IAC-14.C4.4.4

PRELIMINARY MISSION CAPABILITIES ASSESSMENT OF A MAGNETICALLY SHIELDED MINIATURE HALL THRUSTER

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The mission capabilities of a small (~200 - 300 kg) spacecraft utilizing a low-power, long-life Hall thruster are examined. The magnetically shielded miniature Hall thruster is designed to shield the thruster from the key life-limiting erosion mechanism and can thus provide significant improvements to thruster life and delta-V. Such a thruster, which has been demonstrated and is undergoing development, is shown to be an attractive candidate for this spacecraft architecture. Trajectories are calculated to Asteroid 118401 (LINEAR) from a low Earth orbit and to an Earth-Moon L2 halo orbit from a standard geostationary transfer orbit. Results suggest that a small spacecraft using a low-power magnetically shielded Hall thruster is able to accomplish either mission for a variety of scenarios (maximum delivered mass, limited available power, constrained time-of-flight, etc.). Additionally, this novel spacecraft architecture is able to complete missions at significantly lower costs than conventionally sized (>1000 kg) spacecraft, offering considerable flexibility to spacecraft and mission designers.

I. INTRODUCTION

An important class of new and challenging deep space missions would be enabled by a low-power Hall thruster that exhibits high efficiency and a long operational lifetime. Such a device could be used as either a supplement to a conventional chemical propulsion system or as the primary propulsion system for a deep space mission, depending on the requirements. An efficient, long-life, low-power Hall thruster would therefore have a significant and enabling impact on missions utilizing both conventionally sized (~1000 kg) and the more novel low-mass (~200 - 300 kg) spacecraft. For conventional spacecraft, outer-planet missions requiring a significant reduction in velocity (i.e. orbit insertion to Uranus) may be possible, while low-mass (and therefore low cost) spacecraft would be capable of meaningful scientific missions to, for example, Mars, Venus, or the Jovian moons. In an effort to meet the requirements for these missions, the Magnetically Shielded Miniature (MaSMi) Hall thruster is currently under development in a collaborative effort between the University of California, Los Angeles (UCLA), and NASA's Jet Propulsion Laboratory (JPL) [1–3]. MaSMi utilizes a unique magnetic field topology to achieve significantly increased operational life, efficiency, and performance compared to conventional low-power Hall thrusters.

The goal of this investigation is to explore new and challenging deep space missions enabled by a low-powered, long-life miniature Hall thruster. This particular article discusses the preliminary findings of the investigation, broadly demonstrating the utility of the MaSMi Hall thruster by showcasing several mission scenarios that are of interest to the community. A review of low-powered Hall thrusters, magnetic shielding, and the MaSMi Hall thruster is presented in Section II. Additionally, this section contains a brief review of the trajectory optimization tools that are used for this study. Section III discusses the necessary assumptions to generate the mission trajectories and an overview of the missions examined in this study. Section IV presents the results of the mission studies, followed by a discussion of these results in Section V.

Concluding remarks are made in Section VI with a brief Future Work discussion in Section VII.

II. BACKGROUND

Low Power Hall Thrusters

Numerous low power ~3 cm-scale Hall thrusters, such as the BHT-200 and the SPT-30, are commercially available for space missions. While these thrusters deliver attractive thrust and specific impulse performance (>10 mN and >1100 s, respectively) at anode efficiencies of more than 30%, their demonstrated lifetimes (on the order of 1000 h) are insufficient for use as the primary propulsion system for deep space missions [4–9]. In general, the inherently larger surface-to-volume ratio of low-power Hall thrusters (<500 W and <7 cm dia.) causes increased plasma-wall interactions, leading to rapid erosion and electron heating of the discharge channel walls. This results in poor life and low efficiency, ultimately limiting the utility of miniature Hall thrusters. Life-limiting issues involving channel erosion, electron heating, and plasma-wall interactions of high-power Hall thrusters (>4 kW) are significantly reduced, if not eliminated, by the use of a magnetically shielded field topology.

Magnetic shielding of Hall thrusters is achieved through a unique field topology that exploits two key features of Hall thruster behavior: magnetic field line isothermality and magnetic-force-line equipotentialization [10–13]. The field lines passing nearest to (but not intersecting) the channel walls are predominately populated by cold electrons captured in the anode region (through the isothermality along a give field line); holding these cold electrons near the channel surfaces reduces the plasma sheath potential drop near the walls and maintains a plasma potential close to the discharge voltage (equipotentialization). These effects combine to significantly reduce ion-bombardment erosion of and electron power deposition to the channel walls, thereby increasing the thruster's useful life. The concept of magnetic shielding has been successfully applied to multiple Hall thrusters and is well understood [10–16].

In an effort to fill the technology gap between high- and low-power Hall thrusters, the Magnetically Shielded Miniature (MaSMi) Hall thruster is being developed [1–3]. The MaSMi Hall thruster was designed to nominally operate between 300 - 400 W and employs a magnetically shielded field topology. Further specifics on the design and supporting equipment of the thruster can be found in the references [1–3].

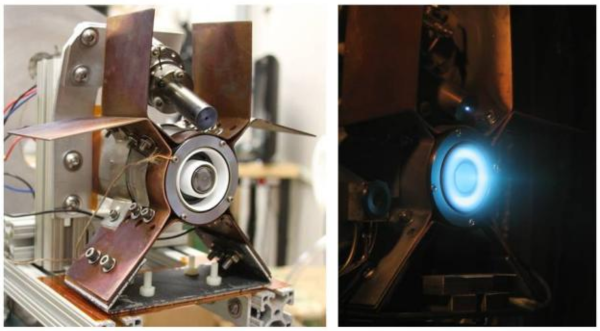

Fig. 1. The magnetically shielded miniature (MaSMi) Hall thruster during setup and operation at 330 W.

The MaSMi Hall thruster, shown before and during operation in Figure 1, has been demonstrated at two vacuum facilities: the UCLA Electric Propulsion Test Facility and the JPL High Bay facility. MaSMi testing at UCLA focused on the nominal operating power of 325 - 330 W, while testing at the JPL facility spanned a range of power levels, from 130 - 640 W. Performance was calculated at the nominal power condition based on plasma plume measurements at UCLA, resulting in a thrust of approximately 19 mN at an anode efficiency of 59% [1,2]. A full diagnostic suite at JPL, including a thrust stand, allowed for a more complete characterization of the thruster's performance at 330 W, resulting in a thrust of approximately 13 mN at an anode efficiency of approximately 24% [3]. The thrust produced by MaSMi over the full operating power range was measured from less than 5 mN to over 20 mN [3].

The performance values at the nominal operating power were observed to differ significantly between the two facilities. During testing, however, the thruster's plasma beam was also observed to have a significantly different structure and behavior at the two facilities. This indicated multiple thruster operation modes, each corresponding to different performance levels; identification of at least two unique modes at the JPL facility, in addition to the mode observed at UCLA, was confirmed with plume measurements [3].

Trajectory Modeling Tools

Two computational solvers were used to model the mission trajectories for this initial investigation. The first is the Low Energy Low Thrust Unified Solver (LOTUS). This tool uses heuristic patch point selection for mating low-thrust and manifold (transfers requiring little or no ΔV) trajectories by use of Q-law, a feedback control law developed by Petropolos [17,18]. A backward propagating solution methodology reduces the problem dimension and enables short computational times [17]. The second trajectory solver used in this study is Mission Analysis Low-Thrust Optimization

(MALTO), a well-documented and widely used trajectory optimization tool at JPL; the fundamentals of this code can be found in the literature [19].

III. INVESTIGATION OVERVIEW

Assumptions

To develop the mission trajectories discussed in this paper, several assumptions must first be made. Although no formal life tests have been completed with the MaSMi Hall thruster, it is assumed that the device is capable of a maximum of 65,000 h of operation based on life-limiting discharge channel erosion approximations [1,2]. While this is a full order of magnitude less than demonstrated lifetimes of larger magnetically shielded Hall thrusters, this assumed lifetime is substantiated in the literature [1,2,20,21].

The MaSMi Hall thruster is a new technology that has not yet been fully optimized. Because the performance results from the two facilities differ, as discussed above, an upper and lower bound of the thruster performance is assumed. The performance lower bound was assumed to be the measured data from the JPL High Bay facility (herein called "measured thrust"). A linearly scaled version of these same data were used as the upper bound (herein called "projected thrust"). The magnitude of the scaling was based on the percent increase of MaSMi's thrust between the JPL and UCLA facilities at the nominal operating condition of ~330 W and 1.2 A (approximately 48% increase). A third-order polynomial was fit to the resulting thrust data and a fourth order polynomial was fit to the anode mass flow data to generate smooth throttling curves for the trajectory software to utilize. These polynomials are as follows:

Measured thrust:

$$\begin{aligned} T = 0.10541321P^3 - 0.14715193P^2 \\ +0.08922424P - 0.00413409 \end{aligned} \quad (1)$$

Optimized thrust:

$$\begin{aligned} T = 0.15601155P^3 - 0.21778486P^2 \\ +0.13205187P - 0.00611845 \end{aligned} \quad (2)$$

Anode mass flow rate:

$$\begin{aligned} \dot{m}_a = -3.106 \times 10^{-5}P^4 \\ +4.434 \times 10^{-5}P^3 - 2.071 \times 10^{-5}P^2 \\ +5.46 \times 10^{-6}P + 3.4 \times 10^{-7} \end{aligned} \quad (3)$$

where T is thrust (N), P is discharge power (kW), and $\dot{m}_a$ is the anode mass flow rate (kg/s). The throttling curves for thrust and propellant mass flow rate vs. thruster discharge power are shown in Figure 2.

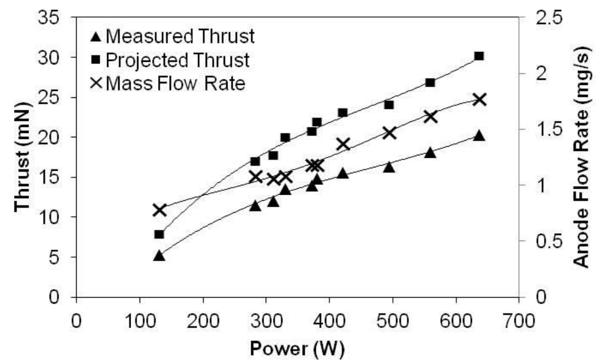

Fig. 2. Throttling curves for the MaSMi Hall thruster, including measured thrust, projected thrust, and anode mass flow rate.

All mission trajectories were generated and optimized using a single MaSMi Hall thruster as the primary propulsion system for the spacecraft with the throttling curves presented in Figure 2. Mission trajectory solutions were optimized to yield the maximum mass delivered to the target for specified mission constraints (available power, time-of-flight, etc.). During the optimization process, the thruster power was bounded between 130 W and 640 W as specified by Figure 2. A thruster duty cycle of 95% of the available power (depending on the spacecraft position relative to the sun) was made available by the propulsion subsystem.

Mission Trajectory Modeling

This preliminary investigation focuses on the use of small spacecraft to complete two challenging deep space missions with launch dates not extending beyond 2025 (i.e. near-term missions). The first is a full rendezvous (approach and orbit insertion with $V_\infty = 0$) with an icy asteroid. Spacecraft initial masses ranging from 200 kg to 550 kg are considered, each with different user-specified power and flight time constraints. The second mission is an Earth geostationary transfer orbit (GTO) - to - Earth-Moon L2 Halo orbit. The range of spacecraft initial masses considered span from under 60 kg to over 300 kg. These two particular mission trajectories were selected because they demonstrate the significantly improved capabilities of small spacecraft enabled by the MaSMi Hall thruster, specifically to carry out meaningful scientific missions.

IV. TRAJECTORY RESULTS

Asteroid Rendezvous

The asteroid rendezvous mission considered for this study targeted Asteroid 118401 (LINEAR), which is an icy body with an orbit path approximately centered between Earth and Jupiter. Five optimized mission trajectories were computed to the asteroid, each demonstrating a unique spacecraft or mission capability. All five missions utilize a Mars flyby. A summary of the spacecraft and mission parameters is found in Table 1, where M_0 is the initial spacecraft mass, M_f is the final (delivered) spacecraft mass, P_0 is the solar array available power at 1 AU, V_∞ is the hyperbolic escape velocity, and *ToF* is the mission time of flight.

M_0 (kg)	M_f (kg)	P_0 (kW)	V_∞ (km/s)	*ToF* (d)
200.0	111.0	1.0	1.5	2,135.4
200.0	105.2	0.8	1.5	2,310.2
211.6	93.9	1.0	2.0	1,600.0
350.0	213.9	1.0	2.0	2,555.0
550.0	353.5	2.0	2.0	2,555.0

Table 1. Summary of spacecraft and trajectory parameters for missions to Asteroid 118401 (LINEAR).

The first two asteroid rendezvous trajectories, shown in Figures 3 and 4, demonstrate the effect of changing the initial available power (at 1 AU) to the spacecraft on the mission time of flight. This simulates differing mission, cost, or packaging constraints that may govern the size of the solar array. These trajectories utilize an initial spacecraft mass of 200 kg and an Earth hyperbolic escape velocity of 1.5 km/s. A complete rendezvous with the asteroid, including orbit capture with V_∞=0, is accomplished in 5.8 years and 6.3 years for each trajectory, respectively.

Figure 5 shows the third mission trajectory presented in Table 1, which prescribes a maximum flight time constraint of 4.5 years. This example demonstrates that demanding missions can be accomplished in very short timeframes using a solar-electric propulsion (SEP) system powering the MaSMi Hall thruster on a small spacecraft. The results of this trajectory analysis suggests that rendezvous can be accomplished in just under 4.4 years using a 211.6 kg spacecraft with a 1 kW solar array and an Earth hyperbolic escape velocity of 2 km/s.

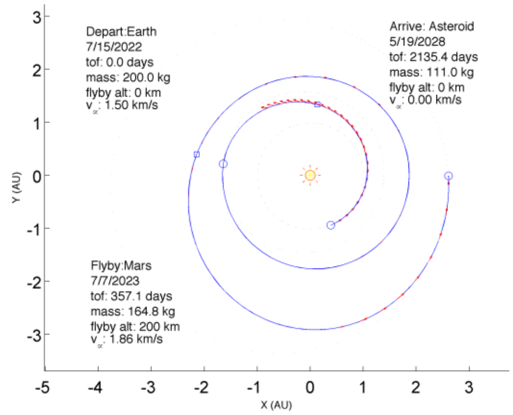

Fig. 3. Mission trajectory to Asteroid 118401 (LINEAR) using a 200 kg spacecraft with an available power of 1.0 kW and an Earth hyperbolic escape velocity of 1.5 km/s.

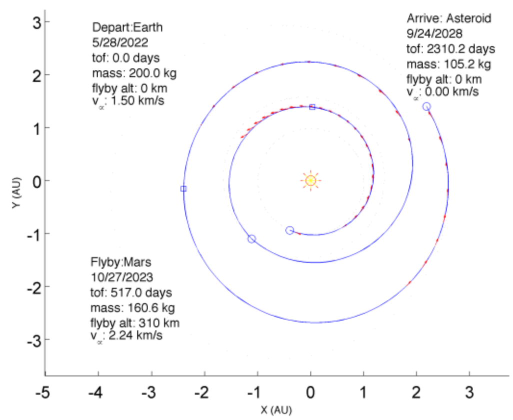

Fig. 4. Mission trajectory to Asteroid 118401 (LINEAR) using a 200 kg spacecraft with an available power of 0.8 kW and an Earth hyperbolic escape velocity of 1.5 km/s.

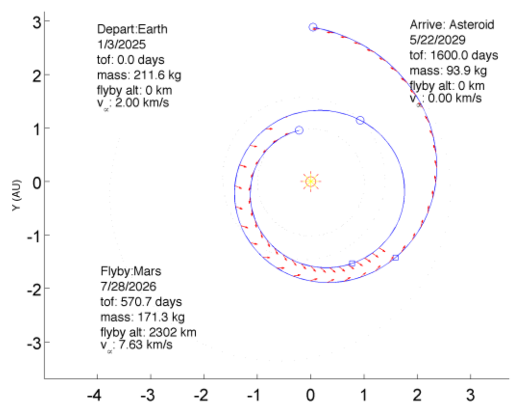

Fig. 5. Minimum flight time mission trajectory to Asteroid 118401 (LINEAR) using a 211.6 kg spacecraft with an available power of 1.0 kW and an Earth hyperbolic escape velocity of 2.0 km/s.

The final two trajectories presented in Table 1 show the impact of changing the initial available power on the spacecraft's possible initial mass for a given time of flight. With an assumed mission time of 7 years and an Earth hyperbolic escape velocity of 2 km/s, a 1 kW solar array can support a 350 kg spacecraft; a 2 kW array can support a 550 kg spacecraft for the same mission constraints. Figures 6 and 7 show plots of these two mission trajectories.

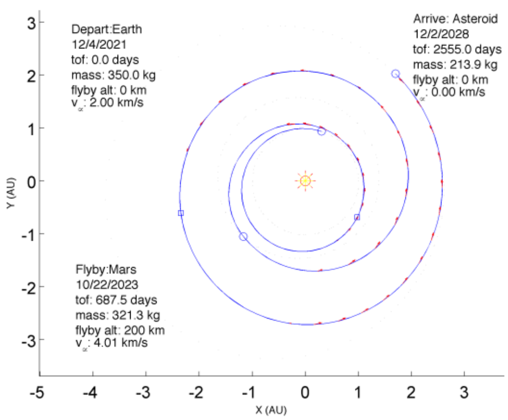

Fig. 6. Mission trajectory to Asteroid 118401 (LINEAR) using a 1.0 kW solar array, an Earth hyperbolic escape velocity of 2 km/s, and a specified mission time of 7 years to support a 350 kg spacecraft.

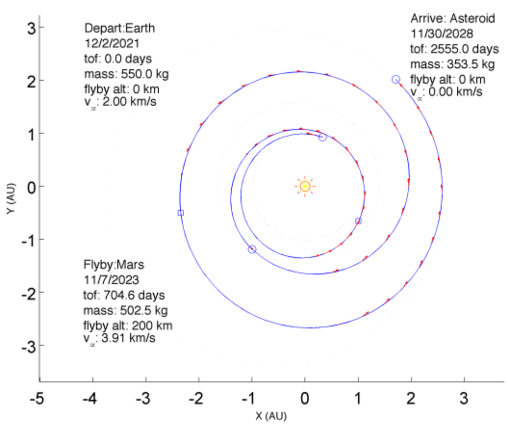

Fig. 7. Mission trajectory to Asteroid 118401 (LINEAR) using a 2.0 kW solar array, an Earth hyperbolic escape velocity of 2 km/s, and a specified mission time of 7 years to support a 550 kg spacecraft.

Earth-Moon L2 Halo Orbit

A series of mission trajectories from a GTO parking orbit to an Earth-Moon L2 Halo orbit were computed to further demonstrate the capabilities of a small spacecraft utilizing the MaSMi Hall thruster. The initial GTO was assumed to have a perigee altitude of 200 km and an apogee altitude of 35,000 km with a 28.5° inclination. For each power level data point presented in Figure 2, trajectories were calculated for delivered masses ranging from 50 kg to 300 kg by increments of 50 kg. If a specified delivered mass required more than a two year transfer time, the result was ignored.

To summarize the results from this portion of the investigation, two examples are shown below. Figure 8 presents the propellant mass as a function of delivered mass for transfer times ranging from 0.5 years to 2 years. These results assume operation of the MaSMi Hall thruster at the nominal power point (330 W). The propellant mass vs. delivered mass for different transfer times and operation of MaSMi at the maximum power point of approximately 640 W is shown in Figure 9.

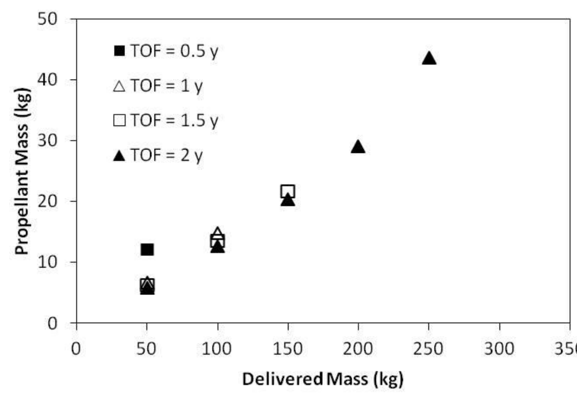

Fig. 8. Propellant mass as a function of delivered mass for a GTO-to-Earth-Moon L2 Halo orbit for specified transfer times with operation of MaSMi at the nominal 330 W power level.

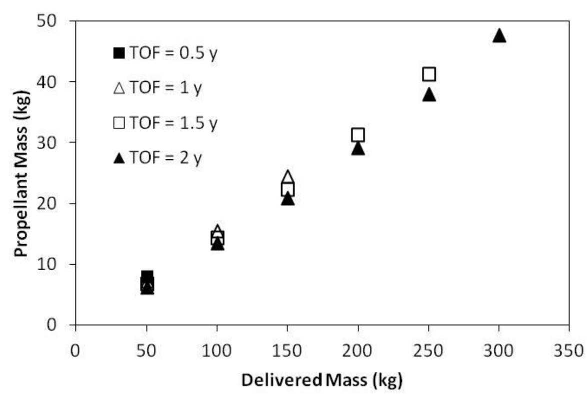

Fig. 9. Propellant mass as a function of delivered mass for a GTO-to-Earth-Moon L2 Halo orbit for specified transfer times with operation of MaSMi at the maximum 640 W power level.

V. DISCUSSION

The MaSMi thruster paired to a low-mass spacecraft enables considerable flexibility in spacecraft architecture and mission design, as illustrated by the asteroid rendezvous mission trajectories above. For example, a 20% decrease in initial available power (1000 W to 800 W) for a 200 kg spacecraft yields less than a 9% increase in mission time (5.8 y to 6.3 y) with nearly identical delivered mass to the target. If delivered mass is slightly less critical, a similarly sized spacecraft using MaSMi can accomplish the same mission in under 4.4 years with a delivered mass penalty of approximately 10-15% (~10 - 15 kg). By contrast, if delivered mass is mission critical and flight time is less of a constraint, a larger spacecraft initial mass can be supported by a MaSMi-based SEP system. A 7 year mission can deliver over 210 kg to Asteroid 118401 with a 1 kW solar array. The delivered mass increases to over 350 kg with a 2 kW array.

The Earth-Moon L2 Halo orbit transfer study revealed similar flexibility in a small satellite using the MaSMi Hall thruster. For all cases examined (and as highlighted in the plots above), the required propellant mass shows a nearly linear dependence on delivered mass, regardless of the resulting transfer time. The large throttling range of the MaSMi Hall thruster allows for significant mission flexibility, enabling a wide range of delivered masses to the L2 halo orbit.

Perhaps the most significant impact of a MaSMi-based small-spacecraft architecture is the capability to complete meaningful scientific missions at significantly lower spacecraft mass, and thus costs, in comparison to conventional (1000+ kg) spacecraft. One option for sufficiently small spacecraft is to share a launch vehicle with a larger spacecraft, where the smaller vehicle may be used as ballast. This reduces launch costs and, due to the noteworthy propulsion capabilities of a MaSMi-based spacecraft, the mission trajectory may not be negatively affected by a less-than-optimal initial low Earth orbit (LEO). Alternatively, a single launch vehicle may be used to launch several small MaSMi-based spacecraft to a LEO or GTO from where each spacecraft can independently thrust to their designated targets. The lower costs associated with small spacecraft also allow for a reasonable increase in acceptable risk for the system, enabling this architecture to be used as a technology demonstration platform as well as a scientific mission platform.

VI. CONCLUSION

The mission capabilities of a novel spacecraft architecture consisting of a low-mass (~200 - 300 kg) bus and a low-power, long-life Hall thruster-based solar-electric propulsion system was examined. Recent developments on a magnetically shielded miniature Hall thruster has provided performance and throttling data to enable this preliminary mission design; the development of the magnetically shielded miniature Hall thruster is an ongoing effort. Optimized trajectories to Asteroid 11841 (LINEAR) and to an Earth-Moon L2 halo orbit were computed using this spacecraft architecture. It was found that this new class of spacecraft grants considerable mission flexibility, enabling mission designers to deliver meaningful payload mass to desirable deep-space destinations while adhering to strict, and possibly changing, mission or spacecraft constraints. Using 200 - 550 kg spacecraft, mission flight times to Asteroid 118401 ranged from 4.4 to 7 years from a low Earth orbit with delivered payloads ranging from just under 100 kg to over 350 kg. Additionally, a similarly designed spacecraft was found to be capable of delivering up to 300 kg to the Earth-Moon L2 halo orbit from an Earth geostationary transfer orbit in no more than 2 years. These meaningful scientific missions are also accomplished at significantly reduced cost compared to conventional missions due to lower initial spacecraft mass, smaller launch vehicle requirements, and the possibility of sharing a launch vehicle. In sum, a low-mass spacecraft employing a low-power magnetically shielded Hall thruster offers a low-cost alternative for meaningful deep-space scientific missions.

VII. FUTURE WORK

In the near future, this investigation will be extended to other mission trajectories. One key study under consideration is a conventionally sized spacecraft (1000-2000 kg) rendezvous and orbit insertion to the outer planets using a low-power, long-life Hall thruster in parallel with a chemical propulsion system. While this mission is extremely challenging and yields low delivered mass using conventional chemical propulsion due to the high propellant mass requirements, preliminary calculations suggest that the use of a low-power, long-life Hall thruster may enable significantly increased delivered mass to the desired target. Other mission studies may include human precursor missions including rendezvous with near-Earth asteroids and the Jovian moons.

In a parallel effort, the magnetically shielded miniature Hall thruster will continue to undergo an optimization process. Changes to the thruster's geometry and/or materials will be made to yield improved performance and ensure sufficient lifetime to complete the missions discussed above. In addition to a full characterization of the optimized thruster, efforts will be made to accurately predict the useful life of the thruster.

ACKNOWLEDGMENTS

This research was funded by the UCLA School of Engineering and Applied Sciences and the NASA Space Technology Research Fellowship, grant number NNX13AM65H. A portion of this research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

The authors would like to thank Dr. Aline Zimmer, Dr. Steve Snyder, Dr. Ioannis G. Mikellides, and Dr. Richard R. Hofer for their contributions to this effort.

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