Assessment of High-Power Electric Multi-Mode Spacecraft Propulsion Concepts

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Multi-mode spacecraft propulsion systems utilizing a chemical monopropellant or bipropellant thruster and high-power electric Hall, arcjet, or pulsed inductive thruster were analyzed and compared in terms of mission capabilities. These systems are most effective compared to an all-chemical system when greater than 25% of the total delta-V is accomplished by the electric system due to the high mass requirements of the power system. Additionally, monopropellant systems are more effective in terms of both reduced system mass as well as decreased burn duration over bipropellant systems despite having 33% lower specific impulse and 1/10th the thrust. The monopropellant/PIT system yielded the lowest total system mass, however the monopropellant/arcjet had the highest transportation rates over all potential chemical/electric delta-V ratios.

Nomenclature

A_c	=	combustion chamber cross sectional area, [m^2]
A_t	=	throat area, [m^2]
C_F	=	thrust coefficient
C	=	effective exhaust velocity, [m/s]
D_c	=	combustion chamber diameter, [m]
D_t	=	throat diameter, [m]
EP	=	electric propulsion usage fraction
F	=	thrust, [N]
F_{tu}	=	ultimate strength of material, [N/m^2]
f_{inert}	=	inert mass fraction
g_0	=	acceleration of gravity, [m/s^2]
I_{sp}	=	specific impulse, [s]
$I_{sp,chem}$	=	chemical mode specific impulse, [s]
$I_{sp,elec}$	=	electric mode specific impulse, [s]
$I_{sp,mm}$	=	multi-mode effective specific impulse, [s]
L_c	=	combustion chamber length, [m]
L^*	=	characteristic combustion chamber length
m_0	=	initial mass of spacecraft, [kg]
m_c	=	combustion chamber mass, [kg]
m_{cables}	=	mass of electrical cables, [kg]
m_{chem}	=	mass of chemical propellant, [kg]
m_{elec}	=	mass of electric propellant, [kg]
m_f	=	final mass of spacecraft, [kg]

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m_{f1}	=	mass of spacecraft after first burn, [kg]
m_{inert}	=	inert mass, [kg]
m_{pay}	=	payload mass, [kg]
m_{PPU}	=	mass of power processing unit, [kg]
m_{prop}	=	propellant mass, [kg]
m_{sa}	=	mass of solar array, [kg]
m_{tank}	=	mass of propellant tank, [kg]
P_b	=	burst pressure, [Pa]
P_c	=	chamber pressure, [psi]
P_e	=	nozzle exit pressure, [Pa]
P_{thr}	=	electric thruster power, [kW]
r_c	=	combustion chamber radius, [m]
r_t	=	throat radius, [m]
T_R	=	transportation rate, [kg/day]
t_b	=	thruster burn time [day]
t_w	=	wall thickness, [m]
α	=	nozzle divergence half-cone angle, [degrees]
ΔV	=	velocity increment, [m/s]
ε	=	nozzle expansion ratio
η_t	=	thrust efficiency
θ_c	=	convergent section angle, [degrees]
γ	=	specific heat ratio
λ	=	nozzle divergence correction factor
φ_{tank}	=	empirical tank sizing parameter
ρ_{prop}	=	propellant density, [kg/m^3]
ρ_w	=	wall material density, [kg/m^3]

I. Introduction

MULTI-mode spacecraft propulsion is the utilization of a combination of high-thrust chemical and low-thrust, high-specific impulse electrical thrusters on a single spacecraft, ideally making use of common propellants and/or integrated hardware. This can be beneficial in two ways. One is through increased mission flexibility in the sense that either chemical or electric propulsive maneuvers can be performed at-will. The second way a multi-mode propulsion system can be beneficial is through a set of precise, pre-determined chemical and electric maneuvers that function to execute a mission in a manner that provides a more optimal trajectory over either a solely chemical or solely electric maneuver. This study presents an assessment and comparison of several multi-mode propulsion system concepts involving a high-power electric thruster along with a chemical monopropellant or bipropellant thruster.

One of the main drivers for research into multi-mode spacecraft propulsion is the potential for flexible spacecraft.[1,2] Since either high-thrust or high-specific impulse maneuvers can be performed at-will, this leads to the possibility of launching a spacecraft without a wholly predetermined mission profile. Propulsion modes can then be selected as mission needs arise in-situ rather than precisely choreographed prior to launch. Additionally, it has been shown that under certain mission scenarios it is beneficial in terms of spacecraft mass savings, or deliverable payload, to utilize two separate propulsion systems, even if there is no common hardware or propellant.[3-5] For example, use of a separate chemical rocket to escape earth gravity avoids a long spiral trajectory characteristic of an electric burn, while a high-specific impulse electric burn in interplanetary space saves propellant mass over a chemical rocket.[6] However, it has been shown that even greater mass savings can potentially be realized through the use of shared propellants or shared hardware.[7,8]

While there have been several studies, as cited, that have focused on combined chemical-electric maneuvers for specific mission scenarios, little work has been done in determining which propulsion technologies are best suited for these missions since in most cases only a single technology was selected for study. These studies also showed that the chemical maneuver had a great effect on the corresponding optimal electric trajectory, which leads to the conclusion that the optimum multi-mode design will involve careful pairing of chemical and electric thruster technologies. Furthermore, it is unclear as to how to quantify 'mission flexibility', rendering it virtually impossible to quantitatively compare and select multi-mode propulsion systems under the mission scenarios envisioned in the flexible spacecraft architecture. This paper focuses on comparing and assessing multi-mode systems in both a defined mission scenario and a flexible mission scenario. The following sections focus on high-power electric propulsion technologies in the electric mode, although the resulting analysis techniques will likely also be applicable to a wider variety of technologies if the design goals are similar. Section II outlines the multi-mode propulsion technologies selected for this study. Section III presents the techniques used to analyze and compare these systems, and Section IV presents the results of these analyses. The results are discussed in Section V, and finally the main conclusions are summarized in Section VI.

II. Multi-Mode Propulsion Systems

Three high-power electric propulsion technologies are selected for this study: hall thruster, arcjet thruster, and pulsed inductive thruster (PIT). For the chemical mode, both monopropellant thrusters and bipropellant thrusters will be considered for each electric propulsion technology yielding a total of six systems. These are shown in Table 1. Each system is designated by two letters, referring to the first letter of the chemical mode and the first letter of the electric mode and will be referred to as such hereafter. Hydrazine is selected as the chemical propellant for all monopropellant systems, as it has been the monopropellant of choice for essentially all of spaceflight history.[9] For the same reason, the hypergolic bipropellant combination of monomethylhydrazine (MMH) fuel and nitrogen tetroxide (NTO) oxidizer is selected for all chemical bipropellant thrusters. The specific impulse and thrust of these propellants shown in the table represents a value typical of these thrusters.[9,10] The chemical thruster mass is determined by scaling relations that will be described in greater detail in the next section.

Multi-mode systems involving hall thruster technology have been studied or used as a baseline for comparison previously.[3,7,11,12] These systems utilize separate propellants for chemical and electric thruster modes. The reason for this is that hall thrusters cannot operate with typical chemical propellants due to either channel erosion or low efficiency issues. As a result, separate propellants are considered for Systems MH and BH, with xenon being chosen as the propellant for the hall thruster, again because it has extensive flight heritage.[10,13] The specific impulse is selected as 1800 seconds, which is again typical of current and future hall thruster technology. The thrust and thruster mass is based on 30 kW thruster power. Thrust is calculated from Eq. 1,

$$\eta_t P_{thr} = \frac{1}{2} F I_{sp} g_0 \tag{1}$$

and the thruster mass is determined from scaling relations for high power hall thrusters developed by Hofer and Randolph.[14]

A multi-mode monopropellant/arcjet has also been studied previously, and is considered a generic multi-mode concept.[1,2] The main benefit of this concept is that hydrazine propellant can be shared by both modes, as hydrazine decomposition products have been utilized in previous arcjet thrusters. The specific impulse and efficiency of the arcjet used in this study is taken from the flight proven Electric Propulsion Space Experiment (ESEX) 26 kW arcjet thruster, which uses ammonia as its propellant.[15,16] Since ammonia is the main decomposition product of hydrazine, it is assumed that the actual performance using hydrazine instead of pure ammonia will be similar. The thrust is calculated from Eq. 1 based on 30 kW thruster power and thruster mass is taken from the ESEX thruster.[15] The same values are used for the corresponding bipropellant system, System BA, but ammonia is used as a propellant since oxidizing exhaust species, such as those of MMH-NTO, tend to erode arcjet electrodes.[17]

PIT devices have the advantage of being able to run on virtually any gaseous neutral propellant since they are electrodeless.[18] Furthermore, ammonia propellant actually increases the efficiency of PIT devices due to lower radiation losses compared to xenon.[19] Therefore, common propellants are selected for Systems MP and BP. The specific impulse, efficiency, and thruster mass are determined from the FARAD thruster, which represents a current measuring stick for anticipated pulsed inductive thruster performance.[20,21] Previous research has shown that it may be beneficial in terms of reduced propulsion system mass to integrate both chemical and electric nozzles into a single geometry.[22] However, this will not be included in this study. The advantages and/or consequences of choosing common propellants will be assessed further in a later section.

Table 1. Multi-Mode Propulsion Systems Selected for Study.

System Designation	MH	MA	MP	BH	BA	BP
			Chemical Mode			
Type	Monopropellant	Monopropellant	Monopropellant	Bipropellant	Bipropellant	Bipropellant
Propellant	Hydrazine	Hydrazine	Hydrazine	MMH-NTO	MMH-NTO	MMH-NTO
I_{sp} (sec)	243	243	243	327	327	327
Thrust (N)	1000	1000	1000	10000	10000	10000
Thruster Mass (kg)	3.2	3.2	3.2	85.4	85.4	85.4
			Electric Mode			
Type	Hall	Arcjet	PIT	Hall	Arcjet	PIT
Propellant	Xenon	Hydrazine	Hydrazine	Xenon	Ammonia	MMH-NTO
I_{sp} (sec)	1800	787	2500	1800	787	2500
Efficiency	0.50	0.28	0.50	0.50	0.28	0.50
Thrust (N)	1.70	2.21	1.22	1.70	2.21	1.22
Thruster Mass (kg)	75.0	12.7	51.3	75.0	12.7	51.3

III. Multi-Mode Propulsion Systems Analysis Methods

Multi-mode propulsion systems enable two primary spacecraft mission benefits: more efficient planned trajectories and flexible mission scenarios. In either scenario, the primary goal of the propulsion system design is to accomplish the given objective with as little mass dedicated to the propulsion system as possible so as to maximize payload capacity or reduce cost. For multi-mode systems, analysis of spacecraft performance and mass is complicated by utilizing an additional propulsion system, since it opens a large design space. Additionally, multi-mode systems must also be assessed in terms of the effectiveness of integrating components, such as propellants, in terms of gains in mission capability or reduction of propulsion system mass. The following paragraphs describe the analysis used in this paper to assess and compare the systems defined previously.

A. Chemical Thruster Sizing

The two chemical propellants selected for study are hydrazine and MMH-NTO for monopropellant and bipropellant systems, respectively. Hydrazine typically decomposes to a chamber temperature of 1350 K, a specific heat ratio of 1.23, and a characteristic velocity of 1345 m/s. The MMH-NTO bipropellant combination combusts to a temperature of 3200 K, a specific heat ratio of 1.15, and a characteristic velocity of 1750 m/s. Given the combustion characteristics of the propellant, a chemical thruster at a desired thrust level can be sized by specifying three additional parameters: chamber pressure, nozzle expansion ratio, and divergence half-cone angle. This study will assume a 300 psi chamber pressure and a nozzle expansion ratio of 200, which are typical values for on-orbit thrusters.[10] The nozzle throat area is calculated from Eq. (2),

$$A_t = \frac{F}{C_F P_C}, \tag{2}$$

where the thrust coefficient is given by Eq. (3),

$$C_F = \lambda \sqrt{\frac{2\gamma}{\gamma-1}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} + \frac{P_e}{P_c}\varepsilon, \tag{3}$$

and the pressure ratio can be solved iteratively using Eq. (4),

$$\frac{1}{\varepsilon}=\left(\frac{\gamma+1}{2}\right)^{\frac{1}{\gamma-1}}\left(\frac{P_e}{P_c}\right)^{\frac{1}{\gamma}}\sqrt{\left(\frac{\gamma+1}{\gamma-1}\right)\left(1-\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}}\,. \tag{4}$$

where the divergence correction factor has been added, shown in Eq. (5),

$$\lambda=\frac{1}{2}\left(1+\cos(\alpha)\right), \tag{5}$$

and for all analysis herein a 15^o half cone divergence angle is used with a 20% reduction in length to estimate the mass of a bell nozzle.

Given the specified parameters, and calculations from Eqs. (2)-(5), the remaining geometry of the divergence section, namely exit area and length are calculated through simple trigonometric relations. The thrust chamber geometry can be calculated through empirical means by Eqs. (6) and (7),[10]

$$A_c=A_t\left(8D_t^{-0.6}+1.25\right) \tag{6}$$

$$L_c=L^*\frac{A_t}{A_c}, \tag{7}$$

where the characteristic length, L^*, historically falls between 0.5 and 2.5, with monopropellant thrusters having characteristic lengths at the high end of this range. Therefore, a characteristic length of 2.5 is chosen for monopropellant thrusters, and a value of 0.8 is chosen for bipropellant thrusters.. Since all of the geometric parameters of the thruster have been calculated, the mass can be estimated by the following equations. The wall thickness is estimated by Eq. (8),

$$t_w=\frac{P_b D_c}{2F_{tu}} \tag{8}$$

and the mass of the thrust chamber is subsequently calculated using Eq. (9),

$$m_c=\pi\rho_w t_w\left[2r_c L_c+\frac{r_c^{\,2}-r_t^{\,2}}{\tan\theta_c}\right]. \tag{9}$$

For the preliminary calculations, the burst pressure is assumed to be twice the chamber pressure and the material is assumed to be columbium ($F_{tu}=310\ MPa$, $\rho_w=8600\ kg/m^3$), a generic thrust chamber material. Additionally, the angle of the convergence section is assumed to be 45^o in all cases, recognizing that it typically comprises only a small percentage of the total thruster mass.

B. The Multi-Mode Rocket Equation

Spacecraft maneuvers are governed by the Tsiolkovsky rocket equation, shown in Eq. (10),

$$\frac{m_f}{m_0}=e^{-\frac{\Delta V}{I_{sp}g_0}} \tag{10}$$

Multi-mode systems utilize two separate thrusters with separate specific impulses. Thus, in order to determine the propellant required for a certain maneuver, the chemical and electric modes must be considered as two separate maneuvers in Eq. (10). If we define a parameter for the percentage of the total delta-V to be conducted by electric propulsion, *EP*, Eq. (11) we can write the two separate rocket equations, (12) and (13),

$$EP=\frac{\Delta V_{elec}}{\Delta V} \tag{11}$$

$$\frac{m_{f1}}{m_0}=e^{-\frac{(1-EP)\Delta V}{I_{sp,chem}g_0}} \tag{12}$$

$$\frac{m_f}{m_{f1}}=e^{-\frac{EP\Delta V}{I_{sp,elec}g_0}} \tag{13}$$

where it is assumed that the chemical burn is conducted first. Multiplying Eqs. (12) and (13) and simplifying yields Eq. (14),

$$\frac{m_f}{m_0} = e^{-\frac{\Delta V}{g_0}\left[\frac{1-EP}{I_{sp,chem}}+\frac{EP}{I_{sp,elec}}\right]} \tag{14}$$

and it can then be easily seen that an effective specific impulse can be defined, which is a function of the chemical and electric mode specific impulse as well as the *EP* usage fraction. The multi-mode specific impulse is then Eq. (15),

$$I_{sp,mm} = \left[\frac{1-EP}{I_{sp,chem}}+\frac{EP}{I_{sp,elec}}\right]^{-1} \tag{15}$$

It is also notable that this equation turns out to be exactly the same regardless of the order or number of chemical or electric thrust maneuvers. Finally, dividing Eqs. (12) and (13) gives an equation for the ratio of chemical propellant to electric propellant as a function of the chemical and electric mode specific impulses and EP usage fraction, Eq. (16)

$$\frac{m_{elec}}{m_{chem}} = \frac{1-e^{-\frac{\Delta V}{I_{sp,mm}g_0}}}{1-e^{-\frac{(1-EP)\Delta V}{I_{sp,chem}g_0}}} - 1 \tag{16}$$

C. Multi-Mode Propulsion System Mass Estimation

The majority of the propulsion system sizing conducted in this study is based on empirical baseline design estimates outlined in Humble.[10] The mass of propellant required to accelerate a spacecraft through a desired velocity change can be calculated from a rearranged form of the rocket equation, Eq. (17),

$$m_{prop} = \frac{m_{pay}\left(\exp\left(\frac{\Delta V}{I_{sp}g_0}\right)-1\right)(1-f_{inert})}{1-f_{inert}\exp\left(\frac{\Delta V}{I_{sp}g_0}\right)} \tag{17}$$

where the inert mass fraction is given by Eq. (18),

$$f_{inert} = \frac{m_{inert}}{m_{prop}+m_{inert}} \tag{18}$$

and the inert mass is composed of the thruster, propellant feed lines and valves, propellant and pressurant tanks, power processing unit (PPU), and structural mounts for the propulsion system. The mass of the tanks can be estimated empirically by Eq. (19),

$$m_{\tan k} = \frac{P_b m_{prop}\rho_{prop}}{g_0\phi_{\tan k}} \tag{19}$$

where the burst pressure is again assumed to be 1.25 times the tank pressure. For hydrazine, MMH-NTO, and ammonia propellant tanks the tank pressure is chosen to be 300 psi plus a 20% injector head loss and 0.35 psi overall line losses for the propellant tanks. For xenon propellant the tank pressure is chosen to be 1100 psi and 1450 psi is chosen for helium pressurant tanks. The density of these propellants at the chosen conditions is shown in Table 2. Also, the empirical tank sizing parameter is chosen to be 2500 m for the hydrazine and MMH-NTO tanks, and 6350 m for the helium, ammonia and xenon tanks. These values correspond to typical stainless steel (compatible with hydrazine, monomethylhydrazine, and nitrogen tetroxide[23]) and titanium tank material, respectively. Since the volume of the pressurant tank is not known beforehand, the pressurant required must be solved iteratively until the mass of pressurant is sufficient to occupy both pressurant and propellant tanks at the desired propellant tank pressure. The mass of lines and valves is estimated as 50% of the thruster mass, a value typical of spacecraft thrusters historically. Finally, the mass of structural mounts is assumed to be 10% of the total inert mass. Eq. (17) is then solved iteratively for the propellant mass.

Table 2. Storage Properties of Propellants.

Propellant	Pressure (psi)	State	Density (kg/m^3)
Hydrazine	300	Liquid	1005
MMH	300	Liquid	878
NTO	300	Liquid	1440
Xenon	1100	Supercritical	1642
Ammonia	300	Liquid	561

In terms of the electric mode of propulsion, the mass of the power processing unit (PPU), associated cables and switches, as well as the powertrain components of the electric thruster itself will have a substantial effect on the overall propulsion system mass. Hofer and Randolph have developed empirical relations to estimate the mass of the PPU, associated cables, as well as the solar array for high-power electric propulsion technologies.[14] These estimates are given in Eqs. (20)-(22),

$$m_{ppu} = 1.7419 P_{thr} + 4.654 \tag{20}$$

$$m_{cables} = 0.06778 P_{thr} + 0.7301 \tag{21}$$

$$m_{sa} = 3 P_{thr} \tag{22}$$

IV. Results

The main results of the analysis methods described previously are presented in this section. The multi-mode specific impulse equation is first analyzed for the systems described in Section II. The effect on system mass of utilizing common propellants for the electric mode is then quantified. The majority of the analysis is then conducted for a payload of 500 kg and a total velocity change of 1500 m/s. These represent values typical of LEO to GEO orbit transfers for communications satellites.

A plot of the multi-mode specific impulse as defined in Eq. (15) is shown as a function of the fraction of delta-V conducted by the electric thruster mode in Fig. 1. The behavior of the function is nonlinear, with most of the effective specific impulse increase occurring at large EP fraction. For example, System MP doubles effective specific impulse moving from an EP fraction of zero to 0.6, but then increases by a factor of four thereafter. All monopropellant systems perform lower than the corresponding bipropellant systems for low EP fraction, but at an EP fraction of 0.6 both Systems MP and MH perform higher than System BA, and at an EP fraction of 0.85, System MP outperforms System BH.

Three systems were chosen to have shared, or common, propellants for both chemical and electric modes. These are systems MA, MP, and BP. In order to quantify the effect of this choice, the payload mass fraction is computed for systems involving the same thruster modes, but with separate propellants. In the case of System MA, the separate electric propellant is ammonia, and for Systems MP and BP, the separate electric propellant is xenon. The payload mass fraction for these systems with a 500 kg payload and a delta-V requirement of 1500 m/s is shown in Table 2. Only the maximum difference computed is shown, which occurs at an EP fraction of one in all cases From the table, it is obvious that utilizing a common propellant produces essentially a net zero benefit for all three systems. This will be discussed further in the next section.

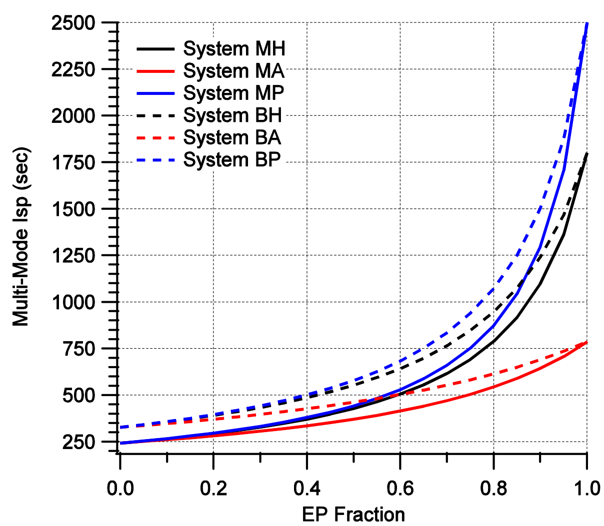

Fig 1. Multi-mode Specific Impulse for Selected Systems as a Function of EP Fraction.

Table 2. Payload Mass Fractions for Systems Involving Common Propellants.

System	Payload Fraction		%Difference
	Separate	Common	
MA	0.572	0.577	0.874
MP	0.656	0.650	-0.915
BP	0.551	0.547	-0.726

The total propulsion system mass and burn time as a function of EP usage fraction required to raise a 500 kg payload through a 1500 m/s delta-V is shown in Figs. 2 and 3, respectively. The propulsion system mass shown in Fig. 2 includes both the inert mass of the propulsion system as well as the total propellant. It is seen that despite a 25 times higher chemical thruster mass, each bipropellant system has lower total mass than the corresponding monopropellant systems for EP fractions less than 0.3. However, beyond this the monopropellant systems have lower total mass, with System MP having the lowest total mass of any system beyond an EP usage fraction of 0.3. At most, System MP has 8% and 33 % less mass than Systems MH and MA, respectively, and is half as massive as System BA. The Hall thruster systems are less massive than the corresponding arcjet systems at an EP fraction of roughly 0.6 and above. From Fig. 3 it is seen that the monopropellant systems have shorter total burn times for virtually every EP fraction compared to the bipropellant systems. In fact, examining the raw data indicates that at only EP fractions below 0.01 do the bipropellant systems have a shorter burn time than the monopropellant systems, despite having ten times the thrust. The burn times essentially follow the electric thrust over all EP fractions, with the arcjet having the shortest durations, and the PIT technology the longest.

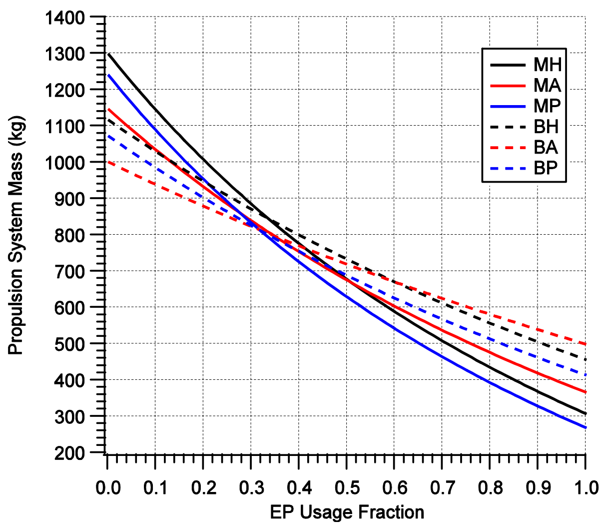

Fig 2. Total Propulsion System Mass as a Function of EP Fraction.

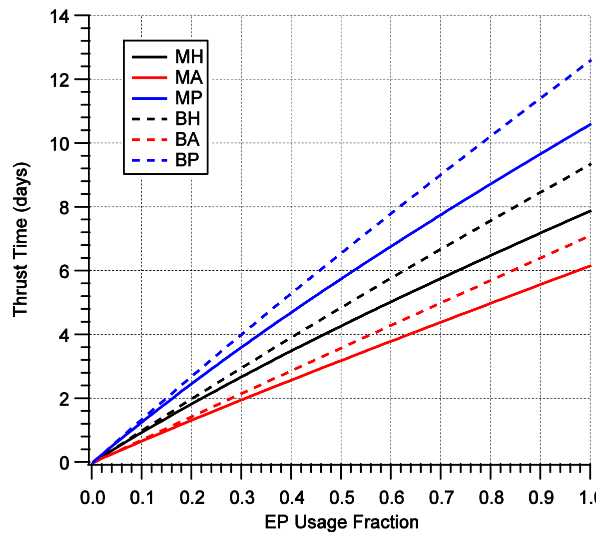

Fig 3. Total Burn Duration as a Function of EP Fraction.

Additional conclusions can be drawn by considering the so-called transportation rate. For purposes of this study, we define this as the extra payload mass capacity gained by using an electric burn compared to an all-chemical burn divided by the increase in burn time, Eq. (23),

$$T_R = \frac{m_{pay,mm} - m_{pay,allchem}}{t_{b,mm} - t_{b,allchem}} \tag{23}$$

An all-chemical burn here does not simply refer to an EP fraction of zero, but a burn as if the spacecraft did not contain any EP components. Any power systems or station keeping thrusters are then considered as part of the payload. The transportation rate for a delta-V of 1500 m/s and a total spacecraft mass of 2000 kg is shown as a function of EP fraction in Fig. 4. First, and perhaps foremost, it is seen that the transportation rate does not pass zero until at least 25% of the total delta-V is dedicated to electric propulsion. Or more succinctly, there is no benefit to adding a high power electric propulsion system unless at least 25% of the total velocity increment is used for electric propulsion. All three monopropellant systems reach positive transportation rate at lower EP fractions than the bipropellant systems. Initially, System MP has the highest transportation rate, but only for a window from 0.25 to 0.30 EP fraction. System MA has the highest transportation rate of all Systems beyond 0.30 EP fraction, with System MH roughly 10% lower.

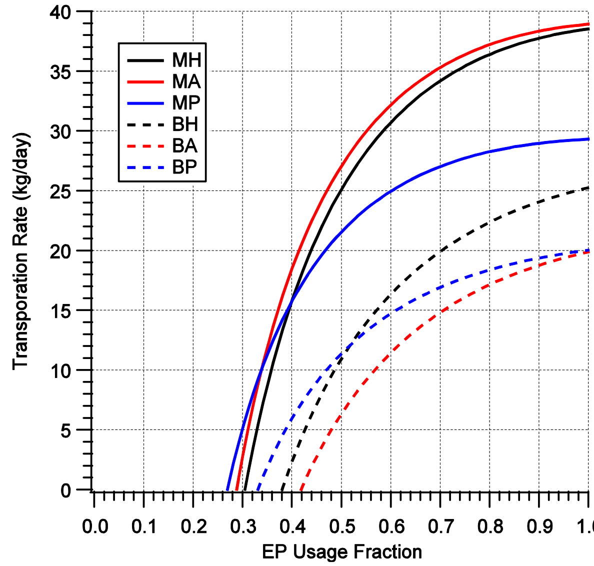

Fig 4. Transporation Rate as a Function of EP Fraction.

V. Discussion

The results presented in the previous section provide clear comparisons of each multi-mode system defined in Section II. However, in order to fully interpret the results and draw relevant conclusions, these must be placed in context of the two primary reasons in which a multi-mode system is beneficial. Again, these are either to enable a more optimal trajectory than can be accomplished by either chemical or electric propulsion alone or to enable flexible mission operations.

A consideration made by most prior multi-mode propulsion studies is whether integrating components of the chemical and electrical propulsion modes can reduce the mass of the spacecraft. For this reason, the use of shared propellants in Systems MP, MA, and BP in this study was examined. Results showed that utilizing a shared propellant provided essentially net zero benefit, and in fact utilizing a common tank with only hydrazine instead of separate tanks with xenon and hydrazine for each separate propulsion mode was actually slightly detrimental in terms of propulsion system mass. This is mainly a consequence of propellant density. Xenon is 60% more dense than hydrazine under typical spacecraft storage conditions, but the tanks must be able to withstand 3.5 times the pressure. The net effect essentially canceled for the conditions presented in this study. More dense multi-mode-specific propellants would be extremely beneficial. Replacements for monopropellant hydrazine are currently being investigated, and these are roughly 20% to 40% more dense than hydrazine.[8,24,25] However, most of these are ionic liquid based and combust to yield oxidizing product species, which as mentioned tend to destroy arcjet electrodes. The PIT thruster, however, could theoretically handle these propellants and would thus provide increased mass benefits over a system utilizing xenon as a separate electric propellant. For flexible mission scenarios, however, shared propellants are essentially a requirement. If separate propellants are budgeted, then effectively the spacecraft will only reach its peak efficiency at a single EP fraction. At all others, it by necessity leaves some unutilized propellant, as discussed in a previous study.[7] Therefore, to achieve inherent flexibility the propellant must be compatible with both propulsion systems.

Computing total propulsion system mass as a function of EP fraction showed that bipropellant systems perform higher at lower EP fractions despite a higher inert mass fraction due to the large size of the 10000 N chemical thruster. However, considering the transportation rate computed and shown in Fig. 4, the multi-mode system is effectively worse than an all-chemical system at low EP fractions. Furthermore, Fig. 3 clearly indicates that there is no advantage in terms of burn duration since in all cases the bipropellant systems had a longer burn duration than the corresponding monopropellant systems at all EP fractions except those below 0.01. Low inert mass on the chemical thruster then is much more important from a multi-mode perspective than both chemical specific impulse and thrust. This may be entirely a high-power electric mode consideration however. The fact that utilizing a multi-mode system below an EP fraction of 0.25 is actually worse than an all-chemical system is due entirely to the massive power source and EP thruster. Since the effective specific impulse increases exponentially, from Fig. 1, the electric mode specific impulse increasingly dominates the propellant requirements as EP fraction increases. The chemical hardware then becomes a larger fraction of the total system mass as EP fraction increases, further taxing propellant requirements while only serving in a more limited capacity. Therefore, while a high specific impulse in the chemical mode is beneficial, for multi-mode systems utilizing a high-power electric propulsion system, smaller chemical thrusters are desirable even at the expense of specific impulse.

The transportation rate, from Fig. 4, essentially serves to combine the effects of decreasing propulsion system mass, but also increasing burn duration in doing so. The system with the most benefit in terms of transportation rate was the monopropellant/arcjet system. Despite having the lowest specific impulse in both modes, the benefit of having the lowest inert mass and highest electric thrust outweighed the extra propellant requirements. This system is therefore most beneficial when time is an important factor. Under a set propellant mass, such as would be designed for in a flexible mission scenario, the MP system would be the most flexible since it would provide low inert mass whilst providing the highest range of delta-V due to the high electric specific impulse.

VI. Conclusions

Multi-mode spacecraft propulsion systems involving separate chemical and electric thrusters were compared and analyzed in terms of mission capability. Hydrazine or MMH-NTO was considered for monopropellant and bipropellant chemical modes, respectively. These propellants enabled propellant sharing with the electric mode for systems with an arcjet thruster or PIT thruster. However, there was a net zero mass benefit for propellant sharing compared to utilizing separate ammonia (arcjet) or xenon (PIT) propellants. Analyzing the propulsion system mass and burn duration for multi-mode systems tasked with providing a delta-V of 1500 m/s with a 500 kg payload showed that utilizing a bipropellant thruster is ineffective by any metric. The lower inert mass of the monopropellant thruster was more beneficial despite lower specific impulse and thrust. This is mainly due to the large power supply requirements of the high-power electric thruster effectively negating any benefit to utilizing a multi-mode system where the electric mode use is lower than a quarter of the total delta-V. Finally, a system consisting of a chemical monopropellant thruster and electric arcjet thruster showed the most benefit in terms of transportation rate despite having the lowest specific impulse in both modes. This indicates that reduction of inert mass is a more important consideration in these types of multi-mode systems than reducing propellant mass through increasing specific impulse.

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