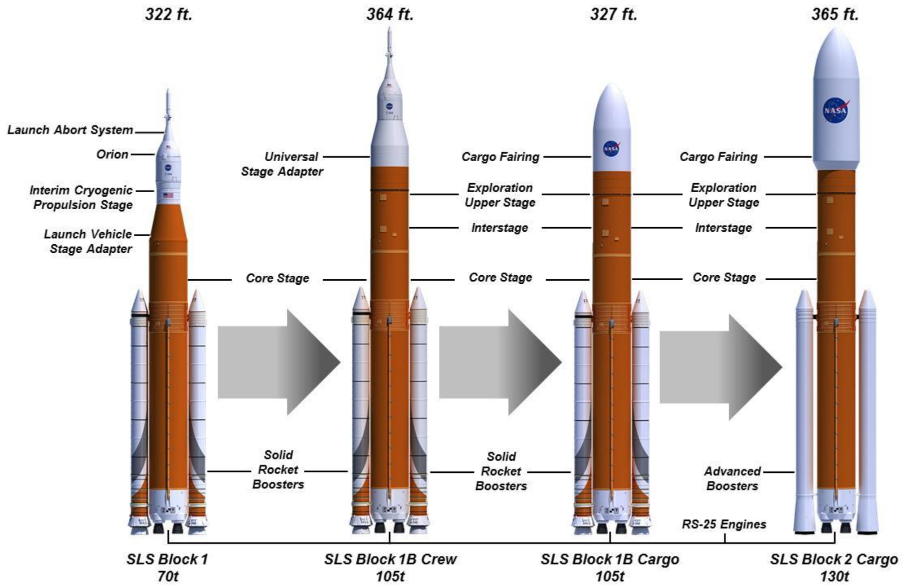


Fig. 3: SLS Block 1, Block 1B, and Block 2B configurations.

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. The key difference between Block 1 and Block 1B is the addition of the LOX/H_2 Exploration Upper Stage. Later the addition of advanced boosters for Block 2B increases the launcher payload capability.

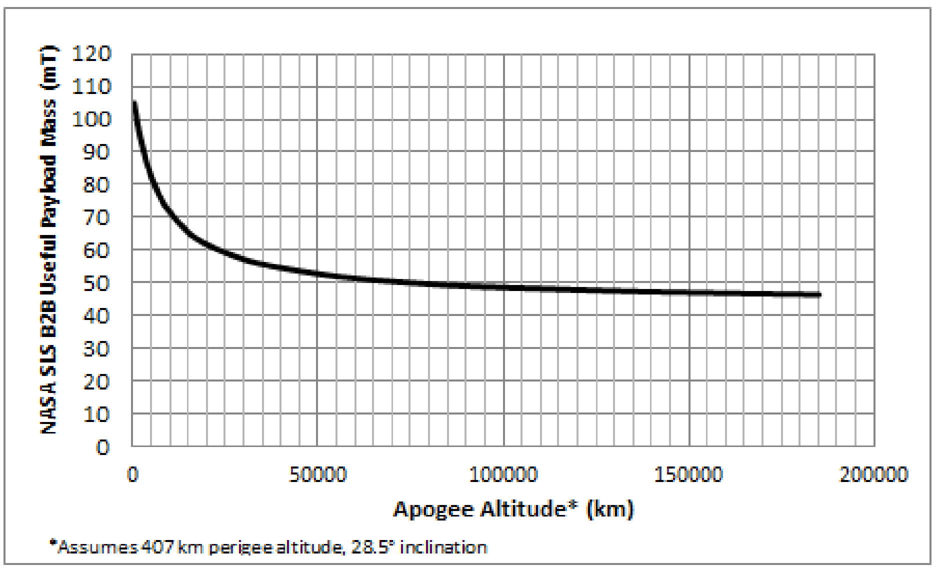


Figure 4: SLS payload capabilities examined in the current AR trade study.

Current plans call for the eventual capability of two to three SLS launches per year to support the human exploration mission objectives of reaching the vicinity of Mars in the early 2030s. The SLS Block 2B performance was assumed for all 2017 study trades. Useful payload system mass (PSM) was taken from the SLS Program Mission Planner's Guide-Version 2.[5] Figure 4 shows the NASA SLS useful payload mass used in the AR SEP cargo study. Useful PSM is the sum of the payload attachment system mass and payload mass. The payload attach fitting mass is assumed to be 2.5% of the payload mass. A payload multiplier is used to adjust the SLS Mission Planner's Guide payload values to align with the 45 mT TLI useful payload system mass not-to-exceed value from EMC.

IV. SEP System Capability Assumptions

Significant advancements have been made in the last five years in the development of higher power SEP systems. First, AR has supported three Advanced Extremely High Frequency (AEHF) satellite missions. Second, continued work by NASA has further illuminated the methods of achieving very long Hall thruster life, along with development of a 12kW long-life xenon Hall thruster. Third, high power solar array technology has advanced considerably with investments by industry and NASA. Notably the work on the Mega-ROSA and the Mega-Flex arrays has advanced the specific power to between 200 and 400W/kg, and increased the packaging density to over 50kW/m^3. [6,7]

For the 2017 Mars cargo study a conservative array specific mass of 5 kg/kW was used per recent SEP spacecraft module design work perfromed at AR. Note that for this mission analysis there is a powered transfer through the van Allen radiation belts since the vehicle is always launched to the optimum orbit for maximum launch vehicle payload capability to maximize Mars payloads. The maximum payload orbit could place the SEP cargo vehicle so that it has to transit the van Allen belts and be exposed to that radiation flux. For the spiral trajectories from the SLS drop-off orbit to an approximate C3= -2 km^2 /sec^2 a degradation between 15-30% occurred to the soalr array system. The stage solar array was oversized to account for this degradation to ensure nominal array output at Earth departure.

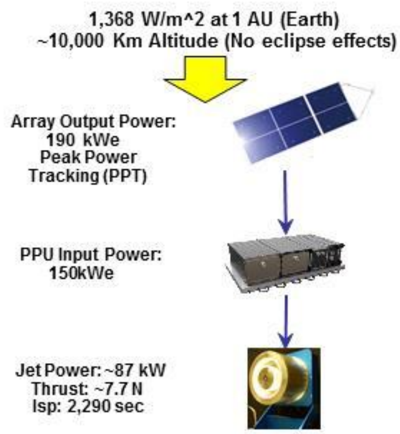
Additionally, for all the SEP cargo vehicles, multiples of the Hall thrusters were used (about 90% active with the rest as spares) to supply the propulsion capability. An example SEP cargo stage dry roll-up for a 100kWe design is shown in Figure 5. Figure 5 provides a summary captured from the SEP sizing model within the ModelCenter integrated multidisciplinary model.

200 kWe SEP Cargo Vehicle Dry Mass	Mass, kg
Structures and Mechanisms	1238
Main Propulsion System (thrusters, gimbal, PPU, cabling, tanks, feed system)	2414
ACS/RCS (dry)	167
Power System (solar array, PMAD, cabling)	2674
Avionics (GN&C, C&DH, Communications)	75
Thermal Control (radiators, insulation, etc.)	1684
Dry Mass w/o growth	**8252**
Reserve (30%)	2476
Total Dry Mass Vehicle	**10727**

Figure 5: Example 200 kWe SEP Cargo Mass Breakdown from SEP Sizing Model Element.

For the purposes of this study, SEP power levels were examined parametrically, scaling between 150 to 200kWe to the Power Processing Unit (PPU) at Earth departure. This was done holding the specific mass constant for the subsystems that scale with power. The sizing worksheet referenced above held to a 5% ΔV margin and sized to an integer number of tanks to account for the Xenon propellant load for a given Earth departure total mass. An integer number of tanks was first determined until the stage ΔV exceeded 5%, then a partial propellant loading was determined to add another tank to bring the ΔV margin down to 5%, in order to provide as much mass as possible for Mars net delviered payload .

Additionally, the AR integrated mission model used electric propulsion thruster data that was correlated to the thrusters being designed on the NASA Advanced Electric Propulsion System (AEPS) project. Under the AEPS contract, AR will develop, qualify and deliver 13 kWe class Hall thruster subsystems including thrusters, Power Processing Units, and xenon flow controllers. The 13 kWe class thrusters are about two-and-a-half times more powerful than AR's state-of-the-art XR-5 Hall thrusters, which are presently used to deliver government and commercial satellites to their geosynchronous orbit. AR will mature the thruster subsystem under development at NASA's Glenn Research Center (GRC) and the Jet Propulsion Laboratory (JPL) to a flight system capable of 50,000 hours of life. This flight propulsion system will form the core of SEP vehicles that NASA can use for efficient transportation of habitats and cargo needed for human exploration of deep space destinations beyond LEO. Data from AR's analysis that is on-going for the AEPS hall thrusters was used for the performance in this study. Figure 6 shows an example of how the Solar power to the arrays is adjusted to arrive at the final values to the PPU and the ISP performance that results for a given voltage.[8]



Study Parameters (Example Values)	
Isp (sec)	2,290
SolarArray BOL Output Power (kWe)	190
PPU BOL Input Power (kWe)	150
String Efficiency	58.0%

Figure 6: Illustration of Method Used for SEP Performance in the Study.

V. Study Approach and Results

A. Integrated Mission Model

AR has been working various elements of SEP systems for orbital and interplanetary missions. This work requires examining many disciplinary aspects of SEP design. In order to capture the requirements that flow into and across the engineering disciplines needed to design a mission optimal SEP system, AR created a multi-disciplinary modeling system. This multidisciplinary model, created using Phoenix Software's ModelCenter™, combines mission payload requirements, launch system performance, orbital mechanics analysis, trajectory analysis, and spacecraft systems sizing (e.g., SEP propulsion, solar arrays, propellant storage, power systems, and structure) to capture interactions of the design drivers.[9]

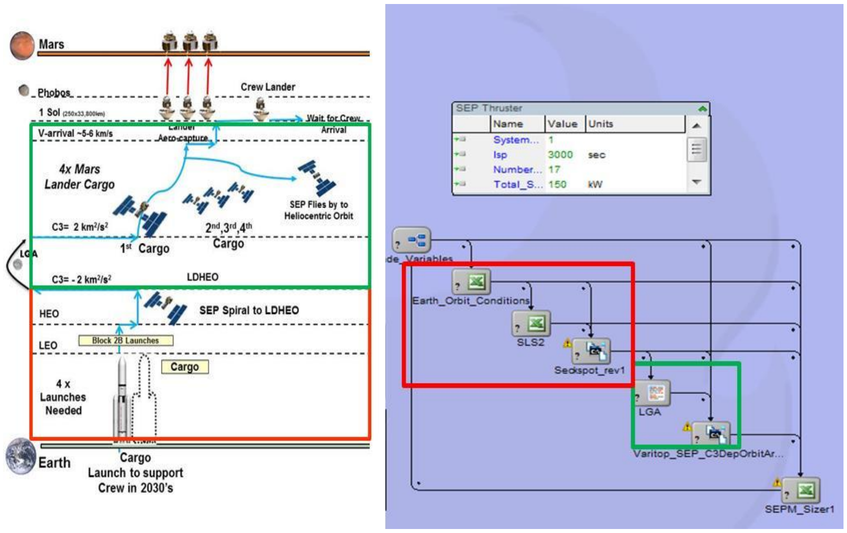


Figure 7: Mars SEP Cargo Mission Architecture Relationship to the ModelCenter Integrated Mission Model.

Figure 7 shows the ModelCenter™ framework and the specific analysis model elements linked to create the multi-disciplinary analysis modeling system. In Figure 7, the red boxes illustrate the connection between the Earth to orbit NASA SLS launches and the Earth departure architecture elements and the integrated model analysis tools that represent those elements. The green boxes in Figure 7 illustrate the connection for the in-space Mars SEP cargo elements and the integrated model analysis tools that represent those elements. The trajectory analysis results are used to then size a SEP cargo vehicle based on the total mass delivery capability of the NASA SLS and the required SEP propellant to get the net payload to Mars.

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 to examine the capability when the SEP cargo drops off the Mars payload (e.g., surface lander) at approximately 5 km/sec for aerocapture and later descent to the surface similar to the architectures presented in Percy[1] and Merril[2].

Initial low thrust trajectories were calculated using the NASA developed Varitop and SECKSPOT codes, and the results for the final "optimized" cases were then analyzed using the NASA Copernicus program to ensure validated low-thrust finite-burn propellant sizing trends. [10,11,12,13]

Early results indicated that for the power trade that Varitop would determine multiple solutions for the same input SEP power and performance. Several feasiblility assessments were performed with Copernicus versus Varitop to understand the results. It was determined that Varitop solution was highly sensitive to the guesses for the primer vector and other terms for the input SEP conditions and mission, It was decided to then use Copernicus as the heliocentric trajectroy analysis model element.

B. Trade Results

AR focused the primary trade study for this paper on the impact of NASA SLS B2B orbital capability for two variations in SEP power to determine what levels of Mars delivered payload were possible when attempting to deliver at within one year of a Mars crew mission in 2037. The SEP cargo mission was launched by a single NASA SLS into the optimum drop-off orbit that was determined iteratively based on maximizing the delivered Mars cargo to less than 5 km/sec arrival velocity at Mars. The mission start times (per the SLS launch) ranged from 2031 to 2032 to get the payload to Mars at least six to twelve months before the crew launch. This assumption was used for the 2037 crew mission to ensure enough time for surface payloads delivered by the cargo landers was fully operational before the Mars surface mission of 2037.

Using the modeling approach described in the last section, a trade study was performed using SEP power into the EP system at ~150 and ~200 kWe. The NASA SLS drop off apogee was varied across three distinct ranges (e.g., 1,000, 4,000, 7,500 km) while holding a perigee of 407 km. The EP power processing performance was held at current Technology Readiness Levels (TRL) around 400 volts to deliver around 2,300 seconds of ISP.

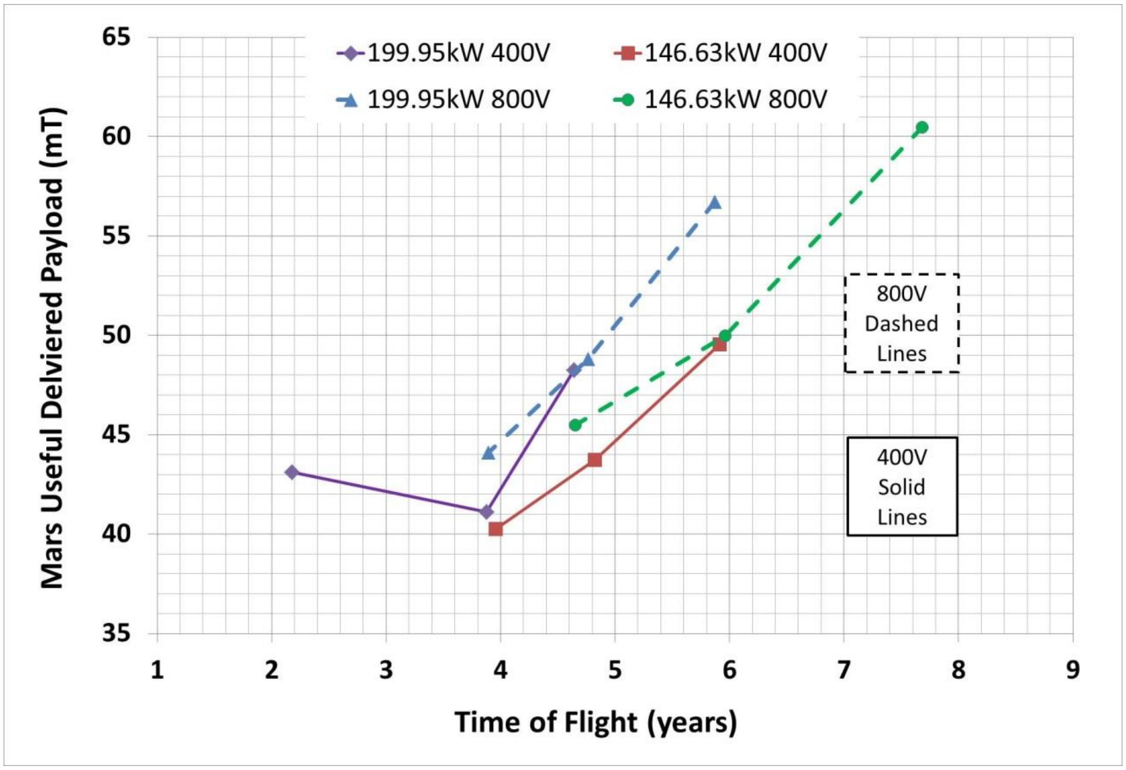


Figure 8: Mars SEP Cargo Trends for Varying SLS Drop-off Apogee and SEP Power.

The results for the SEP power trade are shown in Figure 8. It should be noted that for each point a uniquely optimized trajectory was calculated that permitted Earth departure spiral to the LGA point and then performing a burn-coast-burn low thrust trajectory to Mars vicinity. The thrust on and off times was a Copernicus optimization parameter while maximizing mass inserted at Mars. The results indicate different trends than the previous 2015 paper using the new integrated model. The key change impacting the Mars delivered payload was the capture of current NASA SLS performance to the Highly Elliptical Orbits (HEO) that were determined for best performance and letting Copernicus optimize each heliocentric trajectory for thrust on and off conditions. The results in Figure 8 show that the highest Mars payload occurs for time of flights (earth departure + heliocentric segment) at 4.65 years for the 200 kWe power and at 5.92 years at the 150 kWe power level. The payload capability decreases as time of flight decreases since the NASA SLS payload is lower for the higher drop-off orbit apogees. This decrease provides less SEP mass capability for both levels of power investigated. The trends are showing that even with the SLS B2B payload capability, getting more than 50 mT to Mars vicinity in one launch with the SEP stage and Mars cargo together will be a challenge.

VI. Observations from the SEP Trade Study

As stated in the Integrated Mission Model section, AR combined several model elements to create a multidisciplinary approach for performing the Mars SEP cargo study. One of the initial observations obtained during the analysis of the heliocentric trajectories with Varitop was that the time of flight and net payload mass was highly dependent on the initial guesses and starting state of the SEP cargo spacecraft. Several preliminary trades were performed varying the power and multiple trajectory solutions could be obtained depending on the initial guesses in Varitop affecting the Xenon propellant consumption and thus net payload capability. Time of flight was found to be very sensitive to where the SEP system split the coast and thrusting periods along the trajectory. Based on these preliminary trades, AR decided to then perform all the heliocentric trajectories with the higher fidelity Copernicus model that permitted examination of feasible low thrust trajectories before performing a final optimized trajectory. SEP power and voltage trades were then all analyzed using Copernicus for the Earth to Mars cargo missions. The trends for the voltage variation of 400 and 800 show that the higher performance SEP can exceed a useful delivered payload mass of 55 mT. Exceeding 54 mT is important because that is one of the upper limits for Mars landers being considered from the NASA EMC ground rules and in consideration by the MSC team. The challenge for the 800 volt SEP systems, that provide that higher payload mass, is that the time of flights are exceeding 5.5 years for the 199 kWe and 6.5 years for the 150 kWe power systems. This may prove to be problematic when 3 to 4 landers per crew mission are required and getting them all there in a constrained launch cadence with the crew mission adds risk to the overall mission architecture. An alternative solution that should be examined to get Mars payloads of 45 to 55 mT delivered is to orbit one large SEP stage at high power (e.g. greater than or equal to 200 kWe) to get shorter flight times or multiple medium 200 kWe power SEP stages in one SLS B2B launch to an optimum drop-off orbit that matches SLS capability to deliver 45 to 55 mT. Then follow the SLS B2B SEP launch with one or two SLS B2B launches with the large 45-55mT Mars cargo element that will rendezvous with the SEP stage or stages. This approach would permit use of the SLS maximum payload capability for a higher drop-off orbit and possibly get the time of flights down to two or three years at SEP power levels of 200 kWe or higher while achieving the highest payload delivery to Mars by a cargo system.

VII. Conclusions

Analytical methods and SEP performance data has been defined to search for Mars architecture elements to help determine an affordable path for human exploration of Mars. AR studies are showing that launch system, crew, and cargo in-space architectures elements must be analyzed together in order to find the total optimal Mars architecture and campaign sequence.

AR studies indicate that separating crew from cargo can reduce the required number of launches and permit more efficient lower power SEP transportation systems for cargo and pre-placement. Other AR studies on Mars systems have shown that this split architecture can also enable use of smaller crewed vehicles providing a path for more affordable and robust transportation. Modular propulsive stages optimize launched mass and mission flexibility compared to large monolithic stages and enable commonality between missions. The SEP cargo vehicles leverage recent advances in solar electric power and propulsion. Hall thruster based systems designed for propulsion modules at moderate power levels can perform Mars cargo missions with the timeframe that aligns with the crew mission that follows in later years. Vehicle level trades have shown that getting payloads exceeding 55 mT to Mars for the surface mission needs further analysis since time of flight is exceeding 5 years. What is more realistic is to examine what is required to get the Mars landers in at 45 mT since at a high power around 200 kWe and 800 volts that SEP cargo vehicle has a time of flight of around 4 years. The lower voltage, lower power 150 kWe SEP cargo vehicle can also deliver 45 mT with only a 1 year delay. The 400 volt/150 kWe SEP system may be attractive in regards to cost per SEP vehicle and longer thruster life (mission margin) as Mars architecture planning evolves to see if 45 mT is a good lander design and a 5 year time of flight can match launch system capability within the cadence required for a Mars mission series.

VIII. Future Work

AR has performed a trade study to look at the impact on SEP power level on the SEP stage sizing, Earth orbit to escape spiral time, Earth to Mars time of flight, and the optimum launch vehicle payload capability to maximize the payload performance to Mars orbit. As part of this analysis using several trajectory tools it was concluded that mission year, departure year conditions, trajectory constraints (e.g., launch window, heliocentric trajectory type), and trajectory optimization methods can significantly impact results. During the study it was found that the JPL Varitop trajectory program yielded a wide range of trajectory solutions depending on the guesses, SEP system masses, and the power levels when compared to the Copernicus trajectory analysis. Future analysis will focus on using Copernicus for all the heliocentric trajectory studies due to the greater flexibility in mission configurations and optimization techniques that permit higher fidelity as the SEP design evolves. Future work will examine the impact of the campaign sequencing of the multiple Mars cargo payload deliveries required prior to crew arrival on the overall architecture. Future work will include examination of Mars crew exploration mission year opportunities going out beyond 2040 to determine if those have any impact on selecting a common SEP cargo power level. Future work will include the addition of factors in the Copernicus models to permit higher fidelity array degradation for more detailed examination of the Earth orbit spiral impact on the large solar array area that is typical for the high SEP power levels. The future study activity will continue assessing the impact of SEP system operating voltage on delivered payload and transit time to Mars. AR architecture work will also include an examination of SLS B2B SEP cargo vehicle launches cadence along with launch of only payload for later rendezvous in an aggregation orbit as noted in Section VI. Additionally, with these improvements the Copernicus trajectory program will be interfaced directly within the integrated ModelCenter mission and system model for rapid design and analysis trade studies. Additionally, more architecture study is needed that optimizes the NASA SLS payload capability together with architectural variations for SEP stage size and number launched for a given Mars mission cargo requirement. This study would also look at the concept of operations and orbital phasing to determine timing for the multiple mission elements rendezvous and docking.

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