

# Review of Dual mode/Multimode Space Propulsion

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**Multimode propulsion is the integration of two or more propulsive modes into a single spacecraft propulsion system. The key attribute is shared propellant between the different propulsive modes. Multimode propulsion is emerging as an enabling technology that promises enhanced capabilities for spacecraft and space missions, and can therefore play an important role in the future of in-space propulsion. Specifically, multimode propulsion has potential to provide unprecedented flexibility and adaptability to spacecraft, as well as provide mass savings for certain missions. These benefits extend to both medium and large spacecraft, as well as small satellites. Numerous multimode concepts have been explored and documented in the literature. Concepts combining cold gas, monopropellant, bipropellant, and solid chemical propulsion with electrothermal, electrostatic, and electromagnetic electric propulsion have all been investigated. Electropray electric propulsion paired with monopropellant chemical propulsion has perhaps received the most recent attention. We review the concept of multimode propulsion, mission analyses, benefits, and specific multimode concepts.**

## Nomenclature

$f_{EP}$	=	electric propulsion delta-V fraction, -
$f_p$	=	propellant mass fraction, -
$f_{SI}$	=	system integration factor, -
$g_o$	=	acceleration due to gravity, 9.81 m/s <sup>2</sup>
$I_{sp}$	=	specific impulse, s
$m_e$	=	propulsion system dry mass, kg
$m_f$	=	final mass, kg

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$m_l$	=	payload mass, kg
$m_o$	=	initial mass, kg
$m_p$	=	propellant mass, kg
$P$	=	power, W
$t$	=	time, s
$\Delta V$	=	delta-V, m/s
$\eta_p$	=	propulsion system efficiency, -
$\eta_v$	=	mission planning efficiency, -

### Subscripts

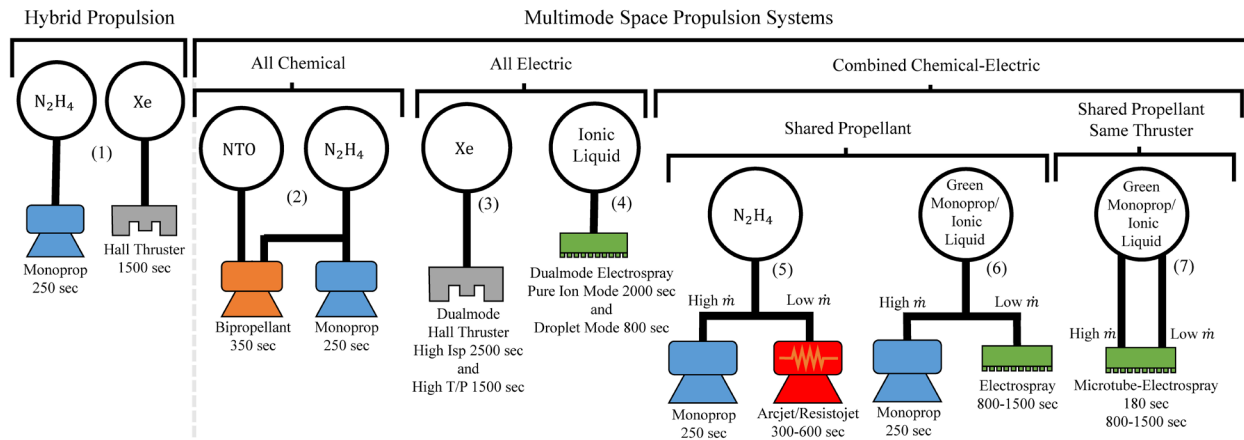
<i>chem</i>	=	chemical
<i>eff</i>	=	effective
<i>elec</i>	=	electric
<i>EP</i>	=	electric propulsion
<i>int</i>	=	integrated
<i>mm</i>	=	multimode
<i>sep</i>	=	separate

## I. Introduction

MULTIMODE space propulsion is the integration of two or more propulsive modes into a single spacecraft propulsion system. In a multimode propulsion system, the key attribute is shared propellant between the different propulsive modes. The multimode propulsion concept is in contrast with ‘hybrid propulsion’ wherein two or more propulsive modes are available on a spacecraft, but do not share propellant. They are completely separate and independent systems. Sharing propellant between different propulsive modes in multimode propulsion has significant benefits in terms of enabling new missions and, perhaps more importantly, in situ mission adaptability. Maximum mission adaptability and flexibility is attained by multimode systems that do *not* pre-allocate propellant to a specific mode of operation. Multimode systems that are entirely chemical, entirely electric, and a combination of chemical and electric propulsion have been investigated. However, combined chemical-electric multimode propulsion has been the subject of most recent interest and is the main focus of this review.

The literature on multimode propulsion also uses the terms ‘dual mode’ and ‘bimodal’ propulsion, which inherently mean a multimode system with two possible propulsive modes. Further, and most confusing, the literature on multimode propulsion also uses the term ‘hybrid’ propulsion. This review focuses on multimode propulsion, including dual mode propulsion and propulsion approaches that may have more than two modes. Grammatically, the literature uses the terms ‘multimode’ and ‘multi-mode’. ‘Multimode’ is perfectly well defined by the Oxford dictionary as an adjective meaning “characterized by several different modes of activity or occurrence”. A hyphenated abbreviation of ‘multiple mode’ is not required. The term ‘multimode’ is the correct grammar, the term adopted throughout this article, and the focus of this review.

An illustration of different types of multimode propulsion systems, their classification, and the notional specific impulse performance of the different modes is shown in Figure 1. Figure 1 shows seven different propulsion system combinations that could be employed on a spacecraft. This figure is *not* an exhaustive list of possible systems, but merely a set of examples illustrating what is, or might be, possible in the near or medium term. System 1 consists of completely separate chemical monopropellant (hydrazine) and electric Hall-effect (xenon) propulsion, and is classified as hybrid propulsion because there is no shared propellant between the two propulsive modes (not to be confused with a hybrid rocket). Hybrid propulsion is quite common on contemporary spacecraft. Deep Space One and DAWN planetary spacecraft used xenon gridded ion engines plus hydrazine monopropellants for roll control. Lockheed Martin A2100 spacecraft bus uses xenon XR-5 Hall-effect thrusters (HETs), hydrazine monopropellants, and hydrazine MR-510 arcjets [1]. Space Systems Loral has a similar system with HETs. Boeing uses a similar system with the xenon ion propulsion system (XIPS) gridded ion engines. While some authors use the term ‘dual mode’ also for this type of system, we find the term ‘hybrid’ regularly used throughout the public domain (see for example the article on the Lockheed Martin A2100 spacecraft bus that describes a ‘hybrid of electrical and liquid technology,’ that is, separate standalone xenon HET and hydrazine monopropellant systems [2]). Also, this type of system, system 1, is regularly referred to as ‘combined chemical-electric’ in the literature [3-6]. Systems 2 through 7 are multimode because they share some propellant between the different propulsive modes. Multimode systems that are entirely chemical, entirely electric, and a combination of chemical and electric propulsion are possible.



**Figure 1: Illustration of hybrid propulsion and multimode space propulsion. Multimode propulsion systems can be categorized as all chemical, all electric, and combined chemical-electric. Multimode propulsion systems share propellant, and may also have common thruster hardware.**

System 2 in Figure 1 is an illustration of an all-chemical partially multimode propulsion system. It consists of monopropellant and bipropellant chemical rocket engines. A common hydrazine ( $N_2H_4$ ) fuel tank feeds both engines, while the nitrogen tetroxide ( $N_2O_4$ , NTO) serves as oxidizer in the bipropellant engine. This system is only partially multimode because fuel and oxidizer must be pre-allocated for specific mission requirements. This type of multimode propulsion system was used on the Mars Global Surveyor in the late 1990's [7]. More recently the BepiColombo mission, a 4100 kg spacecraft launched in 2018 to study Mercury, uses this type of dual mode propulsion system consisting of 22 N NTO/hydrazine bipropellant thrusters for Mercury orbit insertion and 5 N hydrazine monopropellant thrusters for attitude control [8]. All-chemical multimode propulsion is also being explored for small satellites. For example, in the work reported by Ohira et al., a hydrogen peroxide monoprop engine is paired with an ethanol/hydrogen peroxide bipropellant engine [9]. It is interesting to note that in that work, the shared propellant is the oxidizer, not the fuel. Gagne et al. have investigated a mono/biprop system for small satellites that uses hydrogen peroxide and a solution of 15% ferric chloride ( $Fe(III)Cl_3$ ) in propanol [10,11]. Hydrogen peroxide alone is used as the monopropellant, while hydrogen peroxide and the ferric chloride solution are used for bipropellant mode. This concept is also throttleable because the mixture ratio of the hydrogen peroxide to ferric chloride solution can be adjusted, thereby altering the thrust and specific impulse of the bipropellant engine.

Systems 3 and 4 in Figure 1 illustrate two different all-electric multimode systems. These systems are adjustable between a high specific impulse mode and what is often referred to as a high thrust-to-power (and correspondingly lower specific impulse) mode. System 3 is a xenon Hall-effect (HET) system and system 4 is an electro spray propulsion system. Additionally, dual mode ion thrusters [12,13] and hybrid Hall-ion thrusters [14-16] have also been investigated. Modern HET designs tend to be dual mode in order to operate at either high thrust (e.g. for orbit insertion) or high specific impulse (e.g. for orbit station-keeping). Lazurenko et al. provides additional details of these two operational modes for high-power ( $\sim 4$  kW) thrusters [17]. Mode 1 is typically characterized by  $<2000$  s specific impulse and thrust of 200-350 mN, which is sufficient to raise the orbit of Earth-orbiting satellites, and is typically also referred to as high thrust-to-power (T/P). Mode 2 is high specific impulse  $>2500$  s and provides fuel efficient lower thrust for orbit station-keeping of long duration missions. Trade studies of a dual mode HET clearly show the benefits [18]. Magnetic layer (closed-drift, stationary plasma) and anode layer (TAL) Hall thrusters have both been developed for dual mode operation [17,19-26], and, more recently, nested channel Hall thrusters have been developed, wherein a tri-channel thruster would have seven different possible operational modes [27].

System 4 in Figure 1 is an electro spray propulsion system. Electro spray, a.k.a. colloid, propulsion is a type of electrostatic propulsion wherein a conductive or dielectric liquid is subjected to a strong electric field, usually at the tip of a capillary or needle. The resulting imbalance between the surface tension of the liquid and electrostatic force causes an instability of the liquid surface, giving rise to emission of charged molecules/ions and droplets from the liquid. Coffman [28,29] has explored the ability of a dual-grid electro spray system to provide variable specific impulse electro spray at constant power. Theoretical analyses suggested that the specific impulse could be adjusted by as much as 50% at fixed power and still maintain good overall efficiency. It may also be possible to adjust the electro spray mode by switching between a purely ionic emission regime (high-specific impulse) and mixed ion-droplet or purely droplet regime (high thrust-to-power).

Systems 5, 6, and 7 are multimode propulsion systems that integrate together chemical and electric propulsion. Systems 5 and 6 are examples of separate chemical and electric propulsion thrusters that share a common propellant, while system 7 illustrates a multimode system with shared propellant and a common thruster. Combined chemical-electric multimode propulsion has been the subject of most recent interest and is the main focus of the remainder of this review. First the fundamental equations for modeling and assessing chemical-electric multimode systems will be reviewed. Then the benefits of the chemical-electric multimode approach will be described, highlighting the similarity with hybrid propulsion, but also demonstrating the fundamental benefits of the approach: flexibility and adaptability. Next, different chemical-electric multimode concepts and their current status will be reviewed. Finally, we draw conclusions regarding the future potential and current state-of-the-art, and identify areas of future research and development focus.

## II. Multimode System Analysis

### A. Multimode Rocket Equation

Multimode systems use two separate propulsive modes with different specific impulses. The chemical and electric modes are considered to provide separate maneuvers that are each governed by the Tsiolkovsky rocket equation, shown in Eq. (1):

$$\frac{m_f}{m_o} = 1 - \frac{m_p}{m_o} = e^{-(\Delta V / I_{sp} g_o)} \quad (1)$$

Berg [30,31] and Donius [32,33] define a parameter based on the percentage of the total delta-V to be conducted by the electric propulsion (EP) mode, in some instances called the “EP fraction”, which is given by Eq. (2). It is then possible to write the multimode rocket equation as Eq. (3), where the multimode specific impulse is given by Eq. (4) [30,31].

$$f_{EP} = \frac{\Delta V_{elec}}{\Delta V} \quad (2)$$

$$\frac{m_f}{m_o} = e^{-(\Delta V / I_{sp,mm} g_o)} \quad (3)$$

$$I_{sp,mm} = \left[ \frac{1 - f_{EP}}{I_{sp,chem}} + \frac{f_{EP}}{\eta_v I_{sp,elec}} \right]^{-1} \quad (4)$$

In Eq. (4),  $\eta_v$  is the “mission planning efficiency”, defined as Eq. (5), which takes into account the fact that, for practical electric propulsion maneuvers of finite duration, the actual electric delta-V required is higher than a purely impulsive burn. The mission planning efficiency is a function of starting and ending orbits, and steering profile of the non-impulsive maneuver. Typical values range from 0.45 to 0.65 for Earth orbiting mission [34,35]. Since the mission planning efficiency is used in Eq. (3), the delta-V in Eq. (3) is that required to complete a given mission using impulsive maneuvers.

$$\eta_v = \frac{\Delta V_{elec,eff}}{\Delta V_{elec}} \quad (5)$$

It is important to note that Eqs. (3) and (4) do not depend on the number or order in which maneuvers are carried out. While a separate rocket eqn. (Eq. (1)) can be written for each maneuver, the total delta-V of the spacecraft is independent of the number or order of electric vs. chemical maneuvers. It is, however, dependent on the fraction of the total delta-V completed by electric vs. chemical propulsion ( $f_{EP}$ ), the specific impulse of each mode, and, weakly, the mission planning efficiency. Finally, an equation can be written for the fraction of propellant consumed by the EP mode, Eq. (6). The fraction of propellant consumed by the chemical mode would then be  $f_{p,CP} = 1 - f_{p,EP}$ .

$$f_{p,EP} = \frac{m_{p,EP}}{m_p} = \frac{1 - e^{-(f_{EP} \Delta V / \eta_v I_{sp,elec} g_o)}}{1 - e^{-(\Delta V / I_{sp,mm} g_o)}} \quad (6)$$

For multimode propulsion systems it is beneficial to use shared hardware to reduce the mass of the propulsion system and increase deliverable payload. The final mass of the spacecraft consists of both payload and propulsion system dry mass, as shown in Eq. (7).

$$\frac{m_f}{m_o} = \frac{m_l}{m_o} + \frac{m_e}{m_o} \quad (7)$$

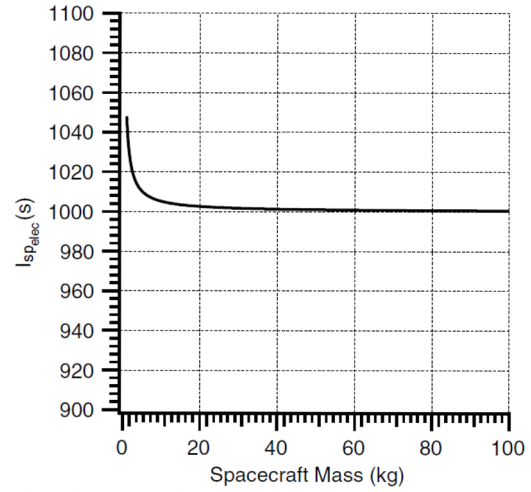
For large spacecraft and launch vehicles, the propulsion system dry mass fraction is small compared to the propellant mass fraction, so any savings or reductions in dry propulsion mass provide minimal benefit. However, dry propulsion mass can be a significant fraction for smaller satellites, such as micro, nano, and Cube satellites. In some cases, it can be upward of half the entire satellite mass. In these situations it can be beneficial to share propulsion system hardware such as tanks, lines, pressurant, valves, thrusters, power processing units, solar arrays, etc. For example, Koizumi et al. show that, for the PROCYON spacecraft, the dominant mass of the I-COUPS multimode propulsion system is the pressurized gas system, which is 40% of the 9.5 kg wet propulsion system mass [36]. They specifically point out that mass savings due to shared propellant and pressurization system is superior to increasing specific impulse. The effect of sharing hardware can be introduced through a system integration factor, Eq. (8), where  $m_{e,sep}$  is the total propulsion system dry mass for separate chemical and electric propulsion systems, and  $m_{e,int}$  is the total propulsion system dry mass when the chemical and electric systems are integrated together. Clearly if there is no mass savings when chemical and electric propulsion are integrated together, the system integration factor is zero. Inserting (7) and (8) into (3) yields Eq. (9), which clearly indicates that the propulsion system dry mass penalty can be reduced by designs that use common hardware between the propulsive modes ( $f_{SI} > 0$ ).

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

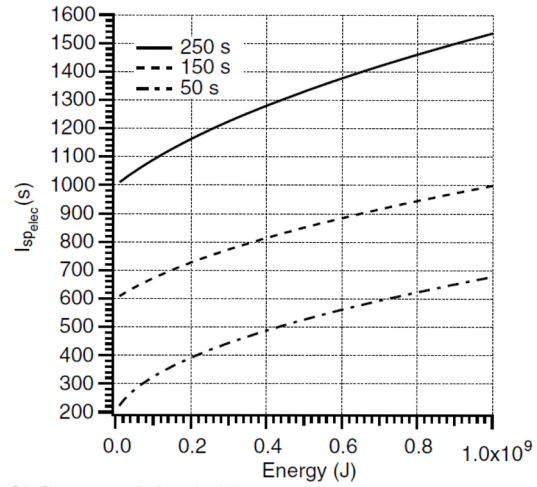
$$\frac{m_l}{m_o} = e^{-(\Delta V / I_{sp,mm} g_0)} - \frac{m_{e,sep}}{m_o} (1 - f_{SI}) \quad (9)$$

## B. Optimum Electric Mode Specific Impulse

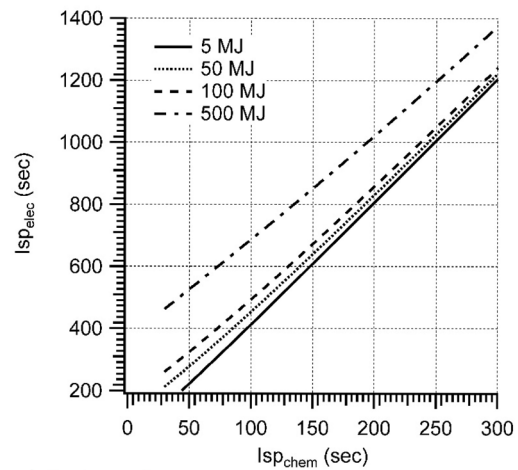
For a multimode propulsion system, there is an optimum electric mode specific impulse that depends on the chemical mode specific impulse. Oh et al. [34] have analyzed chemical-electric orbit raising missions and derive a combined chemical-electric rocket equation similar to Eq. (3) above. They assumed a fixed amount of power and time is available for the electric mode, that the EP system is always on, and that the EP specific impulse is fixed. They linearize the resulting rocket equation and



a) Optimum electric specific impulse



b) Contours of chemical  $I_{sp}$



c) Contours of energy

**Figure 2: Optimum electric specific impulse as a function of a) spacecraft mass, b) energy, and c) chemical mode specific impulse, from Ref. [31]. Reproduced with author permission.**

show that the (optimum) electric specific impulse for maximum transportation rate (payload mass delivered per unit time) is given by Eq. (10):

$$I_{sp,elec}^* = 2I_{sp,chem}/\eta_v \quad (10)$$

This result does not include the power or time (or specific power) because of the linearization, so the result is only valid when the propellant mass is small compared to the spacecraft mass. Since the mission planning efficiency is always less than one, it is clear the optimum electric specific impulse is always at least twice the chemical mode specific impulse. Actually, the early work of Edelbaum (ca. 1962) identified a similar relationship for a field-free space analysis [37], and Edelbaum notes that “the optimum low-thrust exhaust velocity is always at least twice the high-thrust exhaust velocity.”

More generally, Oh et al. [34] show that, for a mission of given power and time constraint, the optimum electric specific impulse (again in terms of transportation rate) is a function of the desired final spacecraft mass, the mission planning efficiency, the electric thruster efficiency, and the chemical mode specific impulse, as shown in Eq. (11).

$$4Pt\eta_p \left(1 - \frac{\eta_v I_{sp,elec}^*}{I_{sp,chem}}\right) + \frac{\eta_v I_{sp,elec}^*}{I_{sp,chem}} \left(2Pt\eta_p + \frac{I_{sp,elec}^{*2}}{g_0^2} m_f\right) \ln \left(\frac{2Pt\eta_p g_0^2}{m_f I_{sp,elec}^{*2}} + 1\right) = 0 \quad (11)$$

Although not immediately intuitive, the optimum electric specific impulse depends on the chemical specific impulse in a multimode system because, under a fixed time constraint, use of the chemical mode will be required to meet the time constraint at the expense of poorer propellant utilization. Berg has explored the effect of spacecraft mass, mission energy, and chemical mode specific impulse on the optimum electric mode specific impulse for small spacecraft [31]. These results are shown in Figure 2, wherein a mission planning efficiency of 0.5 was assumed. Figure 2a shows the minimal effect of spacecraft mass on optimum electric specific impulse for a chemical specific impulse of 250 s and mission energy ( $Pt\eta_p$ ) of 5 MJ. This result is identical to the insight provided by Oh et al. [34] who showed that the optimum specific impulse is independent of the specific power when the electric propellant mass is small compared to the system dry mass. Figure 2b shows the effect of mission energy on optimum electric specific impulse for 50, 150, and 250 s chemical mode specific impulse. As mission energy increases a higher electric mode specific impulse is required to fit within mass constraints. Figure 2c illustrates that, for a fixed mission energy, optimum electric specific impulse increases with chemical specific impulse. This trend is expected because, given a mission of fixed duration, a higher chemical specific impulse will provide more energy gain for the same chemical propellant mass, thereby requiring less energy gain from the electric propulsion system, and correspondingly a higher optimum electric specific impulse for a fixed electric thruster efficiency. From this figure it is clear that, for a typical monopropellant chemical specific impulse of 250 s, the optimum electric specific impulse for a wide range of mission energies is 1000-1200 s.

### III. Benefits of Multimode Space Propulsion

The benefits of combining high- and low-thrust propulsion onboard the same vehicle have been known since the days of Edelbaum [37]. Since then, previous studies have investigated more efficient orbits, and quantified the mass savings and deliverable payload benefits. Most of these previous studies focused on hybrid propulsion systems, where the high-thrust and low-thrust systems are completely separate. The benefits of hybrid propulsion naturally extend to multimode propulsion, and these studies and their results will be reviewed first. But multimode propulsion offers an additional benefit beyond simply new and more efficient orbits and maneuvers; it offers mission flexibility. The same propulsion system can be used for a wide range of missions, without the need for costly redesign and requalification. Further, missions can be adjusted on-the-fly, in situ, post-launch. The flexibility and adaptability of multimode propulsion will be discussed second, followed by a rough order of magnitude cost benefit analysis of a one-size-fits-most multimode propulsion system.

#### A. Hybrid Propulsion Benefits that Extend to Multimode

Combined chemical-electric hybrid propulsion has shown benefits for lunar and interplanetary spacecraft. Gilland analyzed the benefits of high and low thrust capability on a piloted mission to Mars [38]. He found that an “ideal” propulsion system for such missions would have both high specific impulse and high power density, a challenging combination for a single propulsion system. However, he suggests this combination may be approximated through separate high thrust chemical and high specific impulse nuclear electric propulsion systems onboard the same vehicle.

Cluever analyzed combined chemical-electric propulsion for a lunar-interplanetary mission and found the combined approach delivered 15% more payload in the same time as the all-chemical approach [3]. Cluever also considered optimized Earth-Moon trajectories using combined chemical-electric propulsion [5]. He analyzed a chemical boost from LEO with ballistic trajectory followed by a spiral lunar capture trajectory performed by the electric propulsion stage. A Taurus launch vehicle was assumed with either arcjet or stationary plasma (Hall-effect) thrusters for the electric stage. Results showed the arcjet delivered a payload ratio of 0.78 (midlunar orbit mass/translunar injection mass) in 64 days, while the stationary plasma thruster ratio was 0.94 and required 107 days. Both cases delivered more mass in a shorter time compared to all-electric vehicles. Mingotti et al. investigated hybrid propulsion transfers to Mars [39] and noted that hybrid propulsion has lower propellant mass than both all-chemical and all-electric transfers. They also note the hybrid approach provides for an extended launch window due to the flexibility of the propulsive maneuvers available. Percy et al [40] and Chai et al [41] have both investigated hybrid propulsion transfers for a crewed mission to Mars. They show that, for delivering crew, solar electric propulsion alone is not sufficient and must be combined with chemical propulsion. Specifically, chemical propulsion would be used for orbit departure and insertion burns, where higher thrusts are more advantageous, and solar electric propulsion would be used to shorten the coast time between planets. Chai et al [42] also showed that using a hybrid propulsion approach for Mars cargo transfer is more fuel efficient and has minimal travel time penalty. Topputo and Massari investigated the use of hybrid propulsion for transfers from GTO to near Earth objects [43], specifically within the context of the ESA Marco Polo sample return mission. They compare with an all-chemical transfer and show the hybrid approach can return 80 to 100% more mass to Earth. Mani et al. have explored interplanetary CubeSats using hybrid propulsion [44] for the Mars Atmospheric Radiation Imaging Orbiter (MARIO). They investigated an ADN-based bipropellant system [45] and selected an iodine-based RF ion thruster. To complete the MARIO mission requirements, the chemical system would complete 450 m/s delta-V, while the electric system provides 6.9 km/s, and the total thrusting time would be around 1600 days.

Combined chemical-electric hybrid propulsion has shown benefits for commercial spacecraft, specifically for orbit raising missions. Oleson et al. studied the use of advanced onboard propulsion systems to perform both north-south station keeping and part of the orbit transfer for GEO spacecraft [46]. They showed significant payload enhancements are possible, with the use of advanced solar EP for a portion of the orbit transfer providing an increase in delivered mass of 20 to 45% for one- to four-month transfer times, respectively. Mailhe and Heister investigated a hybrid orbital transfer vehicle for LEO to GEO, showing that it reduces trip time and corresponding radiation damage due to the Van Allen belts, and also dramatically reduces vehicle gross weight [47]. They investigated high-low and low-high-low thrusting strategies, showing that the low-high-low approach was the most efficient, but would require solar arrays to be deployed-stored-deployed. Further, they show that this approach can reduce radiation exposure by a factor of 10-50 times compared to an all-electric approach. Of the three EP engines Mailhe and Heister investigated (two Hall thrusters, the SPT100 and HET-220, and an arcjet, ESEX), they show the SPT-100 provides the most mass savings with a small increase in trip time. Oh et al. used a simple analytic multistage model to investigate chemical-electric orbit raising [34]. They included multiple launch vehicles and low-thrust-trajectory optimization to derive optimum orbit-raising profiles to GEO. Their results show increases in the payload mass transfer rate of 6.1 to 7.6 kg/day when using two SPT-140 plasma thrusters, which can result in 680 kg of mass savings for a 90 day electric orbit raising. Jenkin also performed trade studies for GEO transfers that use a combined chemical-electric stage [48]. He used trajectory optimization to maximize GEO-insertion mass, and determined trends among various mission and system parameters, such as the elliptical orbit for the start of the EP phase, input power of the EP system, and EP system specific impulse. Byers and Dankanich investigated chemical-electric transfer from GTO to GEO for COMSATS [49]. They found mass savings of 500-2500 kg is possible using EP with specific impulse 1000-2100 sec. Further, they found that integrating electric propulsion with chemical propulsion has the highest payoff for spacecraft with GTO mass of 2500-5500 kg. Cluever investigated optimal GEO transfers using chemical-electric propulsion [35]. He found that the transfer rate for a chemical-electric vehicle was 4.4 to 10.5 kg/day depending on the size of the launch vehicle. Most recently Cluever developed an analytical algorithm for determining spacecraft mass requirements for a desired electric propulsion system and desired transfer time using combined chemical-electric hybrid propulsion [6]. Ceccherini and Topputo considered a system-trajectory analysis, wherein they couple together both the payload and propulsion system design with trajectory optimization to investigate the benefits of chemical-electric GEO transfer [50].

Other types of missions have also been analyzed and illustrate the benefits of having both high- and low-thrust propulsion onboard the spacecraft. Lee and Hwang [51] investigated small satellite formation flight and showed that the multimode approach is more fuel-efficient than conventional two impulsive high-thrust maneuvers in performing timely reconfiguration tasks. Oland et al. [4] investigated attitude control for small satellites and showed that the multimode thrust history is a viable solution that is able to use chemical thrusters for slew maneuvers, while using the

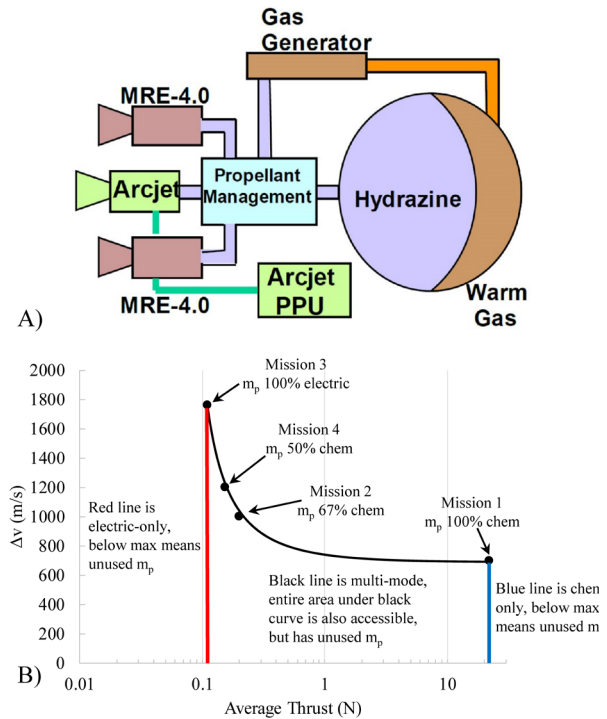
electric thruster for fine attitude pointing. Kemble and Taylor [52] compared different thrust approaches for small satellite missions to Jupiter. They assessed the different approaches in terms of transfer rate (time to get there) and useful mass (final mass at destination) and showed that the best result in terms of transportation rate with only a small (~6%) useful mass reduction was a combined solar-electric and chemical propulsion system. Trawny et al. [53] have investigated mission scenarios for impacting a small satellite into the lunar surface (a la SMART-1) and showed that electric propulsion followed by a chemical impulse for a final aposelene boost yields much more favorable impact conditions. Ulybyshev investigated rendezvous trajectories using multimode propulsion [54], specifically developing new methods for trajectory optimization of spacecraft with both high- and low-thrust in near-circular orbits.

### B. Flexibility of Multimode Propulsion Systems

The literature results described in the previous section have shown that under certain mission scenarios it is beneficial in terms of spacecraft mass savings, deliverable payload, and/or transfer rate, to utilize separate high-specific impulse and high-thrust propulsion systems, i.e., hybrid propulsion. Those benefits extend to multimode systems as well. But even greater mass savings can be realized by using a shared propellant and/or hardware, even if the thrusters perform lower than state-of-the-art in either mode. In order to realize the full potential of a multimode propulsion system, it is necessary to utilize a single shared propellant for both modes; this allows for a large range of possible maneuvers (flexibility), while still allowing for all propellant to be consumed regardless of the specific choice or order of maneuvers. The following paragraphs describe analyses focused on specific multimode systems, and have quantified potential mass savings as well as illustrated the flexibility of the multimode approach.

Haas and Holmes illustrate the mass savings and flexibility of multimode propulsion using four design reference missions [55]. They studied the benefits of a multimode propulsion system for small spacecraft <200 kg, wherein they assumed a 100 kg spacecraft with maximum 80 kg of propulsion and propellant mass shared between a 235 s specific impulse chemical thruster and 1kW 600 s electric arcjet thruster. A schematic of the propulsion system is shown in Figure 3A, and is similar to that shown as system 5 in Figure 1. The four missions considered include: (1) a rapid 12 hr, 180 deg orbital phase shift followed by a 12 hr return to original phase; (2) a rapid 48 hr, 1000 km altitude rise followed by a 30 day return; (3) a large plane change mission of 15 deg in 90 days; and (4) a 48 hr drop to 300 km altitude for 300 days followed by a 30 day return to original orbit. Results show that chemical-only and electric-only spacecraft cannot complete all of these missions. Specifically, a chemical-only spacecraft cannot complete mission 3 or 4, while an electric-only spacecraft cannot complete mission 1 or 2. A hybrid system (both chemical monoprop and 1 kW arcjet onboard, but no shared propellant) can complete all four missions. However, the multimode system, with its shared propellant, can complete all four missions and provide significant increases in deliverable payload mass because of the decreased system dry mass enabled by propellant sharing. Specifically, their results show the multimode system provides 6.9, 1.3, 2.5, and 5.8 times more delivered payload mass for each of the four missions, respectively. Rexus and Holmes [56] provide additional details of mission 3. Specifically, they investigated the 15 degree plane change maneuver. The chemical thruster alone was unable to complete the mission with the maximum propellant, while the multimode system required only 37 days.

A comparison of the delta-V requirements for the four missions of Haas and Holmes, as well as the delta-V capability of their system, is illustrated in Figure 3B. The multimode system capability is shown as the black curve in Figure 3B. The multimode approach enables the same propulsion system to be used on drastically different



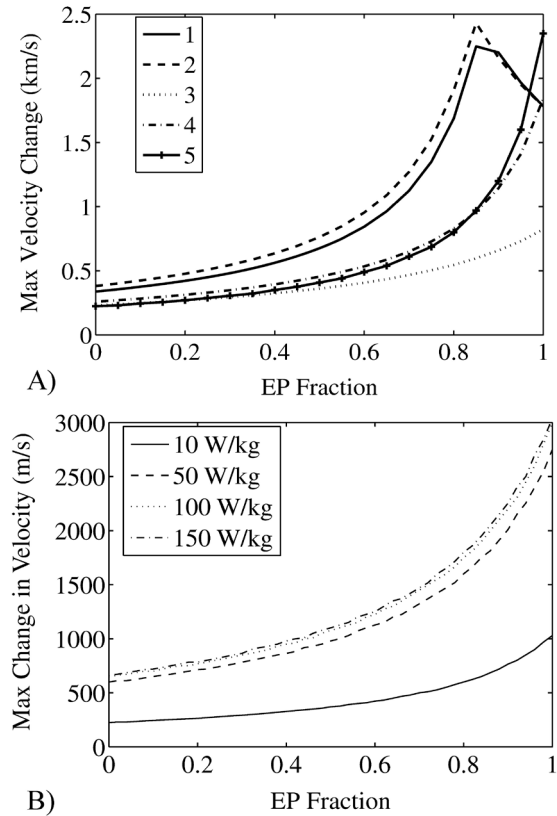
**Figure 3: A) Multimode system investigated by Haas and Holmes [55] consisting of monopropellant chemical propulsion and arcjet electric propulsion systems fed by shared hydrazine propellant tank; B) their mission analysis illustrating flexibility to enable a wide range of missions. Reproduced with author permission.**



missions, from the high delta-V requirement of mission 3 to the high thrust requirement of mission 1, and everything in between. All missions lying below the black curve are also possible, but only the missions lying on the black curve represent 100% propellant utilization. In contrast, a purely electric system is confined to the vertical red curve on the left of the figure, and a purely chemical system is confined to the vertical blue curve on the right of the figure. Further, recognize that only the peak delta-V value for the purely electric or purely chemical systems represents 100% propellant utilization. Finally, missions 2 and 4 are only possible when both chemical and electric propulsion are available, and are achievable by allocating propellant mass 50:50 and 67:33 in the chemical:electric propulsion modes, respectively.

Donius and Rovey assessed different multimode spacecraft propulsion concepts [33]. Specifically, they considered five different chemical and electric propulsion combinations, three of which are multimode. Their combinations 1 and 2 were hybrid propulsion concepts that paired monopropellant and bipropellant chemical propulsion with a xenon Hall thruster, respectively. Their combinations 3 and 4 paired bipropellant chemical propulsion with an electro spray thruster, and in combination 3 the electro spray only used the fuel component of the bipropellant. Their combination 5 was monopropellant with an electro spray thruster. Analyses were done assuming a 100 kg total spacecraft mass, and the max delta-V as a function of the electric propulsion fraction ( $f_{EP}$ , EP Fraction, as given in Eq. 2) is shown in Figure 4A. Figure 4A shows the hybrid systems (combinations 1 and 2) provide larger delta-V, not the multimode systems (combinations 3, 4, and 5). This is due directly to the authors' assumption of the EP power supply (PPU) specific power. They assumed a 200 W xenon Hall thruster with specific power of 150 W/kg, but they assumed a 200 W electro spray with specific power of 10 W/kg. This results in an electro spray PPU that is almost 15 times more massive than the xenon Hall thruster PPU, which decreases the available propellant mass from about 25 kg (for combinations 1 and 2) to only 10 kg (for combinations 3, 4, and 5). With a heavier PPU and corresponding less propellant, combinations 3, 4, and 5 provide the 100 kg spacecraft with less delta-V. The effect of an increase in PPU specific power was studied, and, as expected, higher specific power increased the available propellant mass and provides larger delta-V. This is shown in Figure 4B. Their analyses suggest that the electro spray specific power must be greater than about 15 W/kg for the electro spray multimode combinations to perform as well as the xenon Hall thruster combinations. Finally, it is important to recognize the propulsion system flexibility illustrated by Figure 4B. A purely chemical system is confined to the vertical line at an EP fraction of zero, while a purely electric system is confined to the vertical line at an EP fraction of one. The multimode system can perform any mission along and below the curve, and missions below the curve have unused propellant.

Berg and Rovey assessed high power multimode systems [30]. They investigated monopropellant and bipropellant chemical rockets paired with either an arcjet, Hall thruster, or pulsed inductive thruster (PIT). The electric propulsion systems had a nominal 30 kW electric power, and the focus was on larger spacecraft with a 500 kg payload. Some of the system combinations were multimode and shared a common propellant, specifically, the monopropellant-arcjet and monopropellant-PIT pairings used hydrazine, while the bipropellant-PIT pairing used monomethylhydrazine fuel with nitrogen tetroxide oxidizer. Results show that these multimode 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 electric power supply. When assuming a 1500 m/s delta-V for a 500 kg payload, results indicate that a bipropellant thruster is ineffective by any metric. The lower inert mass of the monopropellant thruster was more beneficial to the overall mission capability despite its lower specific impulse and thrust. This result is



**Figure 4: (A) Comparison of delta-V as a function of EP fraction for different electric and chemical propulsion combinations 1 through 5. (B) Effect of EP specific power on combination 5. Reproduced from [33] with author permission.**

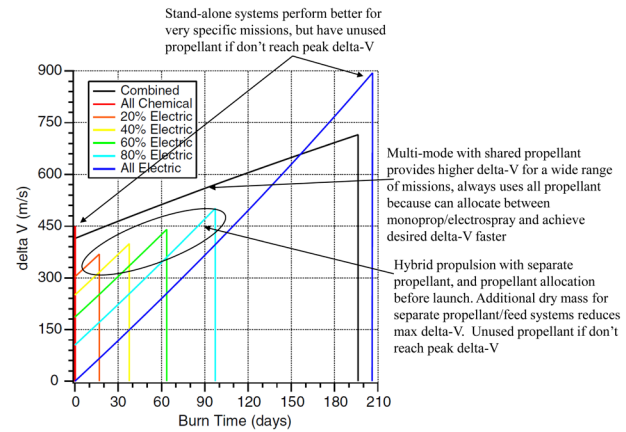
indicative of the tradeoffs within multimode propulsion systems where enhanced overall mission capability is possible with a lower performance system. Systems that use a common shared propellant have flexibility and adaptability, but results indicated no benefit in terms of overall mass savings. This was attributed to the propellant storage properties (density) and corresponding tank sizing. Analysis of the transportation rate indicated that the monopropellant-arcjet pairing provided the highest value, despite having the lowest specific impulse in both modes. Basically, the benefit of having the lowest inert mass and highest electric thrust outweighed the extra propellant requirements.

Berg and Rovey also assessed multimode micropropulsion systems [31]. They investigated cold gas and monopropellant chemical rockets paired with either a pulsed plasma thruster (PPT), electro spray thruster, or an ion thruster. The focus was on small satellites with a mass of 7.8 kg, and the electric propulsion systems required less than 100 W.

Missions requiring 250, 500, and 1000 m/s delta-V were considered. An interesting trade-off was identified where the highest payload ratios are achieved with monopropellant chemical propulsion, but only when the delta-V performed by the electric propulsion is below about 55-70%. If greater than 70% of the delta-V is to be performed by electric propulsion, then cold gas provides a higher payload ratio because of its lower inert system mass. The thrust of the electric mode determines the minimum burn duration for a multimode system, so multimode systems using the lowest thrust electric propulsion (the PPT) will have the longest burn duration. Some of the systems investigated by Berg and Rovey [31] have shared propellant. Specifically, the cold gas-PPT (butane), cold gas-ion (xenon), and monoprop-electrospray (ionic liquid [Emim][EtSO<sub>4</sub>]/[HAN]) have shared propellants (propellant given in parentheses). Of these, the monoprop-electrospray was shown to have the highest mission capability in terms of delta-V for missions lasting less than 150 days. This result was directly attributed to the combination of shared propellant, low inert mass, high electric thrust, and an electric specific impulse near optimum for the system (optimum based on the chemical mode specific impulse, as shown in Figure 2).

A clear example of the flexibility and adaptability of multimode propulsion is illustrated by Berg and Rovey for the monopropellant-electrospray combination using a [Emim][EtSO<sub>4</sub>]/[HAN] propellant [31]. They compared the monopropellant alone, electro spray alone, multimode, and hybrid propulsion approaches. Specifically, the monopropellant alone approach was assumed to use the new 'green' AF-M315E propellant, while the electro spray alone approach uses a typical electro spray ionic liquid propellant [Emim][Im], with corresponding performance of 230 s, 500 mN and 800 s, 0.7 mN in each mode, respectively. The hybrid propulsion approach assumes separate AF-M315E monopropellant and [Emim][Im] thrusters with propellant mass allocated to each mode before launch. The monopropellant-electrospray multimode concept assumes performance of 226 s, 500 mN and 780 s, 0.6 mN in each mode respectively. Their results assume a 7.8 kg small satellite with a 5.2 kg payload and are shown in Figure 5. From Figure 5 it is clear that the multimode (i.e., 'combined') approach has a higher delta-V at a given burn duration for all missions except the monopropellant alone (all chemical) and electro spray alone (all electric) cases, because in those cases the mass of the opposite mode thruster is not included in the analysis. Above about 150 days, the extra mass of the chemical mode thruster is no longer worth the benefit and an electric-only system provides higher delta-V. The multimode approach provides higher delta-V than the hybrid propulsion systems for the same burn duration. This is due to the use of shared hardware (lines, valves, and associated structural mount mass) in the multimode combined system. Further, it is important to recognize that any hybrid propulsion or stand-alone system mission that does not reach the peak delta-V, has less than 100% propellant utilization. In contrast, for the combined multimode system, the entire area under the topmost (black) curve is available for mission applications, even after launch. Further 100% propellant utilization is achieved for all missions on the topmost curve.

It is clear from the previous analyses that an advantage to utilizing a shared propellant for both modes is the mission flexibility, which is possible even post-launch. The system can take advantage of reallocating propellant on-the-fly to perform either quick, impulsive maneuvers or large delta-V orbit changes. For example, from Figure 5, a spacecraft could achieve 700 m/s in 195 days using 100% propellant in electric mode, or 600 m/s in 120 days using propellant



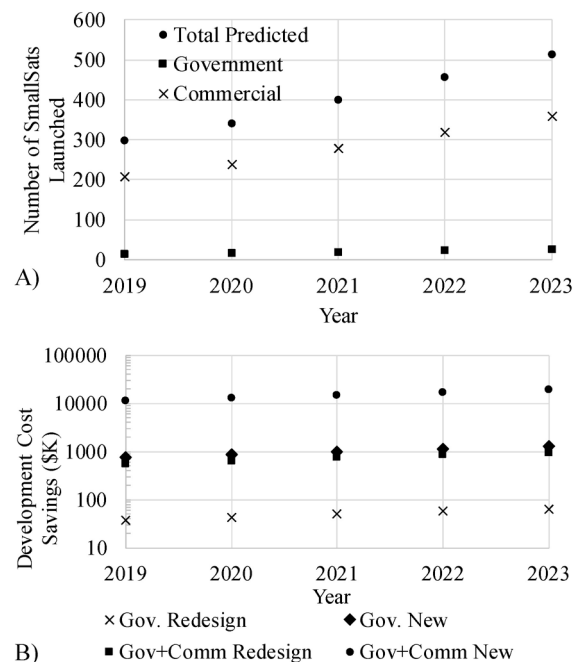
**Figure 5: Reproduced from Berg and Rovey [31] with author permission. Their comparison of chemical-only, electric-only, hybrid, and multimode (combined) propulsion systems to illustrate the flexibility of the multimode approach.**

80:20 electric:chemical, or 450 m/s in 20 days using 50:50, or 415 m/s with 100% propellant in an impulsive chemical mode. Four identical spacecraft could be launched, and they could each execute drastically different missions with the exact mission even determined post-launch. Even if a multimode system has lower performance than that of a single mode thruster, utilizing a shared propellant has the potential to enable a large mission design space. This flexibility and adaptability to perform well (not the optimized best, but still excellent) over a wide range of mission scenarios may be beneficial, especially within the burgeoning small satellite community. The tenet for small satellite design is rapid concept-to-flight using plug-and-play architecture. It is conceivable that a multimode system could be constructed with a common, known interface, and thereby enable a plug-and-play architecture that would allow such a system to be quickly integrated onto a small satellite. This would enable faster concept-to-flight since small satellite designers could avoid the time-consuming task of selecting and designing a new propulsion system for a specific mission. Further, it could eliminate the even worse, but all too common, reality of degrading and de-scoping mission capabilities to settle for a quickly-available single-mode propulsion system.

### C. Potential Impact on Recurring Development Costs

The last section highlighted results and analyses that have shown flexibility and adaptability are key benefits of multimode propulsion. These benefits may enable a single propulsion system to be used on a wide range of missions, thereby eliminating the need to re-design, re-qualify, or de-scope a propulsion system. These are time-consuming tasks and therefore there is potential for multimode propulsion to provide cost savings, specifically, lower recurring cost. But cost models are complex. Even when applied to specific technologies with known historical development and recurring costs, cost models can be inaccurate. Here, we use existing data and trends to make at least an order of magnitude prediction of the possible benefit, and focus solely on forecasts related to the small/microsatellite market.

A recent SpaceWorks report predicts the number of nano/microsatellites (1-50 kg) to be launched in the next few years [57], predicting government will be about 5% of launches and the commercial sector about 70% of launches. These data are shown in Figure 6A. The cost to develop new space propulsion is high. For high-power electric propulsion systems (even those based on existing flight-proven technologies), it is in the +\$100M range. For example Aerojet-Rocketdyne has won +\$10M's grants to develop high-power electric propulsion systems based on already flight proven ion and Hall technology [58], and NASA's NEXTStep program has invested +\$10M's in developing high-power deep-space electric propulsion [59]. But small satellite propulsion systems are attractive because they are supposed to be 1/10<sup>th</sup> to 1/100<sup>th</sup> less expensive to develop, so here we assume, very conservatively, the development and qualification costs of a new smallsat propulsion system is \$1M, and modification or re-design of an existing propulsion system is only 1/20<sup>th</sup> of that cost, \$50K. Further, we assume conservatively that multimode propulsion will save development costs on only 5% of missions. The resulting predicted development cost savings are shown in Figure 6B. Clearly the savings have the potential to be significant. For government missions alone the estimated savings is between \$50K-\$1M/year. If commercial smallsats are included, the estimated savings is \$1-10M/yr. These savings are substantial because of the sheer number of small satellites predicted in the near term. A small or modest cost savings with a one-size-fits-most multimode propulsion system can be compounded into a significant cost benefit.



**Figure 6: (A) Predicted smallsat launches [56]. (B) Recurring development cost savings if 5% of smallsats use multimode propulsion instead of requiring a new (assumed to be \$1M cost) or re-designed (assumed to be \$50K cost) propulsion system.**

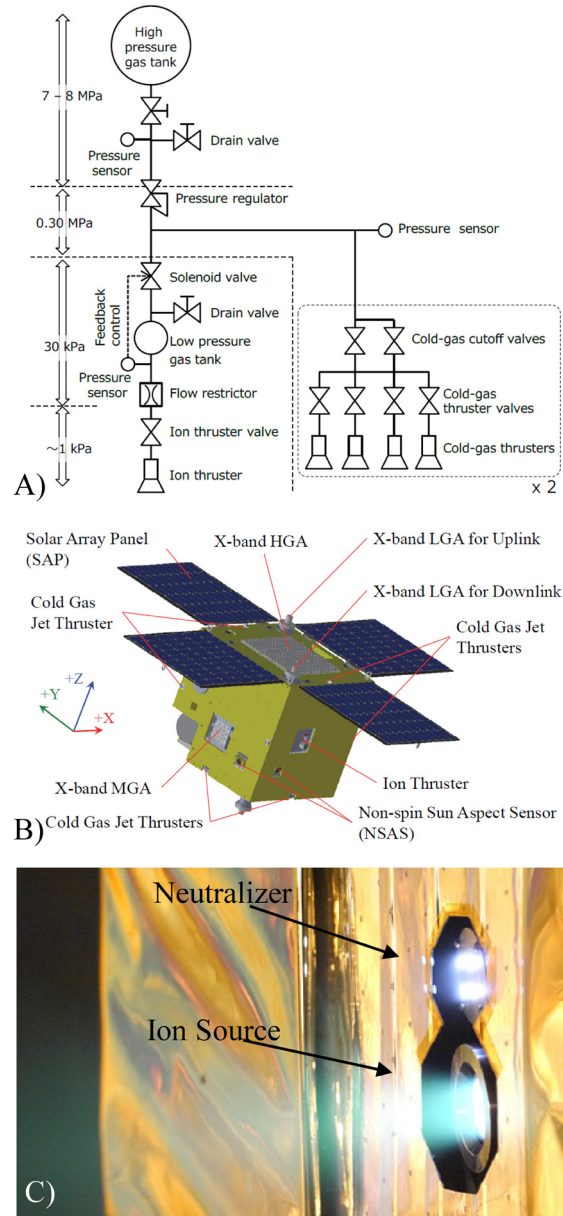
## IV. Multimode Propulsion System Concepts

Recent efforts have been focused on integrating together high thrust chemical and high specific impulse electric propulsion systems into what is called multimode propulsion. In the following sections we review the different technologies that have been studied and/or developed for this application. Of course we are limited to concepts that are publicly available within the literature, so proprietary or otherwise restricted concepts cannot be included. We do not include in this review the combination of cold-gas and warm-gas thrusters, which offer modest improvements in performance (e.g., 70 vs. 140 s Isp) and have been available for some time. Rather our focus is on new concepts that propose previously unexpected and unanticipated pairings and combinations of chemical and electric propulsion. The review is organized by the different chemical propulsion techniques (cold-gas, mono/biprop, solid propellant) paired with different electric propulsion techniques (electrothermal, electrostatic, electromagnetic).

### A. Cold-gas with Electrostatic Propulsion

#### 1. Cold-gas with Ion Thruster

The University of Tokyo developed and operated a multimode propulsion system consisting of eight xenon cold-gas thrusters and a single xenon gridded ion thruster called the Ion thruster and Cold gas thruster Unified Propulsion System (I-COUPS) [36,60,61]. This system has shared propellant and shared propellant pressurization system and was flown on the PROCYON, a 50-kg-class micro-space probe orbiting around the Sun [62]. A schematic of the feed system, model of the PROCYON spacecraft, and photograph of the ion thruster operating are shown in Figure 7. The ion thruster is the main engine to provide delta-V and the cold-gas thrusters provides attitude control. The ion thruster utilizes a microwave discharge plasma for both the ion source and neutralizer. The ion source and neutralizer have identical design, and both are driven at 4.25 GHz and have a 0.15 T magnetic field for electron cyclotron heating of the plasma. The total mass of the I-COUPS is 9.95 kg, of which 2.57 kg is xenon propellant mass. Operation of the ion thruster consumes 40 W, while operation of two of the cold gas thrusters requires only 11.5 W electrical power. Cold gas thrusters each produce about 30 mN with 24 s specific impulse, while the ion thruster produces about 300  $\mu$ N at 1000 s specific impulse. The I-COUPS flexibility with shared propellant in an approximately 10 kg package on a 60 kg spacecraft means the propellant can be used in drastically different ways. For example it could be used for main propulsion ion thruster mode to provide about 430 m/s max delta-V, or as angular momentum for cold-gas attitude control providing at most 208 N-m/s, or for high-thrust cold-gas trajectory correction maneuvers providing about 42 mN (not  $2 \times 30 = 60$  mN due to non-axial orientation) for at most 9,800 s for



**Figure 7: Ion thruster and Cold gas thruster Unified Propulsion System (I-COUPS). A) Feed system schematic of Ref. [36] showing the common propellant tank for the ion thruster and cold gas thrusters, four of the eight cold gas thrusters are shown, B) model of the PROCYON spacecraft from Ref. [61] showing the location of the ion thruster and six of the eight cold gas attitude control thrusters, and C) vacuum test of the ion thruster integrated onto the PROCYON spacecraft, from Ref. [62]. Reproduced with author permission.**

a total impulse of 410 N-s. PROCYON was launched on 3 December 2014 as a secondary payload of the main Hayabusa-2 spacecraft.

The PROCYON spacecraft is clearly unique for a variety of reasons. First, it is the first small satellite to have both main propulsion and attitude control. Second, it is the first use of micropropulsion on an interplanetary orbit. Third, its I-COUPS propulsion is the first to use shared propellant between cold-gas and ion thruster propulsion. Finally, and perhaps most importantly, there are in-space data available on the performance and operation of the propulsion system [61-63]. The high pressure xenon gas system at 7.75 MPa has performed well throughout the launch and space operations. The reaction control cold-gas thrusters have been successfully used to unload the wheel momentum accumulated by the ion thruster operation and solar radiation pressure. Additionally, the cold-gas thrusters have been used for several tests of translational thrust with duration as long as 600 s. Operation of the ion thruster showed in-space thrust nearly identical to ground based measurements of 300  $\mu$ N, however, some anomalous phenomena were found. A leak was found in the ion thruster valve, there was an error in the control of the pressure regulation valve, the I-COUPS controller was found to occasionally freeze, and finally a gradually increasing high neutralizer voltage was measured. The leaking ion thruster valve necessitated the ion thruster to be operated at abnormally high flow rate, and eventually caused a short between the neutralizing and accelerating grids, ending the ion thruster operation altogether. The ion thruster was operated for a total of 223 h.

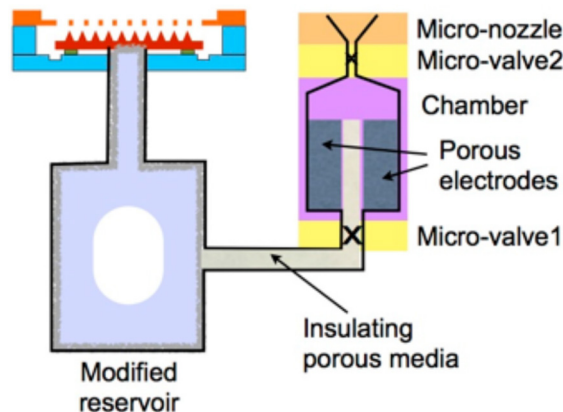
## 2. Cold/Warm-gas with Electro spray

Masuyama and Lozano explored an electro spray and cold-gas propulsion system fed by a common propellant tank [64], a schematic of which is shown in Figure 8. A common reservoir feeds both the electro spray thruster (left) and a cold gas thruster (right). The electro spray thruster is assumed to be the ion Electro spray Propulsion System (iEPS) developed at MIT, wherein ionic liquid propellant is fed to and transported through a porous substrate to an array of pointed emission tips. A strong electric field applied between the liquid in the substrate and an extraction electrode creates ion emission sites (Taylor cones) at the many pores of the substrate. The same propellant reservoir feeds a cold-gas thruster, wherein propellant was assumed to be electrolytically decomposed to high pressure and expanded through a converging-diverging nozzle. They considered common ionic liquid electro spray propellants [EMI][IM] and [EMI][BF<sub>4</sub>]. The performance and operation of these propellants in existing porous emitter array electro spray thrusters (i.e., iEPS) is well documented, but their potential as cold-gas propellants was unknown, and was therefore the focus of the work. Experimental results on the composition of electrolytically generated gases using a residual gas analyzer showed numerous possible hydrocarbon and nitrogen functional groups. Additional experiments measured the pressure change in a closed volume due to electrolysis of the propellant. In both cases where the initial pressure was moderate vacuum or atmospheric pressure, no significant pressure increase was measurable. However, the propellants did exhibit significant color change, from clear to orange to brown, and increases in conductivity due to electrolysis, which is indicative of electrochemical changes within the liquid. Cold-gas thruster performance predictions, based on assumed electrolytic propellant behavior since experiments were inconclusive, suggest that 65 s specific impulse at almost 1 mN of thrust is possible. The overall conclusion was that incorporating the cold-gas thruster would indeed enhance the available performance of the propulsion system beyond what is achievable with iEPS alone.

## B. Monopropellant with Electrothermal Propulsion

### 1. Monopropellant with Arcjet

A monopropellant catalytic hydrazine thruster with a 1 kW arcjet for small satellite applications was investigated by Haas and Holmes [55], and reviewed in detail above (i.e., Figure 3).



**Figure 8: Electro spray-cold gas multimode system wherein a common electro spray liquid propellant reservoir feeds both thrusters. To operate in the cold-gas thruster, the propellant is electrolytically decomposed. Reproduced from [63] with author permission.**

## 2. Monopropellant with Plasma-heated Gas

Wada et al. have explored hydroxylammonium nitrate (HAN)-based combustion thrusters with an integrated plasma discharge, which may enable dual mode operation [65,66]. They use the propellant SHP163 developed by the Japan Aerospace Exploration Agency (JAXA) in a 1-N-class thruster with plasma-based ignition system. The combustion chamber is cylindrical and the swirl injector serves as the cathode electrode, while an anode ring electrode is placed 3.5 mm downstream. The method of operation is typically as follows. First, argon gas ( $\sim 0.15$  g/s) flows into the chamber, then electrical power is applied between the electrodes generating an argon plasma discharge. Next, SHP163 propellant is injected, subsequently ignited and exothermically decomposed to high temperature gaseous products that may be exhausted through a nozzle. With a single-hole injector, at an SHP163 feed pressure of 800 kPa, a maximum thrust of 0.37 N was achieved with a power consumption of 527 W and an SHP163 mass flow rate of 0.34 g/s, in conjunction with a C-star efficiency of 98%. The authors theorize that the system can operate in a dual mode configuration, specifically a low thrust electrothermal mode when only high temperature plasma-heated argon gas is exhausted, and then high thrust combustion mode when the monopropellant is also injected. At an  $L^*$  of 508 mm, a C-star efficiency of 88% was obtained with only the argon plasma.

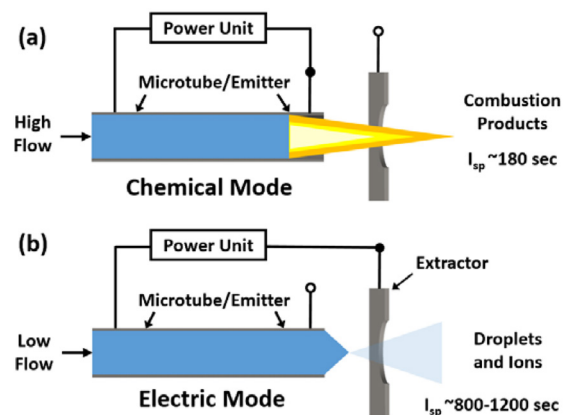
## C. Monopropellant with Electrostatic Propulsion

### 1. Monopropellant with Electrospray

Another possible combination of chemical and electric propulsion is a monopropellant catalytic combustion thruster and electrospray thruster. This type of combination was shown schematically as system 6 and 7 in Figure 1. In the configuration of system 6 of Figure 1, separate state-of-the-art combustion and electrospray thrusters are fed from a single propellant tank. More recently, a monopropellant-electrospray thruster concept has been proposed that integrates the combustion thruster and electrospray thruster together, and still shares a common propellant tank, and illustrated by system 7 in Figure 1 [67]. This thruster concept is further illustrated in Figure 9. The thruster consists of a multiplexed array of microchannels or microtubes ( $\sim 100$   $\mu\text{m}$ ). Further, the microtubes are coated or lined with catalyst material, e.g., platinum. If the array of microtubes is heated, propellant exothermically decomposes within the microtube and is exhausted as high temperature gaseous decomposition products (Figure 9a). If instead, a potential difference is applied between the array of microtubes and a downstream extractor electrode, ions and droplets of the propellant are electrostatically extracted and accelerated to high speed (Figure 9b).

A major challenge with these monopropellant-electrospray concepts is the propellant, which must be electrosprayable and chemically reactive. Much recent research has centered on ionic liquid-based green energetic liquid propellants that may be capable of operating in both modes. Donius and Rovey assessed ionic liquid-based multimode spacecraft propulsion [33]. Their work surveyed ten different potential ionic liquids for application as both chemical propulsion and electric electrospray propulsion propellants. Specifically, they predicted chemical propulsion performance when these ionic liquids are used as fuels paired with common oxidizers, and compared the results with traditional chemical propellant fuels hydrazine and UDMH. In terms of specific impulse, the novel ionic liquid combinations performed lower by about 1-4%. But when storability was considered, the density specific impulse of the ionic liquids performed better, especially when paired with hydroxylammonium nitrate (HAN) oxidizer. In some cases, an improvement of 25% in the density specific impulse was predicted. The same ionic liquid fuels and oxidizers were considered as propellant in an electric electrospray system.

Fonda-Marsland and Ryan investigated thirteen ionic liquids for potential suitability as chemical and electric electrospray propellants [68,69]. They compared propellants based on electrospray liquid parameters (surface tension, conductivity, viscosity) and chemical monopropellant parameters (plume composition, atomic combination and dissociation enthalpy). They suggest that ideal propellants for this concept will have no graphite in the decomposition products. Further they identify the tradeoffs between selecting a propellant that is best for both modes, in terms of maximizing the liquid surface tension



**Figure 9: Schematic example of a microtube-electrospray propulsion system that is both a chemical microtube thruster and electrospray thruster fed with a common green ionic liquid propellant.**

and conductivity for electrospray,  $\sqrt{\gamma\kappa}$ , versus maximizing the enthalpy to molecular weight ratio for chemical mode,  $\sqrt{\Delta H/MW}$ . They identify ethylammonium formate ( $[EA][(C_1)O_2]$ ) as a promising candidate.

de la Mora has highlighted ionic liquids (ILs) with a nitrate anion as potential candidates for multimode monopropellants. Specifically, he identified  $[Emim][NO_3]$  and ethyl ammonium nitrate (EAN) as energetic ILs potentially suitable for electrospray extraction and/or chemical decomposition [70]. EAN has shown an ability to be ignited at pressures in excess of 50 atmospheres, despite its oxygen deficiency, and an ability to be electrosprayed in vacuum [71]. EAN is formed from a protonated reaction between an amine and an acid. ILs from such a reaction are of interest for electrospray [70]. However, the reversibility of the reaction poses volatility concerns for operations within vacuum.

Berg and Rovey [72,73] investigated imidazole-based ionic liquids as potential candidates for multimode chemical monopropellant and electrospray propulsion. They investigated these liquids as standalone monopropellants, and as a fuel component in a binary mixture with hydroxylammonium nitrate (HAN). Results showed that standalone monopropellant performance would be poor, but HAN-based mixtures are promising, with predicted specific impulse and density specific impulse on par with state-of-the-art green monopropellants and state-of-the-art electrospray propellants. Follow-on research synthesized and evaluated the decomposition of these mixtures on heated catalyst beds [74,75]. Specifically, tests included mixtures of fuels 1-butyl-3-methylimidazolium nitrate ( $[Bmim][NO_3]$ ) and 1-ethyl-3-methylimidazolium ethyl sulfate ( $[Emim][EtSO_4]$ ) with oxidizer HAN. Comparison of pressure rise rate and light emission for these mixtures with standard hydrazine monopropellant indicated strong decomposition of these mixtures on heated rhenium catalyst (160 °C). Follow-on research has focused on the  $[Emim][EtSO_4]$ -HAN mixture combination. Specifically, decomposition studies of the  $[Emim][EtSO_4]$ -HAN mixture have shown platinum to be an excellent catalyst [76]. Recent studies by Berg [77] have also shown that this mixture can be decomposed within a sub-millimeter heated platinum tube, that is, a microtube. These results suggest this propellant may be suitable in a microtube micropropulsion system.

Mundahl et al. explored the linear burning rate of potential multimode chemical-electrospray propellants [78-80]. Using pressure rise rate data in a pressurized microreactor, Mundahl et al. measured a burn rate for  $[Emim][EtSO_4]$ -HAN mixture of 22.8 to 26.5 mm/s at 1.5 MPa background pressure [79,80], which is very similar to other HAN-based chemical monopropellants. More recently Rasmont et al. correlated high-speed images of the propellant burning with the pressure rise rate showing excellent agreement between the two methods, and showing a linear burn rate of 10-40 mm/s for the mixture at a pressure of 1 MPa (145 psia) [81]. Mundahl et al. also explored custom designed propellants for multimode chemical-electrospray propulsion [78]. Specifically, they investigated choline nitrate – glycerol as a fuel component in a binary mixture with oxidizers ammonium nitrate (AN) and HAN. The AN-based propellants required significantly higher preheat temperatures to initiate decomposition and were excluded from further tests. Mixtures of choline nitrate-glycerol with HAN had predicted performance 10% higher than  $[Emim][EtSO_4]$ -HAN mixture. However, the choline nitrate-glycerol also required 26-88% higher decomposition temperature.

The combination of  $[Emim][EtSO_4]$ -HAN has also been shown to be a promising electrospray propellant. Berg and Rovey [82], and Wainwright et al. [83-85] have shown stable electrospray of this mixture. The relatively high flow rates of Berg [82] suggested performance in the electric mode of around 412 s, but also hypothesized that lower flow rates would enable higher specific impulse operation. The more recent work of Wainwright et al. has shown for the first time the presence of HAN species in an electrospray plume [85]. Specifically, this work identified the presence of both ionic and covalent (proton-transferred) forms of HAN, as well as ion swapping between HAN and the  $[Emim][EtSO_4]$  in the electrospray.

The interest in ferrofluids as possible electrospray propellants motivated Berg et al. to investigate the chemical decomposition of ionic liquid ferrofluids for multimode propulsion [86]. Results indicated that addition of 10-30% by mass iron nanoparticles to  $[Bmim][NO_3]$  and  $[Emim][EtSO_4]$  enabled rapid decomposition of those liquids at 30% lower preheat temperature. This could be advantageous for lower preheat power requirements. However, it was concluded that ferrofluids may not be conducive for electrospray multimode application. Specifically, ferrofluid electrospray propellants require at least 50% addition of iron nanoparticles to exhibit the Rosensweig instability that is fundamental to this application. Further, oxidation reaction with the iron nanoparticles causes the liquid to decompose relatively quickly, on the order of 24 h.

## D. Mono/Bipropellant with Electromagnetic Propulsion

### 1. Mono/Bipropellant with Pulsed Inductive Thruster

Pulsed inductive electric propulsion techniques use an inductive coil to ionize and accelerate the working fluid (gaseous plasma). Because an induced electromagnetic field is used, the electromagnetic components of the thruster can be shielded and prevented from contacting the working fluid. This makes an inductive thruster amenable to operation with chemically-reactive gases that would normally degrade and erode metallic electrodes. Within the context of multimode propulsion, this means inductive thrusters can be operated with the gaseous decomposition products of mono- and bipropellants. The decomposition products of these propellants are often chemically-reactive polyatomic molecular species. For instance, monopropellant hydrazine ( $\text{N}_2\text{H}_4$ ) decomposes into nitrogen ( $\text{N}_2$ ) and ammonia ( $\text{NH}_3$ ), a fraction of which may also be dissociated into monatomic nitrogen and hydrogen. Green ionic liquid-based propellants, such as AF-M315E and LMP-103S, decompose into water vapor ( $\text{H}_2\text{O}$ ), carbon dioxide ( $\text{CO}_2$ ), carbon monoxide ( $\text{CO}$ ), and may even contain some hydrogen sulfide ( $\text{H}_2\text{S}$ ). The decomposition temperatures of these propellants are high enough that a fraction of these molecules will also be dissociated into their monatomic constituents. Oxidized species are known to be highly reactive and degrade/erode metallic electrodes exposed to the gas. Hence one of the main motivating factors for using the pulsed inductive technique with these reactive molecular gases (decomposition products) is the absence of metallic electrodes in contact with the working fluid. An example multimode system may consist of a common hydrazine propellant tank that feeds a catalytic combustion thruster and an inductive thruster. The hydrazine (liquid) would need to be decomposed into gaseous form before being injected into the inductive thruster.

The pulse inductive thruster (PIT) developed in the late 1980's and early 1990's was tested with both nitrogen and ammonia molecular gases (simulating hydrazine decomposition products), and tested directly on hydrazine decomposition products. While the motivation at that time was not necessarily multimode propulsion, but rather the flight heritage of hydrazine propellant, the results illustrate the potential of inductive thrusters to serve as the EP component of a multimode system. Polzin reviews the state-of-the-art of the PIT [87]. Multiple versions of the thruster were tested, with typical results highlighted here. With hydrazine and its decomposition products, at specific energies of 200-1200 J/mg, the measured specific impulse linearly increased from 1000-3500 s. Typical pulse energies were 2-5 kJ/pulse and efficiency on ammonia was as high as 50%. It is interesting to note that, while the PIT has been tested with inert gases such as argon and xenon, it has the highest performance ever measured when operating with hydrazine decomposition products.

Berg and Rovey investigated an integrated monopropellant inductive plasma thruster that used a conical induction coil as the diverging nozzle [88]. The concept therefore has both shared propellant and a shared thruster geometry. Regardless of the integrated thruster geometry, results indicated that having shared propellant between both modes will be beneficial in terms of mass savings and system flexibility. However, integrating a conical induction coil as the chemical mode nozzle may not be advantageous. This is due to the conflicting trade-offs between nozzle geometry and thruster performance. In chemical mode, high performance is achieved for relatively low nozzle divergence half-angles, e.g., the standard 15 deg conical nozzle, whereas for inductive thrusters the highest efficiency is achieved with large divergence angles, e.g., a 90 deg flat plate. Specifically, results indicated that integrating the thruster geometry is not beneficial for divergence half-angles of 20-38 deg. Benefits in terms of mass savings compared with state-of-the-art separate thrusters may be possible with a nozzle/inductive coil half-angle greater than 55 deg., but combustion thrusters with such a large diverge angle have not been investigated.

### 2. Mono/Bipropellant with Field-reversed Configuration Thruster

Another type of inductive electric propulsion system is the field-reversed configuration (FRC) thruster. Like the PIT, it may be considered to operate on the gaseous decomposition products of a mono- or bipropellant. While molecular decomposition products have not been specifically investigated, there has been significant previous research on other molecular gases, including air [89], the Martian atmosphere ( $\text{CO}_2$ ), and water [90]. Results with  $\text{CO}_2$  showed performance of 500-4000 s specific impulse. In addition to directly using these gases in the pre-ionization, formation, and acceleration of the FRC plasma, it is also possible to ionize and entrain neutral gas [91]. Specifically, as an FRC plasma is accelerated and expelled from the thruster, it can be used to ionize and entrain neutral gas through charge-exchange collisions. This has the benefit of enabling molecular gases to be entrained into the FRC, and can also be used to tailor and control the ejected FRC mass and energy, and corresponding propulsion performance.

### 3. Mono/Bipropellant with Pulsed Plasma Thruster

Work by Thrasher et al. has investigated a novel HAN-based green electric monopropellant (GEM) in a pulsed plasma thruster [92]. In a chemical catalytic thruster, the GEM is predicted to have higher specific impulse and density



specific impulse than other green monopropellants such as LMP-103S and AF-M315E. Further, it has lower volatility, making it less toxic and easier to handle. The research presented in [92] focused on operation of the GEM in a pulsed plasma thruster (PPT) electric propulsion mode. It is interesting to note that this is a liquid propellant being used in a PPT. Traditionally PPTs have used solid Teflon propellant. Experiments focused on measuring thrust of a prototype liquid PPT (LPPT) operating with GEM. The thruster was operated at 40 J per pulse with a charging voltage of 300 V. Impulse bits of 300-400  $\mu\text{N}\cdot\text{s}$  were measured, and, when the 2% water content of the propellant is accounted for, the predicted specific impulse was 200-400 s. Increasing charging voltage to 390 V at an energy of 38 J increased the impulse bit to 800-1000  $\mu\text{N}\cdot\text{s}$ .

## E. Solid Propellant with Electromagnetic Propulsion

### 1. Solid Propellant with Pulsed Plasma Thruster

Another multimode concept proposed by Glascock et al. [93,94] leverages advances in electric solid propellants. Electric solid propellants (ESPs) are solid chemical propellants that ignite and decompose only when electric power is applied at sufficient current and voltage [95-99]. The decomposition is a highly exothermic process that generates hot gas at a burn rate that can be throttled by varying the applied current. Removal of the voltage and current extinguishes the reaction, which may be restarted by reapplication of electric power. This behavior of ESPs may facilitate a dual mode propulsion system using the solid propellant. Mode one is a high thrust chemical mode where direct current electric power is applied to incite pyroelectric gas generation. This gas is accelerated gas-dynamically through a conical nozzle to generate thrust like in a solid rocket motor. Mode two uses a second circuit configuration to operate like a coaxial ablation-fed pulsed plasma thruster (APPT). Coaxial APPTs and solid rocket motors can be designed with similar tradeoffs with respect to propellant grain and nozzle geometry. Thus, this combination of modes favors a shared geometry and shared propellant multimode design.

Electric solid propellants are known to the chemical rocket community [95-99], so recent work has focused application of these propellants in the electric propulsion mode, specifically as a propellant in an APPT. Work by Glascock et al. has focused on the ESP called HIPEP [95,100]. PPT microthrusters of HIPEP have been shown to behave very similar to traditional Teflon propellant PPTs. Specifically, plume properties of a HIPEP microthruster have been shown to have temperature of 1.7 eV and plasma density of  $10^{11}\text{-}10^{14}\text{ cm}^{-3}$ , and have exhaust velocity of 1500-1600 m/s [101,102]. These microthrusters were shown to have a non-equilibrium plume consisting of high temperature electrons and low temperature neutrals, and to be operating in a predominantly electrothermal mode (as opposed to higher efficiency electromagnetic mode). Additionally, these microthrusters were shown to exhibit significant late-time ablation similar to traditional Teflon PPTs, with an estimated 50% of the mass loss occurring at late time and contributing negligible thrust [103]. More fundamental arc ablation mass loss experiments have shown the HIPEP to ablate with over double the specific ablation (mass loss per pulse per energy per pulse) of traditional Teflon, and the differences have been attributed to the fundamental thermochemistry of the HIPEP [93,104,105].

More recent investigation by Glascock et al. [106] found that the impulse bit (impulse-per-pulse) of a HIPEP-fueled coaxial APPT laboratory thruster is  $100\pm 20\text{ }\mu\text{N}\cdot\text{s}$  with 5 J initially stored energy. This impulse bit increased linearly with stored energy by about  $30\text{ }\mu\text{N}\cdot\text{s}/\text{J}$  up to  $575\pm 20\text{ }\mu\text{N}\cdot\text{s}$  at 20 J. Further, tests with Teflon showed little change (<10%) in impulse bit between propellants, but a significant reduction of specific impulse. Using the same device, HIPEP specific impulse was measured to be 225 s, which was only 50% of specific impulse when using Teflon as propellant. This performance reduction was attributed to the increased specific ablation of HIPEP. The authors also surmised that moisture absorbed by the hygroscopic propellant evaporated early in the impulse tests and skewed mass loss measurements artificially high on a per-pulse basis. If true, the actual specific impulse of HIPEP in the coaxial APPT may be higher if no water is absorbed into the propellant.

## V. Conclusions

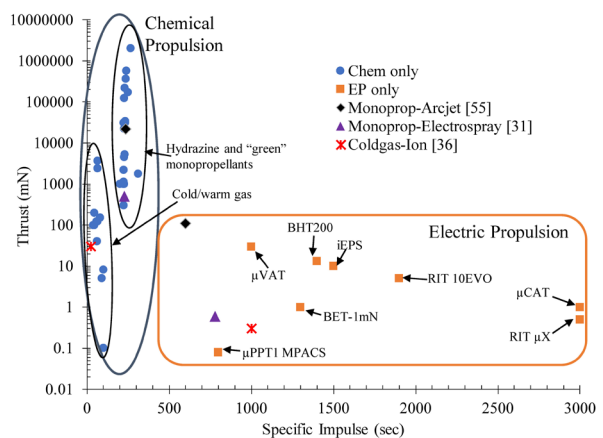
In-space propulsion of medium to large spacecraft (>500 kg) in traditional orbits (>200 km) using stand-alone electric or chemical propulsion is well-developed. Chemical bi- and monopropellant performance (450, 250 s  $I_{sp}$ , respectively) is already close to theoretical maximums, and has not significantly changed in decades. Further, the toxicity, volatility, and general safety hazard of chemical propellants has been reduced with the advent of high performance, green propellants, e.g., LMP-103S and AF-M315E. In-space chemical propulsion is and has been readily available from numerous global suppliers for decades. At the same time, electric propulsion for these satellites is also well-developed. Hall-effect thrusters in the 0.5 to 5 kW range are now readily commercially available from numerous global suppliers, and will continue to be used to fulfill commercial industry and government needs for the

foreseeable future. For very high energy missions of typical interest to academic and government customers, DC and RF gridded ion thrusters are readily commercially available from numerous suppliers. There is interest in even higher power spacecraft with higher power stand-alone electric propulsion (e.g., 50 kW). Clusters of Hall thrusters or geometry variations of Hall thrusters (e.g., nested channels) are clear front runners to fulfill this need because they are built on decades of in-space demonstration and flight heritage. Alternative high-power EP concepts like the field-reversed configuration (FRC) and VASIMR may be capable of also fulfilling this higher power need. But these concepts must overcome the steep TRL climb to in-space demonstration and proven flight. Modification of already flight-proven commercially-available technology (i.e., HETs) is already possible and a promising path forward. It is likely that, for these medium to large spacecraft, existing Hall thrusters, or their higher power variants, will fulfill most government and commercial industry needs for the next 10-20 years.

*Where then lies the future of in-space propulsion?* We conclude from our review that multimode propulsion is emerging as an enabling technology that promises enhanced capabilities for spacecraft and space missions, and can therefore play an important role in the future of in-space propulsion. Specifically, multimode propulsion has potential to provide unprecedented flexibility and adaptability to spacecraft, as well as potential mass savings and cost savings. And these benefits extend to both medium and large spacecraft, as well as small satellites. Multimode propulsion is closely related to hybrid propulsion (separate high- and low-thrust propulsion systems on-board the same spacecraft), and the benefits of that approach, which can include enhanced payload transfer rate, faster trajectories, and lower initial spacecraft mass, naturally extend to multimode. However, the key attribute of multimode propulsion is shared propellant, and this provides flexibility. Even if a multimode system has lower performance than that of a single mode thruster, utilizing a single shared propellant has the potential to enable a large mission design space. This flexibility and adaptability to perform well (not the optimized best, but still excellent) over a wide range of mission scenarios may be beneficial, especially within the burgeoning small satellite community.

Small satellite propulsion is still in its infancy and multimode propulsion may have the greatest impact on small satellite capabilities. Small satellite propulsion is still, clearly, an active and growing area of major global attention. Multiple recent review papers highlight the numerous chemical and electric propulsion technologies being explored for these spacecraft [107-109]. Also recently, The Aerospace Corporation released a propulsion technologies survey, in which they compiled over 100 micro, nano, pico, and Cube-satellite propulsion systems including the technology readiness level and performance of each system [110]. We have plotted the performance of those propulsion systems whose TRL is 7 or higher in Figure 10. Both chemical (cold/warm gas and monopropellant) and electric propulsion systems have been developed. Additionally in Figure 10 we plot the performance of three multimode propulsion systems previously described. Specifically, the monoprop-arcjet [55] (section III.B.), monoprop-electrospray [31] (section III.B.), and coldgas-ion (I-COUPS) [36] (section IV.A.1.) performance are shown. The multimode systems have performance points in both the chemical and electric propulsion ranges. The wide array of challenges with integrating meaningful propulsion, not only limited by mass and volume constraints, but also secondary payload restrictions (e.g., pressure, energetic materials restrictions), are multifaceted and will continue to be explored by researchers around the globe for the foreseeable future. Of all classes of spacecraft, small spacecraft may benefit the most from multimode propulsion, especially if the chemical and electric propulsion hardware can be integrated together. For small spacecraft, propulsion system dry mass is typically a large fraction of the overall propulsion system mass. Multimode concepts are being investigated that integrate together, not only the propellant and feed system, but also the chemical and electric propulsion hardware, such that it may be possible to provide a small satellite with both chemical and electric propulsion with minimal increase in propulsion system mass.

Numerous multimode concepts have been explored and documented in the literature. Concepts combining cold gas, monopropellant, bipropellant, and solid chemical propulsion with electrothermal, electrostatic, and electromagnetic electric propulsion have all been investigated. Electrostatic electric propulsion paired with monopropellant chemical propulsion has perhaps received the most recent attention. Notably absent from the



**Figure 10: Some of the small satellite chemical and electric propulsion systems identified by The Aerospace Corporation survey [110] along with three multimode propulsion systems.**

reviewed concepts is Hall and gridded ion thrusters operating with monopropellant decomposition products. While molecular gases have been explored as Hall thruster propellants [111-115], these previous works focused on atmospheric gasses (oxygen, nitrogen) as opposed to chemical propellant species. Main challenges with any of these reactive molecular gases will be decreased thruster performance and lifetime.

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