AIAA SciTech Forum
January 3-7, 2022, San Diego, CA & Virtual
AIAA SCITECH 2022 Forum
10.2514/6.2022-1354

Preliminary Screening of Multimode Spacecraft Propulsion Systems for Interplanetary Missions

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Multimode spacecraft propulsion combines two or more propulsive modes into a single system and has been shown to be beneficial for geocentric missions and for specific interplanetary missions. This study develops an analytical method for preliminarily screening all-chemical, all-electric, hybrid, and multimode propulsion systems for small spacecraft (100 kg class) completing interplanetary missions. Key parameters of comparison include payload mass delivered, transfer time, transfer rate, and excess propulsion capability for a given required payload mass. It is shown that present term multimode propulsion systems with a chemical mode specific impulse of 170 seconds and 16% electric mode efficiency provide lower transfer rates than all-electric and hybrid architectures for multiple concepts of operations for Earth-Mars missions. Increasing the multimode chemical specific impulse to 230 seconds and electric mode efficiency to 30% results in transfer rates greater than any other architecture considered can provide. It is also shown that multimode electric mode specific impulse values between 1,000 and 1,500 seconds produce the greatest multimode transfer rates across the scenarios considered. For a fixed payload mass of 40 kg, it is shown that multimode and hybrid systems have the potential for significant propulsive capability at the target planet. Although all-electric, hybrid, and multimode architectures can complete the considered transfers, multimode propulsion provides mission designers increased operational flexibility that may be compelling for some applications with present technology despite lower transfer rates. Future improvements in multimode performance and system dry mass reductions will provide transfer rates greater than all-electric or hybrid architectures in addition to flexibility.

I. Introduction

Multimode spacecraft propulsion combines two or more propulsive modes into a single system using a propellant common to both modes. This has the potential to reduce the required mass and volume of the propulsion system in comparison to all-chemical and hybrid (where multiple propulsive modes are present on a spacecraft but are entirely separate and do not share propellant) architectures while providing the flexibility of having a comparatively high thrust, low specific impulse chemical mode and a low thrust, high specific impulse electric mode available. One promising architecture currently being studied and developed within the Electric Propulsion Laboratory (EPLab) at the University of Illinois Urbana-Champaign (UIUC) combines a monopropellant chemical mode with an electrospray electric mode using a single thruster. The focus of this study is performing rapid, preliminary comparisons of this multimode approach to all-chemical, all-electric, and hybrid systems for small spacecraft (100 kg class) completing interplanetary missions. In particular, the methods developed are easily implemented by propulsion engineers working

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on defining requirements of future systems in addition to mission designers. These results may be used to guide future studies and serve as a starting point for higher fidelity analysis. Key performance metrics include the amount of payload mass delivered (i.e. all mass *not* related to the propulsion system) and the transfer time. These may be jointly assessed through the transfer rate (payload mass per transfer time, commonly with units of kg/day).

Several previous studies have examined the potential benefits of hybrid and multimode approaches for interplanetary missions, commonly using optimal control techniques. These studies are briefly summarized here and additional discussion may be found in Ref. [1]. Gilland [2], Mingotti, Topputo, and Massari [3], Percy, McGuire, and Polsgrove [4], and Chai, Merrill, and Qu [5] have noted benefits for Mars transfers using hybrid propulsion architectures. These include the potential for reduced propellant mass than all-chemical or all-electric transfers as well as shortened interplanetary phases. Percy et al and Chai et al. suggested hybrid approaches are necessary for crewed missions to Mars [4,5].

The Moon and near-Earth objects (NEO) have also been examined as potential targets for multimode missions. Kluever has shown hybrid approaches to be beneficial for lunar missions, with hybrid architectures delivering more payload than all-chemical or all-electric architectures [6,7]. Topputo and Massari showed hybrid architectures for a NEO sample return mission may simplify operations and allow for a larger sample to be returned in comparison to an all-chemical approach [8].

The University of Tokyo and JAXA's PROCYON was the first interplanetary small spacecraft (50 kg-class) to include multimode propulsion. PROCYON included xenon cold gas propulsion as well as a xenon ion propulsion mode [9–11]. Other multimode and hybrid small spacecraft have been considered, including a CubeSat with a monopropellant chemical mode and a radiofrequency ion thruster. This architecture was studied in detail by Mani, Cervone, and Topputo in the context of a 16U CubeSat mission to Mars. Their work showed a suitable balance between propellant mass and overall transfer time [12]. The EPLab at UIUC is currently investigating lunar trajectories for a 12U CubeSat with a multimode propulsion system combining monopropellant and electrospray modes [1].

Despite this body of work, little has been done to *broadly* assess multimode propulsion for interplanetary missions. This is likely due to the complexity of analyzing such missions, especially in a given optimal sense (e.g. payload delivered, transfer time, transfer rate, etc.). While all-chemical architectures can be readily estimated using basic orbital mechanics, electric maneuvers within an inverse-square gravity field are more difficult to analyze due to the lack of an analytical, closed-form solution for arbitrary low thrust maneuvers. Traditional methods of designing electric interplanetary maneuvers include optimal control techniques or use of a numerical optimizer, such as NASA JPL's Mission Analysis Low-Thrust Optimizer (MALTO) [13]. These methods, however, are often time consuming, computationally expensive, and require good initial guesses. These characteristics are not ideal for preliminary, large design space studies nor are they appropriate for designers and propulsion engineers to develop initial propulsion system requirements during conceptual design efforts.

Oh and Landau developed a semi-analytic model for optimal low-thrust interplanetary trajectories using curve fitting of higher fidelity results that is appropriate for conceptual design efforts [14]. This method, however, has a limited range of applicability and because it was developed using curve fitting techniques, it cannot be readily extended. These limitations made this approach insufficient for this investigation. The primary interplanetary transfer analysis technique used in this study was developed by Zola for rapidly assessing electric propulsion system requirements [15,16]. While Zola's analytical method is low-fidelity, it provides estimates of the propellant requirements and transfer time for electric interplanetary transfers. This makes it appropriate for preliminary screening of concepts for further consideration and higher fidelity analysis. To further assess the impact of the considered propulsion architectures on the spacecraft, the propulsion system dry mass was estimated using a variety of empirical estimates. This allows estimation of the payload mass that may be delivered to the target orbit.

The remainder of this paper is organized as follows: section II describes the methodology, section III presents results and discussion, and section IV recommends future work. The paper is concluded in section V.

II. Methodology

The emphasis in this study is rapidly estimating the amount of payload mass which may be delivered as well as the transfer rate using a given propulsion system to complete a specified interplanetary mission. In this study, payload mass refers to all mass not related to propulsion (i.e. spacecraft bus and other subsystems). Solar arrays, batteries, and other power handling components are considered part of the payload mass despite being necessary for electric propulsion. Multimode, hybrid, all-electric, and all-chemical propulsion systems are analyzed. These results are intended to be used to screen propulsion system concepts for further consideration and higher fidelity analysis as well as inform requirements for future systems.

A. Fundamental Assumptions

This study was intended to produce preliminary results. As such, several fundamental assumptions were made which apply to each scenario. Namely, the solar system was considered coplanar with all planetary bodies having circular orbits. Gravitational perturbations due to non-spherical central bodies were ignored. Other assumptions, in particular those related to an electric interplanetary phase, are discussed below.

B. All-Chemical Approach

The all-chemical architecture was modelled using the traditional impulsive patched conics method (in which the central body changes from the initial planet to the Sun and finally to the final planet throughout the trajectory) and assumed a Hohmann transfer beginning in geosynchronous (GEO) orbit. Monopropellant thrusters were considered for this approach.

C. Approaches using Electric Propulsion (All-Electric, Hybrid, and Multimode)

1. *Electric Propulsion Interplanetary Phase*

Charles Zola's [15,16] method for estimating propulsion requirements for interplanetary missions was used. This method is analytical and thus can be used to perform rapid trade studies. Unlike optimal control or other numerical approaches, it does not provide trajectory information beyond the transfer time (i.e. neither trajectory geometries nor launch windows may be found with this method).

This method assumes all planetary orbits are circular and coplanar. It also assumes constant thrust and constant specific impulse with optimized coasting periods (i.e. the thruster is either on or off without any throttling or degradation). This model is based on the "dynamic similarity between flight on an optimum trajectory in the inverse-square field and rest-to-rest flight on a rectilinear path in gravity-free space" [16]. Accordingly, the spacecraft is assumed to start and end in an orbit about the relevant central body with zero hyperbolic excess speed.

To avoid more complex modelling, this approach relates the propulsion time (i.e. thrusting time) to a "straight line" path length which the spacecraft would traverse in gravity-free space. This length can be estimated using complete numerical simulations. In this study, a value of 6×10^{10} meters was used as a representative length for Earth-Mars transfers. This value was estimated from plots included in Ref. [16]. Additional analysis could refine this value to increase the accuracy of the results.

Assuming minimum time transfers are desired (i.e. no coasting period), the following expression can be solved for the transfer time or propulsion "on time," t_p, given the path length, L, specific impulse of the system, I_{sp}, and initial acceleration at the beginning of the maneuver, a_0 (i.e. thrust per mass at start of the maneuver). The acceleration due to gravity on the Earth's surface is represented by g_0.

$$L = \frac{(I_{sp}g_0)^2}{a_0}\left(1 - \sqrt{1 - \frac{a_0 t_p}{I_{sp}g_0}}\right)^2 \tag{1}$$

That value, along with the other known values, can then be used in eqn. 2 to determine the required ΔV and thus propellant requirements via the rocket equation.

$$\Delta V = -I_{sp}g_0 \ln\left(1 - \frac{a_0 t_p}{I_{sp}g_0}\right) \tag{2}$$

Errors for this method are on the order of errors in the straight line length value used. Full numerical simulations would be required to accurately assess the error, though Zola claimed the method could predict ΔV with an error of 10% or less [16].

2. *Planetocentric Phases*

Planetocentric phases (escape and capture) were modelled using basic orbital mechanics relations. Since Zola's method does not patch each phase together, the escape phase was concluded upon reaching escape velocity using an impulsive maneuver or upon achieving zero velocity relative to the planet using a low-thrust maneuver. For low-thrust maneuvers, Edelbaum's equation (eqn. 3) was used. In this equation, v_1 and v_2 correspond to the circular orbital velocities of the spacecraft in the initial and final orbit, respectively, and θ represents any required plane change (in degrees).

$$\varDelta V = \sqrt{v_{1,\,circ}^2 + v_{2,\,circ}^2 - 2v_{1,\mathrm{circ}}v_{2,\mathrm{circ}}\cos(\frac{\pi}{2}\theta)} \tag{3}$$

The magnitude of the velocity change for an impulsive capture was modelled using eqn. 4 given the final orbit's radius of periapsis and eccentricity. Here, v_∞ is the hyperbolic excess speed, μ is the gravitational parameter for the central body, r_p is the radius of periapsis, and e is the final orbit's eccentricity.

$$\varDelta V = v_{\mathrm{p,hyperbolic}} - v_{\mathrm{p,capture}} = \sqrt{v_\infty^2 + \frac{2\mu}{r_p}} - \sqrt{\frac{\mu(1+e)}{r_p}} \tag{4}$$

Since Zola's method assumes "rest to rest" motion, the hyperbolic excess velocity is zero, reducing the equation to:

$$\varDelta V = \sqrt{\frac{2\mu}{r_p}} - \sqrt{\frac{\mu(1+e)}{r_p}} \tag{5}$$

For a low-thrust capture, which was assumed to be possible, eqn. 3 was used to model the propulsion requirements of the spacecraft spiraling from an orbit with zero velocity relative to the planet to the final orbit.

The burn time, t_p, for each planetocentric phase was determined using equation 6. In this expression, $m_{\mathrm{propellant}}$ is the mass of the propellant expended during the maneuver and $\dot{m}$ is the corresponding mass flow rate.

$$t_p = \frac{m_{propellant}}{\left(\frac{Thrust}{I_{sp}g_0}\right)} = \frac{m_{propellant}}{\dot{m}} \tag{6}$$

D. Propulsion System Mass Estimation

Mass estimation for each of the propulsion systems considered was performed using estimates from the NASA State-of-the-Art Small Spacecraft Technology report [17], ongoing work at the UIUC EPLab, and empirical relations and estimates [18–20].

Thruster mass values were estimated from Ref. [17] and were selected to be representative of typical thruster masses. The total mass of all required valves was estimated as 50% of the thruster mass [19,20]. Relationships from chapter 18 of Ref. [18] were used to estimate the mass of components including any required propellant tank(s), pressurant tank, and nitrogen pressurant assuming a beginning-of-life pressure of 300 psi, an end-of-life pressure of 100 psi, and a nominal pressurant temperature of 300 K. These values were chosen as first estimates and may be refined with further analysis. For systems with an electric mode, the power processing unit was estimated to be approximately 0.42 kg based on estimates from Ref. [20]. The mounting hardware mass was assumed to be 10% of the total dry mass of the propulsion system [20]. See Table 1 for a summary of the performance parameters and mass estimates used throughout the study.

E. Algorithm Summary

The following algorithm briefly summarizes the analysis method used in this study.

1) Select the spacecraft total mass, power, and propulsion system. Choose the initial trajectory (circular Earth orbit or escape trajectory) and final orbit.
2) Choose the concept of operations (i.e. order and type of maneuvers).
    a. If an all-chemical architecture is used, complete patched conics Hohmann transfer analysis.
    b. Otherwise, use the methods described in section II.C.
3) Calculate the required propellant mass and transfer time.
4) Given the propulsion system and propellant mass required, estimate the dry propulsion system mass and pressurant mass (if required). Subtract this mass and the propellant mass from the spacecraft initial mass to determine the payload mass delivered. If the required payload mass is known, determine the excess mass that may be used for additional propulsive capability (e.g. orbital transfers or station keeping) at the target body.
5) Divide the delivered mass by the transfer time to determine the transfer rate.
6) Repeat for other scenarios of interest.

III. Results and Limitations

One potentially advantageous concept of operations (CONOPs) for a spacecraft using a multimode propulsion system is to use the chemical mode to escape the Earth's gravity well, perform an electric interplanetary cruise, and perform an impulsive (chemical) capture maneuver at the destination planet. Figure 1 represents this CONOPs schematically for a Mars mission (not to scale). Other CONOPs, including trajectories which start with sufficient energy to escape Earth, are possible. This section includes discussion of several mission scenarios, including the aforementioned, as well as limitations of the analysis method.

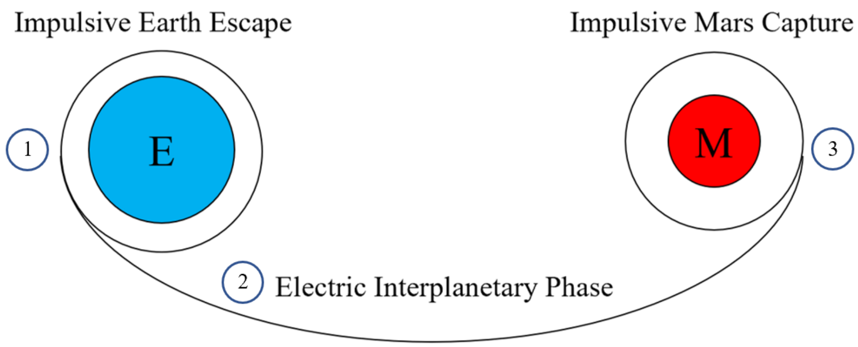

Fig. 1 Possible Earth-Mars Interplanetary Mission CONOPs Schematic

A. Performance and Mass Estimates

The focus of this study is comparing the performance of near-term multimode systems to current state-of-the-art all-chemical, all-electric, and hybrid architectures and future multimode systems for 100 kg spacecraft. As a result, a variety of specific impulses has been considered for multimode propulsion in both modes. These range from 170 sec (near-term) to 230 sec (future-term) for the chemical mode and 1,000 sec (near-term) to 2,500 sec (future-term) in electric mode. Electric mode efficiencies of 16% (near-term) and 30% (future-term) were also considered. This efficiency, η, relates the input power to the jet power as shown by eqn. 7. In this equation, T represents thrust in Newtons, I_{sp} is the specific impulse, and g_0 is the gravitational acceleration at Earth's surface.

$$\eta P_{input} = P_{jet} = \frac{1}{2} T I_{sp} g_0 \tag{7}$$

Input power levels, P_{input}, of 100 W and 200 W were explored (n.b. the sizing of solar panels or other power elements was outside the scope of this investigation). For comparison to the multimode systems, representative ranges for thrust and specific impulse were selected from Ref. [17] for the all-chemical, all-electric, and hybrid architectures.

The specific architectures considered in this study include an ionic-liquid multimode system combining a monopropellant chemical mode with an electrospray electric mode, a monopropellant all-chemical system completing a Hohmann transfer, an all-electric xenon Hall thruster system, and a hybrid system with the aforementioned hydrazine monopropellant thruster and xenon Hall thruster. The multimode system was assumed to produce 1 N of thrust in chemical mode and have a mass of 2 kg. It was also assumed to use a novel ionic propellant, FAM-110A ([Emim][$EtSO_4$]-HAN in a 41:59% ratio by mass), specifically designed for multimode propulsion [21–25]. The chemical monopropellant thruster was assumed to be capable of producing 22 N of thrust with a specific impulse of 230 seconds. The Hall thruster was considered to be 30% efficient with a specific impulse between 800 and 1,900 seconds. These systems were selected due to their maturity for small spacecraft [17]. The chemical and electric thrusters were assumed to have masses of 0.6 kg and 1.2 kg, respectively.

Table 1 summarizes the performance values and mass estimates used throughout this investigation. The propellant tank mass, pressurant tank mass, pressurant mass, and mounting hardware mass were each determined iteratively as described in section II.D. Note that the xenon propellant used in the all-electric and hybrid cases was assumed to not require additional pressurization beyond its storage pressure and the corresponding propellant tank mass was found using the pressurant tank (i.e. gas pressure vessel) relations in Ref. [18]. In this table and all following figures, CP refers to chemical propulsion (or chemical mode) and EP to electric propulsion (or electric mode).

Table 1 Propulsion System Performance Values and Mass Estimation

	All-Chemical	All-Electric	Hybrid	Multimode
Total Spacecraft Mass (kg)	100	100	100	100
Total EP Power Available (W)	-	100, 200	100, 200	100, 200
Multimode Thruster Mass (kg)	-	-	-	2
CP Thruster Mass (kg)	0.6	-	1.2	-
CP Specific Impulse (s)	230	-	230	170, 200, 230
CP Thrust (N)	22	-	22	1
CP Propellant/Density (kg/m^3)	Hydrazine/1,021	-	Hydrazine/1,021	FAM-110A/1,422
EP Thruster Mass (kg)	-	1.2	0.6	-
EP Specific Impulse (s)	-	800-1,900	800-1,900	1,000-2,500
EP Efficiency (%)	-	30	30	16, 30
EP Propellant/Density (kg/m^3)	-	Xenon/5.89	Xenon/5.89	FAM-110A/1,422
PPU Mass (kg)	-	0.42	0.42	0.42
Valve Mass (kg)	0.3	0.6	0.9	1
Propellant Tank Mass (kg)	*	*	*	*
Pressurant Tank Mass (kg)	*	-	*	*
Nitrogen Pressurant Mass (kg)	*	-	*	*
Mounting Hardware Mass (kg)	*	*	*	*

*Iteratively Determined

One way to quantify the mass savings a multimode approach provides over a hybrid system is through the system integration factor [20] that is described by eqn. 8.

$$f_{SI} = 1 - \frac{m_{e,int}}{m_{e,sep}} \tag{8}$$

In this expression, f_{SI} is the system integration factor, $m_{e,int}$ is the dry propulsion mass (no propellant or pressurant) of an integrated (i.e. multimode) system, and $m_{e,sep}$ is the dry propulsion mass of an entirely separate (i.e. hybrid) system. System integration factors greater than zero represent mass savings provided by a multimode approach over a hybrid approach. Due to the variety of CONOPs and performance levels explored as well as the masses of some propulsion system components being calculated iteratively, it is not possible to provide a single system integration factor that represents all of the possible multimode systems considered in this study. It was found, however, that the system integration factor ranges between 0.18 and 0.26 across all parameters explored in this investigation.

B. Geosynchronous orbit to Mars

One CONOPs potentially of interest to small spacecraft operators is that of a "rideshare" mission in which the spacecraft is initially placed in geosynchronous orbit. To escape Earth, an impulsive or low-thrust maneuver can be used to achieve escape energy prior to the interplanetary phase. A similar option is available upon reaching Mars (the target body). The results for the multimode, all-electric, and hybrid architectures are compared to the two-dimensional chemical Hohmann transfer.

Results for this CONOPs, assuming 100 W of power available, are shown in Fig. 2, which includes four subplots. The top left subplot displays the total propellant mass required while the top right shows the payload mass remaining after the completion of the transfer (i.e. remaining mass after accounting for propellant mass and propulsion system dry mass). The bottom left displays the total transfer time (total of all three phases) and the bottom right shows the transfer rate. The initial orbit is geosynchronous while the final orbit about Mars is a circular orbit with a radius of 23,463 km, which is similar to that of Deimos. Other final orbits can readily be explored.

The all-chemical system is assumed to complete a Hohmann transfer and is thus represented by a single point on each subplot. For the other systems, a range of electric specific impulses are considered. Any single point on the plot could represent a given propulsion system. In this figure, 100 W of power is assumed to be available for the electric propulsion systems and the escape and capture maneuvers are impulsive for the hybrid and multimode systems (q.v. Fig. 1). The multimode system's electric mode efficiency is assumed to be 16% (n.b. sections III.C. and III.D. considered the impact of increased multimode efficiency). Multimode systems are described in the legend by their chemical specific impulse and electric mode efficiency.

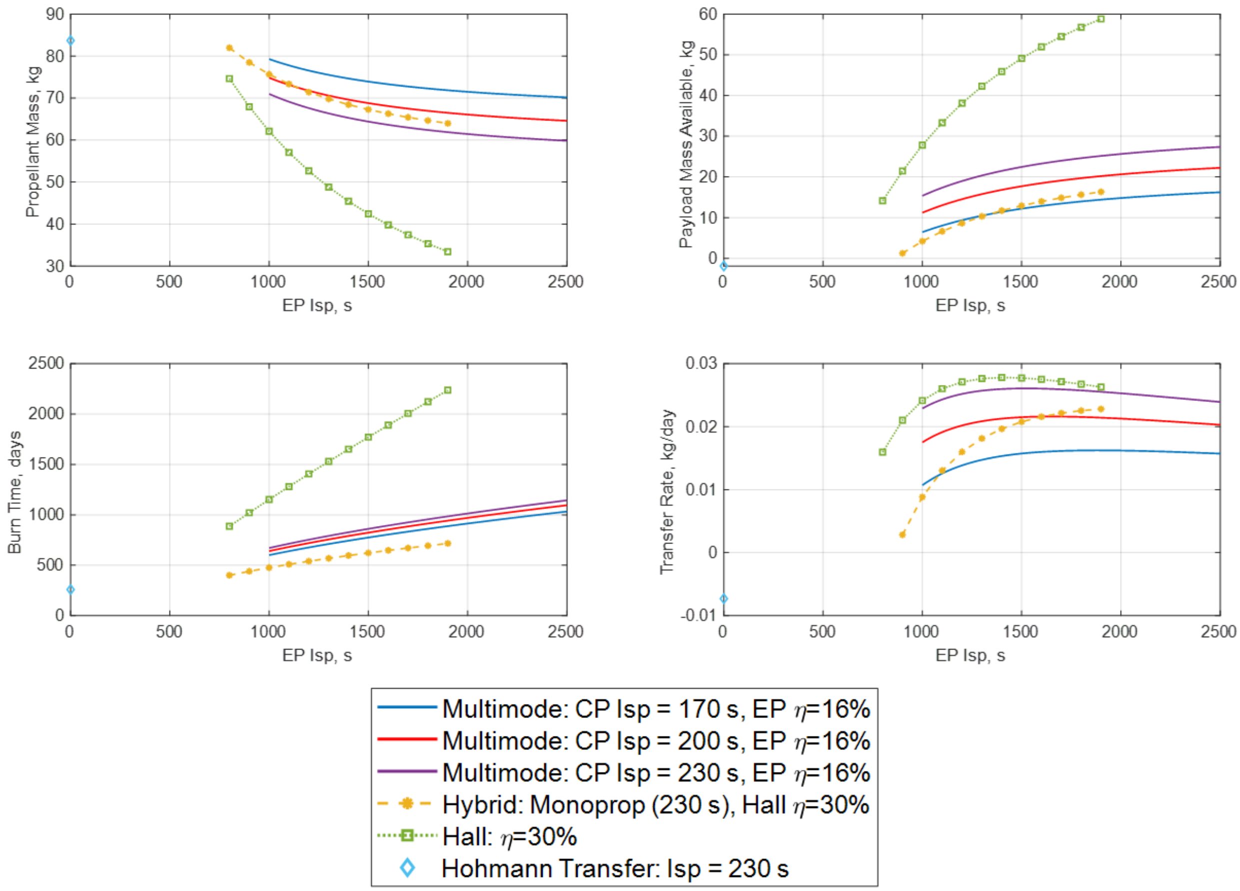

Fig. 2 Geosynchronous Orbit to Mars, 100 W Available. Hybrid and Multimode CONOPs: Impulsive Escape, Low-Thrust Interplanetary Phase, Impulsive Capture
CW from top left: a) Propellant Mass Required b) Payload Mass Available c) Burn Time d) Transfer Rate

Figure 2 shows a large quantity of propellant mass is required for systems using chemical maneuvers (all-chemical, multimode, and hybrid systems). This greatly reduces the amount of payload mass available. In the case of the Hohmann transfer, the combined propellant mass and dry propulsion mass is greater than 100 kg (hence the negative payload mass available). As a result, this system cannot be used for a 100 kg spacecraft with this CONOPs. While the Hall thruster system delivers the greatest payload mass for most scenarios due to its comparatively higher efficiency, its transfer rate is similar to that of the multimode system with a chemical specific impulse of 230 seconds due to the long escape and capture phases (and thus long overall burn time). For the multimode systems, the chemical specific impulse has a non-negligible impact on the transfer rate. An increase in chemical specific impulse from 170 seconds to 230 seconds, however, may not result in sufficient mass savings for this CONOPs to be viable for a given mission (i.e. potentially too little payload mass delivered).

This figure also reveals the maximum transfer rate for the multimode systems occurs with an electric mode specific impulse of approximately 1,500 seconds. This is in contrast to the common belief that greater specific impulse always improves performance. While higher specific impulse values reduce the propellant mass required, this is at the cost of longer transfer times (due to lower levels of thrust) that increase faster than the mass is reduced. This reduces the possible transfer rate. Multimode electric mode specific impulse values producing that maximum transfer rate typically lie between approximately 1,000 and 1,500 seconds across all scenarios explored in this study.

In the figure below, all initial parameters are identical to those used in Fig. 2, though the CONOPs has changed for the multimode and hybrid systems. In this case, the escape is assumed to be impulsive while the interplanetary and capture phases are completed using low-thrust maneuvers.

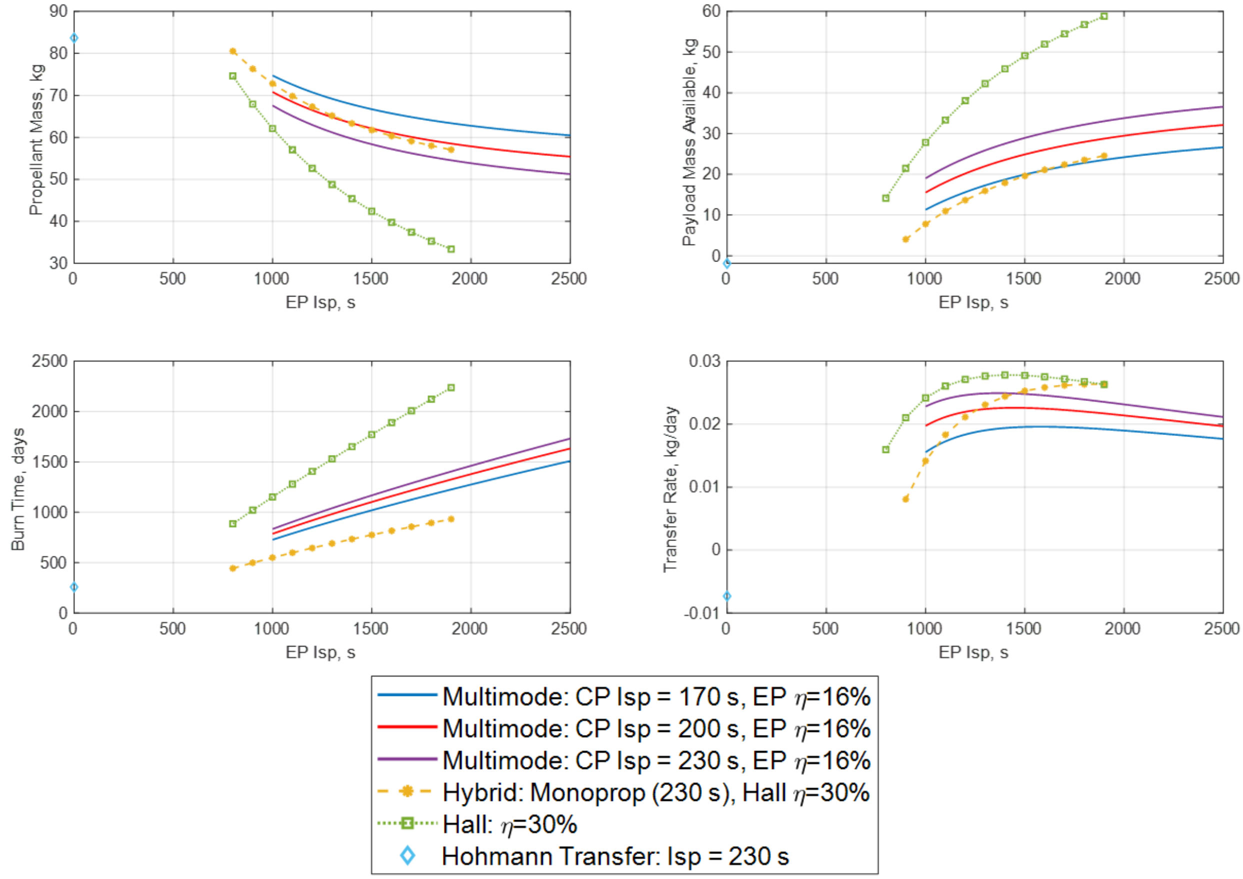

Fig. 3 Geosynchronous Orbit to Mars, 100 W Available. Hybrid and Multimode CONOPs: Impulsive Escape, Low-Thrust Interplanetary Phase, Low-Thrust Capture
CW from top left: a) Propellant Mass Required b) Payload Mass Available c) Burn Time d) Transfer Rate

This figure reveals using a low-thrust capture saves approximately 8-10 kg of propellant mass for the hybrid and multimode systems due to the expanded use of electric propulsion modes. This is translated into increased payload mass available at the cost of increased transfer times. Despite the long transfer time, the all-electric architecture delivers a transfer rate similar to the other systems due to significantly larger amounts of payload mass being delivered.

These two figures suggest the impulsive escape (and capture) greatly limit the amount of payload which may be delivered by the multimode and hybrid systems. Low-thrust escape from the Earth is also exceedingly time consuming and may minimize the potential benefits of an all-electric system. If extremely long transfer times can be tolerated, an all-electric system with high specific impulse will allow the greatest payload mass to be delivered to the target. Long transfers, however, may not be desirable for small satellite missions. While the all-chemical system completes the transfer in a shorter time frame than the other systems, it requires more total propulsion mass than the allowable total spacecraft mass (100 kg).

C. Escape Trajectory ($C_3 = 0$) to Mars

A second possible CONOPs eliminates the Earth escape phase by assuming the spacecraft is initially on an escape trajectory. Due to the limitations of Zola's method, this trajectory is assumed to have a characteristic energy (i.e. the square of the hyperbolic excess speed or twice the specific energy), C_3, of 0. Thus, for the architectures using electric propulsion, the interplanetary phase is the first propulsive maneuver. Upon reaching the target (Mars), a capture maneuver is required. For the multimode and hybrid systems, this can potentially be an impulsive maneuver or a low-thrust maneuver. In this section, impulsive capture is assumed. Brief consideration of low-thrust capture is included in the following section. The impacts of available power and multimode electrospray efficiency are also presented.

Figure 4 is identical to Fig. 2 with the exception that the initial escape from Earth has not been included. The all-chemical Hohmann transfer considered previously is included for reference. The electric mode of the multimode systems has again been assumed to be 16% and the final orbit is identical to that of the previous section. This results in significant mass savings for all systems (with the exception of the impulsive Hohmann transfer system). A single solid curve representing the burn time for the multimode system is visible because all three multimode systems considered (with chemical specific impulses of 170 sec, 200 sec, and 230 sec, respectively) complete the transfer in the same amount of time per eqn. 1 (i.e. each has the same initial acceleration, a_0). The transfer rate for all systems (again, excluding the Hohmann transfer) is greater than that in Fig. 2. Here, greater parity is seen between the transfer rates of the electric and hybrid systems, though the all-electric architecture delivers more payload. Generally, this scenario results in lower transfer rates for the multimode systems than for the all-electric and hybrid approaches primarily due to the lower electric mode efficiency resulting in comparatively lengthy transfers. The multimode system also has greater dry mass than the all-electric approach and thus delivers lower payload mass than the Hall thruster system.

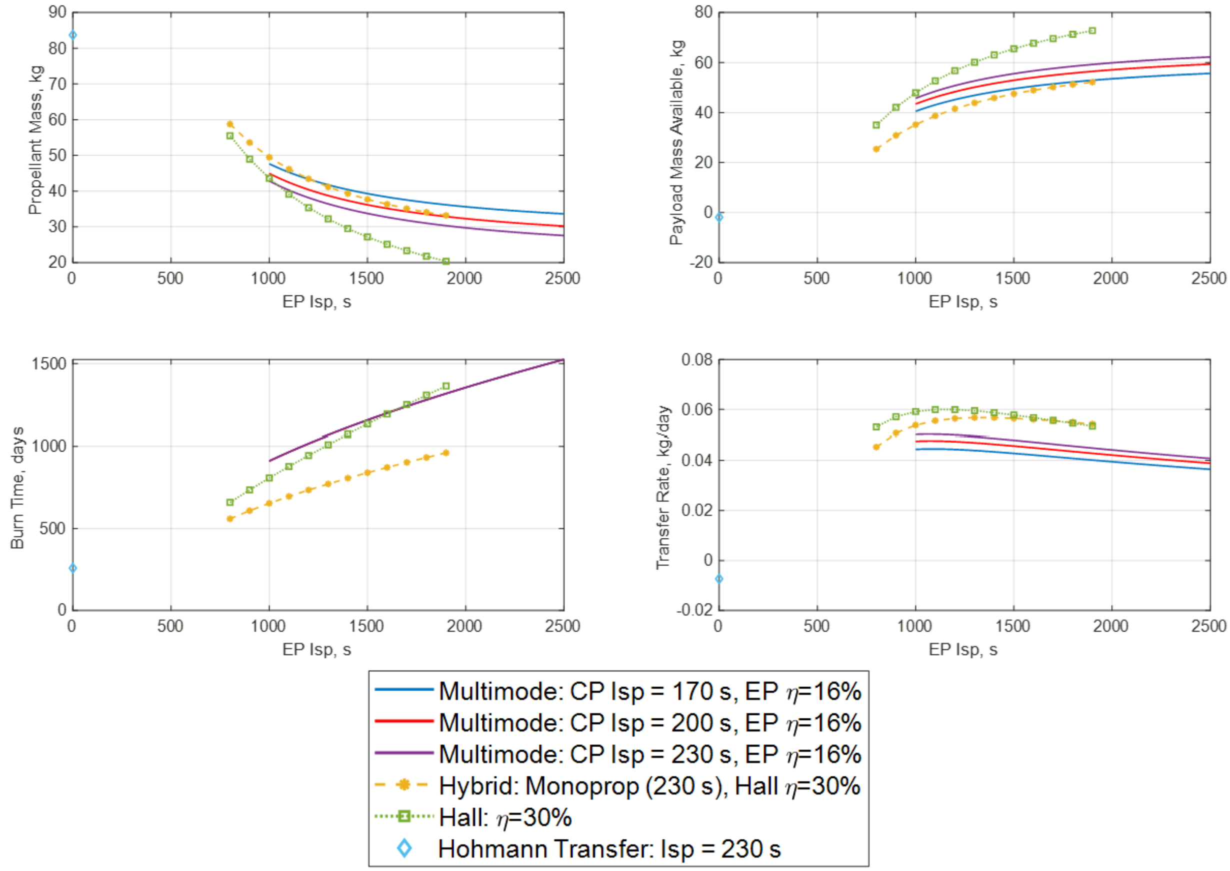

Fig. 4 Earth-escape Trajectory to Mars, 100 W Available. Hybrid and Multimode CONOPs: Low-Thrust Interplanetary Phase, Impulsive Capture
CW from top left: a) Propellant Mass Required b) Payload Mass Available c) Burn Time d) Transfer Rate

In Fig. 5, a comparison between the all-chemical, all-electric, and hybrid systems is shown between multimode systems with a chemical specific impulse of 230 seconds (representing future technology) and electric mode efficiencies of 16% and 30% (expected to be feasible with further development of multimode systems). It should be noted that a multimode chemical mode specific impulse of 230 seconds and a 30% efficient electric mode corresponds to the level of performance assumed for the hybrid system throughout this study. This figure reveals the 30% efficient multimode system provides the overall greatest transfer rate of the architectures considered for most electric mode specific impulse levels and provides a transfer rate approximately 0.01 kg/day greater than the 16% efficient multimode system. In comparison to the hybrid system, the 30% efficient multimode system requires the same propellant mass and transfer times (due to the identical thruster performance values) but delivers a greater payload mass because of the lower propulsion system dry mass. This results in an increased transfer rate for the multimode system. Finally, the transfer rate for the 16% efficient multimode system is lower than any of the other architectures, despite greater payload mass delivered than the hybrid and 30% efficient multimode system, due to generally longer transfer times stemming from lower thrust levels for fixed specific impulse in electric mode.

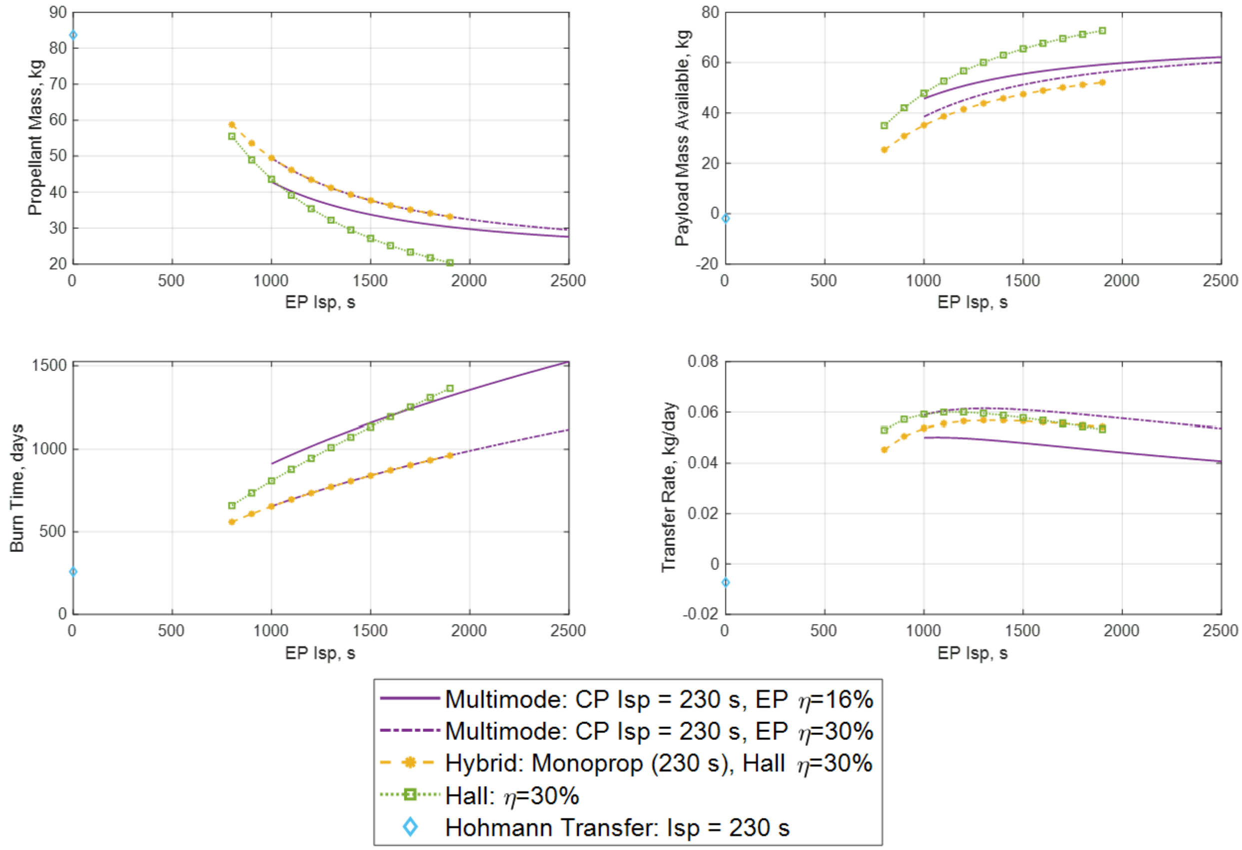

Fig. 5 Earth-escape Trajectory to Mars, Multimode Efficiency Comparison, 100 W Available. Hybrid and Multimode CONOPs: Low-Thrust Interplanetary Phase, Impulsive Capture CW from top left: a) Propellant Mass Required b) Payload Mass Available c) Burn Time d) Transfer Rate

Finally, the impact of propulsion power available is considered. In Fig. 6 below, the multimode systems are assumed to have an efficiency of 30% in electric mode and a chemical specific impulse of 230 seconds. Generally, more power available leads to greater transfer rates despite increases in propellant requirements (increased power leads to increased thrust and thus increased mass flow rate for fixed power and specific impulse in electric propulsion modes). This benefit, however, results in the reduction of payload mass available in comparison to the lower power systems. The all-electric approach delivers the greatest transfer rate when 200 W of power are available while the hybrid and multimode systems provide nearly identical transfer rates.

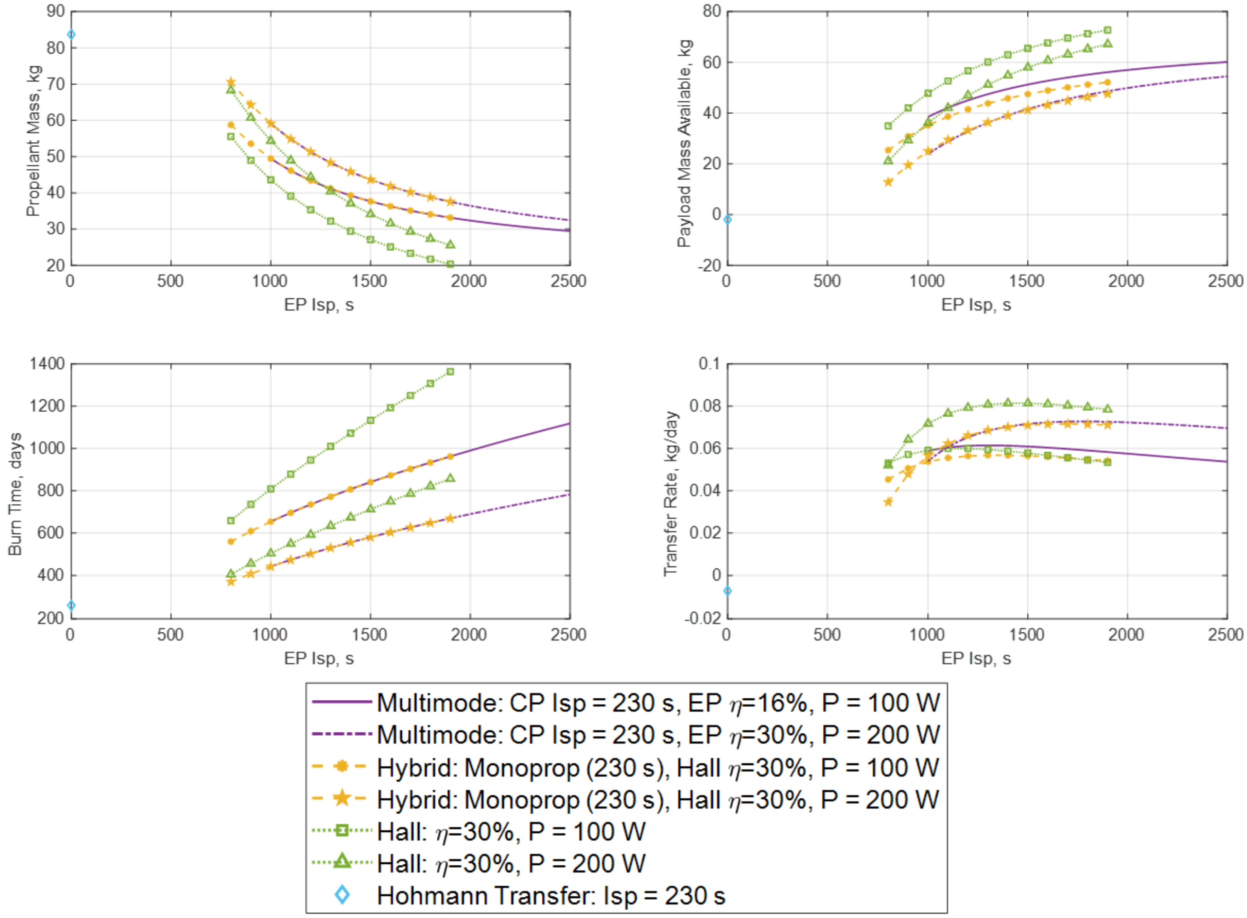

Fig. 6 Earth-escape Trajectory to Mars, Power Comparison. Hybrid and Multimode CONOPs: Low-Thrust Interplanetary Phase, Impulsive Capture
CW from top left: a) Propellant Mass Required b) Payload Mass Available c) Burn Time d) Transfer Rate

Although this section did not explore all possible combinations of multimode chemical specific impulse and electric mode efficiency as well as propulsion power available, the impacts of each have been considered. Generally, increased electric mode efficiency and power increase the transfer rate while reducing payload mass delivered. The parity between the all-electric, multimode, and hybrid systems in terms of transfer rate suggests similar missions can be performed with hybrid or multimode approaches with shorter transfers than an all-electric approach requires with some reduction in payload mass delivered.

D. Fixed Payload Mass: Capability at Target

In this section, an Earth-Mars transfer that begins on an escape trajectory with low-thrust interplanetary *and* capture phases is considered. The spacecraft is assumed to be generating 100 W for propulsion. Multimode systems with 16% and 30% efficient electric modes are considered. The results for this scenario are shown in Fig. 7. Due to the hybrid system being used as though it is a pure electric system, no mass is assigned to the chemical mode and thus the performance measures overlap for the hybrid and electric systems (i.e. the hybrid and all-electric systems are identical). Since the modes are separate in a hybrid system, some payload mass could readily be reassigned to a chemical propulsion mode.

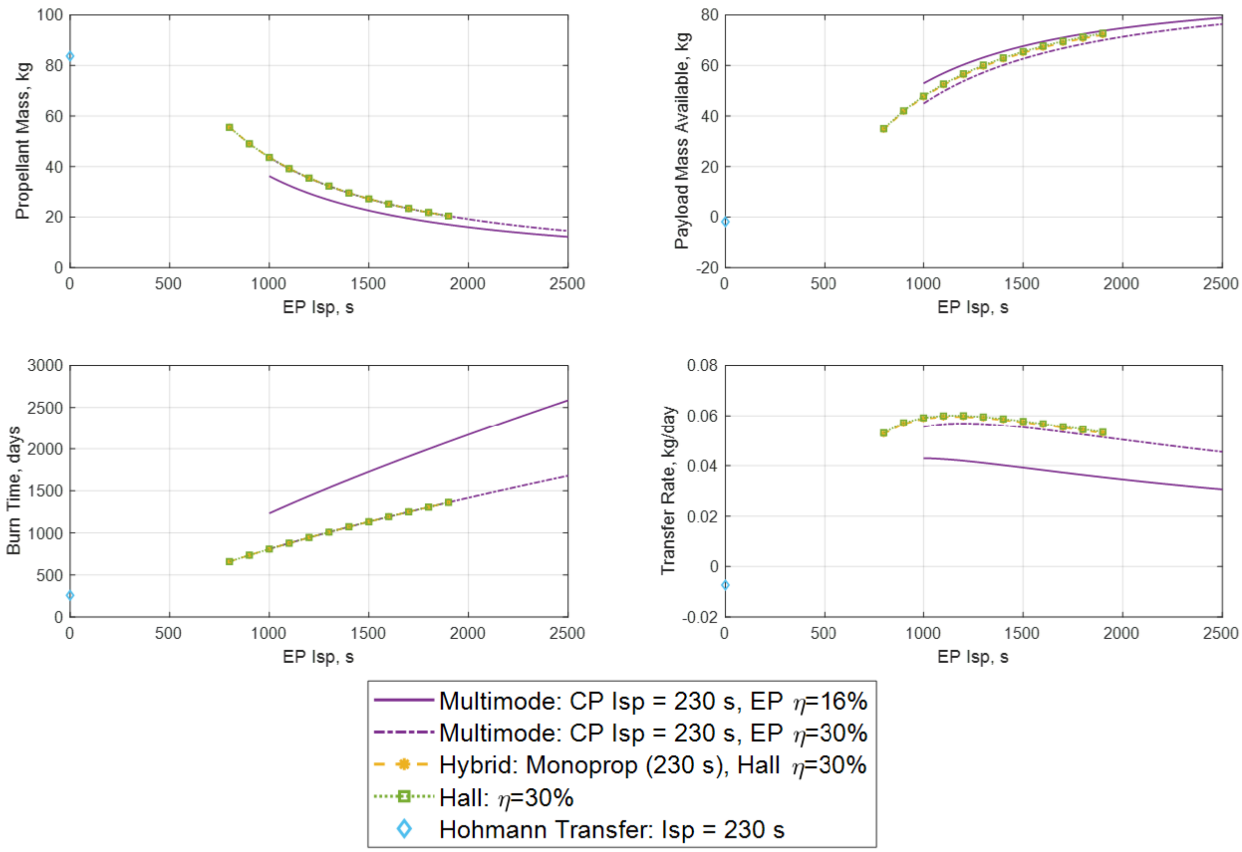

Fig. 7 Earth-escape Trajectory to Mars, 100 W Available. Hybrid and Multimode CONOPs: Low-Thrust Interplanetary Phase, Low-Thrust Capture at Mars
CW from top left: a) Propellant Mass Required b) Payload Mass Available c) Burn Time d) Transfer Rate

If the required payload mass to be delivered is known, any mass remaining after the transfer above the required value can be used for propulsion at the target body. Here, it was assumed 40 kg of payload mass was required. Fig. 7 reveals the all-electric, multimode, and hybrid architectures have excess mass available (i.e. more than 40 kg of payload) upon reaching the final orbit for some specific impulse values.

Figure 8 depicts the excess the excess propellant mass and ΔV capability (found using the ideal rocket equation) available at the target body for the multimode and hybrid architectures (the all-electric approach is omitted due to its comparatively limited operational flexibility). Curves corresponding to both 16% and 30% efficient electric modes are included for the multimode systems.

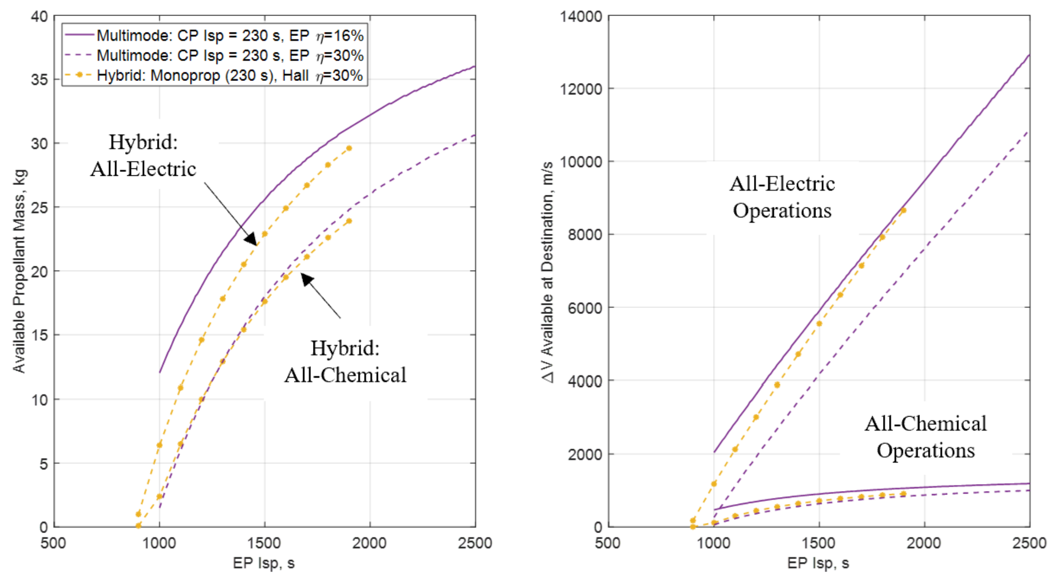

Fig. 8 Earth-escape Trajectory to Mars, 100 W Available. Hybrid and Multimode CONOPs: Low-Thrust Interplanetary Phase, Low-Thrust Capture at Mars a) Available Propellant and b) ΔV Capability at Target Body

For the multimode systems, a single curve represents the available propellant mass in the left side of Fig. 8. The hybrid system, however, requires two curves because the usage ratio must be predetermined to appropriately allot the propellant and dry propulsion mass (i.e. propellant can be assigned to chemical or electric mode but not both). Here, the extremes (all-chemical or all-electric) are shown. Any value lying between the two hybrid curves is also possible. Note the 30% efficient multimode system has lower propellant mass available (and correspondingly a smaller range of ΔV) than the 16% efficient system due to more propellant being consumed during the electric interplanetary phase.

On the right side of the figure, the available ΔV capability is shown. In the case of the multimode systems, any value between the curves may be achieved depending on which mode is used. The hybrid system can similarly achieve any value of ΔV between the two relevant curves, but that must be predetermined. While the multimode and hybrid systems have similar performance envelopes according to this metric, the primary benefit a multimode system affords mission designers is flexibility. Maneuvers can be planned while operations are ongoing as opposed to needing to be predetermined. This may be advantageous for some missions and would allow operators and scientists to respond to in situ needs and discoveries.

E. Limitations

While this method rapidly provides results, it is low-fidelity due to a large number of fundamental assumptions, particularly for architectures using an electric interplanetary phase, and the highly imprecise path length (L, in eqn. 1) value. These assumptions include assuming a circular, coplanar solar system (this assumption holds somewhat better for Earth-Venus transfers than Earth-Mars transfers). Additionally, this method does not consider planetary alignment or launch windows—the transfer is always assumed to be feasible. Further, the departure and arrival energies are assumed to be zero (i.e. this analysis method does not allow for a $C_3 > 0$).

Other limitations of this analysis include only a consideration of minimum time transfers, the phases are not smoothly patched (in comparison to the typical patched conics method), and all chemical maneuvers were assumed to be impulsive. The latter assumption may not be valid low-thrust chemical systems. Thruster lifetime and degradation are not considered. Constant power is also assumed, though this assumption is akin to having sufficient battery life during eclipses and sufficient solar arrays at the target body. It is also assumed an electric propulsion capture is feasible at the target (this may be a poor assumption).

For the purposes of comparison, the final orbit in all cases has been assumed to be circular in this study. This allows application of Edelbaum's equation for electric captures. Impulsive captures, however, allow analysis of the propellant impacts of non-circular orbits. As is well known, elliptical orbits reduce ΔV requirements in comparison to circular orbits. Thus, the final orbit has a non-negligible impact on the propellant required.

Finally, the accuracy of the results is unknown due to the variety of estimates used for both the transfers and dry propulsion system. It should be noted, however, that early mission concept studies often carry 40% mass margins [14]. Therefore, these results may be sufficient for preliminary studies.

IV. Future Work

Future work should include validation of the results herein using more sophisticated techniques, such as optimal control methods or existing numerical optimizers. Exploration of electric transfers that include a coast phase is also recommended. While the transfer time increases, more payload mass can be delivered in this scenario. Exploring electric captures to elliptical orbits and developing a more sophisticated mass build-up are also priorities. The analysis method used in this study may also be improved by allowing for variation in power available between Earth and the destination as well as variations in the thrust and/or specific impulse of the system.

V. Conclusion

An analytical method for comparing multimode, hybrid, all-electric, and all-chemical propulsion systems for small satellites completing interplanetary missions has been presented along with limitations. Primarily qualitative trends were discussed due to the low fidelity of the model. While only transfers to Mars were presented, this methodology can readily be used to explore transfers to other targets. The particular concept of operations, system performance parameters (e.g. specific impulse, efficiency), and available electric power significantly impact the transfer rate for the scenarios considered. It was determined a two-dimensional Hohmann transfer using a chemical monopropellant system requires more than 100 kg to be dedicated to propulsion and thus is not feasible.

One promising CONOPs for a spacecraft using multimode propulsion begins with the spacecraft on an escape trajectory and using low-thrust propulsion for the interplanetary and capture phases. For a fixed payload mass of 40 kg, significant mass is available for maneuvers at the target body.

Chemical mode specific impulse and electric mode efficiency are key parameters for maximizing the benefits of multimode propulsion. Present multimode systems with a chemical specific impulse of 170 seconds and 16% electric mode efficiency generally provide lower transfer rates than all-electric or hybrid architectures. It was found that multimode electric mode specific impulse values between 1,000 and 1,500 seconds generally produce the greatest transfer rates for all scenarios considered. Analysis of trajectories that begin on an Earth-escape trajectory with an electric interplanetary phase and a chemical mode capture at Mars showed raising the chemical mode specific impulse to 230 seconds and the electric mode efficiency to 30% significantly improves the system's transfer rate (beyond that of the hybrid system) at the cost of some payload mass. Doubling the power available for propulsion from 100 W to 200 W was also shown to increase the transfer rate for all systems using at least one electric phase at the cost of payload mass delivered.

Despite the low fidelity of the results, it is believed this study can be used to inform future studies considering multimode propulsion for small satellite interplanetary missions and for preliminary requirements definition of future multimode systems. To maximize the benefits of multimode propulsion for Earth-Mars missions, chemical mode specific impulse and electric mode efficiency must increase over present levels while also reducing dry mass. Significant improvement will be achieved when multimode performance levels surpass those of hybrid systems while also achieving greater system integration factors. While present multimode technology generally provides lower transfer rates in comparison to hybrid systems, the multimode approach provides mission designers and operators greater flexibility in that all maneuvers do not need to be predetermined prior to launch and assigned to a given mode. This flexibility may be beneficial to some science missions, such as a mission investigating a temporally or spatially-varying phenomenon, and may be sufficiently compelling for some applications with present technology. Improved multimode thruster performance beyond present hybrid systems will provide the greatest benefits in terms of both transfer rate and in situ flexibility.

Acknowledgments

The authors thank Thomas Liu from NASA Glenn Research Center and Khary Parker from NASA Goddard Space Flight Center for their time, expertise, and insight. This work was completed at the University of Illinois Urbana-Champaign as part of the NASA SmallSat Technology Partnerships program, grant number 80NSSC20M0089.

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