ENGINEERING REALITIES OF DEBRIS MITIGATION

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ABSTRACT

Global orbital debris mitigation guidelines were established in the mid-1990s based upon a simple framework of (1) limiting the amount of debris created by each launch/mission, (2) preventing explosions and collisions, and (3) reducing the amount of time space hardware is allowed to reside in Earth orbit after its mission is completed.

The sequence of initiatives that flowed from this activity were critical to establishing the philosophy of responsible behavior in space but were also tempered by concerns of imposing burdensome requirements on spacecraft designers and space operators. That concern was very relevant for that time when there was limited debris on orbit, few countries actively operating in space, space technology was in early stages of maturity, and few commercial ventures depended on reliable space systems.

Over the last 25 years, space has become critical to every aspect of our national security and daily lives and our ability to build capable space systems has also dramatically improved. However, the guidelines to mitigate debris and avoid collisions, which could threaten our ability to operate our space systems reliably, have not kept pace with these technological changes.

This paper shows that the engineering realities now are such that two key components of the mitigation guidelines can, and should, be immediately, and substantially, updated by use of electric propulsion systems:

1. Collision avoidance capability can be incorporated in all spacecraft operating over 400 km in altitude.
2. The 25-year rule can be reduced to 5-year (or even 1-year) rule with minor impact on design and operations.

1 TECHNICAL APPROACH

This paper presents engineering analysis related to collision avoidance and post-mission disposal (PMD) with uncontrolled reentry of three classes of spacecraft (15 kg (6U) cubesat, 300 kg smallsat, and 2,000 kg satellite) operating up to a 1,500 km altitude in low Earth orbit (LEO). Each will incorporate a five-year collision avoidance capability (i.e., ~1 m/s) and a fuel reserve sufficient for supporting a minimum of a 5-year PMD threshold (i.e., move to 500 km circular orbit immediately upon mission completion). Analysis of additional requirements for a 1-year rule (i.e., move to 400 km circular orbit) is also provided.

This analysis examines the size, weight, and power (SWAP) requirements; the fuel needed to support the two new mitigation guidelines; and other spacecraft design benefits and liabilities incurred by satisfying these two debris mitigation requirements. Off-the-shelf, flight-proven electric propulsion (EP) systems are identified in this assessment highlighting that more stringent and more responsible debris mitigation guidelines can easily be satisfied.

2 DEBRIS MITIGATION GUIDELINES

The Orbital Debris Mitigation Standard Practices (ODMSP) [1] codifies the standards and guidelines used by Government space operators in the United States. While the debris mitigation guidelines were established in the 1990s, the first ODMSP was issued in 2001, and updated in 2019.

Highlights from the guidelines are:

- The goal of the ODMSP is to limit the generation of new, long-lived debris by the control of debris released during normal operations, minimizing debris generated by accidental explosions, the selection of a safe flight profile and operational configuration to minimize accidental collisions, and PMD of space structures.
- Spacecraft and upper stages should be designed to eliminate or minimize debris released during normal operations. Each instance of planned release of debris larger than 5 mm in any dimension that remains on orbit for more than 25 years should be evaluated and justified. For all planned released debris larger than 5 mm in any dimension, the total debris object-time product in low Earth orbit (LEO) should be less than 100 object-years per upper stage or per spacecraft. The total object-time product in

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LEO is the sum, over all planned released objects, of the orbit dwell time in LEO.

- In developing the design of a spacecraft or upper stage, each program should demonstrate, via commonly accepted engineering and probability assessment methods, that the integrated probability of debris-generating explosions for all credible failure modes of each spacecraft and upper stage (excluding small particle impacts) is less than 0.001 (1 in 1,000) during deployment and mission operations.
- All on-board sources of stored energy of a spacecraft or upper stage should be depleted or safed when they are no longer required for mission operations or PMD.
- In developing the design and mission profile for a spacecraft or upper stage, a program will estimate and limit the probability of collision with objects 10 cm and larger during orbital lifetime to less than 0.001 (1 in 1,000).
- Spacecraft design will consider and limit the probability to less than 0.01 (1 in 100) that collisions with micrometeoroids and orbital debris smaller than 1 cm will cause damage that prevents the planned PMD.
- The probability of successful PMD should be no less than 0.9 with a goal of 0.99 or better.

Two key points from the 2019 update are: (1) collision avoidance capability is not required (simply a written explanation showing that the probability of collision for hardware is less than 1/1000 over the entire orbital lifetime) and (2) the PMD requirement (i.e., the 25-year rule) stayed the same as established in the early 1990s, in the original ODMSP. Complicating the potential drive to reduce the PMD time threshold from 25 years to something much shorter is that the current global compliance rate to the 25-year rule is less than 50%. [2] However, poor adherence to a requirement should not be the rationale that prevents making the requirement more stringent.

3 OFF-THE-SHELF CHEMICAL AND ELECTRIC PROPULSION SYSTEMS

As a starting point for the argument that current debris mitigation guidelines could easily be made more stringent, propulsion systems that can easily integrated into a range of satellite systems are reviewed. Six off-the-shelf propulsion systems are analyzed and system features of each are detailed in Tab. 1. Four systems are EP systems, and two are chemical systems, one monopropellant system and one bipropellant system.

EP systems have been flying since the 1980s, however, it has only been recently that there has been widespread use of EP by satellite designers and manufacturers. There

Table 1. The six propulsion systems used in this assessment have proven to be highly reliable or are in the process of being space-qualified (ACE-Max and Accion).

System Name	Type of System	Target Satellite	Mass, kg (Power Processing Unit [PPU] and harness for EP)	Power Requirement, W
Aerojet Rocketdyne XR-5	Hall Thruster	2,000 kg class	60	3000
Apollo Fusion ACE-Max	Hall Thruster	300 kg class	20	1400
Enpulsion	FEEP	15 kg class	0.7	33
Accion TILE	Electrospray	15 kg class	1.4	33
AR MR-103G	Monoprop	All	1.3 (four thrusters assumed for 2,000 and 300 kg satellites); 0.7 (two thrusters assumed for 6U cubesat)	22.5
AR R1-E	Biprop	2,000 kg and 300 kg classes	2	36

has been a marked increase in their use for collision avoidance, stationkeeping (primarily in geosynchronous orbit (GEO)), and orbit acquisition, including large GTO to GEO transfers. Several types of EP systems are flying today on a variety of government and commercial satellites.

Typically, EP systems use a single working fluid as a propellant and require power from the spacecraft to ionize and accelerate the propellant. EP systems' ability to perform at different power levels make them more tailorable than chemical systems for different de-orbit maneuver scenarios.

While six systems were studied, for purposes of the PMD calculations, the performance assessment (i.e., propellant mass required) can be performed on three general families of specific impulse (I_{sp}): monopropellant at ~230 s; bipropellant at ~280 s; and electric thrusters at ~1,800 s.

Another parameter of importance to the analysis is the power available to the EP system. Since electric thrusters use power to achieve the thrust and I_{sp} they generate, power can be thought of as a resource just as propellant is.

The more power available, the higher the thrust that can be achieved and, therefore, the shorter the time to complete the maneuver. This also requires the addition of dedicated electronics to process the spacecraft power and provide it at the voltages and currents required (*i.e.*, a Power Processing Unit, PPU).

Historically, monopropellant systems based on hydrazine propellant are the most commonly used satellite propulsion, dating back to the 1960s. It should be noted that while hydrazine was used in this analysis, the authors are aware that this type of propellant is being phased out for environmental reasons but the new types of propellants are of similar specific impulse.

They are simple, proven systems with thrusters that have flown everywhere in the solar system and beyond (*e.g.*, Voyagers 1 and 2 launched in 1977 continue to operate today beyond the heliopause).

Bipropellant systems perform more effectively, but are more complex than monopropellant propulsion. Satellites typically use bipropellant systems for large maneuvers such as insertion from Geosynchronous Transfer Orbit (GTO) into GEO. The most common propellant and oxidizer combination is mono-methyl hydrazine (MMH) and nitrogen tetroxide (NTO).

Three types of EP systems are examined. Hall thrusters at power levels of 3 kW and 1 kW for the 2,000 kg and 300 kg satellites, respectively. Field Emission Electric Propulsion (FEEP) thrusters and electrospray thrusters were both considered for the 6U cubesat missions. These were selected because they are commercially available today.

4 TECHNICAL PERFORMANCE ASSESSMENT

The anticipated ΔV budget required for the three "typical" spacecraft classes are summarized in Tab. 2. The collision avoidance budget was derived from an anticipated 20 firings a year of 1 cm/s each for a five-year mission.

Orbit raising for the two smaller spacecraft was included since they would likely have to use a rideshare or be launched in a multi-satellite deployer that would require some thrusting to put them in their desired orbit (i.e., orbit acquisition).

The larger spacecraft would likely be placed directly into its desired orbit. Deorbit ΔV values are calculated based on moving the three satellite classes from 1,500 km to the worst case PMD scenario, 400 km (*i.e.*, 1-year rule).

The PMD propulsion requirement will depend on both the starting altitude of the target spacecraft and the objective orbit for PMD compliance, e.g. 650 km for a 25-year deorbit threshold.

The other major physical parameter to determine is the mass fraction for a given scenario which requires both the dry mass of the spacecraft and the propellant mass consumed for the operations.

The values are calculated using a mission analysis code for both high and low thrust cases for each of the chemical propulsion system types, as well as for the EP systems, which were all low thrust. The mass breakdown used for each class of spacecraft is shown in Tab. 3.

Table 2. The fuel budget for each class of satellite examined is depicted below with the maximum maneuver budget (i.e., move from 1,500 km circular orbit to ~400 km circular or ~1-yr orbital lifetime).

Spacecraft	Orbit Raising (ΔV, m/s)	Collision Avoidance (ΔV, m/s)	Post Mission Disposal (ΔV, m/s)	Total (ΔV, m/s)
15 kg, cubesat	20	1	561	341
300 kg, smallsat	20	1	561	341
2,000 kg	---	1	561	321

Table 3. The mass budget for the three classes of satellites examined in this engineering assessment establishes the baseline the mass fraction calculations.

		15 kg "Average" 6U Cubesat			300 kg "Average" Satellite			2000 kg "Average" Satellite		
			Electric		Chemical			Chemical		
		Chemical Monoprop	Electrospray	FEEP	Monoprop	Biprop	Electric Hall	Monoprop	Biprop	Electric Hall
BOL Power	W	22	33	33	336	336	1522	1861	1861	3261
Dry Mass Total	kg	11.3	12.2	11.4	225.7	225.7	245.3	1504.7	1504.7	1518.7
Payload	kg	2.4	1.4	1.8	58.9	58.9	47.2	410.5	410.5	406.4
Structure	kg	2.6	2.8	2.6	49.0	49.0	53.2	326.5	326.5	329.5
Thermal	kg	0.2	0.2	0.2	7.7	7.7	8.3	51.2	51.2	51.6
Power	kg	2.8	3.5	3.5	63.0	63.0	78.5	419.8	419.8	432.2
TT&C	kg	1.4	1.5	1.5	16.9	16.9	18.4	112.9	112.9	113.9
ADCS	kg	1.3	1.4	1.3	18.1	18.1	19.6	120.4	120.4	121.5
Propulsion	kg	0.6	1.4	0.7	12.2	12.2	20.0	63.4	63.4	63.4
Total Loaded Propellant	kg	3.7	3.6	2.8	74.3	74.3	54.7	495.3	495.3	481.3

The collision avoidance (CA) fuel budget is very small, less than a kg for all mass classes, and less than 10 g for cubesats. Therefore, if a propulsion system is added to a satellite in order to execute orbit raising and/or PMD, the use of this system for CA during satellite operations presents very little engineering burden. It can also be seen that the PMD requirement is the primary driver for fuel which will, in turn, determine the mass fraction. The resultant propellant utilization for each type of satellite, including for PMD is shown in Tab. 4.

We examined how much of an effort it is for the propulsion system (i.e., ΔV needed) to move the three “typical” satellites selected from LEO to altitudes consistent with different PMD thresholds. For an intact derelict object, the orbital altitude that equates to orbital lifetimes of 25, 5, and 1 years are ~650, ~500, and ~400 km, respectively. Note that the Isp for the Electrospray and FEEP systems use their exact Isp values (2500 s and 1500 s, respectively) and not 1800 s. The variation in the lifetimes due to changing solar activity will increase for the lower altitudes. In reality, the average altitudes provided above will be higher for low periods of solar activity or lower for high periods of solar activity. [3]

These results are now compared against a 1999 review of the efficacy of the initial orbital debris mitigation guidelines, stating that it was important to keep the mass fraction to satisfy the 25-yr rule to 2-5% in order to not be too cumbersome. [4] This perspective is important in the next three figures below showing the propulsive requirement for moving the target satellites to altitudes consistent with the 25-year, 5-year, and 1-year rules.

Using the off-the-shelf propulsion described previously and the ΔV required to execute the PMD maneuvers, Fig. 1 provides insights on the mass fraction for PMD compliance for the 300 kg satellite example. The three lines correspond to 400 km, 500 km, and 650 km final orbit destinations which correspond to 1-year, 5-year, and 25-year compliance thresholds, respectively. Calculations are performed using the NASA Copernicus trajectory analysis tool. [5] Copernicus is a generalized

Table 4. The fuel budget for the three satellite classes and three operational modes plus residuals shows the illustrates the fuel efficiencies of the EP systems.

		15 kg "Average" 6U Cubesat			**300 kg "Average" Satellite**			**2000 kg "Average" Satellite**		
			Electric		Chemical			Chemical		
		Chemical Monoprop	Electrospray	FEEP	Monoprop	Biprop	Electric Hall	Monoprop	Biprop	Electric Hall
Orbit Raising (20 m/s)	kg	0.17	0.01	0.02	2.6	1.9	0.3	N/A	N/A	N/A
Collision Avoidance (1 m/s)	kg	0.008	0.001	0.001	0.13	0.09	0.02	0.9	0.6	0.1
Deorbit (Max 1500 --> 400 km)	kg	4.104	0.336	0.562	65.61	54.98	9.41	436.6	366.4	58.8
Residuals (2.5%)	kg	0.09	0.09	0.07	1.9	1.9	1.4	12.4	12.4	12.0

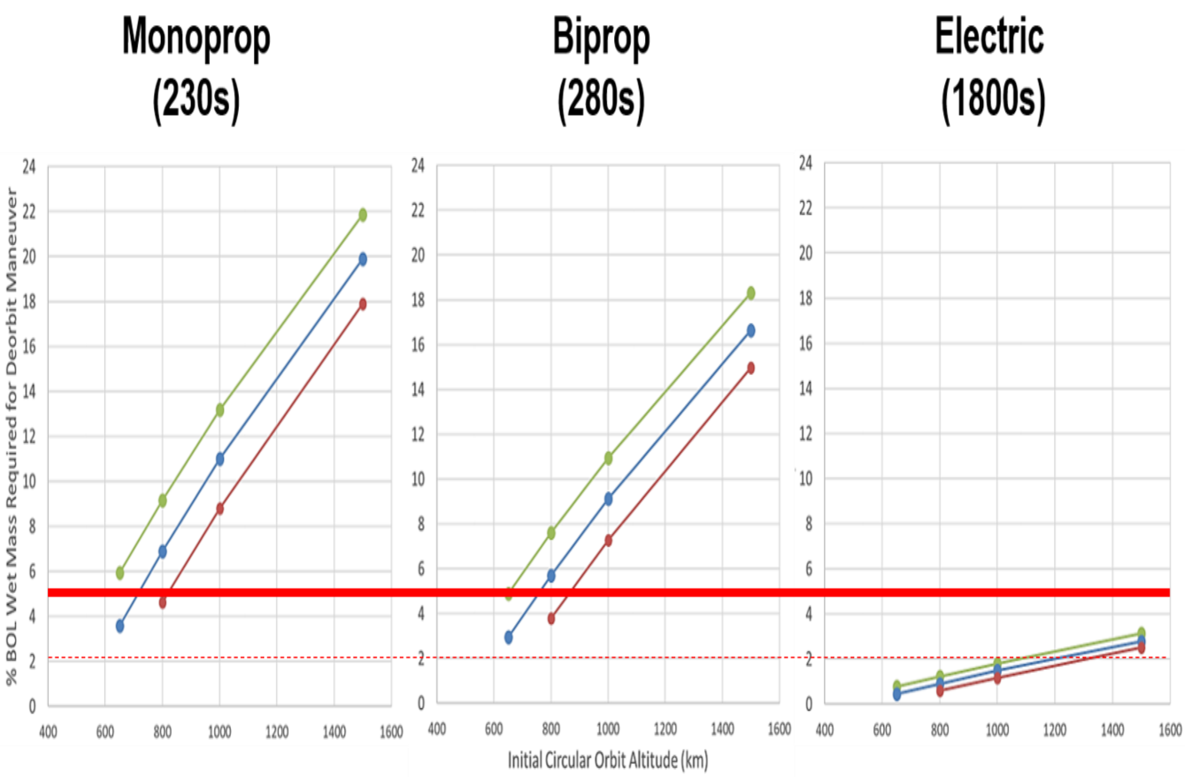

Figure 1. The mass fraction for a 300 kg satellite as a function of mitigation threshold (i.e., 25-yr [650 km], 5-yr [500 km], and 1-yr [400 km]) versus starting circular LEO altitude shows the resulting much lower mass fraction for the EP systems.

capable of handling both high thrust and low thrust orbit transfers.

The electric thruster can easily meet the 2-5% mass fraction challenge for all PMD thresholds while the chemical propulsion systems will not be able to meet the requirements for most orbits. However, if instead of recircularizing the satellite at the lower altitude for the chemical propulsion cases, the PMD maneuver strives to reduce the perigee such that the new orbital lifetime is equivalent to the PMD thresholds, significant fuel may be saved.

For the analysis described above we used the NASA Copernicus code for both Chemical and the EP cases. The analysts set the starting orbit and the final orbit and allowed for continuous burns to lower the altitude. Because both the Chemical and EP were low thrust / mass, the burns resulted in higher delta V than a traditional Hohmann transfer. Also, it probably is not a realistic operational strategy. So, we went back and ran a set of cases where we limited the chemical (bi-prop) burns to short arcs around apogee to reduce the perigee altitude until it decayed. The result is what is shown in the chart with the red line. The delta V and required propellant are reduced but it takes more time because the burn time is limited to a fraction of each orbit.

Fig. 2 shows the new calculations for this maneuver strategy using a continuous burn for the same three I_{sp} levels for the 300 kg satellite. As is evident from the results, this has a greater impact for satellites starting in higher initial orbits. Further optimization of burn profiles and orbit adjustments may reduce fuel requirements even more.

This process is developed for all of the satellite classes; Fig. 3 depicts all of the mass fractions for all scenarios listed in Table 4 to be deposited at 400 km (i.e., worst case 1-year rule). It should be noted that the modularized electric propulsion systems are the only viable options for 15 kg cubesats due to the SWAP issues and the fact that low thrust is perfectly fine for cubesats to be effective at changing their orbit.

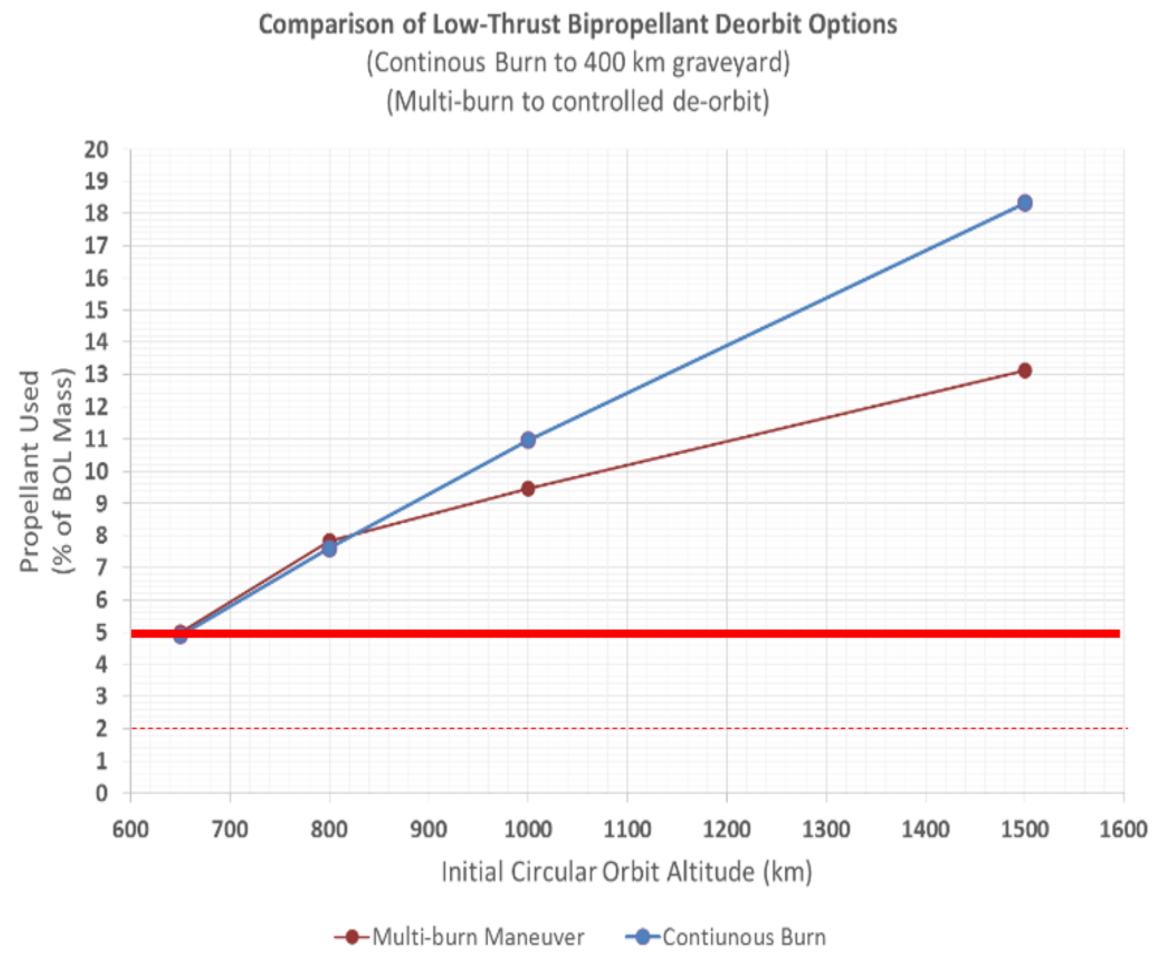

Figure 2. Different burn scenarios can reduce the propellant needed for the larger PMD maneuvers necessary for PMD operations of LEO satellites.

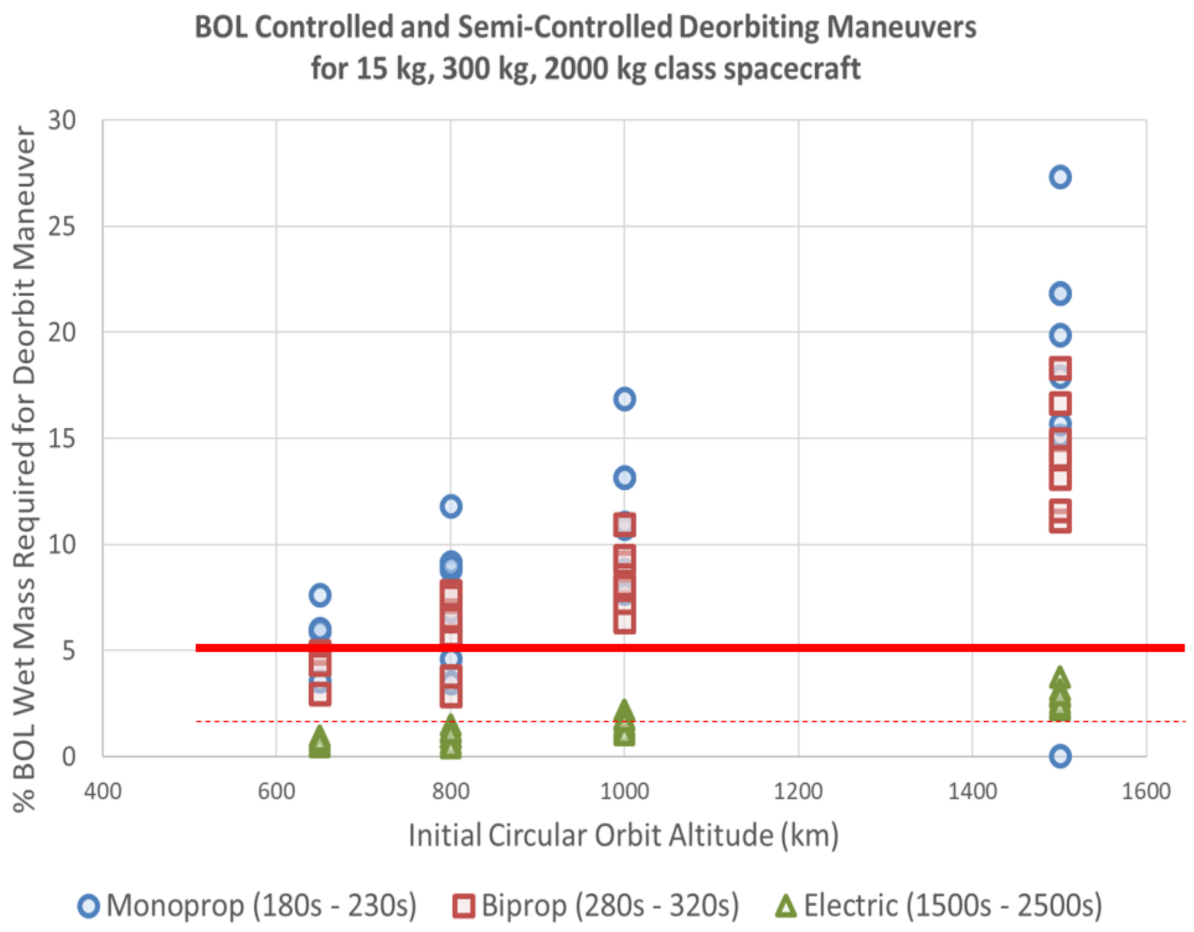

Figure 3. The summary figure for mass fraction for different source altitudes and satellite classes shows that EP systems can satisfy even the most stressing objectives.

While the suggestion of putting EP systems on everything from cubesats to large satellites may seem challenging and risky, there are many instances of successful, space-proven applications:

- UWE-4 1U cubesat: The University of Wurzburg 1U cubesat successfully used EP to perform a collision avoidance maneuver in 2020. [6]
- STRaND: 3U cubesat with plasma thruster with 2 m/s ΔV budget for CA. In 2012, this was the first cubesat that employed EP successfully.[7]
- IceEye: Iceye is obtaining interferometric data with a single satellite weighing less than 100 kilograms that follows a precise ground track, thanks to electric thrusters from Enpulsion of Austria. [8]
- TechDemosat: The first satellite to perform two collision avoidance maneuvers within the same week was a smallsat (157 kg TechDEMOSat) using an electric thruster in 2014. [9]
- OneWeb: Their 145 kg satellite uses electric Hall thrusters for orbit raising, constellation maintenance, CA, and planned PMD. [10]
- Starlink: 287 kg satellite uses EP system for orbit raising, constellation maintenance, CA, and PMD. [11]
- MEV-1: In 2020, MEV-1 used Hall effect thrusters to move itself to GEO over several months, then to rendezvous with Intelsat 901 in a graveyard orbit above GEO, where it became the first commercial satellite to dock with another satellite. [12]
- AEHF-1 was almost lost due to a failure of the apogee propulsion system to perform the GTO-GEO burn. However, it was rescued by using the Hall thrusters it carried for orbit maneuvering and GEO stationkeeping and successfully reached GEO with many years of fuel life remaining. [13]

5 ENGINEERING ASSESSMENT

Now that the technical efficacy of EP systems has been shown for the wide range of challenging PMD scenarios, the engineering realities of integrating these solutions into satellites needs to be examined. The incorporation of EP systems into the three satellite classes are discussed below with an emphasis on size (i.e., volume), weight (i.e., mass), power, and cost.

For a 6U cubesat, the EP system is a simple bolt-on system with minimal interface requirements to the satellite. Primarily, it requires power and a structural interface. Secondarily, the EP system requires some amount of thermal dissipation. Alignment of thrust vectors with spacecraft axes must be considered, as well as plume interactions with spacecraft surfaces and deployed appendages. Another factor influencing the adoption of a propulsion system is cost. The cost for the electrospray and FEEP systems is expected to range between $20,000 - $80,000 depending on requirements such as performance and lifetime. Some missions may require multiple units to meet total impulse requirements.

For the 300 kg smallsat, it is likely that the propulsion system, to include the propellant tank, power processing, and flow regulation would need to be integrated into the structure of the satellite so would require more upfront engineering effort and time. Other interface considerations are similar – power, structural, thermal, and alignment / plume considerations. The cost of the EP systems would be most strongly influenced by the requirements for performance and lifetime, as well as the total quantities procured. However, a ballpark figure of between $150,000 - $500,000 per system is expected.

For the 2,000 kg satellite, the integration time will follow the same trend of the 300 kg smallsat – the EP system needs to be integrated sooner and is more a part of the overall assembly, integration, and test (AI&T) flow of the satellite build. The requirements may also vary more widely because of the customer and mission differences. This class of satellite is more likely to be tailored to meet specific customer and mission needs which may have a strong effect on the EP system costs. Rough estimates of these systems may range from $2M - $5M, however, they remain a small fraction of the overall satellite systems cost.

Overall, the addition of an EP system imposes a slight burden on the operator from the perspective of mass or volume, and can be tailored to fit within the expected power needs of the payload on a given spacecraft. However, the benefit for the spacefaring community is substantial and unique. For these reasons, EP has become the more accepted solution for on-orbit satellite maneuvers and has been shown in this paper capable of executing the most demanding PMD maneuvers.

6 CLOSING COMMENTS

The analysis contained in this paper highlights the capability of EP systems to satisfy the most stringent CA and PMD requirements. The EP systems easily surpassed the 2-5% mass fraction anecdotal threshold identified when debris mitigation guidelines were established in the late 1990s. The last major hurdle for widespread use of EP systems for all spacecraft (i.e., from 1U cubesats to 6,000 kg satellites) is likely the cost. However, prices are dropping as quickly as capabilities are increasing.

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