Integrated Optimization of Mars Hybrid Solar-Electric/Chemical Propulsion Trajectories

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NASA's Human Exploration and Operation Mission Directorate is developing a reusable hybrid transportation architecture in which both chemical and solar-electric propulsion systems are used to deliver crew and cargo to the Martian sphere of influence. By combining chemical and solar-electric propulsions into a single spacecraft and applying each where it is the most effective, the hybrid architecture enables a series of Mars trajectories that are more fuel efficient than an all chemical propulsion architecture without significant increase to trip time. Solving the complex problem of low-thrust trajectory optimization coupled with the vehicle sizing requires development of an integrated trajectory analysis framework. Previous studies have utilized a more segmented optimization framework due to the limitation of the tools available. A new integrated optimization framework was recently developed to address the deficiencies of the previous methods that enables higher fidelity analysis to be performed and increases the efficiency of large design space explorations.

Nomenclature

BOL	Beginning of Life
ΔV	Velocity Change
LDHEO	Lunar Distant High Earth Orbit
LDRO	Lunar Distant Retrograde Orbit
LGA	Lunar Gravity Assist
MALTO	Mission Analysis Low-Thrust Optimizer
MSCT	Mars Study Capability Team
NRHO	Near Rectilinear Halo Orbit
SEP	Solar Electric Propulsion
V_∞	Hyperbolic Excess Velocity

I. Introduction

NASA's Mars Study Capability Team (MSCT) continues the agency's effort to study and refine NASA's ambitious plan to field a sustainable human Mars campaign. Building upon the success of the Evolvable Mars Campaign[1,2] (EMC), the MSCT is developing more capabilities to improve the fidelity of the Mars campaign study and to broaden the design trade space to understand the impact of technology investments and architecture decisions for missions to Mars in the 2030s. Many different mission design concepts have been studied and proposed over the past three decades,[3–5] and many more are currently being investigated. In the majority of these studies, chemical propulsion has been assumed for the crewed Mars missions because

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solar electric propulsion, despite being much more fuel efficient, produces less thrust, and is more suitable for cargo pre-deployment missions when the transit time can be much longer.

The MSC Team is continuing the refinement of the new Hybrid transportation architecture in which both chemical and electric propulsion are combined in an integrated design.[6] By combining chemical and solar-electric propulsion (SEP)[7,8] into a single spacecraft and applying each 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 allows the spacecraft to complete the round-trip journey to/from Mars in less than 1,100 days, which enables the reuse of the transportation system for multiple trips and eliminates the need to develop separate transportation systems for crew and cargo.

The utilization of both SEP and traditional chemical propulsion systems together requires a completely new method to solve the integrated trajectory optimization problem. This paper provides a summary of the methods utilized to analyze and optimize the Hybrid transportation system. The integration of the trajectory optimization, the vehicle sizing, and performance closure process is nontrivial. A discussion of two different methods utilized to solve the problem is presented in this paper. Discussion of the deficiency of the old analysis framework is provided, and a demonstration of the updated trajectory optimization framework shows the improvement of the new framework. Finally, a short demonstration of the capability of the updated system is presented with a design space exploration of the Hybrid transportation system.

II. Trajectory Breakdown

Solving the complete end-to-end trajectory for human Mars missions with the hybrid transportation system is more complex than the traditional chemical transportation system. 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 and closure.

Figure 1: Mars Hybrid Crew Mission Concept of Operation

To solve the trajectory optimization problem, the complete Mars roundtrip trajectory mission can be broken into seven distinct phases as shown in Figure 1. The primary challenge of solving the integrated trajectory optimization is the circular nature of the iteration problem, as solving each phase of the problem requires inputs from other phases. Additionally, the combination of the low-thrust optimization and the traditional orbital mechanics closed-form solution of the Lambert problem makes the optimization more challenging.

A. Earth Departure and Arrival

A traditional conjunction class Mars transfer is a simple two-burn Lambert solution that requires an Earth departure hyperbolic excess velocity (V_∞) between 2.7 km/s and 4 km/s and is typically achieved by a trans-Mars injection (TMI) maneuver close to Earth. A traditional crewed Mars mission also needs to carry an Earth entry capsule roundtrip to avoid a costly chemical burn at the end of the mission because the Earth arrival V_∞ of similar magnitude applies as well. The large TMI burn typically requires a large chemical propulsion stage to be assembled at Earth departure, as the stage needs to be able to provide the necessary velocity change (ΔV) to push the roundtrip stack to the required heliocentric velocity. To mitigate the large departure stage, most traditional missions pre-deploys most, if not all, of the return propellant and supplies to Mars to minimize the payload mass at Earth departure. For the Hybrid architecture with sufficient SEP power, the Earth departure maneuver requirement can be reduced significantly because the SEP system can continuously thrust during the long heliocentric transit to achieve the orbital energy required to reach the destination. By reducing the departure and arrival requirements, the Hybrid architecture can eliminate the need to pre-deploy return assets.

Solving the Earth departure and arrival portions of the trajectory requires some assumptions for the mission architecture. The Hybrid spacecraft is assumed to be deployed and aggregated in a lunar distant retrograde[9] (LDRO) or a near rectilinear halo orbit[10] (NRHO). When the crew mission is ready to be deployed, the spacecraft transits from the aggregation orbit to a Lunar Distant High Earth Orbit (LDHEO) where it rendezvous with the crew and allows the crew to perform final check out. The LDHEO is selected and oriented so that the spacecraft will encounter the moon after the crew rendezvous to help increase the orbital energy to prepare for departure. Multiple Lunar Gravity Assist (LGA) maneuvers may be required to achieve Earth escape velocity of 1.4km/s with proper direction to match the velocity requirement for the heliocentric transit. If the the SEP system is underpowered for the heliocentric transit and a V_∞ higher than 1.4 km/s is required, a small impulsive maneuver can be added during the LGA's perilune passage to boost the V_∞ by a small amount. This dependency creates the first circular nature of the trajectory optimization problem as the LGA/powered LGA requirement is dependent on solving the low-thrust trajectory, which also depends on the final Earth departure V_∞ magnitude and direction.

B. Heliocentric Transits

After achieving the appropriate Earth escape V_∞, the Hybrid spacecraft utilizes the SEP system to thrust for the majority of the heliocentric transit to target a low V_∞ arrival at the destination. SEP system thrusting is limited by the array power that is generated as the spacecraft moves away from the Sun and also by the limited time it has to thrust, as the Hybrid architecture uses standard conjunction style transit durations. Chemical maneuvers are required at Mars arrival and departure to compensate for the SEP system's deficiency. This creates a second circular dependency in the trajectory optimization problem as the chemical maneuver required is dependent on the low-thrust trajectory and the low-thrust trajectory is dependent on the chemical maneuver, as the chemical propellant is a significant fraction of the payload. Finding a balance between the chemical and SEP elements is key to optimizing the Hybrid transportation architecture.

C. Mars Sphere of Influence

Once the Hybrid spacecraft arrives at the Mars parking orbit, it must reorient itself into the parking orbit such that it can meet the Mars departure velocity direction and magnitude. Additionally, the parking orbit much be aligned such that it can allow the crew to access their chosen landing site for every mission opportunity. For elliptic parking orbits, the geometry of the orbits becomes rather complex because it involves lining up not only the orbital plane but also the line of apsides to ensure an efficient burn at escape. Furthermore, orbit precession and third body perturbations over the long period of stay play major roles in the orientations of the orbits. Solving this portion of the trajectory is extremely complex and is highly dependent on multiple other phases of the trajectory designs. A more detailed study and discussion on the Mars sphere of influence maneuvers can be found in previous work.[11]

III. Optimization Tools & Methods

To solve the overall trajectory optimization problem, different tools and analysis methods need to be integrated together. To size the elements in the system, the overall objective of the optimization problem for the hybrid architecture is to minimize the fully fueled transportation system size for a given set of Mars payloads across the entire mission manifest. As there are no closed form solution for low-thrust trajectory optimization, specialized analysis codes are required.

A. MALTO Based System

The initial analysis capability that was utilized to study the Hybrid architecture is based on JPL's Mission Analysis Low-Thrust Optimizer (MALTO).[12] The developed tool integrates MALTO with a vehicle sizing routine to provide vehicle closure and ensure the vehicle's performance meets the specified requirements. The analysis tool is represented visually in Figure 2. 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

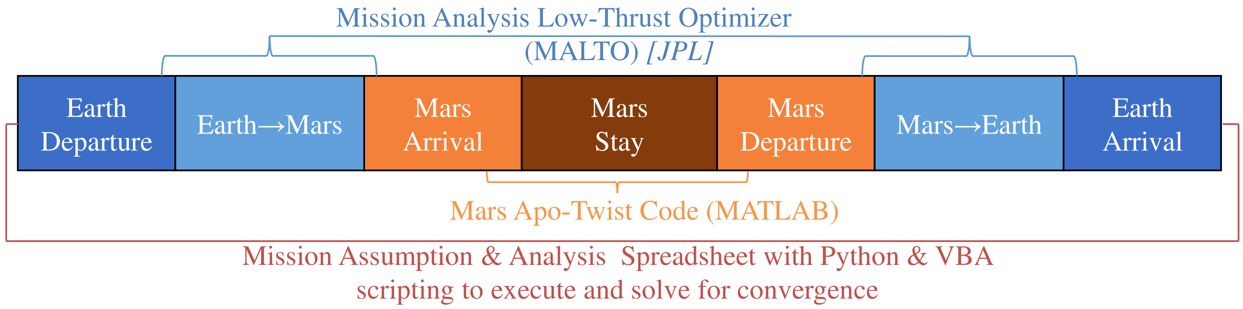

Figure 2: JPL's Mission Analysis Low-Thrust Optimizer Based Hybrid Trajectory Analysis and Optimization Framework

utilized to evaluate and optimize the Mars sphere of influence maneuvers that are required.[11] 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 utilized to support the Evolvable Mars Campaign[1] analysis cycle for the Hybrid architecture and a significant amount of the previously published Hybrid architecture results.[13,14]

One of the primary limitations in this analysis framework is the inherent segmentation of the 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 consideration. 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 time. 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,[15] require hundreds of cases to be run over a period of months.

B. Copernicus Integrated System

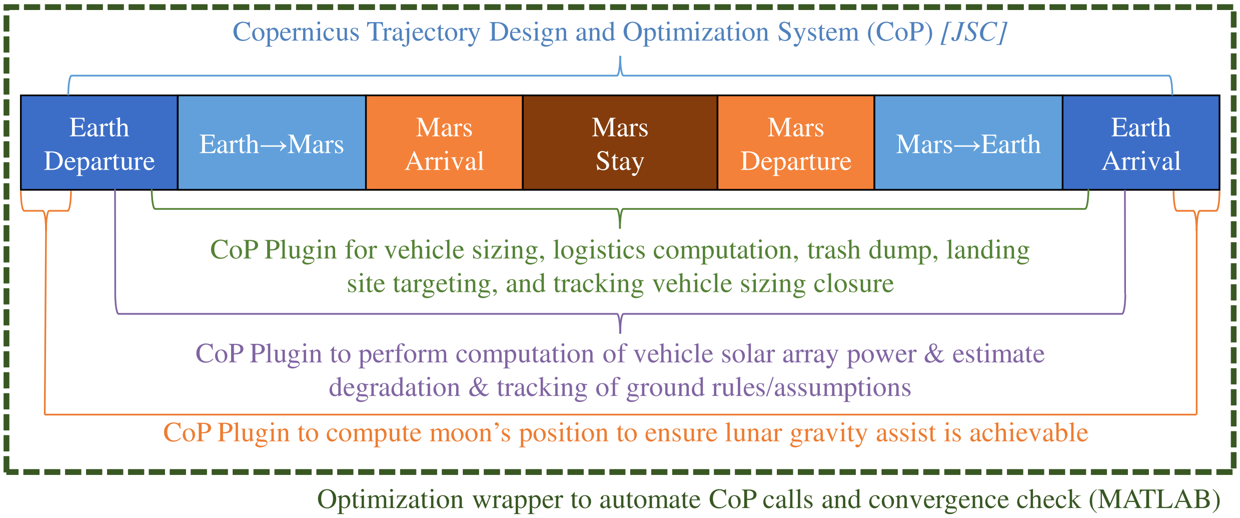

Figure 3: JSC's Copernicus Trajectory Design and Optimization System-Based Hybrid Trajectory Analysis and Optimization Framework

To remedy the deficiency of the previous analysis method, an integrated optimization method is 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.[16] 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 in house at NASA Langley and compiled to calculate the parking orbits 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[17] and the logistics that are both functions of the transit duration. A second plug-in allows for tracking of the vehicle's power consumption, the array degradation across multiple mission opportunities, and tracking of the various ground rules and assumptions, such as various 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.

A final piece of the analysis framework centers around the automation of the input deck generation, trajectory convergence, and output post-processing. A Matlab script was developed to automatically generate the Copernicus input decks for specified analysis runs, perform Copernicus function calls and check for convergence, and read the Copernicus output files to generate the results in a plain text file for use. The optimization wrapper reduces the user input time for generating Hybrid transportation optimal trajectory data point by orders of magnitude compared to the previous method because the analysis can be set up to allow for parallel processing of multiple cases. The same 800 case sensitivity analysis performed by the previous analysis method[15] that took three months to complete can be completed using the new framework in about 100 hours using a single core and less than 24 hours with multi-core support. Additionally, the new analysis framework provides inherently higher fidelity results as many of the areas that were left as assumptions in the previous analysis method can now be modeled and solved as part of the overall optimization problem. This new analysis capability allows for a significantly more comprehensive sensitivity analysis to be performed to allow NASA to fully understand the Hybrid transportation spacecraft and architecture so better decisions can be made on how to best set up and deploy the system to field a sustainable human Mars campaign.

IV. Baseline Results Comparison

Table 1 shows the baseline comparison of the results from the two analysis frameworks. The assumption and vehicle sizing for the baseline comparison are taken from the 2017 sensitivity analysis study.[15]The system utilizes xenon as the primary propellant for the SEP system and non-cryogenic chemical propellant (Monomethylhydrazine and Nitrogen Tetroxide) for the chemical propulsion system. The MALTO solutions were taken directly from the results from the 2017 study. The Copernicus solutions were achieved using the same baseline assumptions provided in the 2017 study; the lunar location constraint was not utilized for this comparison as it would increase the propellant demand significantly and would not be an equal comparison.The primary difference between the two methods shown in this table is the Mars sphere of influence maneuver requirements and the ability for the Copernicus tool to globally optimize the trajectory.

Examining the propellant and vehicle dry mass for the four mission opportunities reveals that using the Copernicus analysis methods does not immediately result in reduction in the system and propellant mass. For two of the mission opportunities examined, the total mass of the system actually increased as compared to the sub-optimal solutions. This increase can be attributed to the higher fidelity of the analysis with improved accuracy of the orbit determination and improvement in the computation of the Mars sphere of influence maneuvers. The previous method only allocated a fixed ΔV budget for the Mars

	2033		2037		2041		2045		
	MALTO	COP	MALTO	COP	MALTO	COP	MALTO	COP	
SEP Earth-Mars	13,700	21,200	14,800	15,800	15,400	16,800	16,400	18,100	kg
SEP Mars-Earth	7,400	6,900	7,700	7,900	7,600	8,200	8,200	7,200	kg
SEP Maneuvering	1,760	490	1,900	490	1,900	490	2,000	490	kg
Total SEP	*22,860*	*28,590*	*24,400*	*24,190*	*24,900*	*25,490*	*26,600*	*25,790*	*kg*
CP TMI	0	0	8,260	6,900	0	0	0	0	kg
CP MOI	11,900	5,530	4,830	5,900	9,400	11,230	8,000	7,190	kg
CP TEI	5,120	4,780	5,230	6,520	6,520	7,480	4,730	3,620	kg
CP EOI	0	0	0	0	0	0	6,090	8,050	kg
CP Maneuvering	10,050	10,490	10,190	10,360	10,240	10,690	10,740	10,810	kg
Total CP	*27,070*	*20,800*	*28,510*	*29,680*	*26,160*	*29,400*	*29,560*	*29,670*	*kg*
Vehicle Dry Mass	*24,350*	*23,940*	*24,720*	*24,920*	*24,240*	*25,280*	*25,250*	*25,350*	*kg*
HPS Wet Mass	**74,280**	**73,330**	**77,630**	**78,790**	**75,300**	**80,170**	**81,410**	**80,810**	kg

Table 1: Hybrid Trajectory Propulsive Propellant Mass Breakdown and Comparison Between the MALTO Analysis Method and the COP Analysis Method

sphere of influence maneuvers as the computation of the parking orbit reorientation is separate from the interplanetary optimization; thus the result could under predict the propulsive requirements.

The 2033 mission opportunity is of particular interest because of how the global optimization changes the balance between the chemical and SEP system as compared to the suboptimal solution. Using the suboptimal routine, the 2033 mission opportunity trajectory has a relatively short heliocentric transit and uses significantly less SEP thrusting as compared to the other mission opportunities. The suboptimal routine is unable to find a better solution that utilizes more of the SEP system, and thus has to compensate for the loss of SEP thrusting with large chemical maneuvers. This can be observed in both Table 1 and Table 2, which show the propellant breakdown and the ΔV breakdown for each of the missions using the two optimization framework, respectively. For the suboptimal solution, the 2033 crew mission requires less than 23t of SEP propellant and more than 27t of chemical propellant. Compared that to the global optimal solution, which requires significantly more SEP propellant (more than 28t), but reduces the chemical propellant requirement to less than 21t. This results in a system wet mass reduction of almost 1t. Examining Table 2, the global optimal solution trades nearly 1km/s of ΔV for the SEP system to reduce the MOI chemical maneuver by more than 50%. The ability for the optimizer to change both legs of the low-thrust trajectory also reduces the propellant demand and ΔV of the Earth return portion of the trajectory in most cases.

		CP	*SEP*	*CP*	*CP*	*CP*	*SEP*	*CP*	
		TMI	Earth-Mars	MOI	Reorient	TEI	Mars-Earth	EOI	
2033	MALTO	0	2,680	387	150	209	3,218	0	m/s
	COP	0	3,684	179	0	298	2,831	0	m/s
2037	MALTO	220	3,049	163	150	190	3,266	0	m/s
	COP	179	2,874	195	0	204	3,213	0	m/s
2041	MALTO	0	2,983	305	150	257	3,226	0	m/s
	COP	0	2,809	345	0	283	3,284	0	m/s
2045	MALTO	0	3,031	245	150	172	3,138	276	m/s
	COP	0	3,052	220	0	130	2,625	356	m/s

Table 2: Hybrid Trajectory Propulsive ΔV Comparison Between the MALTO Analysis Method and the COP Analysis Method

Overall, the new Copernicus-based trajectory optimization framework can produce similar results to the old MALTO based framework, but with significant increases to the flexibility of the problems it can solve and increases to the overall fidelity of the solution. The new framework also enables more rapid analysis of different mission opportunities, SEP power level, targeting of specific landing sites, and varying payload mass. The automation of the analysis framework enables thousands of cases to be completed with little to no user input and allows for parallel processing to improve the convergence and data generation for large design space explorations.

V. Design Space Exploration

Opportunities:	2033, 2035, 2037, 2039, 2041, 2043, 2045, 2048, 2050, 2052, 2054
BOL Array Power:	650, 675, 700, 725, 750 kW
SEP Thruster Power:	372, 386, 400, 412, 426, 439, 452 kW
Array Degradation:	0% (new), 7.5% (5yr Old), 15% (10yr Old)
Landing Site Latitude:	18.8^o (Jezero Crater)
Habitat Dry Mass:	22.3t (plus ∼22-23t logistics)

Table 3: Input Variables for the Design Space Exploration to Update the Hybrid Transportation System

Utilizing the Copernicus based integrated trajectory optimization framework, a SEP and solar array power sensitivity analysis was conducted to evaluate how the updated framework handles large design space sweeps and provide an update on the Hybrid transportation architecture. Table 3 shows the design space input for this analysis. Currently, decisions on which particular Mars mission opportunity to field crew missions are still being investigated. For the Hybrid spacecraft to be able to accommodate changing mission dates, the design trade space will include every mission opportunity across the Earth-Mars synodic cycle. The SEP thruster currently under consideration is rated at 13.3kW per thruster.[7,8] The thrusters are installed in pairs to balance out the thrusting moment. Previous work[13,14] has utilized 24 thruster pairs for the Hybrid spacecraft for a total thruster power of approximately 320kW. For this analysis, because of the additional analysis capability and fidelity, the thruster power trade space was increased to between 28 and 34 thruster pairs, or between 372kW and 452kW. To accommodate the increased thruster power, the BOL array power will range from 650kW and 750kW.

The array and SEP thruster power trade is critical for the Hybrid spacecraft's design. The BOL array power represents the power that the solar array can generate at Earth distance (1AU) at the start of the spacecraft's life. As the spacecraft moves away from Earth, the power available to the thrusters to provide thrust decreases with the standard $1/R^2$ law. If the arrays are sized too small, it would limit the amount of SEP thrust available to the spacecraft as it moves away from Earth, requiring the less efficient chemical system to make up the difference. However, increasing the array power to provide more thrust as the spacecraft moves away from Earth will increase the dry mass of the vehicle, decreasing the thrust-to-weight ratio of the SEP thruster, which would also require the chemical system to be utilized more. Additionally, because the spacecraft is designed to be reusable, the array must be sized to be able to to field any mission opportunity during its lifetime. Thus a full factorial system power level trade is required to fully understand the design space.

The sizing of the propellant tanks is highly dependent on the mission opportunities and the SEP thruster and array power as well. Certain opportunities are more taxing on the SEP system than the chemical system, depending on the orientation and the positions of the planets. When the vehicle is new, the array generates full power, the SEP system can be fully utilized, and the propellant tanks must be sized appropriately. As the vehicle gets older, the solar array degrades and produces less power to the SEP thrusters. This increases the burden on the chemical system, resulting in more chemical propellant to be required to field the missions. Thus, to design and size the vehicle to be able to field any mission opportunities at any stage of the vehicle's life, every possible combination of the design inputs must be considered.

This is the same method utilized for the full Hybrid transportation sensitivity analysis performed in 2017.[15] Utilizing the MALTO analysis framework, a total of 880 cases were completed to in 2017 for the analysis. It took three members of the MSCT three months at roughly quarter effort to complete the trajectory runs and one member an additional two weeks to fully post process the results. In comparison, for this analysis, a total of 1,155 cases were required to complete the design space exploration, and with the automation of the analysis framework, it took a 40 core computer just under 12 hours to complete the runs and provide the post processed results.

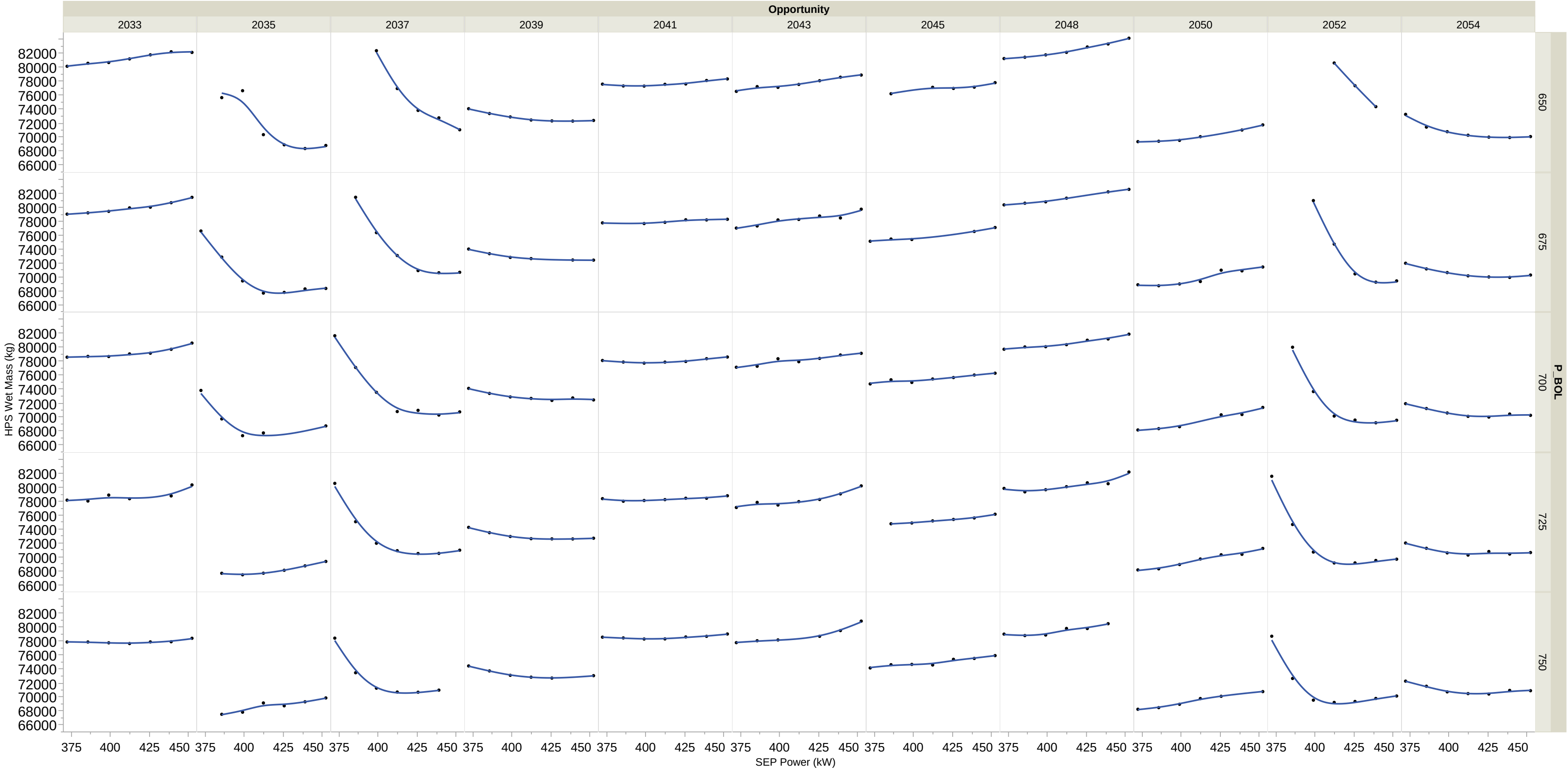

Figure 4: Hybrid Spacecraft Wet Mass as Functions of SEP Thruster Power, BOL Array Power, and Mission Opportunities with 0% Array Degradation

Figure 4 shows an overview of the results of the design space exploration. The plots show the Hybrid wet masses, which includes all propellant mass and the spacecraft's dry mass, as a function of the SEP thruster power for all of the mission opportunities and BOL array power. To simplify the plot, this particular figure only shows the result for the non-degraded spacecraft. From the plot, the large variation of the system performance across different mission opportunities can be seen clearly. Several mission opportunities' spacecraft wet mass (2035, 2037, 2052 in particular) are very sensitive to changing SEP thruster power. Those opportunities are more taxing for the chemical system, which exhibits the exponential nature of the rocket equation.

2035 is an interesting opportunity to examine. For lower BOL array power, the wet mass is very sensitive to the SEP thruster power. But as the array power increases, the sensitivity is reduced significantly. This shows how having enough array power to be utilized by the SEP thruster can reduce the dependency on the chemical propulsion system, which in turn reduces the mass sensitivity to other changing variables. This same trend can also be observed in the 2054 mission opportunity, though the effect is not as pronounced. In contrast, the wet mass of the mission opportunities in the 2040s are all relatively insensitive to the array and SEP power.

For the 2037 and the 2052 mission opportunities, the wet mass is still relatively sensitive to the SEP thruster power even with the array power at the top of range of the design trade space. Even with 750kW of array power at beginning of life, the wet mass of the vehicle still grows exponentially as the SEP thruster power falls below 400 kW. This shows how difficult it is to design a spacecraft to accommodate every possible mission opportunity, as some opportunities are inherently more difficult for each propulsion system to handle. The balance between the high efficiency SEP system and the low efficiency chemical systems is a challenging problem to solve. The ability for the new integrated trajectory optimization framework to rapidly run cases with different inputs enables the full trade space to be better understood so the appropriate analysis can be conducted to better inform decision makers on the impact of design decisions.

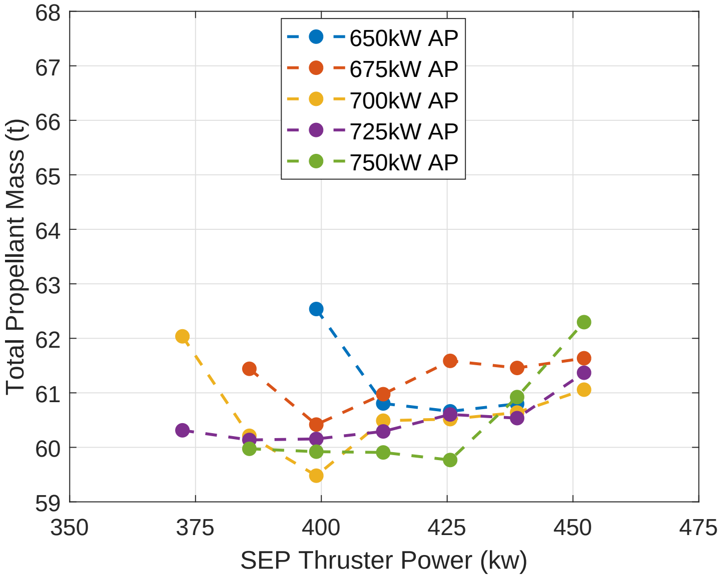

Figure 5: Maximum Propellant Mass Across All Mission Opportunities and Spacecraft Age as Functions of SEP Thruster Power and BOL Array Power

Using the data generated by the design space sweep, the mass optimal combination of SEP thruster and array power can be investigated. Figure 5 shows the maximum propellant mass across all mission opportunity and spacecraft age as a function of the SEP thruster power and BOL array power. As the figure shows, the variation of the maximum required propellant mass can be pretty drastic for different BOL array power. For the 650kW BOL array power vehicle, the propellant mass required has a very small mass optimal valley around 425kW SEP. The propellant mass grows exponentially as the SEP power decreases from the optimal design, as the the power deficient vehicle makes up the performance difference with the chemical system. This represents an optimal but not robust design point, as deviation from the optimal design point results in significant increase in the metric variable, which in this case is propellant and vehicle mass. As the BOL array power increases, the robustness of the design, as it pertains to propellant mass, increases as seen in the figure. Overall, for this design payload of roughly 45t, the optimal combination of system power is in the 400-425kW SEP and 700-750kW array range. It is important to note that as the mission parameter changes (i.e. payload mass, landing site, mission opportunities), the optimal system power combination can be significantly different from what is shown in Figure 5. Thus, the newly developed method is critical to allow for rapid analysis to understand the impact of mission parameter changes to system performance.

VI. Summary

NASA's Mars Study Capability Team continues to work on the refinement of different architecture concepts for fielding human Mars campaigns. The Hybrid transportation system is a promising concept that combines the highly efficient solar electric propulsion system with the high thrust of a traditional chemical propulsion system, utilizing each system when it is most effective. Solving the complex problem of low-thrust trajectory optimization coupled with the vehicle sizing requires the development of integrated trajectory analysis framework. Previous studies have utilized a more segmented optimization framework due to the limitation of the tools available. A new integrated optimization framework was recently developed to address the deficiencies of the previous methods. The new framework is significantly more capable both in terms of fidelity of the analysis and the speed in which large design space exploration can be completed.

A demonstration of the new analysis framework shows the sensitivity of the hybrid transportation system to changes in thruster and array power across multiple mission opportunities. The result shows the difficulty in designing a reusable transportation system that can operate across multiple mission opportunities: the propulsive requirement for the SEP system and the chemical propulsion system can be significantly different. Optimal SEP thruster power and BOL array power for one mission opportunity can be wholly inadequate for a different opportunity. However, from a campaign planning and technology investment standpoint, it is infeasible to design a new vehicle that is optimal for each mission opportunity when the vehicle is being reused. With the newly developed analysis framework, the Hybrid architecture can be further analyzed to understand the impact of different architecture decisions such as landing site, payload mass, and timing of the missions.

References

[1] Craig, D. A., Herrmann, N. B., and Troutman, P. A., "The Evolvable Mars Campaign - Study Status," *IEEE Aerospace Conference*, March 2015, 2015-8.0101.

[2] Goodliff, K., Troutman, P. A., Craig, D. A., and Herrmann, N. B., "Evolvable Mars Campaign 2016 Analysis Update," *AIAA SPACE 2016 Conference and Exposition*, 2016.

[3] Mars Architecture Steering Group and Drake, B. G., "Human Exploration of Mars Design Reference Architecture 5.0," Special Publication 2009-566, National Aeronautics and Space Administration, July 2009.

[4] Stanley, D. O., Cook, S., Connolly, J., Hamaker, J., Ivins, M., Peterson, W., Geffre, J., Cirillo, B., McClesky, C., Hanley, J., et al., "NASA's Exploration System Architecture Study," Technical Memorandum 2005-214062, National Aeronautics and Space Administration, November 2005.

[5] Olson, J., "Human Exploration Framework Team Phase I Closeout," National Aeronautics and Space Administration, September 2, 2010.

[6] Merrill, R. G., Strange, N., Qu, M., and Hatten, N., "Mars Conjunction Crewed Missions with a Reusable Hybrid Architecture," *IEEE Aerospace Conference*, March 2015, 2015-8.0104.

[7] Hofer, R. and Gallimore, A., "High-Specific Impulse Hall Thrusters, Part 1: Influence of Current Density and Magnetic Field," *Journal of Propulsion and Power*, Vol. 22, No. 4, 2006, pp. 721–731.

[8] Hofer, R. and Gallimore, A., "High-Specific Impulse Hall Thrusters, Part 2: Efficiency Analysis," *Journal of Propulsion and Power*, Vol. 22, No. 4, 2006, pp. 732–740.

[9] Murakami, N. and Yamanaka, K., "Trajectory design for rendezvous in lunar Distant Retrograde Orbit," *2015 IEEE Aerospace Conference*, 2015.

[10] Davis, D., Bhatt, S., Howell, K., Jan, J.-W., Whitley, R., Clark, F., Guzzetti, D., Zimovan, E., and Barton, G., "Orbit Maintenance and Navigation of Human Spacecraft at Cislunar Near Rectilinear Halo Orbits," *AAS Space Flight Mechanics Meeting*, 2017.

[11] Qu, M., Merrill, R. G., Chai, P. R., and Komar, D. R., "Optimizing Parking Orbits for Roundtrip Mars Missions," *AAS Space Flight Mechanics Meeting*, 2017.

[12] Sims, J. A., Finlayson, P. A., Rinderle, E. A., Vavrina, M. A., and Kowalkowski, T. D., "Implementation of a Low-Thrust Trajectory Optimization Algorithm for Preliminary Design," *AIAA/AAS Astrodynamics Specialist Conference and Exhibit*, 2006, AIAA 2006-6746.

[13] Chai, P. R., Merrill, R. G., and Qu, M., "Mars Hybrid Propulsion System Trajectory Analysis Part I: Crew Missions," *AIAA SPACE 2015 Conference and Exposition, Pasadena, CA*, August 2015, AIAA 2015-4443.

[14] Chai, P. R., Merrill, R. G., and Qu, M., "Mars Hybrid Propulsion System Trajectory Analysis Part II: Cargo Missions," *AIAA SPACE 2015 Conference and Exposition, Pasadena, CA*, August 2015, AIAA 2015-4444.

[15] Chai, P. R., Joyce, R. T., Kessler, P. D., and Merrill, R. G., "Sensitivity Analysis of Hybrid Propulsion Transportation System for Human Mars Expeditions," *AIAA SPACE 2017 Conference and Exposition, Orlando, FL*, 2017.

[16] Ocampo, C. and Senent, J., "The Design and Development of Copernicus: A Comprehensive Trajectory Design and Optimization System," *2006 International Astronautical Congress*, 2006, IAC-06-C1.4.04.

[17] Ewert, M. K., Broyan, J. L., Semones, E. J., Goodliff, K. E., Chai, P. R., Singleterry, R. C., Abston, L., Clowdsley, M. S., Wittkopp, C. J., and Vitullo, N. A., "Comparing Trash Disposal to Use as Radiation Sheilding for a Mars Transit Vehicle," *47th International Conference on Environmental Systems*, 2017, ICES-2017-178.