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Assessment of Multi-Mode Spacecraft Micropropulsion Systems

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Multi-mode spacecraft micropropulsion systems which include a high-thrust chemical mode and high-specific impulse electric mode are assessed with specific reference to cubesat-sized satellite applications. Both cold gas Freon-14 propellant and ionic liquid chemical monopropellant modes were investigated alongside pulsed plasma, electrospray, and helicon electric thruster modes. Systems involving chemical monopropellants have the highest payload mass fractions for a reference mission of a 500 m/s delta-V and 6U sized cubesat for electric propulsion usage below 55% of total delta-V. For higher electric propulsion usage, cold gas thrusters delivered a higher payload mass fraction due to lower system inert mass. Due to the combination of utilizing a common propellant for both propulsive modes, low inert mass, and high electric thrust, the cold-gas chemical/helicon-type electric combination had the highest mission flexibility, able to achieve a delta-V 10% lower than that of the largest delta-V system, but at roughly 500 days less burn time. A System utilizing a monopropellant thruster and electrospray thruster can achieve the largest delta-V, but with a burn time of over 600 days. This same system, however, can achieve the largest delta-V for missions requiring a thrust time of less than roughly 10 days.

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_{chem}	=	mass of chemical propellant, [kg]
m_{elec}	=	mass of electric propellant, [kg]
m_f	=	final mass of spacecraft, [kg]
m_{f1}	=	mass of spacecraft after first burn, [kg]

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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_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 use of two or more propulsive devices on a spacecraft, specifically making use of a high-thrust, usually chemical, mode and a high-specific impulse, usually electric mode. This can be beneficial in two primary ways. The first is to increase the mission flexibility of a single spacecraft architechture in that both high-thrust and high-specific impulse maneuvers are available to mission designers at will, perhaps even allowing for drastic changes in the mission plan while on-orbit or with a relatively short turnaround from concept to launch. The second way a multi-mode propulsion system can be beneficial is by designing a mission such that the high-thrust and high-specific impulse maneuvers are conducted in such a way that it provides a more optimum trajectory over a single chemical or single electric maneuver. This study will use methods developed in a previous analysis of high-power electric multi-mode systems,[1] extending them to multi-mode micropropulsion systems.

One of the main drivers for research into multi-mode spacecraft propulsion is the potential for flexible spacecraft.[2, 3] 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, or simply reducing the length of time from development to launch. 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 separate high-thrust and high-specific impulse propulsion systems even if there is no common hardware or propellant.[4-6] For example, use of a 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.[7] However, it has been shown that even greater mass savings can potentially be realized through the use of shared propellants or shared hardware.[8, 9] The use of shared propellants is essential in order to realize the full potential of the multi-mode system under the flexible mission scenario since utilizing separate propellants for each mode fixes the possible delta-V from each mode, whereas there is a wider range of possible delta-V if propellants are shared. The only possible deviations under the separate propellants architechture inherently lead to underutilization of propellant.[1]

Recent efforts have placed a greater emphasis on smaller spacecraft, specifically microsatellites (10-100 kg) and nanosatellites (1-10 kg), including the subset of cubesats.[10] Many different types of thrusters have been proposed to meet the stringent mass and volume requirements placed on spacecraft of this type. A few multi-mode systems have been proposed as well. One includes the use of an ionic liquid propellant for chemical combustion or decomposition as well as for electrospray.[8, 9, 11] A specific propellant for this purpose is even under development.[12] This study will examine this type of system, as well as others, specifically to compare these systems in reference to their multi-mode performance in reference to both mission-defined and flexible-mission scenarios. Analyses will focus on cubesat-type architechture since required system mass data are readily available, but there is no specific reason why these analyses could not also apply to larger satellites. Section II will introduce the systems to be examined in this study. Section III will describe the analysis methods and assumptions made in reference to developing the multi-mode system comparisons. Section IV will present the results of analysis. Section V will discuss the results and Section VI presents the relevant conclusions from all analyses.

II. Multi-Mode Propulsion Systems

Two chemical thrust modes and three electric thrust modes are selected for this study. The chemical thrusters include cold gas with Freon-14 as propellant[10,13] and monopropellant with either AF315E or the [Emim][EtSO$_4$]/HAN dual-mode propellant.[9, 12] The three electric thrusters are the pulsed plasma thruster (PPT), the electrospray thruster, and the helicon thruster. Combining these yields six multi-mode systems, shown in Table 1. Teflon is chosen as the electric propellant for the systems involving PPT thrusters. Although gas-fed PPTs have been investigated, the solid propellant PPTs are more compact and less massive, providing the most extreme comparison in reference to the other systems, especially cubesats where volume is a greater concern. For the flexible-mission scenario, however, the propellant will be changed to nitrogen in the case of System CP since, from previous insights, the flexible-mission designed system is most applicable when a single propellant is used for both modes. System CE will be retained for the mission-defined analysis, but will not be included in the flexible-mission analysis since cold gas and electrospray are not compatible with the same propellant.[11] Finally, the helicon thruster is an electrodeless device, meaning there is no fundamental reason why any gaseous propellant could not be used. In the case of System MH, this will require an additional gas generator to decompose the AF315E prior to injection into the electric thruster. This will be accounted for in the mass and volume analyses. It should also be noted that the helicon-type thruster is not yet a flight-proven type of thruster, so it's performance is still somewhat uncertain. However, it is included in this study to discern what effect an extremely high-performance electric thruster that can make use of chemical propellants will have on a multi-mode propulsion system design.

Thrust and specific impulse values for nitrogen cold gas systems are based on typical values from flight heritage thrusters.[10, 14] The performance of the AF315E monopropellant thrusters is based on a commercially available design from Busek, Inc.[15] The performance of the [Emim][EtSO$_4$]/HAN propellant is scaled from the theoretical specific impulse of 251 seconds from CEA computations to match the same reduction in performance between the theoretical specific impulse of AF315E[16] and the Busek thruster. The PPT selected is a commercially available thruster from Clyde Space, Inc.[17] The [Emim][Im] performance values are taken from a commercially available thruster from Busek, Inc.[18] The performance of the [Emim][EtSO$_4$]\HAN blend in the electrospray device is scaled in a similar manner as described for the chemical monopropellant performance. Values for the helicon type thruster are taken from the Cubesat Ambipolar Thruster (CAT) and are theoretical only.[19]

Table 1. Performance of Multi-Mode Propulsion Systems.

System Designation	CP	CE	CH	MP	ME	MH
			Chemical Mode			
Type	Cold Gas	Cold Gas	Cold Gas	Monopropellant	Monopropellant	Monopropellant
Propellant	Freon-14	Freon-14	Freon-14	AF315E	[Emim][EtSO$_4$]/HAN	AF315E
I_{sp} (sec)	45	45	45	230	226	230
Thrust (N)	0.1	0.1	0.1	0.5	0.5	0.5
			Electric Mode			
Type	PPT	Electrospray	Helicon	PPT	Electrospray	Helicon
Propellant	Teflon	[Emim][Im]	Freon-14	Teflon	[Emim][EtSO4]/HAN	AF315E
I_{sp} (sec)	600	800	2000	600	1280	2000
Thrust (mN)	0.14	0.7	2	0.14	0.43	2

III. Multi-Mode Propulsion Systems Analysis Methods and Subsystem Sizing

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. And, since this enables flexible mission design scenarios where perhaps there is a loosely defined mission that includes as yet undetermined requirements, comparing multi-mode systems for use in such a scenario becomes difficult. Finally, 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. The Multi-Mode Rocket Equation

Spacecraft maneuvers are governed by the Tsiolkovsky rocket equation, shown in Eq. (1),

$$\frac{m_f}{m_0} = e^{-\frac{\Delta V}{I_{sp} g_0}} \tag{1}$$

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. (1). If we define a parameter for the percentage of the total delta-V to be conducted by electric propulsion, *EP*, Eq. (2) we can write the two separate rocket equations, (3) and (4),

$$EP = \frac{\Delta V_{elec}}{\Delta V} \tag{2}$$

$$\frac{m_{f1}}{m_0} = e^{-\frac{(1-EP)\Delta V}{I_{sp,chem} g_0}} \tag{3}$$

$$\frac{m_f}{m_{f1}} = e^{-\frac{EP \Delta V}{I_{sp,elec} g_0}} \tag{4}$$

where it is assumed that the chemical burn is conducted first. Multiplying Eqs. (3) and (4) and simplifying yields Eq. (14),

$$\frac{m_f}{m_0} = e^{-\frac{\Delta V}{g_0}\left[\frac{1-EP}{I_{sp,chem}} + \frac{EP}{I_{sp,elec}}\right]} \tag{5}$$

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. (6),

$$I_{sp,mm} = \left[\frac{1-EP}{I_{sp,chem}} + \frac{EP}{I_{sp,elec}}\right]^{-1} \tag{6}$$

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. (3) and (4) 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. (7)

$$\frac{m_{elec}}{m_{chem}} = \frac{1-e^{-\frac{\Delta V}{I_{sp,mm} g_0}}}{1-e^{-\frac{(1-EP)\Delta V}{I_{sp,chem} g_0}}} - 1 \tag{7}$$

B. Chemical Thruster Sizing

The two chemical propellants selected for study are Freon-14 for cold gas and either [Emim][$EtSO_4$] or AF315E for monopropellant systems as defined in Table 1. For chemical propellants, relevant parameters for thruster sizing include chamber temperature and specific heat ratio. The [Emim][$EtSO_4$] propellant combusts to a temperature of 1900 K, a specific heat ratio of 1.22, and a characteristic velocity of 1330 m/s.[9] AF315E combusts to a chamber temperature of 2300K.[16] The exact composition of AF315E is not given in the literature, so a specific heat ratio of 1.2 is chosen based on typical values for combustion products of HAN-based ionic liquid propellants.[9, 20] 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.[14] The nozzle throat area is calculated from Eq. (8),

$$A_t = \frac{F}{C_F P_C}, \tag{8}$$

where the thrust coefficient is given by Eq. (9),

$$C_F = \lambda \sqrt{\frac{2\gamma}{\gamma-1}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}\right]} + \frac{P_e}{P_c}\varepsilon, \tag{9}$$

and the pressure ratio can be solved iteratively using Eq. (10),

$$\frac{1}{\varepsilon} = \left(\frac{\gamma+1}{2}\right)^{\frac{1}{\gamma-1}}\left(\frac{P_e}{P_c}\right)^{\frac{1}{\gamma}}\sqrt{\left(\frac{\gamma+1}{\gamma-1}\right)\left(1-\frac{P_e}{P_c}\right)^{\frac{\gamma-1}{\gamma}}}. \tag{10}$$

where the divergence correction factor has been added, shown in Eq. (11),

$$\lambda = \frac{1}{2}\left(1+\cos(\alpha)\right), \tag{11}$$

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. (8)-(11), 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. (12) and (13),[14]

$$A_c = A_t\left(8D_t^{-0.6}+1.25\right) \tag{12}$$

$$L_c = L^* \frac{A_t}{A_c}, \tag{13}$$

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 0.5 is chosen for cold gas thrusters since they only require essentially a convergent nozzle section and tubing thick enough to withstand the chamber pressure. 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. (14),

$$t_w = \frac{P_b D_c}{2F_{tu}} \tag{14}$$

and the mass of the thrust chamber is subsequently calculated using Eq. (15),

$$m_c = \pi \rho_w t_w \left[2r_c L_c + \frac{r_c^2 - r_t^2}{\tan\theta_c}\right]. \tag{15}$$

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*, ρ_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.

C. Multi-Mode Propulsion System Mass Estimation

1. Propellant Tankage

The majority of the propulsion system sizing conducted in this study is based on empirical baseline design estimates outlined in Humble.[14] 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. (16),

$$m_{prop} = \frac{m_{pay}\left(\exp\left(\frac{\Delta V}{I_{sp}g_0}\right)-1\right)(1-f_{inert})}{1-f_{inert}\exp\left(\frac{\Delta V}{I_{sp}g_0}\right)} \tag{16}$$

where the inert mass fraction is given by Eq. (17),

$$f_{inert} = \frac{m_{inert}}{m_{prop}+m_{inert}} \tag{17}$$

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. (18),

$$m_{\tan k} = \frac{P_b m_{prop}\rho_{prop}}{g_0\phi_{\tan k}} \tag{18}$$

where the burst pressure is again assumed to be 1.25 times the tank pressure. For [Emim][Im], [Emim][$EtSO_4$]-HAN, and AF315E 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 and 1450 psi is chosen for the helium pressurant tanks. The density of these propellants at the chosen conditions and teflon is shown in Table 2. Also, the empirical tank sizing parameter is chosen to be 2500 m for the AF315E , [Emim][Im], and [Emim][$EtSO_4$]-HAN tanks, and 6350 m for the helium and nitrogen tanks. These values correspond to typical stainless steel (compatible with HAN-based propellants[21]) 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. (16) is then solved iteratively for the propellant mass.

Table 2. Storage Properties of Propellants.

Propellant	Pressure (psi)	State	Density (kg/m^3)
Freon-14	300	Liquid	1603
[Emim][$EtSO_4$]-HAN	300	Liquid	1419
AF315E	300	Liquid	1460
[Emim][Im]	-	Liquid	1519
Teflon	-	Solid	2200

2. Power Processing Systems

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. Mass and volume of the power processing unit and cables are taken from the commercially available PPUs manufactured by Clyde Space, Inc.[22] These are shown in Table 3. For the solar panels, a constant value of 15.5 g/W is used. This is a typical value for current state of the art solar cell technology.[10]

Table 3. Mass and Volume of Cubesat PPUs.

Power (W)	Volume (U)	Mass (g)
9	0.127	83
12	0.127	85
15	0.127	87
27	0.153	129
39	0.153	133
42	0.153	137
72	0.153	139

IV. Results

The results of the analysis methods and system sizing estimates are presented in this section. First, the thruster chemical and electric thruster masses are computed using the equations described in Section III. B. Then, performance is computed for each multi-mode thruster, and the mass and volume of each multi-mode propulsion system is computed in order to draw comparisons between each system.

A. Thruster Mass and Volume

Thruster mass and volume of the chemical thrusters was computed using the equations described in Section III. B. The results are shown in Table 4. Additionally, mass and volume of the PPU is also shown since it depends only on thruster power. The PPT thruster used in Systems CP and MP requires 2 W of power.[17] The electrospray thruster requires 9 W,[18] and the helicon thruster requires 100 W.[19] Additionally, the monopropellant thrusters require 20 W to preheat the catalyst bed to high enough temperature to initiate decomposition of the propellant.[15] The mass and volume of the PPU and solar panel mass for each system are sized according to these values. Solar panel volume is not included since it is typically located on the outside of the cube or deployed, and therefore does not affect propulsion system volume in the same manner as that of the rest of the components.

Table 4. Mass and Volume of Thrusters and Associated Power Units.

System Designation	CP	CE	CH	MP	ME	MH
Chemical Thruster Mass (g)	200	200	200	500	500	500
Electric Thruster Mass (g)	190	900	500	190	900	500
PPU Mass (g)	50	83	200	105	130	240
Solar Array Mass (g)	31	139	1550	341	449	1860
Total Mass (g)	471	1322	2450	1136	1979	3100
Chemical Thruster Volume (U)	0.25	0.25	0.25	0.5	0.5	0.5
Electric Thruster Volume (U)	0.21	0.5	0.5	0.21	0.5	0.5
PPU Volume (U)	0.105	0.127	0.183	0.136	0.153	0.202
Total Volume (U)	0.565	0.877	0.933	0.846	1.153	1.202

B. Multi-Mode Propulsion System Performance

The multi-mode specific impulse for each system as defined by Eq. (15) was computed and is shown in Fig. 1 as a function of EP usage fraction. Obviously, the bounds of the multi-mode specific impulse are the specific impulses of the chemical and electric thrusters chosen for the system. However, as seen in Fig. 1, the behavior of the function between these bounds is nonlinear. Furthermore, it is seen that most of the benefit of the high-specific impulse electric thruster is utilized at EP fractions close to unity. For example, system MH doubles in multi-mode specific impulse from EP usage fraction of 0 to 0.55, then increases by a factor of 4 from 0.55 to 1.0. All systems utilizing Freon-14 cold-gas thrusters perform lower than systems utilizing a monopropellant thruster for EP fractions lower than 0.85, where the performance of System CH overtakes that of system MP.

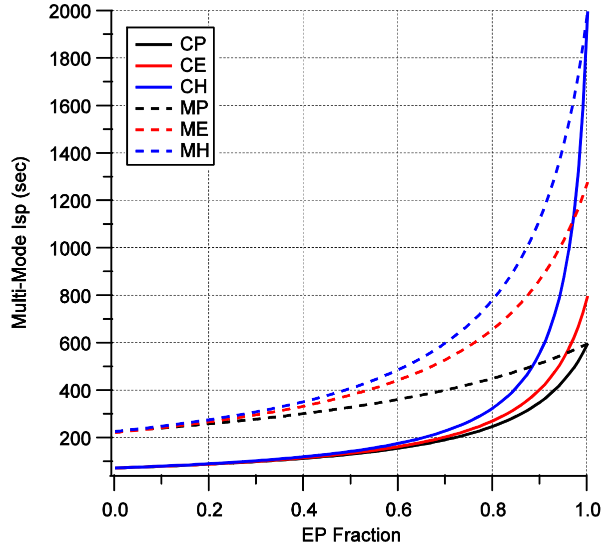

Fig. 1. Multi-Mode Specific Impulse.

C. Multi-Mode System Sizing

As mentioned in the introduction, there are two main approaches to preliminary design and selection for multi-mode propulsion systems. The first approach is more traditional in that maneuvers are planned at an early design stage. The propulsion system is then tailored to that set of maneuvers. This is especially true for electric propulsion systems, since the continuous thrust maneuver could be more or less efficient depending on the start and stop points on the trajectory. For a multi-mode system, this is even more complex because conducting an impulsive maneuver via a high-thrust chemical burn could effectively instantaneously change the efficiency of the next planned electric maneuver, as previous research has shown.[6, 23] Thus, simply defining a reference delta-V and payload mass and sizing the propulsion system may not tell the entire story, as other mission needs could dictate propulsion system choice. However, by loosely defining a reference mission one can eliminate obviously poor candidate systems, as well as gain an understanding of the strengths and potential weeknesses of the multi-mode system prior to fully defining the mission scenario. Additionally, as will be discussed, this can provide insight into the second approach to multi-mode system design, which is where a mission is not defined prior to spacecraft design maturation or even launch itself.

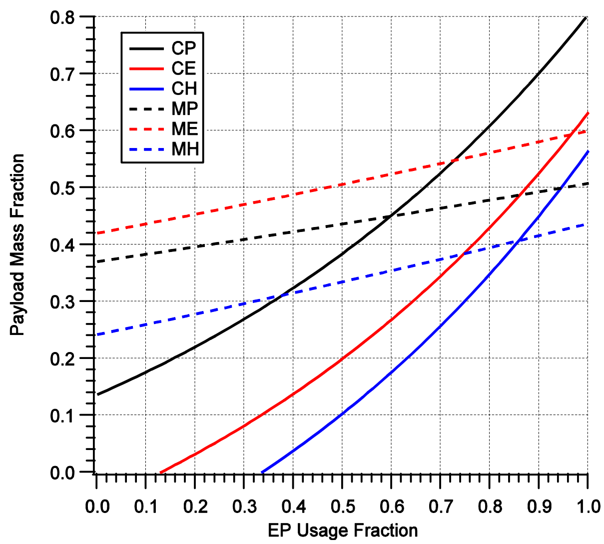

Fig. 2. Payload Fraction as a Function of EP Usage Fraction.

For a design reference mission, a delta-V of 500 m/s and a total satellite mass of 6.9 kg is chosen. The latter corresponds to a 6U cubesat. The payload fraction for each system defined in Table 1 is shown in Fig. 2. Clearly, systems involving the cold gas thruster have a clear disadvantage compared to their corresponding monopropellant systems for low electric propellant fractions. Only the cold gas system also utilizing a PPT is able to complete a 500 m/s delta-V without the required propellant pushing the satellite over the limit of 6.9 kg. System CH is unable to complete the defined mission unless at least 32% of the total delta-V is dedicated to electric propulsion. For EP usage below 70%, System ME is able to complete the mission with the highest payload fraction of all systems, while System CP has the highest payload fraction for missions allowing more than 70% of the delta-V to be accomplished via an electric thrust maneuver.

Volume is also a concern in the design of micropropulsion systems, particularly cubesats, since they have stringent volume requirements for inclusion as a secondary payload in launch systems. Fig. 3 shows the volume of the propulsion system for the same reference mission as the previous paragraph. The cold gas systems require much more volume than the monopropellant systems. At very high EP fractions (> 0.95) the cold gas propulsion systems actually require slightly less overall volume than the monopropellant systems. It should be noted, however, that if one considers the density of the remaining payload and volume, (i.e. the payload mass divided by the volume unoccupied by the propulsion system) that this is under the general cubesat requirement of 1.15 kg/U, meaning volume is not a limiting factor.

An additional important consideration in electric propulsion systems, and thus multi-mode propulsion systems also, is the required time to expel all propellant carried onboard the spacecraft. This can serve as a comparison for how long the mission will take with a given propulsion system. However, this may not describe the entire scenario as the length between burns is not defined. Furthermore, the selection of burn type and duration could play a significant role such that the time of unpowered flight is significantly longer than the burn duration in one case, but not in another. So, while simply comparing burn duration required of a propulsion system does not come close to describing the actual mission scenario, it does at minimum serve as a lower bound. The burn duration for the reference mission described previously is shown in Fig. 4. For all three electric propulsion systems, the burn duration is longer when using a monopropellant thruster compared to a cold gas thruster. Systems using a PPT in the electric mode require the longest burn durations, while systems involving the helicon-type thruster have the lowest overall burn times, requiring only about 10% of the total time required to perform the 500 m/s delta-V compared to the PPT systems.

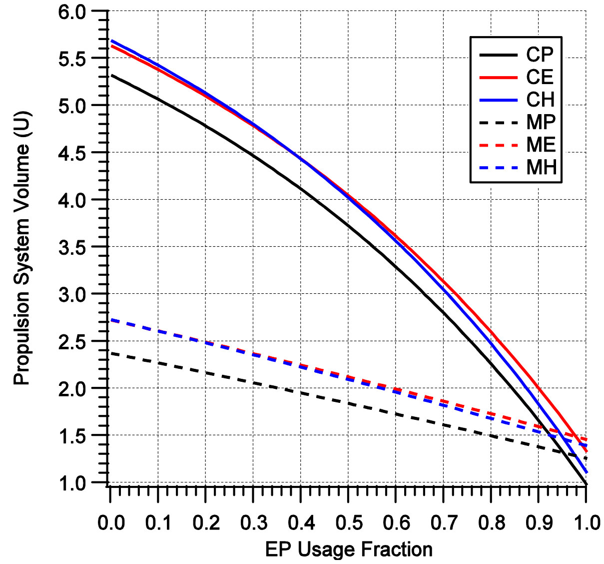

Fig. 3. Propulsion System Volume as a Function of EP Usage Fraction.

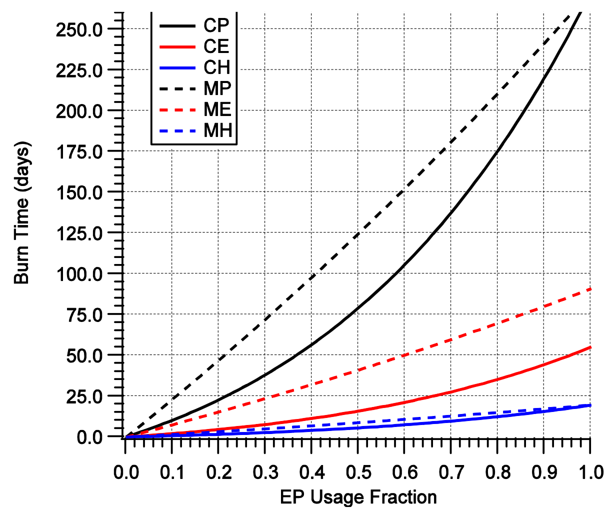

Fig. 4. Burn Duration as a Function of EP Usage Fraction.

As mentioned, one of the main drivers toward multi-mode propulsion usage is the ability to design a system to meet a large number of mission scenarios. Additionally, multi-mode propulsion systems utilizing a single propellant for both modes offer the highest flexibility since any give EP usage fraction may be chosen as mission needs arise rather than defined to a strict ratio as would be the case if two propellants had to be loaded into two separate tanks. The mission trade space for Systems CH, ME, and MH is shown in Fig. 5 since these systems involve utilization of a common propellant. The burn duration versus delta-V is shown for a 6U (6.9 kg) satellite with a 2 kg payload. This may be viewed as the mission trade space with the same caveats applied to the use of burn duration as a comparison tool as described in the previous paragraph. System ME can achieve the highest delta-V of any system, but requires nearly 500 more days compared to System CH to produce only 10% more delta-V. System CH encompasses nearly the entire trade space of System MH, except for missions requiring the majority of delta-V be produced via a chemical burn. For missions requiring less than about 25 days total burn time, System ME can achieve the highest delta-V.

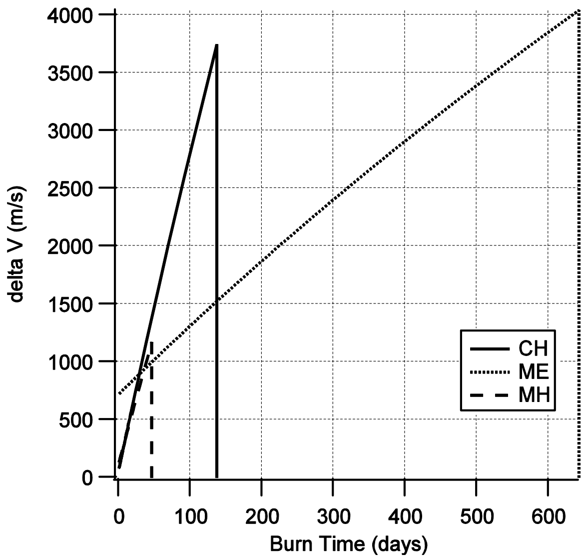

Fig. 5. Mission Trade Space for Systems Utilizing Common Propellant.

V. Discussion

For missions requiring a majority of the total delta-V to be performed through quick, impulsive chemical maneuvers, specific impulse of the chemical mode is the most important consideration, since from Fig. 2 all of the monopropellant systems have a higher payload mass fraction for EP usage below 55% of the the total delta-V. Of the monopropellant systems, System ME has the highest payload fraction. Despite having a higher thruster and powertrain mass than System MP, the fact that the common propellant requires only one propellant tank and one set of lines and valves means it has less overall inert mass compared to System MP. Systems involving the helicon-type thruster have the lowest payload fractions, despite having the highest electric thruster specific impulse. This is due to the massive power unit required to operate the thruster.

Considering the volume of the propulsion systems shows that it is not the limiting factor in terms of these analyses, since the total system volume including payload falls under 1.15 kg/U for systems that can achieve a 500 m/s delta-V with 6.9 kg total system mass. However, this analysis does not include factors such as empty space due to design form factors. While the moderate pressures typically employed in cubesat propulsion systems may allow for space saving tank-designs, such as a toroidal tank, simply applying the results of Fig. 3 is not conclusive in final determination of a preliminary design. However, the comparisons drawn may be somewhat useful. In general, the monopropellant systems have the lowest required volumes. This is tied directly to the fact that the specific impulse of the cold gas systems is only 45 seconds compared to 230 seconds for the monopropellant systems. Thus, to perform the same delta-V as the monopropellant systems requires much more propellant, and thus tank volume. Furthermore, because the propellant densities are relatively similar, the cold gas systems require more volume even up to 95% EP usage despite having less inert mass as evidenced by Fig. 2.

In terms of required burn duration to expend all onboard propellant, the cold gas systems have the longest burn times. This is due directly to the fact that they have by far the lowest electric mode thrust of any propulsion system considered in this study. The thrust of the electric mode effectively dictates the burn duration of the entire multi-mode propulsion system since any low-thrust maneuver utilizing electric propulsion takes significantly longer than a chemical burn even to expel 100% of the propellant. Thus, systems involving the higher-power helicon-type thrusters could be advantageous for multi-mode systems where spacecraft lifetime or mission lifetime is a critical factor.

For spacecraft designs involving flexible mission scenarios, the system with the cold gas thruster combined with the helicon thruster could be the most advantageous since it offers a relatively high delta-V capability in a short amount of time for a given payload mass and total spacecraft mass. The monopropellant system does not perform as well since it has a higher inert mass, and thus less available propellant. This can also be explained directly from the insights of the multi-mode specific impulse computed in Fig. 1. As mentioned in the results section, the shape of the multi-mode specific impulse function is exponential, with most of the specific impulse benefit of the electric system being realized through high values of EP usage. In a flexible mission design scenario such as that defined in the results section where the only constraints are payload mass and total spacecraft mass, limiting propulsion system inert mass is more important than high specific impulse in either chemical or electric mode. Or stated differently, limiting inert mass of the propulsion system allows for a greater fraction of the onboard mass to be propellant. The fact that specific impulse grows at a rate greater than linear according to the multi-mode specific impulse function means that more available propellant will result in an exponentially growing delta-V availability as propulsion system inert mass is reduced. It is therefore highly advantageous to reduce propulsion system inert mass as much as possible through hardware integration or careful selection of components.

VI. Conclusions

Multi-mode spacecraft propulsion systems involving separate chemical and electric thrusters were compared and analyzed in terms of mission capability and overall system sizing. Propulsion systems involving chemical monopropellant thrusters generally outperformed their cold-gas counter parts in terms of both payload mass fraction and propulsion system volume required to perform a 500 m/s delta-V with a 6U scale spacecfraft (6.9 kg). The thrust of the electric mode effectively determines minimum burn duration directly, and as such the systems utilizing the PPT had the highest burn durations since they also had the lowest thrust of all electric propulsion systems considered in this study. For flexible propulsion system design, a multi-mode system utilizing a common propellant is the most important consideration, followed by reduction of propulsion system inert mass through the use of common hardware. The electric system thrust level also plays an important role, effectively determining the scale at which the maximum delta-V can be accomplished.

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