Stage-Based Electrospray Propulsion System for Deep-Space Exploration with CubeSats

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Abstract—Independent deep-space CubeSat missions require efficient propulsion systems capable of delivering several km/s of ΔV. The ion Electrospray Propulsion System under development at MIT’s Space Propulsion Laboratory is a high ΔV propulsion system that is a promising technology for propulsion of independent deep-space CubeSat missions due to its mechanical simplicity and small form factor. However, current electrospray thrusters have demonstrated lifetimes up to an order of magnitude lower than the required firing time for a mission to a near-Earth asteroid starting from geostationary orbit. A stage-based concept is proposed where the propulsion system consists of a series of electrospray thruster arrays. When a set of thrusters reaches its lifetime limit, it is ejected from the spacecraft exposing new thrusters thereby increasing the overall lifetime of the propulsion system. Such a staging strategy is usually not practical for in-space thrusters. However, the compactness of micro-fabricated electrospray thrusters means that their contribution to the overall spacecraft mass and volume is small relative to other subsystems. Mechanisms required for this stage-based approach are proposed and demonstrated in a vacuum environment. In addition, missions to several near-Earth asteroids with orbital elements similar to those of Earth are analyzed with a particular focus on the escape trajectory. With a stage-based approach, independent deep-space CubeSat missions become feasible from a propulsion standpoint.

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1. Introduction
The standardization of small satellites through CubeSats has allowed for more affordable space exploration. However, this progress in affordability has often been limited to Earth orbit with the recent launch of Mars Cube One [1] representing the first deep-space mission performed with CubeSats. There are several other missions that could take advantage of the CubeSat form factor. In particular, missions to near-Earth asteroids lend themselves to the use of CubeSats. Our current understanding of these asteroids in terms of composition and characteristics is limited and the paradigm of using a single large spacecraft restricts the number of visits to once every few years. By using fleets of CubeSats, all of which could be launched on the same launch vehicle, the frequency of asteroids visits could be dramatically increased all while significantly decreasing the cost per visit.

However, there are a number of technical challenges that are preventing such missions from being performed including the miniaturization of hardware, autonomy, and guidance, navigation, and controls systems. This work aims towards an independent deep-space CubeSat mission from the propulsion standpoint. Specifically, given the current propulsion technology available, can a propulsion system compatible with the CubeSat form factor be developed that can propel CubeSats from Earth orbit into deep-space and to a near-Earth asteroid?

A mission to a near-Earth asteroid from Earth orbit would require at least 3 km/s of ΔV. Figure 1 shows the fuel mass fraction for a 3 km/s ΔV mission for various specific impulses. We can see that from a fuel mass fraction standpoint such a mission is not prohibitive. For a standard chemical propulsion system with a specific impulse of 450 s the required fuel mass fraction is 0.5. However, for a CubeSat mission the fuel mass fraction can be misleading and it might be more informative to view the problem from an absolute dry mass perspective as shown in Figure 2. For a propulsion system with a specific impulse of 450 s on a 3U CubeSat, the available dry mass is only 2 kg. This is a significant constraint on the mission considering that a typical integrated attitude determination and control system has a mass of around 1 kg [2]. This leaves only 1 kg of mass for all other hardware including any systems associated with the propulsion system such as fuel tanks and propellant management.

From a dry mass perspective, a near-Earth asteroid CubeSat mission does not become feasible until specific impulses of 1000 s or more, which calls for an electric propulsion system. Within electric propulsion systems, electrospray propulsion holds many other advantages that reduce the dry mass of the propulsion system. As such, in this work we will consider a propulsion system based on electrospray propulsion.

While electrospray propulsion is a promising technology for its high efficiency and low mass, its current limitation is the lifetime of an individual thruster. In their current state, electrospray thrusters have lifetimes of around 500 hours, or about an order of magnitude lower than the required firing times for deep-space missions. While current efforts are aimed at extending the lifetime of these devices, it is relevant to find alternatives to overcome this limitation with existing versions of electrospray thrusters. Therefore, we propose a stage-based approach where the propulsion system consists of multiple stages of thrusters. As a stage of thrusters reaches its lifetime limit, it is ejected and exposes a set of new thrusters.
to continue the mission. Such an approach is normally not feasible as it significantly increases the total mass of the propulsion system. However, electrospray thrusters have low size and mass compared to other systems onboard the spacecraft. Therefore, multiple stages can be added without constraining other systems.

This paper considers a potential configuration for an electrospray thruster based CubeSat propulsion system as well as the requirements in terms of firing time based on trajectories to various near-Earth asteroids. Mechanisms for a stage-based approach are proposed and demonstrated in a vacuum environment. Lastly, the impact of the stage-based approach on the spacecraft’s escape trajectory is analyzed, which in most cases represents the most demanding mission stage for the propulsion system.

2. ELECTROSPRAY PROPULSION

Electrospray thrusters produce thrust through electrostatic acceleration of ions. Ions are evaporated from an ionic liquid propellant with a strong electric field. The ionic liquid propellant is a molten salt at room temperature that is non-reactive, readily available, and has low toxicity. Electrospray thrusters and their ionic liquid propellants hold three main advantages that make them an excellent choice for propulsion of small satellites such as CubeSats. Firstly, the ionic liquid is “pre-ionized” and does not need an ionization chamber. Second, ionic liquids have near-zero vapor pressure and therefore do not need any form of pressurized containment. Lastly, propellant is fed to the thruster by passive capillary forces through a porous liner embedded into the fuel tank thereby eliminating the need for any pumping systems. These three advantages allow electrospray thrusters to be incredibly compact and suitable for the CubeSat form factor.

To produce a strong enough electric field to evaporate the ions, the ionic liquid is fed to a sharp emitter tip. A voltage is applied to the ionic liquid with respect to an extractor grid. The sharp tip of the emitter allows for a strong electric field to be produced that both extracts the ions from the ionic liquid and accelerates the ions to produce thrust [3]. A diagram of a single emitter and extractor is shown in Figure 3. The thrust produced by a single emitter is only on the order of 10s of nano-Newton. Therefore, multiple emitters are arranged in an array to produce a single thruster. Since a single emitter is on the 100 µm scale, arrays of 100s of emitters can be manufactured on a 1 cm scale.

The ionic liquid is composed of both positively and negatively charged ions. By changing the polarity of the voltage applied between the emitter tip and extractor, the thruster will either evaporate and accelerate negative ions or positive ions. To prevent spacecraft charging, thrusters are operated in pairs with one thruster firing positive ions and the other thruster firing negative ions. This paired operation eliminates the need for a neutralizer further reducing the mass of the propulsion system.

**Ion Electrospray Propulsion System**

For this work we will consider the ion Electrospray Propulsion System (iEPS) under development in the Space Propulsion Laboratory (SPL) at the Massachusetts Institute of Technology. Each iEPS thruster consists of an array of 480 emitter tips made from porous glass. The emitter array is housed in a 13 x 12 x 2.4 mm silicon frame with a gold-coated silicon extractor grid. Figure 4 shows an iEPS thruster mounted on a single-thruster fuel tank. Due to the passive propellant feed system, the same iEPS thruster can be mounted on a variety of fuel tanks as long as a porous material connection exists between the ionic liquid and emitter array.

Each iEPS thruster array can produce thrust in the range of 2-20 µN with a specific impulse close to 1000 s when using EMI-BF₄ as the ionic liquid propellant [4]. However, the
thrust and specific impulse are heavily dependent on the ionic liquid used as well as the material of the emitter array. Ongoing research at the SPL is investigating different materials for the emitter array that can contribute to an increased thrust and thruster lifetime [5]. For this work, we will assume that a single iEPS thruster produces 20 \( \mu \)N of thrust with a specific impulse of 1000 s to reflect currently demonstrated performance metrics.

To further improve the thrust produced by the entire propulsion system, arrays of thrusters are used. Thrusters are clustered in groups of four on top of a single fuel tank to maximize the fuel volume ratio while maintaining structural integrity during launch. A 1U CubeSat face can hold up to nine of these clusters in a 3 x 3 square pattern for a total of 36 thrusters. For this work, we will consider a configuration where a thruster in the center row is omitted to provide space for the staging electronics and mounting on either side of the thrusters as shown in Figure 5.

This configuration, with 32 thrusters, can produce 0.64 mN of thrust. While this thrust is still relatively small, the low mass of the CubeSat form factor means that the net acceleration is comparable to that of other electric propulsion based missions such as Dawn [6].

3. Trajectory Analysis

For all missions considered in this work, the spacecraft starts in geostationary orbit. This maximizes the starting orbital radius of the spacecraft, relatively reducing the \( \Delta V \) required for escape, while still being a common launch destination for primary payloads thereby not significantly limiting launch opportunities. While a mothership or “kicker” stage could be used to propel the spacecraft into an escape trajectory, we will consider the case where the spacecraft independently escapes Earth in order to decrease the cost of the mission and to use a standardized form factor for ease of launching.

The primary target for the considered missions are near-Earth asteroids with similar orbital elements to those of Earth. For these asteroids, the majority of the required thruster firing, and therefore \( \Delta V \), occurs during the escape from Earth. Therefore, detailed trajectory analysis will be considered for the escape portion of the mission while the required \( \Delta V \) to match the asteroid’s orbit will be analyzed with JPL’s Small-Body Mission-Design Tool [7] which gives estimates of \( \Delta V \) requirements for continuous low-thrust transfers from Earth escape to asteroid rendezvous.

A 3U CubeSat propelled by 32 iEPS thrusters will be used as the representative spacecraft. This gives an initial spacecraft mass of 4 kg with a propulsion system producing 0.64 mN of thrust with a specific impulse of 1000 s. The trajectory for this spacecraft from geostationary orbit around Earth (42164 km) until escape is achieved is shown in Figure 6. Throughout the trajectory, thrust was applied in the direction of the velocity vector at a constant level of 0.64 mN, and the effects of the Moon are ignored. During the trajectory, the spacecraft spends a considerable amount of time on the last leg of the escape near the orbit of the Moon. The effect of the Moon on the escape time depends on the initial phasing between the spacecraft and the Moon as shown in Figure 7. Excluding the regions near phase angles of 0 degrees and 330 degrees in which the spacecraft passes near to, or even impacts, the Moon, the 2-body simulation serves as an average estimate for the 3-body escape time and therefore the 2-body firing time estimates will be used for propulsion system analysis.

In total, the 2-body escape trajectory requires 168 days of thruster firing and consumes 950 g of fuel. The total \( \Delta V \) for the escape is 2.66 km/s which is over double the \( \Delta V \) requirement for an impulsive escape trajectory from geostationary orbit (1.3 km/s). More fuel efficient escape trajectories such as firing in a restricted thrust arc around
Figure 7. Effect of the Moon phasing on escape time.

Table 1. Predicted ∆V and thruster firing time required to match near-Earth asteroid orbits from Earth’s orbit

<table>
<thead>
<tr>
<th>Asteroid</th>
<th>∆V [km/s]</th>
<th>Firing Time [hr]</th>
</tr>
</thead>
<tbody>
<tr>
<td>2009 BD</td>
<td>0.77</td>
<td>982</td>
</tr>
<tr>
<td>2010 UE51</td>
<td>1.25</td>
<td>1556</td>
</tr>
<tr>
<td>2008 EA9</td>
<td>1.49</td>
<td>1833</td>
</tr>
<tr>
<td>2007 UN12</td>
<td>1.51</td>
<td>1856</td>
</tr>
<tr>
<td>2011 MD</td>
<td>2.31</td>
<td>2729</td>
</tr>
</tbody>
</table>

periapsis could be devised but would increase the total time of the mission significantly. By firing only in a 90 degree arc around periapsis, the total firing time can be reduced to 90 days but the total time to achieve escape increases to 79 years. Decreasing the firing arc further pushes the total firing time to the impulsive limit of 88 days where the impulse limit is defined as the firing time required to achieve the ∆V of the equivalent impulsive trajectory and places a lower bound on the possible firing time. However, the total mission time increases at a greater-than-exponential rate. Figure 8 shows the required firing time for various thrust arcs and the total mission time to escape from Earth.

In addition to the required ∆V to escape from Earth, additional ∆V is required to match the orbit of a near-Earth asteroid. The JPL Small-Body Mission-Design Tool [7] provides estimates for the ∆V requirements of continuous low-thrust transfers to various asteroids starting from Earth’s orbit. Table 1 shows the predicted ∆V requirements and associated thruster firing times to match the orbits of several near-Earth asteroids with orbital elements similar to those of Earth. The initial mass of the spacecraft for the Earth-asteroid transfer trajectory was assumed to be the final mass of the continuous thrust escape trajectory. For the easiest to reach asteroid, 2009 BD, the total required firing time is 209 days.

Electrospray thrusters currently have demonstrated laboratory lifetimes of up to 500 hours [8]. This is much less than the predicted 5000 firing hours required to reach the easiest to reach near-Earth asteroid starting from geostationary orbit. Therefore, in their current state, iEPS thrusters cannot be used for main propulsion of an independent deep-space CubeSat mission. In order to make such a mission feasible, a method of increasing the total lifetime of the propulsion system needs to be developed.

4. STAGING CONCEPT

Two approaches could be used to increase the total lifetime of the propulsion system. Either the lifetime of individual thrusters could be increased, or an alternative system could be developed that bypasses the lifetime limit of the thrusters. Near-term developments at the SPL aim to increase the lifetime of individual thrusters through improvements in materials and manufacturing techniques. However, those developments only expect to increase the lifetime by a factor of 2 or 3, still less than the required firing time to escape from geostationary orbit. Therefore, we propose a staging concept with sequential stages of thruster arrays. One set of thrusters is fired until the lifetime limit before being ejected from the spacecraft and exposing a new set of thrusters. Through staging, the lifetime limit of individual thrusters is bypassed in order to increase the overall lifetime of the propulsion system. Figure 9 shows concept image of a CubeSat staging a set of thrusters mid-flight.

Development of a staging concept also provides two additional benefits. Firstly, stages can be used to provide redundancy for the propulsion system. Second, as thrusters are fired they begin to decay and their performance decreases. Current efforts to minimize the effect of this decay involve increasing the input voltage to the thruster [8]. However, increasing the input voltage further accelerates thruster decay and requires more power. By staging thrusters, a new, fresh set of thrusters will be used thereby avoiding the effects of thruster degradation.
Normally this staging concept would not be feasible for most propulsion systems. However, the compact nature of iEPS thrusters and the lack of need for ionization chambers, pressurized propellant containment, and propellant feed systems means that the contribution of a single thruster array to the overall spacecraft mass and volume is small relative to other spacecraft systems. A complete array of thrusters with enough fuel for 500 hours of firing only occupies approximately 0.2U (200 cm³) of volume and weighs less than 200 grams.

To develop the stage-based propulsion system, two mechanisms are required. The first is the staging mechanism itself which holds together successive stages during flight and separates the outermost stage at the time of staging. The second is a routing mechanism that passively routes control signals to the active stage. All thruster stages will use the same control electronics. Therefore, it is necessary to route the control signals to the correct stage. By performing the routing mechanically and passively, the control electronics can remain “stage blind” and no electrical addressing of individual stages will be required.

**Staging Mechanism**

The staging mechanism is based on a fuse wire approach. Successive stages are held together with a thin stainless steel wire. At the time of staging, a high current (10 A) provided by a high power density (≤ 200 mΩ ESR) ultra-capacitor is run through the wire, heating it up until it melts. After the wire melts and the stage is released, a compression spring is used to eject the stage from the spacecraft. The wire is housed in a machinable glass-ceramic casing and attaches to standard 4-40 standoffs allowing for easy integration to existing iEPS electronics boards. Figure 10 shows the staging mechanism in operation. The entire fusing procedure takes approximately 100 ms. A gap between machinable ceramic pieces is intentionally added during this test in order to provide visibility to the fuse wire.

Similar approaches have been used previously for miniature release mechanisms for small satellites. A nichrome burnwire mechanism is explored in [9] where a nichrome wire is heated up in order to cut through a Vectran tie down cable. The release mechanism in this work differs from [9] in that the wire itself is melted to activate release rather than used to cut through a second wire therefore simplifying the mechanism design.

Multiple miniature release mechanisms are explored in [10] including a fuse wire based mechanism. The fuse wire mechanism in [10] is based on beryllium-copper wire and has the wire loop through a retainer mechanism. The mechanism in [10] uses five unique components and requires the fuse wire to be etched in order to control the fusing location. The mechanism developed in this work keeps the fuse wire straight and uses four unique components, only one of which, the ceramic casing, requires fabrication. By keeping the wire straight and securing it between two screw terminals, no wire etching is required as the wire can fuse in any location and still release the two stages.

**Routing Mechanism**

The routing mechanism is a custom made, normally closed, momentary push button. When a preceding stage is present, the mechanism is opened preventing control signals from entering the stage. When the preceding stage is ejected the mechanism is closed and the stage becomes active. After the thrusters on the stage reach the end of their lifetime the stage is ejected, automatically activating the next stage. With this mechanism the control electronics remain “stage blind” and do not need to track which stage is active. This greatly simplifies the electronics design as existing iEPS electronics boards can continue to be used without needing to add extra electronics for addressing of individual stages. It also allows for greater flexibility when adding or removing stages as the number of stages does not impact the electronics boards.

**Stage Configuration**

Each stage will contain 2 routing mechanisms and 2-4 staging mechanisms as shown in Figure 11. All control electronics required for thruster firing and staging mechanism activation
will be mounted on a separate electronics board below the stage stack and have their signals routed to the stage through the routing mechanisms.

5. ANALYSIS AND TESTING

Wire Material Selection

Material selection for the fuse wire consists of balancing mechanical and thermal properties. Desirable mechanical properties are low density and high tensile strength while desirable thermal properties are low specific heat capacity and low melting point. These properties can be combined to form a fusing metric, \( \Gamma \), which is a function of the wire’s initial temperature and defined as

\[
\Gamma(T_0) = \frac{\rho c (T_m - T_0)}{\sigma}
\]  

(1)

where \( \rho \) is the wire’s density, \( c \) is the specific heat capacity, \( T_m \) is the melting point, \( T_0 \) is the initial wire temperature, and \( \sigma \) is the ultimate tensile strength. The fusing metric is also equal to the ratio of the required energy to fuse the wire \( (E_f) \) to the maximum load the wire can carry \( (F) \) and the length of the wire \( (l) \)

\[
\Gamma = \frac{E_f}{F l}
\]  

(2)

For a given maximum load and wire length, set by the form factor of the spacecraft and staging mechanism, the fusing metric gives the minimum energy required for fusing. Materials with a lower fusing metric are therefore more desirable.

Figure 12 shows the ultimate tensile strength versus the fusing energy per unit volume for various materials starting from an initial temperature of 0°C. Lines originating from the origin represent lines of constant fusing metric where lines with a larger slope correspond to lower fusing metrics. The line for \( \Gamma = 10 \) is shown for reference. Per this analysis, good material choices are the stronger aluminum and beryllium-copper alloys. In addition, pure tin is a particularly good choice with a fusing metric of 1.3.

Beyond mechanical and thermal properties, the electrical properties of the material determine its ability to be integrated into the spacecraft system. While pure tin has a low fusing metric, its low conductivity prevents its use given current capacitor capabilities. A 6.35 mm long pure tin wire capable of holding 40 N of force has a resistance an order of magnitude lower than the internal resistance of commercially available high power density capacitors (~200 mΩ). This discrepancy in resistance prevents power dissipation in the wire and therefore prevents fusing. Figure 13 shows the available energy output from a capacitor with energy capacity of 50 J and internal resistance of 200 mΩ. Required energies for fusing for materials shown in Figure 12 as well as total wire resistances are shown for 6.35 mm long wires capable of holding 40 N of load (equivalent to 4 wires each holding 10 N of load). We can see that most materials have very low resistances (~5 mΩ) due to their low resistivity and therefore do not dissipate enough energy for fusing.

For this work stainless steel 304 (SS304) is used as the fuse wire material. Although SS304 has a poor fusing metric (~11), we can see from Figure 13 that its relatively high resistivity means that more of the capacitor energy is dissipated in the wire compared to most other materials considered in this study thereby allowing for wire fusing with sufficient safety margin. In addition, SS304 is readily available which allows for rapid prototyping. As capacitors with lower internal resistances are developed, materials such as tin which offer better fusing metrics can be used. Figure 13 shows the available energy for fusing if the internal resistance of the capacitor is reduced by half to 100 mΩ.

Select beryllium-copper alloys are the only materials surveyed that have lower required fusing energy and higher...
wire resistance than SS304. However, given that SS304 is more readily available and can be fused with the capacitor considered in this study, SS304 is used for all tests in this work. Future work will consider using beryllium-copper alloys as the fuse wire material.

The fuse wire itself is 6.35 mm long and has a diameter of 0.203 mm (0.008 in). Each fuse wire can therefore hold \( \sim 16 \) N of load. Given the fusing metric of SS304 is \( \sim 11 \), the estimated energy to fuse the wire is 1.15 J. As the melting point of SS304 is high \( (1400^\circ C) \) the effect of initial wire temperature is minimal and so a reference of \( 0^\circ C \) is used.

**Vibration Analysis**

With the addition of the staging mechanism, successive stages are only held together by a 2-4 thin metal wires thereby influencing the vibration characteristics of the spacecraft during launch. While the entire spacecraft structure could be constrained, in the worst-case scenario the entire vibrational load will have to be absorbed by the fuse wires.

The stack of stages is approximated by stacked spring-mass systems where the spring constant, \( k \), is approximated as

\[
k = N \frac{AE}{L}
\]  

where \( N \) is the number of staging mechanisms per stage, \( A \) is the wire cross-sectional area, \( L \) is the relaxed wire length, and \( E \) is the Young’s modulus of the wire material, SS304 in this case. Figure 14 shows the first resonant mode of the staging stack versus number of stages for both 2 and 4 staging mechanisms per stage. In all cases the first resonant mode is greater than 100 Hz and will therefore avoid dynamic coupling between the low frequency dynamics of the launch vehicle and the stages.

In actuality, the resonant modes will be higher than in this analysis. The spring-mass approximation allows for compression of the fusing wire. On the staging mechanisms, compression of the fusing wire is constrained by the stiffer machinable ceramic casing. Future work will involve experimental measurement of the resonant modes of the staging mechanisms as well as their response to random vibration.

**Staged Trajectory Analysis**

Using a stage-based propulsion system changes the escape trajectory of the spacecraft. As stages are ejected, the dry mass of the spacecraft decreases and therefore increases the propulsive acceleration of the spacecraft. Figure 15 shows the propulsive acceleration during escape for an assumed stage lifetime of 500 hours and dry mass of 75 g. The stage dry mass accounts for the iEPS thrusters and tanks as well as the thruster electronics board, staging mechanisms, and routing mechanism. The power processing unit for the thrusters is assumed to be common for all stages and therefore not ejected during staging. After the first staging, the acceleration for the staged trajectory is always greater than the acceleration for the continuous trajectory due to the lower spacecraft dry mass. The acceleration increases after each staging and the final acceleration for the staged trajectory is 17% greater than the continuous trajectory.

The increased acceleration over the vast majority of the escape results in a different escape trajectory. Figure 16 shows a comparison of the continuous thrust trajectory and staged trajectory. We can see that due to staging and the resulting change in acceleration, the staged spacecraft leaves Earth in the opposite direction of the continuous trajectory. In addition, the staged spacecraft achieves escape 8% faster than the un-staged spacecraft resulting in a 309 hour (12.9 day) decrease in escape time.

With the current iEPS performance specifications on the
configuration shown in Figure 11 (0.64 mN thrust, 1000 s specific impulse, 500 hr stage lifetime) 8 stages are required for escape. Future developments in the SPL predict a 4x increase in thrust, 2.5x increase in specific impulse, and 2x increase in stage lifetime. Figure 17 shows the required number of stages for escape from Earth for thrust and lifetime values between their current values and the predicted future values for specific impulses of 1000 s and 2500 s. We can see that the effect of specific impulse is minimal. If all target performance values are achieved then the number of stages can be reduced to 2.

Interestingly, the number of stages required for a given thrust and lifetime increases with specific impulse. This is due to the lower mass flow rate and therefore a decreased propulsive acceleration throughout the trajectory. Therefore, it is possible that increasing the specific impulse actually lowers the payload mass. Figure 18 shows the payload mass delivered at escape versus specific impulse for a fixed thrust and stage lifetime. We can see a clear drop in output voltage during fusing due to the equivalent series resistance of the capacitor module. Approximately 0.55 J of energy was dissipated in the internal resistance of the capacitor meaning the actual energy dissipated in the wire was ∼2.4 J, around double the theoretical prediction of 1.15 J due to losses into the rest of the staging mechanism.

The data in Figure 17 was generated by posing the 2-body escape problem as a multiple objective optimization. For a given number of available stages, the objective was to minimize the required thrust and lifetime while still achieving escape from Earth. A genetic algorithm was used to determine the Pareto front. Each population member was ranked based on the number of other population members that dominated it, members that had both lower thrust and lifetime while still achieving escape. After the Pareto front stabilized, a second order exponential curve was fitted to the front. The optimization was run for all possible number of stages and produces the data shown in Figure 17. As the front represents the minimum allowable performance in terms of thrust and lifetime such that a spacecraft can achieve escape with a given number of stages, it also represents the break between required number of stages. Any thruster that has slightly lower thrust or slightly lower lifetime will require an extra stage to achieve escape.

Mechanism Testing

A PBL-4.0/5.4 passively balanced ultracapacitor module with 200 mΩ equivalent series resistance from Tecate Group was charged to 4.4 V and used as the fusing capacitor. A single staging mechanism with a 6.35 mm length, 0.203 mm diameter stainless steel 304 fusing wire was tested at a vacuum chamber pressure of ∼70 µTorr. Fusing took approximately 83 ms and required approximately 2.95 J of energy. Figure 19 shows the capacitor output voltage during charging and fusing. We can see a clear drop in output voltage during fusing due to the equivalent series resistance of the capacitor module. Approximately 0.55 J of energy was dissipated in the internal resistance of the capacitor meaning the actual energy dissipated in the wire was ∼2.4 J, around double the theoretical prediction of 1.15 J due to losses into the rest of the staging mechanism.

The staging and routing mechanisms were tested together in the same vacuum environment with dummy stages. Figure 20 shows the combined test where the signals are represented by LED lights. Initially the signal is only routed to the first stage (left). The staging mechanisms are then activated from control electronics below the staging stack (center). After the staging mechanisms actuate and eject the first stage, the signal is passively rerouted to the second stage (right). Future testing for the staging and routing mechanisms will integrate the mechanisms into actual thruster electronics board and combine thruster firing and mechanism usage.
Figure 19. Capacitor output voltage during charging and fusing of a single staging mechanism.

Figure 20. Combined staging and routing mechanism test under vacuum with LEDs representing thrusters.

ACKNOWLEDGMENTS

This work was supported by a NASA Space Technology Research Fellowship, the Small Spacecraft Technology program within the Space Technology Mission Directorate, and the Miguel Alemán-Velasco foundation. In addition, the authors like to thank Alexander Peraire-Bueno for his contributions to the development of the signal routing mechanism.

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BIOGRAPHY

Oliver Jia-Richards received his S.B. in aerospace engineering from MIT in 2018. He is currently a graduate student and NASA space technology research fellow in the Space Propulsion Laboratory at MIT. His research interests include guidance and control of satellites with low-thrust propulsion systems and low-thrust trajectory design.

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