AAS 19-226

HYBRID TRANSPORTATION SYSTEM INTEGRATED TRAJECTORY DESIGN AND OPTIMIZATION FOR MARS LANDING SITE ACCESSIBILITY

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NASA's Mars Study Capability Team continues the agency's efforts to study and refine the nation's plan to field a sustainable human Mars campaign. One of the primary open issues left unanswered during previous analysis cycles was the uncertainty of the impact to the vehicle performance requirement to deliver crew and cargo to the same landing site across multiple mission opportunities. The Mars Study Capability Team has recently developed an integrated trajectory optimization and system closure model to solve the complex interplanetary trajectory optimization using both low-thrust and high-thrust maneuvers. This paper demonstrates the capability of this new integrated trajectory design and optimization method as it applies to the landing site accessibility problem for the Hybrid transportation architecture. The results showed that the current vehicle is capable of reaching up to +20 degree latitude and down to -20 degree latitude in every mission opportunity from 2033 to 2052. However, reaching latitudes beyond +/-20 required more propellant than the spacecraft is currently designed to carry for many of the mission opportunities.

INTRODUCTION

NASA's Mars Study Capability Team continues the agency's efforts to study and refine the nation's plan to field a sustainable human Mars campaign. Building upon the success of the Evolvable Mars Campaign,[1] the Mars Study Capability Team is further developing capabilities to improve the fidelity of the Mars campaign and to continue exploring the design trade space to assess the impact of technology investments and architecture decisions for missions to Mars in the coming decades. One of the transportation options currently under consideration by the Mars Study Capability Team is the Hybrid transportation architecture. The Hybrid transportation architecture combines a chemical propulsion system with a solar electric propulsion system into a single integrated design. By applying each propulsion system where it is most effective, the Hybrid architecture enables a series of Mars trajectories that are more fuel efficient than an all chemical propulsion architecture without significant increases to trip time. The Hybrid style trajectory enables the reuse of the transportation system for multiple trips to Mars and eliminates the need to develop separate transportation systems for crew and cargo.

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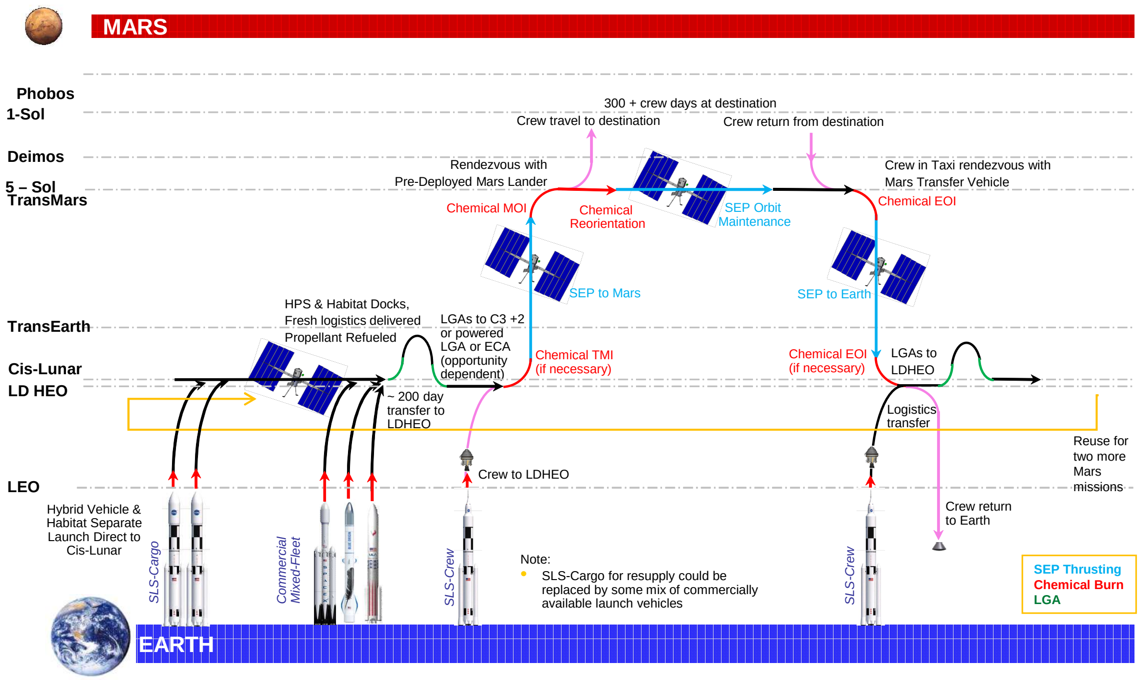


Figure 1. Mars Hybrid Crew Mission Concept of Operation

One of the primary open issues left unanswered during the Evolvable Mars Campaign analysis cycles was the uncertainty of the impact on the system concept design for the Hybrid architecture to deliver crew and cargo to the same landing site across multiple mission opportunities. To minimize the overall propellant requirement, the Hybrid architecture utilizes relatively low-energy transfers between the Earth and Mars. The spacecraft arrives at Mars with relatively low hyperbolic excess velocity, which limits the ability for the transportation system to insert into a specific parking orbit based on landing site requirements. Due to the performance limitations of the descent and ascent stages, the Hybrid transportation vehicle must arrive and depart Mars with the perigee of the parking orbit over the landing site for direct descent and due east ascent. Due to the limitations of the previous analysis capability, the Mars sphere of influence trajectory optimization problem was solved separately from the interplanetary trajectory optimization The results from the analysis show the high cost of the maneuvers that are required to reorient the spacecraft's parking orbit both to have access to different landing site latitudes and to enable proper orientation for the Earth departure maneuver when the maneuvers are not optimized with the interplanetary trajectories.

The Mars Study Capability Team has recently developed an integrated trajectory optimization and system closure model to solve the complex Hybrid trajectory problem. Typically, the trajectory optimization of a traditional all chemical transportation system is independent of the vehicle sizing and staging optimization of the particular transportation vehicle. For the hybrid system, because the thrust delivered by the SEP system is so low, the trajectory optimization is highly coupled with the vehicle sizing and optimization. This requires the trajectory optimization to be solved simultaneously with the vehicle sizing to achieve design closure. The development of this analysis capability enables additional architecture level trades to be completed with higher fidelity.

This paper demonstrates the capability of the new integrated trajectory design and optimization method as it applies to the landing site accessibility problem for the Hybrid transportation architecture. A full system level analysis shows the limitations of the current vehicle concept in reaching higher latitudes. Sensitivity of the system mass to increased landing site accessibility across the full Earth-Mars synodic cycle is presented to show the additional performance requirements on the system given the current constraint of system power.

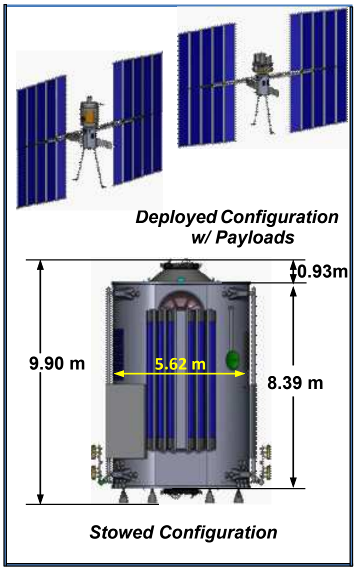
HYBRID TRANSPORTATION ARCHITECTURE

The initial Hybrid crew mission is depicted in Figure 1. Additional crew missions that reuse the integrated Mars spaceship begin with the vehicle in Lunar Distant Retrograde Orbit (LDRO) after the previous use. The crewed Mars mission begins with initial deployment and checkout of the integrated Hybrid Propulsion Stage (HPS) and the deep space transit habitat.[2] The stack launches separately on NASA's Space Launch System (SLS) directly to a characteristic energy (C3) of -2 km^3/s^2, targets a Lunar Gravity Assist (LGA) and performs a six month weak stability boundary transit to a stable orbit in cis-lunar space. A transfer orbit with C3 of -2 km^3/s^2 has an apogee altitude roughly equal to the moon's orbit. Upon arrival in a LDRO, the HPS and the habitat rendezvous with existing cis-lunar infrastructure, and resupply modules are launched (or are already waiting in LDRO) to transfer the propellant and logistics required for the Mars missions. The transit of the resupply modules from Earth to LDRO requires a more direct transit as compared to the Hybrid's weak stability transit due to the limited lifetime of the vehicle. Depending on the launch opportunity and the performance capability of the resupply module, a direct transit or a powered LGA transit is utilized.

After the HPS/Habitat stack has been fully fueled and stocked with logistics, the stack performs another six month weak stability boundary transit from LDRO to lunar distant high Earth orbit (LDHEO) via a solar perturbation loop with a pair of LGAs. The Mars crew is launched on an SLS directly to the LDHEO, where they rendezvous with the HPS/habitat stack, transfer final logistics, and depart Earth in the HPS/habitat stack to Mars. From LDHEO, one or two LGA propels the crewed HPS stack to a C3 of +2 km^3/s^2. After Earth departure, the SEP system produces thrust for much of the interplanetary trajectory to increase the vehicle's orbital energy to reach Mars. The crewed HPS stack arrives at Mars 300-400 days after Earth departure targeting a Mars close approach at 250 km altitude. The Hybrid's chemical engines performs a three-burn insertion maneuver to capture into a highly elliptical Mars orbit with a period of 5-Sol.

Upon arrival at Mars, a pre-deployed Mars taxi or a lander rendezvous with the crew HPS stack, then transfers the crew to their exploration destination. After the crew departs for their destination, the un-crewed HPS stack performs a series of maneuvers to reorient itself into the proper orbit for the return trip. After a minimum stay of 300 days in the Martian sphere of influence, the crew completes its exploration mission and returns to the HPS stack using the Mars taxi or a Mars ascent stage. From there, the HPS performs another three-burn maneuver to depart Mars. After Mars departure, the SEP thrusters produces near continuous thrust to reduce the spacecraft's energy to target an Earth arrival C3 of less than +2 km^3/s^2. The stack captures back into LDHEO via a one or two LGA sequence similar to Earth departure, but in reverse.

An SLS launches an empty Orion to LDHEO to rendezvous with the crewed HPS stack and return the crew to Earth. After crew return, the HPS stack transits from LDHEO to LDRO using either a slow transfer ($\approx$ 6 months) or fast transfer ($\approx$ 10 days) depending on the departure window and fuel availability of the next mission. The fast transfer would require additional fuel to be carried by the SLS that brought the empty Orion capsule. Once in LDRO, the HPS rendezvous with existing



Design Constraints/Parameters

Element Designed Lifetime		5500+ days
Destination		Mars
Stage Diameter		5.62 m
Stage Length		9.90 m
Main Propellant Type	Xenon	LOx/CH4
# Engines / Type	8 x 50kW Hall	6 x Cyro Thruster
Engine Thrust (100%)		4.5kN Each
Engine Isp (100%)	2600 sec	351 sec
# of Tanks	12x 2000 psi Tanks	1x LOx 1xCH4
Tank Material	COPV	Al/Ti
RCS Propellant Type		
# Engines / Type	52x Supercritical Ox/CH4 Gas RCS	
Engine Thrust (100%)	445N each	
Engine Isp (100%)	330 sec	
Power System		
Arrays	300V MegaROSA + 120V Body Mounted	
BOL Generation	675kW Main + 1.4kW Commissioning	
Structure	ISS SARJ Gimbals	
Cell Type/ Efficiency	IMM Solar Cells, 32% Efficient	

Category	Mass, kg
Structure	5,805
Thermal Control	2,036
Electric Propulsion	4,116
Chemical Propulsion	3,010
Electrical Power	6,299
Avionics & Control	138
Growth	6,426
DRY MASS SUBTOTAL	27,830
Max Xenon Load	34,000
Max LOx/CH4 Load	24,000
MAXIMUM WET MASS	85,830
Deep Space Habitat Payload	45,000
Mid-L/D Lander Payload	53,000

Figure 2. Hybrid Transportation System Vehicle Summary

cis-lunar infrastructure to perform refuel and resupply activities in preparation for the next trip to Mars.

Figure 2 shows a summary of the HPS vehicle's characteristics. The Hybrid vehicle utilizes both Solar Electric Propulsion (SEP) and liquid-oxygen (LOx)/liquid methane (LCH4) cryogenic chemical propulsion system to perform the roundtrip mission to Mars. The vehicle carries two wings of mega-ROSA (Roll Out Solar Arrays) capable of producing 675 kW of power at beginning of life at 1AU. The vehicle uses eight 50 kW class SEP thrusters and six 4.5 kN class chemical thrusters to produce thrust for the in-space maneuvers. The twelve composite over-wrapped pressure vessels (COPV) at 2000 psi can carry up to 34,000 kg of Xenon, and the two liquid propellant tanks can carry up to 24,000 kg of LOx and LCH4.

TRAJECTORY ANALYSIS FRAMEWORK

The initial analysis capability that was utilized to study the Hybrid architecture is based on JPL's Mission Analysis Low-Thrust Optimizer (MALTO).[3] The developed tool integrates MALTO with a vehicle sizing routine to provide vehicle closure and ensure the vehicle's performance meets the specified requirements. Low-thrust Earth-Mars and Mars-Earth trajectories are modeled separately and integrated with the sizing algorithms in which the chemical and SEP elements are sized independently based on their propellant requirements. A separate, independent analysis code is utilized to evaluate and optimize the Mars sphere of influence maneuvers that are required.[4] An outer loop mission assumption and analysis tool was created with Visual Basic scripting and Python codes to assist with the input and output of the analysis. The tool also serves as the primary vehicle sizing and closure analysis. This analysis framework was used to support the Evolvable Mars Campaign[1] analysis cycle for the Hybrid architecture and a significant amount of the previously published Hybrid architecture results.[5,6]

One of the primary limitations of this analysis framework is the inherent segmentation of the

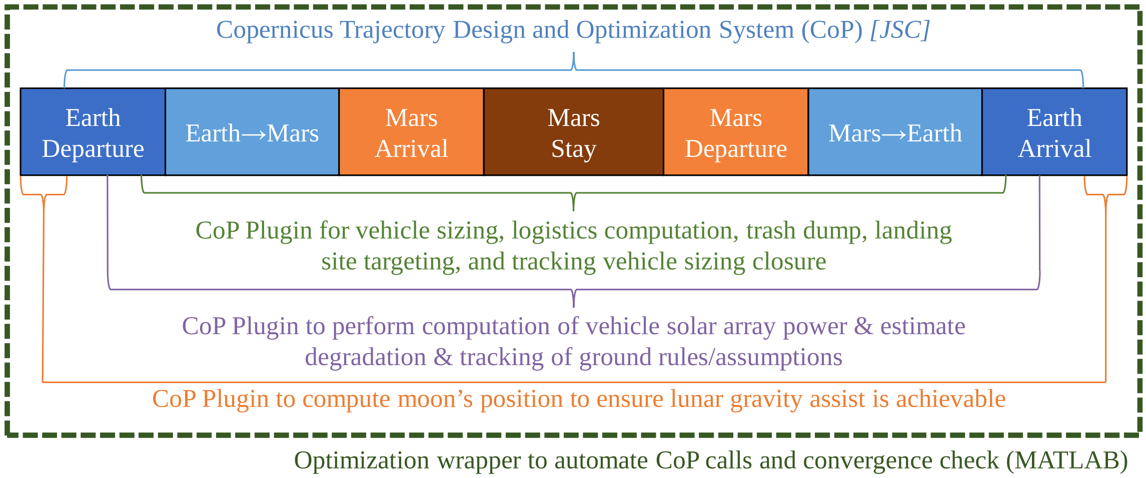


Figure 3. Copernicus Trajectory Design and Optimization System-Based Hybrid Trajectory Analysis and Optimization Framework

optimization problem. Because multiple analysis tools and methods are required to solve the overall problem, each of the tools is optimizing its own set of functions and metrics, and there is little to no overall optimization. The separation of the low-thrust trajectory optimizations from each other and from the planetary departure/arrival optimization results in solutions that are inherently suboptimal. Additionally, the circular nature of the trajectory optimization and vehicle sizing problem makes it impossible to perform global optimization in the framework. Solving any particular problem in this framework requires significant computational time and user input. The solution requires a large number of initial guess and propagation cycles to achieve overall closure. Large design space sweeps, like one required to understand the sensitivity of the spacecraft to changing payload and/or power,[7] require hundreds of cases to be run over a period of months.

To remedy the deficiency of the previous analysis method, an integrated optimization method was desired. To achieve more global optimization results, the two low-thrust trajectories must be solved simultaneously with the Mars sphere of influence problem. Ideally, the vehicle sizing and closure should be part of the overall optimization as well, as the vehicle dry mass has a significant impact on both the low-thrust and chemical propulsive requirements. A new analysis framework has been developed using the Copernicus Trajectory Design and Optimization System.[8] The overall integrated framework is illustrated visually in Figure 3. In this framework, the entire trajectory from Earth departure to Earth arrival can be modeled as a single integrated trajectory with multiple segments. Copernicus solves the trajectory problem by connecting the different segments while tracking the global optimization variable, which is typically minimizing the initial mass. This method provides a significant increase in the ability for the optimizer to find the globally optimal solution, as it is able to have control of the optimization variables.

To allow the vehicle sizing and closure to be integrated into the Copernicus trajectory optimization, a plug-in was developed at NASA Langley and compiled to calculate the parking orbit of the transportation system, compute the orbit reorientation maneuvers required at Mars sphere of influence, provide the Copernicus optimizer access to the different variables that are required to size the vehicle, and to ensure enough propellant is available to perform the roundtrip mission. The plug-in also allows for computation of the trash dumps[9] and the logistics that are both functions of the

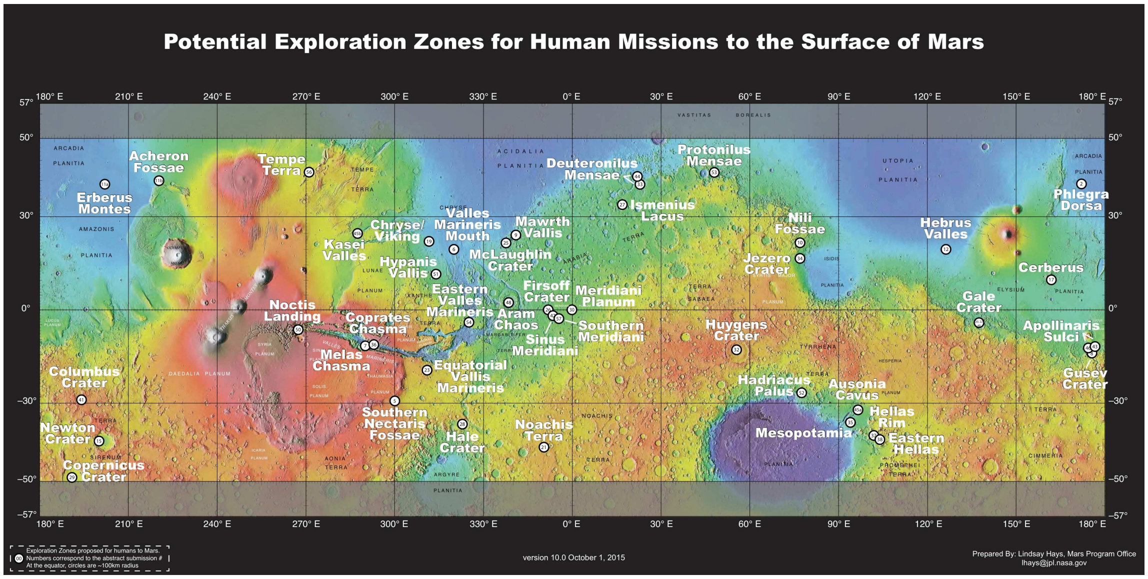


Figure 4. Potential Exploration Zones for Human Missions to the Surface of Mars from NASA's 2015 Mars Human Landing Site Workshop

transit duration. A second plug-in allows for tracking of the vehicle's power consumption, the array degradation across multiple mission opportunities, and the ground rules and assumptions, such as maneuvers' ΔV budgets, and vehicle sizing parametric variables. Finally, a third plug-in was developed to track the Earth departure and arrival V_∞ direction to ensure the Moon is in the proper location during departure and arrival to allow for the LGAs. The third plug-in sets a constraint for the departure and arrival dates to ensure the LGAs are possible.

SURFACE LANDING SITE ACCESS

Landing site latitude plays a crucial role in determining the optimal trajectories for traveling to Mars. Figure 4 is a chart presented at NASA's 2015 Mars Human Landing Site Workshop showing all of the potential sites that are of interest for human exploration. The sites range from negative 50 to positive 50 degrees latitude and across all longitudes. In all previous NASA Mars exploration studies, the limitation of the landing site latitude was not a major factor in the design constraints because previous studies assumed new landing sites for each mission opportunity. With NASA's focus on setting up a permanent presence on the Martian surface, the ability for the chosen mission architecture to provide access to the same landing site across multiple mission opportunities is essential. Due to the limitations of the descent and ascent stage, the burden of providing this capability falls on the in-space transportation system. Without performing a costly plane change maneuver, the orientation and the latitude of the parking orbit achieved by the trajectory optimization are governed by the geometry of the Earth-Mars orbit alignment.

Figure 5 shows the latitude of the parking orbit that is achievable with the optimal interplanetary trajectory. As the figure show, the latitude varies between plus or minus 20 degrees from opportu-

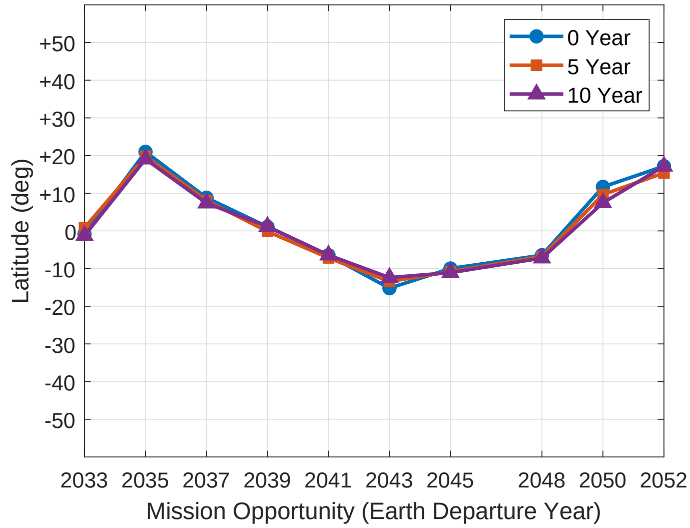


Figure 5. Latitude of Parking Orbit Periapsis of Unconstrained Optimal Trajectory Minimizing Total Propellant Mass Across Multiple Mars Mission Opportunities and Varying Hybrid Propulsion System Age

nity to opportunity. In order to reach a landing site that is not the optimal for any particular mission opportunity, traditional chemical propulsion-based mission architectures would utilize the propulsion system to perform plane change maneuvers or, if the parking orbit precesses fast enough, wait for the orbit to align properly for landing site access. For the Hybrid mission architecture, neither of these options are feasible. First, because the Hybrid mission architecture is an "all-up" architecture, carrying additional propellant to perform the plane change maneuver will grow the system exponentially. Second, to reduce the total energy required to perform the roundtrip mission, the Hybrid architecture utilizes a 5-sol orbit as its parking and staging orbit at Mars. The 5-sol orbit does not precess fast enough to allow the crew to wait for the orbit to align for landing site targeting. Lower altitude parking orbit have been considered in the past; however, the propulsive cost to achieve lower orbit is significant especially for the "all-up" nature of the Hybrid architecture.

The Hybrid Transportation System does have an advantage over the traditional chemical transportation system. Once the chemical propulsion transportation system performs its trans-Mars injection maneuvers, the trajectory is relatively set with minor mid-course maneuvers performed to clean up any thrust errors or to correct the course for navigation errors. The Hybrid propulsion system can perform a similar (but smaller) trans-Mars injection maneuver, but can utilize the SEP system to contentiously thrust during the transit to change the interplanetary trajectory to target a proper insertion maneuver to align the parking orbit properly for any particular landing site.

Figure 6 shows the propellant loading requirement comparison between the unconstrained trajectory from Figure 5 and a set of constrained trajectories that allows the optimizer in Copernicus to target a landing site latitude of 18.8^o (Jezero Crater, the current reference landing site), and also targets a departure parking orbit inclination of 18.8^o to support a direct, due east ascent from the Mars ascent stage. As the figure show, in nearly every mission opportunity, additional propellant is required to allow the orbit to target the landing site latitude of 18.8^o. More propellant is required the further away 18.8^o is from the unconstrained latitude seen in Figure 5. 2035 and 2052's un-

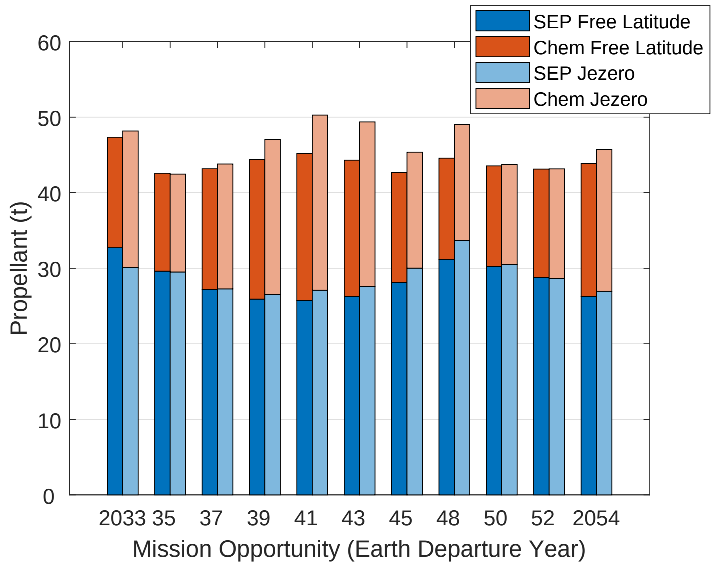


Figure 6. Propellant Requirement Comparison Between Unconstrained "Free Latitude" Trajectory and Constrained Trajectory Targeting Landing Site Latitude of 18.8^o (Jezero Crater) For Mission Opportunities from 2033 to 2054

constrained orbit latitudes are nearly the same as Jezero crater, and thus the propellant demand is nearly the same. One interesting note from this analysis is that most of the additional propellant is chemical, and not SEP. The SEP thrusting, and in turn propellant usage, is limited by the electrical power of the system as well as the time it has to thrust. As the Hybrid mission architecture utilizes conjunction class missions, the SEP does not have additional time to thrust beyond small perturbations to the optimal trajectory duration, especially with a minimum 300 day stay time constraint. Thus, the system will utilize more chemical propulsion to enable targeting of the specific parking orbit orientation for landing site accessibility.

HIGHER LATITUDE ACCESSIBILITY

As Figure 4 shows, the primary points of interest for human exploration on Mars are between plus or minus 50 degrees latitude. In an ideal mission design cycle, the landing site of the permanent outpost for human exploration would be determined prior to the design of the transportation system so that the system can be optimized to target that particular landing site across all the mission opportunities of interest. However, the reality of the mission design cycle is that the vehicle design must be robust enough to accommodate different landing sites, as those decisions typically happen much later in the architecture and mission design cycle. Therefore it is desired to understand the performance requirements across the range of the possible landing sites.

Figure 7 shows the propellant demand to reach plus or minus 50 degrees latitude for the 2033 mission opportunity for the current HPS as well as the SEP and chemical propellant capacity. For the 2033 mission opportunity, the current HPS is capable of reaching every latitude from plus 50 to minus 50 degrees. From Figure 5, the optimal trajectory for the 2033 mission opportunity has a parking orbit latitude that is nearly equatorial, which shows Figure 7 as 0 degrees latitude is the minimal propellant demand solution. Additional propellant, mostly in the form of chemical propellant, is required to reach higher latitude landing sites. As seen in the previous discussions,

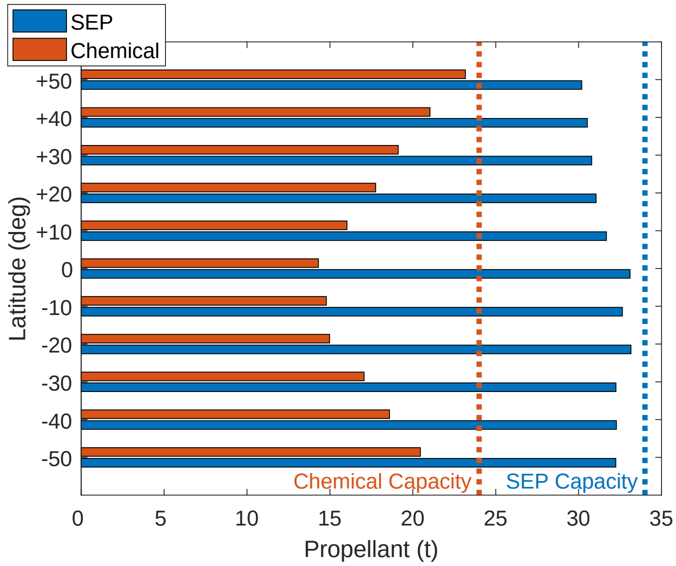


Figure 7. Propellant Demand for 2033 Mission Opportunity to Reach $\pm$ 50 Degrees Latitude for the Current Hybrid Transportation System with No Solar Array Degradation (0 Year Vehicle)

the SEP system is limited by time available for thrusting, and thus chemical propellant is utilized to reach higher latitudes.

The performance split between the SEP system and the chemical system can be seen more explicitly in the vehicle's ΔV comparison. Figure 8 shows the main propulsion system ΔV for the 2033 mission opportunity. For the SEP system, the plot depicts an effective ΔV based on the system's propellant use and the effective specific impulse. For the primary burns of the chemical systems, the maneuvers are the Trans-Mars Injection (TMI), the Mars Orbit Insertion (MOI), the parking orbit reorientation maneuvers which are required to orient the parking orbit between the arrival declination and the departure declination, the Trans-Earth Injection (TEI), and finally the Earth Orbit Insertion (EOI). Note that the TMI and the EOI maneuvers are absent from the 2033 mission opportunities due to the fact that the lunar gravity assist maneuvers are capable of producing enough Earth departure and arrival energy to eliminate the burns. As the figure shows, the near equatorial solutions all result in higher effective SEP ΔV compared to higher latitude solutions. Conversely, the equatorial solutions minimize the use of chemical ΔV to minimize the overall propellant mass that is required. This is because the optimal trajectory, from the planetary alignment perspective, yields solutions that do not have large out of plane components to the thrusting vector, staying relatively equatorial, as seen in Figure 5. To reach higher latitude, more chemical ΔV is required to achieve the desired parking orbit orientation. For the 2033 mission opportunity, this manifests itself mostly in the form of the MOI maneuver and the reorientation maneuver. The TMI and EOI maneuvers are not necessary for the 2033 mission opportunity given the current HPS configuration, and the TEI maneuver doesn't change much across different latitudes. The constant TEI maneuver is likely due to the reorientation maneuver being applied to align the parking orbit properly for an optimal Earth departure velocity direction.

As Figures 5 and 6 showed, the propellant demand for each of the propulsion system can vary both across mission opportunity and targeted landing site latitude. Figure 9 shows the propellant

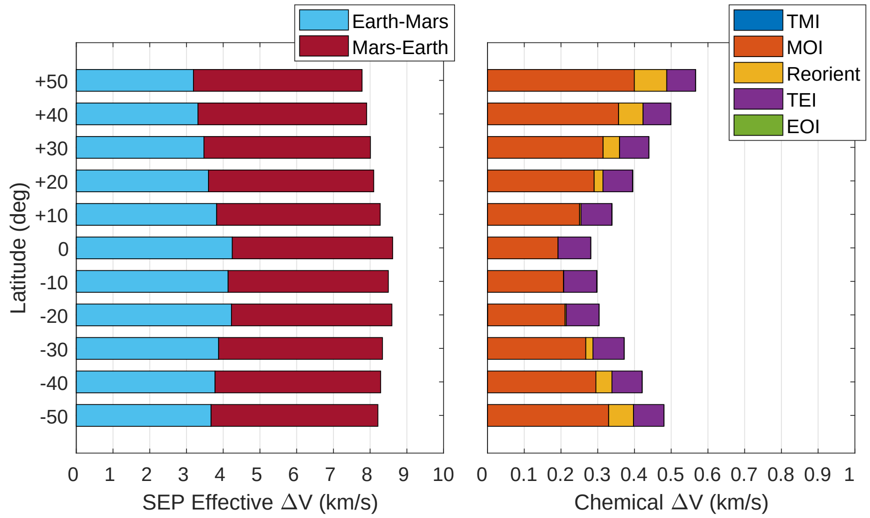


Figure 8. SEP and Chemical Propulsion System ΔV for the 2033 Mission Opportunity to reach $\pm$ 50 Degrees Latitude with No Solar Array Degradation (0 Year Vehicle)

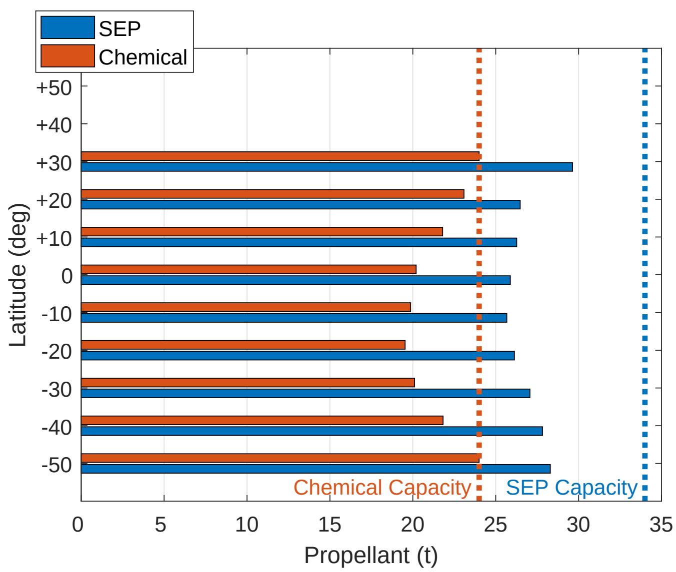


Figure 9. Propellant Demand for 2041 Mission Opportunity to Reach $\pm 50^o$ Latitude for the Current Hybrid Transportation System with No Solar Array Degradation (0 Year Vehicle)

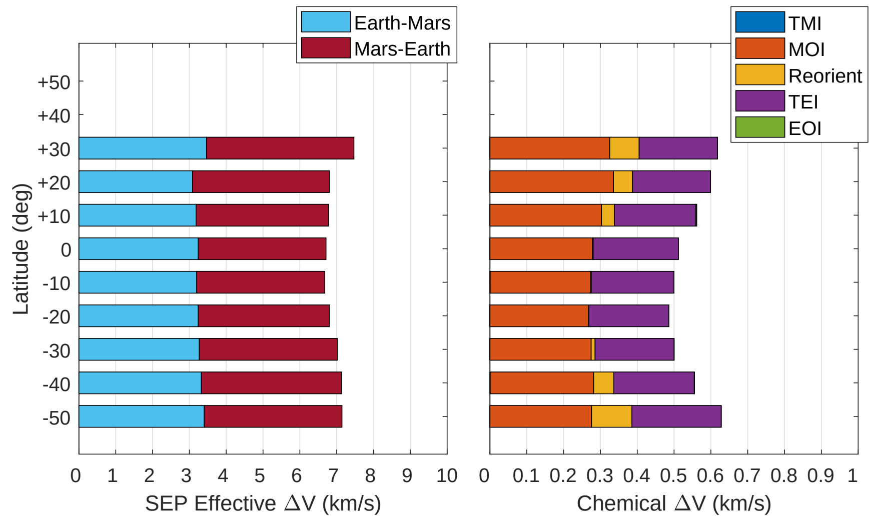
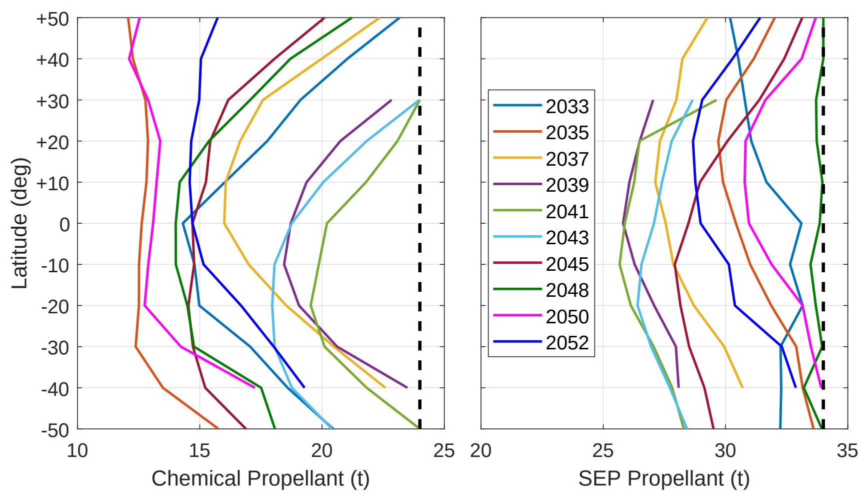


Figure 10. SEP and Chemical Propulsion System ΔV for the 2041 Mission Opportunity to reach $\pm$ 50 Degrees Latitude with No Solar Array Degradation (0 Year Vehicle)

loading requirements for the 2041 mission opportunity to reach plus or minus 50 degrees latitude. The figure shows that with the current HPS design parameters, the vehicle does not have enough propellant capacity to reach latitudes higher than plus 30 degrees, as the plus 30 degrees solution has already leveraged the full chemical propellant tank capacity. Even though the SEP system still has capacity in its propellant tanks, the interplanetary trajectory optimization was unable to find a solution given the current mission constraint that was able to utilize more of the SEP propulsion system to reach greater than plus 30 degrees latitude for the 2041 mission opportunity.

Examining the ΔV breakdown for the 2041 mission opportunity shows how much the system performance requirements can vary from one mission opportunity to another. As in the 2033 mission opportunity case, the different latitude targeting did not significantly affect the SEP portion of the effective ΔV for the 2041 mission opportunity, shown in Figure 10. However, comparing the 2033 and 2041 mission opportunity SEP effective ΔV shows a dramatic decrease in the use of the SEP system in the 2041 case, from 7.5 - 8.5 km/s in 2033 to 6.5 - 7.5 km/s in 2041. The geometry between the Earth and Mars plays a primary role in determining how effective the SEP system can be, given the current HPS design parameters. Comparing the chemical system ΔV between the 2033 and 2041 mission opportunities also shows the dramatic change in how the system performs the overall mission. In the 2033 case, higher latitude access requires additional MOI maneuver ΔV to achieve; however, in the 2041 case, the MOI maneuver is relatively insensitive to the higher latitude targeting. The additional ΔV required to achieve higher latitudes is mostly applied in the parking orbit reorientation maneuver.

These figures and discussions illustrate one of the primary challenges of mission design for the Hybrid architecture, as each opportunity presents different challenges to the two propulsion systems. Figure 11 shows the propellant demand to reach plus or minus 50 degrees latitude for mission opportunities between 2033 and 2052. The top row of figures show the propellant demand by type, while the bottom figure shows the total propellant combined. Note that even though the maximum capacity of the Hybrid vehicle's propellant tanks (between both SEP and chemical) is 58t, no solutions required that cumulative limit. However, there are multiple opportunities that reached the capacity



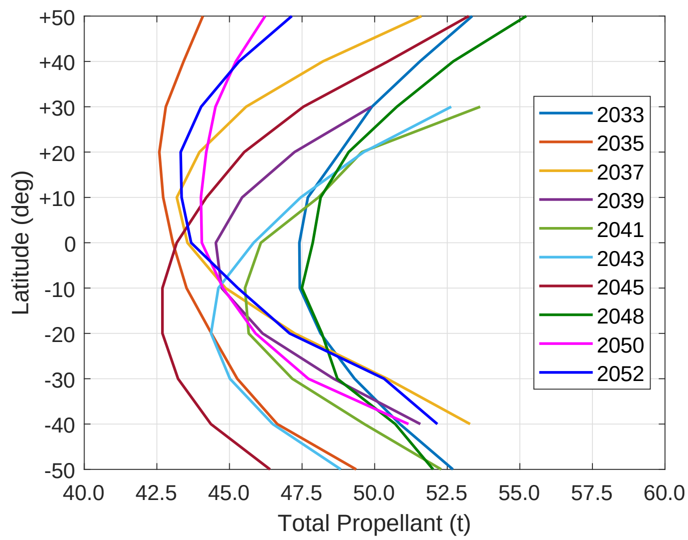


Figure 11. Propellant Demand to Reach $\pm$ 50 Degrees Latitude with No Solar Array Degradation (0 Year Vehicle) for Mission Opportunities Between 20330 and 2052

Table 1. Hybrid Propulsion System Age and Solar Array Degradation Assumptions

Mission	1st	2nd	3rd
HPS Age	0 Year	5 Year	10 Year
Array Degradation	0%	7.5%	15%
Array Power at TMI	675 kW	624 kW	574 kW

of the individual propellant types when trying to reach higher latitudes. This further demonstrates how different each mission opportunity can be for the HPS and how different opportunities stress different aspect of the trajectory when reaching higher latitudes. Some opportunities, like 2033, 2035, 2048, and 2050, can reach all latitudes from -50 to +50 with the current HPS design parameters and no array degradation. All other opportunities do not support full latitude range access. The 2039 mission opportunity is the most restrictive, as it is only able to reach from -40 degrees to +30 degrees, although it has one of the lowest SEP propellant uses across all opportunities.

Additionally, as seen in Figure 5, the minimum propellant latitude changes across the different mission opportunities. The 2033, 2039, and the 2048 mission opportunities all have optimal solutions near the equator. 2035, 2037, 2050, and 2052 have propellant optimal solutions in the northern hemisphere, while the other opportunities have optimal solutions in the southern hemisphere. Again, this presents a challenge to mission architecting, as the landing site selection can have a dramatic impact on the overall propulsion system performance requirements. Figure 11 also shows the overall sensitivity of propellant mass to higher latitude landing sites can be diverse for different mission opportunities. From an architecture decision standpoint, it would be easy to select those mission opportunities that have relatively benign propulsion sensitivity to landing site latitude accessibility. However, this could present other challenges, as any mission date slip could move the mission from a benign opportunity to a harsh opportunity. Additionally, the mission architecture calls for deployment of surface access using HPS in cargo modes, and thus regardless of the benign opportunities that are chosen for the crew missions, there will be instances in which the HPS has to perform a mission during harsher opportunities. It is imperative that a mission designer takes into consideration the challenges of the difficult opportunities and does not focus solely on the easy ones.

ARRAY DEGRADATION CONSIDERATION

The final consideration for landing site accessibility for HPS mission design is the effect of the HPS age and the impact of solar array degradation. So far, all of the discussion and the figures shown in the paper assumes a brand new HPS, leaving Earth will a fully functioning solar array producing 675kW of power at 1 AU. The HPS is designed as a fully reusable spacecraft, making three roundtrip mission to Mars. The current assumption for the HPS design is that the solar array power production will degrade at an approximate rate of 1.5% per year. Table 1 shows the current solar array degradation rate assumption for the HPS mission architecture. As the solar array degrades, there is less power available to the SEP system thus reducing the overall efficiency of the SEP System. The degradation to the array is the most impactful at Earth departure, when the HPS is heaviest and there is sufficient insolation to fully utilize the solar cells.

The effect of the solar array degradation can be seen in Figure 12. The figure expands on the data shown in Figures 7 and 9 by showing how the propellant demand changes as the HPS array

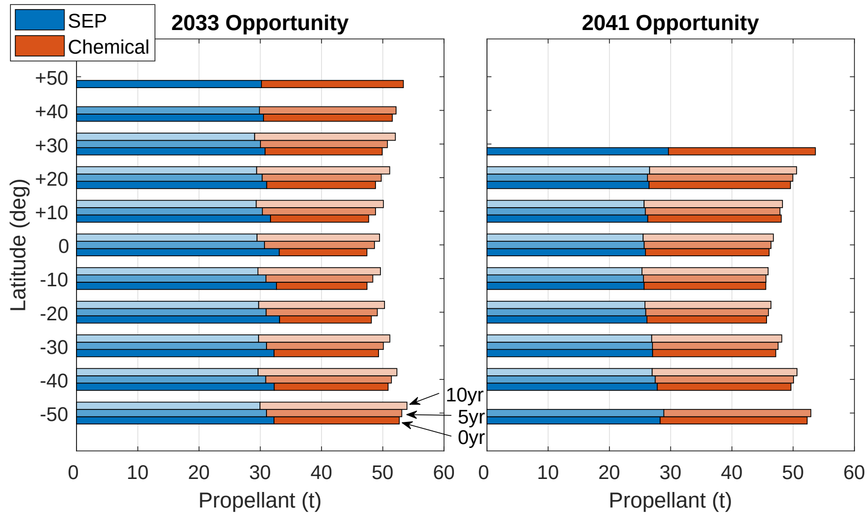


Figure 12. Impact of Array Degradation on Propellant Demand for 2033 & 2041 Mission Opportunity to Reach $\pm 50^o$ Latitude for the Current Hybrid Transportation System

degrades. The grouped bars with varying gradients show the three array degradation levels for each of the latitudes. The figures show the 2033 and the 2041 mission opportunities side-by-side. As the array degrades, less power is available to the SEP system, and this effect can be seen in most of the cases as less SEP propellant is used to perform the mission. To make up the lost performance, more chemical propellant is required. The loss of SEP performance resulting from the array degradation has a direct impact on the accessibility of the landing site. For the 2033 mission opportunity, the HPS is able to reach all latitudes between plus or minus 50 degrees. However, as the array degrades, the higher latitude becomes unreachable. On the first reused trip (second overall) with 7.5% array degradation, the HPS loses the ability to reach higher than plus 50 degrees latitude in 2033. On the second reused trip (third overall) with 15% degradation, the HPS loses the ability to target higher than plus 40 degrees latitude. Because more chemical propellant is required to make up for the reduced performance from the SEP thrusters, less chemical propellant is available to target different latitude landing sites. Similar trends can be seen in the 2041 mission opportunity. With no degradation, the HPS can reach between -50 degrees and +30 degrees latitude. However with 15% degradation, the HPS can only reach -40 degrees to +20 degrees latitude, losing access to many of the higher latitude sites.

Figure 13 provides a more complete view of the overall trade space with all three variables (landing site latitude, mission opportunity, and spacecraft age) on the same plot. The figure clearly shows the latitudes for which the current HPS design point cannot reach based on mission opportunity and array degradation. With the current HPS design point, the only mission opportunity in which the HPS is capable of reach all latitudes from minus 50 to plus 50 degrees is 2045. The 2048 mission opportunity comes very close to having full coverage, only missing the plus 50 latitude at the 3rd use of the HPS.

Looking at Figure 13, some interesting observations can be made about the performance of the HPS. As mentioned previously, each mission opportunity can stress the two propulsion systems differently, and this is made clear in this figure. The 2048 mission opportunity is quite different from the other opportunities in that SEP propellant is almost always fully utilized regardless of the

Hybrid Mission to Mars Landing Site Latitute Trade - Chemical Propellant

Latitude	2033			2035			2037			2039			2041			2043			2045			2048			2050			2052		
	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y
+50°	23.2	X	X	12.1	13.0	16.5	22.4	21.6	23.6	X	X	X	X	X	X	X	X	X	20.1	22.1	24.0	21.2	22.2	X	12.5	18.3	21.4	15.7	16.9	18.6
+40°	21.0	22.3	X	12.3	13.2	16.1	20.0	18.9	22.0	X	X	X	X	X	X	X	X	X	18.1	19.1	20.6	18.7	19.2	21.8	12.1	14.9	19.6	15.0	16.3	18.0
+30°	19.1	20.7	23.0	12.8	13.2	15.9	17.6	18.6	19.9	22.8	23.9	24.0	24.0	X	X	24.0	24.0	X	16.2	17.6	18.4	17.1	17.4	19.8	12.9	13.6	17.8	15.0	16.3	17.4
+20°	17.8	19.4	21.7	12.9	13.3	16.0	16.6	17.3	19.0	20.7	21.4	22.5	23.1	23.7	24.0	21.8	22.7	23.5	15.4	16.3	17.2	15.4	15.6	18.3	13.4	14.5	16.9	14.7	15.3	17.2
+10°	16.0	18.5	20.8	12.8	13.6	16.8	16.1	16.8	18.3	19.4	19.8	20.8	21.8	22.0	22.6	20.0	20.7	21.5	15.3	15.5	16.6	14.2	14.9	16.5	13.2	14.4	16.8	14.6	15.7	17.5
0°	14.3	18.0	20.0	12.6	13.4	18.1	16.0	17.2	19.3	18.7	19.2	20.0	20.2	20.7	21.3	18.8	19.4	20.1	14.7	15.2	15.9	14.0	15.4	15.3	13.1	14.9	17.5	14.7	16.1	18.1
-10°	14.8	17.4	20.0	12.5	13.9	20.5	17.0	18.4	20.2	18.4	19.0	20.2	19.9	20.0	20.6	18.1	18.7	19.4	14.8	15.1	15.8	14.0	15.3	18.1	12.9	16.3	18.4	15.2	17.3	18.9
-20°	15.0	18.1	20.5	12.5	14.9	22.0	18.5	20.2	21.5	19.1	19.9	21.1	19.5	20.0	20.6	18.0	18.4	19.1	14.5	15.7	15.9	14.5	15.1	18.6	12.7	17.8	19.5	16.7	17.6	21.2
-30°	17.1	19.1	21.4	12.4	17.5	`	20.5	21.0	X	20.6	21.7	23.2	20.1	20.5	21.2	18.1	18.8	19.5	14.7	15.4	16.8	14.8	16.3	19.2	14.2	19.6	21.1	18.0	19.7	X
-40°	18.6	20.5	22.7	13.5	24.0	X	22.6	24.0	X	23.5	24.0	X	21.8	22.6	23.6	18.8	19.4	20.4	15.2	15.9	17.2	17.5	17.7	20.5	17.3	21.3	23.1	19.3	23.7	X
-50°	20.5	22.1	24.0	15.8	X	X	X	X	X	X	X	X	24.0	24.0	X	20.4	21.2	22.5	16.9	17.3	18.9	18.1	20.0	22.2	X	X	X	X	X	X

Hybrid Mission to Mars Landing Site Latitute Trade- SEP Propellant

Latitude	2033			2035			2037			2039			2041			2043			2045			2048			2050			2052		
	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y	0y	5y	10y
+50°	30.2	X	X	32.0	32.0	30.6	29.3	30.3	29.7	X	X	X	X	X	X	X	X	X	33.1	32.3	31.5	34.0	34.0	X	33.7	30.6	28.9	31.4	31.0	30.6
+40°	30.5	29.9	X	31.2	30.7	29.7	28.2	29.6	28.1	X	X	X	X	X	X	X	X	X	32.4	31.4	30.8	34.0	34.0	33.4	33.1	32.0	28.9	30.3	29.9	29.1
+30°	30.8	30.0	29.1	30.0	30.1	29.0	28.0	27.6	27.3	27.0	26.7	28.0	29.6	X	X	28.7	29.4	X	31.4	30.8	30.2	33.7	34.0	34.0	31.6	32.1	29.0	29.0	28.5	28.4
+20°	31.1	30.3	29.4	29.7	29.8	28.7	27.3	27.2	26.7	26.5	26.4	26.0	26.5	26.2	26.6	27.8	27.3	27.0	30.1	29.9	29.3	33.7	34.0	34.0	30.8	30.1	28.9	28.7	28.5	28.1
+10°	31.7	30.3	29.3	29.9	30.1	28.3	27.1	27.0	26.7	26.1	26.0	25.7	26.3	25.9	25.7	27.4	27.0	26.7	29.0	29.0	28.8	34.0	34.0	34.0	30.8	30.2	28.7	28.8	28.4	28.2
0°	33.1	30.7	29.5	30.4	31.0	28.5	27.6	27.2	26.8	25.8	25.7	25.4	25.9	25.7	25.5	27.1	26.8	26.5	28.5	28.3	28.2	33.9	33.4	34.0	31.0	29.8	28.6	29.0	28.7	28.7
-10°	32.6	30.9	29.6	31.0	31.3	27.6	27.9	27.6	27.6	26.3	26.2	25.7	25.7	25.6	25.3	26.6	26.2	25.9	27.9	27.8	27.6	33.5	33.9	31.5	31.9	29.5	28.5	30.1	29.2	29.6
-20°	33.2	31.0	29.7	31.9	31.7	27.9	28.7	28.5	29.0	27.1	26.9	26.6	26.1	25.9	25.8	26.4	26.2	26.0	28.2	27.8	27.5	33.7	33.7	31.4	33.2	29.3	28.5	30.4	31.0	30.0
-30°	32.2	31.0	29.7	32.9	32.0	X	29.9	30.5	X	28.0	27.6	27.1	27.1	27.0	26.9	27.0	26.6	26.5	28.5	28.2	27.5	34.0	33.4	31.6	33.5	29.0	28.4	32.3	31.5	X
-40°	32.3	30.9	29.6	33.2	32.6	X	30.7	30.9	X	28.1	28.4	X	27.8	27.5	27.0	27.7	27.5	27.2	29.1	28.9	28.3	33.2	33.4	31.5	33.9	29.3	28.4	32.9	30.5	X
-50°	32.2	31.0	30.0	33.6	X	X	X	X	X	X	X	X	28.3	28.9	X	28.4	28.1	27.7	29.5	29.4	28.5	33.9	32.9	32.1	X	X	X	X	X	X

Figure 13. Hybrid Mission Propellant Demand as Functions of Landing Site Latitude, Mission Opportunity, and Spacecraft Age

target latitude. The unconstrained solution for the 2048 opportunity yielded a optimized latitude of +10 degrees (Figure 5), and examining Figure 13 it is clear that the near equatorial solutions in 2048 utilize the minimum amount of chemical propellant, even though the SEP propellant load is quite high compared to the other opportunities.Thus, 2048 is a mission opportunity that heavily relies on the SEP system. Note that this does not mean that the 2048 mission opportunity is more challenging from the SEP propulsion system perspective; rather it is an opportunity in which the planetary geometry allows for more of the overall roundtrip ΔV to be performed by the SEP system, which ultimately makes it a more mass optimal trajectory. Therefore, 2048 mission oppotunity was able to provide more landing site latitude coverage, as using the SEP to perform the interplanetary portion of the trajectory allows for the utilization of the chemical propulsion system to perform the maneuvers required for higher latitude access.

The 2039, 2041, and the 2043 mission opportunities use significantly less SEP propulsion propellant than the 2048 mission opportunities and consequently have to use more chemical propulsion propellant to perform the roundtrip mission. This is likely due to the constraint on the available transit time given the alignment of Earth and Mars coupled with the 300 day minimum stay time constraint. In this case, because a larger portion of the chemical system was utilized for the interplanetary portion of the trajectory (also observed in the unconstrained case in Figure 6), the HPS as designed is not able to reach the more extreme ends of the latitude range. This type of analysis is critical in the understanding of how landing site selection can impact the overall mission design.

SUMMARY

The results show that the current vehicle is capable of reaching up to +20 degree latitude and down to -20 degree latitude in every mission opportunity from 2033 to 2052. However, reaching latitudes beyond +/- 20 required more propellant than the spacecraft is currently designed to carry for many of the mission opportunities, especially after the array has degraded. The Mars campaign requires delivery of crew and cargo in almost every mission opportunity (with pre-deployment of surface assets and supplies prior to crew missions); thus it is imperative that the transportation system be designed to be able to reach the landing site in all mission opportunities. If the landing site selected is beyond +/- 20 degrees latitude, the current baseline vehicle will not be able to support missions to the surface in every opportunity.

If the landing site is not selected prior to final vehicle design, the alternative option is to design a vehicle with larger tanks which allows for more robustness in targeting different latitudes. Increasing the latitude range that the system can access increases the cost to reach any latitude, and choice of landing site accessibility requirements should be weighed against the additional cost and risk associated with maintaining a broad range of landing site options. Both the landing site latitude and the Hybrid transportation system designs constrain the mission architecture leading to a choice between a propellant optimal design that poses potential schedule and accessibility risks versus a more robust schedule and accessibility design that requires increased propellant amounts for every mission.

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