Evaluation of a 4.5 kW Commercial Hall Thruster System for NASA Science Missions

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The readiness of commercial Hall thruster technology is evaluated for near-term use on competitively-award, cost-capped science missions like the NASA Discovery program. Scientists on these programs continue to place higher demands on mission performance that must trade against the cost and performance of propulsion system options. Solar electric propulsion (SEP) systems can provide enabling or enhancing capabilities to several missions, but the widespread and routine use of SEP will only be realized through aggressive cost and schedule risk reduction efforts. Significant cost and schedule risk reductions can potentially be realized with systems based on commercial Hall thruster technology. The abundance of commercial suppliers in the United States and abroad provides a sustainable base from which Hall thruster systems can be cost-effectively obtained through procurements from existing product lines. A Hall thruster propulsion system standard architecture for NASA science missions is proposed. The BPT-4000 from Aerojet is identified as a candidate for near-term use. Differences in qualification requirements between commercial and science missions are identified and a plan is presented for a low-cost, low-risk delta qualification effort. Mission analysis for Discovery-class reference missions are discussed comparing the relative cost and performance benefits of a BPT-4000 based system to an NSTAR ion thruster based system. The BPT-4000 system seems best suited to destinations located relatively close to the sun, inside approximately 2 AU. On a reference near Earth asteroid sample return mission, the BPT-4000 offers mass performance competitive with or superior to NSTAR at much lower cost. Additionally, it is found that a low-cost, mid-power commercial Hall thruster system may be a viable alternative to aerobraking for some missions. Suggestions for the near- and far-term implementation of commercial Hall thrusters on NASA science missions are discussed.

I. Introduction

SINCE the mid-1990s, solar electric propulsion (SEP) technology has enjoyed a string of successes on-board commercial, military, and science spacecraft. These activities have been driven by the rapid increase of in-space power, the rise of the global telecommunications industry, and the realization of commercial and government investment in electric propulsion (EP) technology dating to the early 1960s. Commercial satellites in the United States (US) began using Hall and ion thrusters in 1997, and now 32 commercial geosynchronous earth orbiting (GEO) communication satellites of US origin have flown. Commercial programs in Europe, Russia, and Japan have

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also flown SEP or are nearing their first flights. The maturity of commercial product lines worldwide provides a sustainable base from which EP systems can be cost-effectively obtained for application on NASA science missions. In this paper, we evaluate the potential benefits that an EP system based on the 4.5 kW Aerojet BPT-4000 Hall thruster offers to cost-capped science missions such as the National Aeronautics and Space Administration (NASA) Discovery program. Efforts to expand the qualified power range of the BPT-4000 for application to NASA missions are discussed by Welander, et al.[1] The potential benefits of commercial ion thruster technology have also been evaluated by Oh and Goebel and Tighe, et al.[2,3]

Deep space science missions using SEP have proliferated in recent years. The 2.3 kW NSTAR ion thruster on the NASA technology demonstration mission Deep Space 1 (DS1) marked the beginning of these recent efforts. The success of DS1 was followed by the selection of the science mission Dawn. Dawn will use a significantly expanded version of the DS1 ion propulsion system (IPS) and is scheduled to launch in 2007.[4,5] Additionally, the Japan Aerospace Exploration Agency (JAXA) has demonstrated the use of microwave ion thruster technology on the Hayabusa asteroid sample return mission, while the European Space Agency (ESA) has demonstrated the use of commercial Hall thruster technology on the lunar orbiting SMART-1 technology demonstration mission. [6,7]

The high-specific impulse (1000-4000 s) of SEP systems primarily benefit missions by reducing propellant mass. This mass savings can be used to accommodate greater payloads or reduce launch vehicle class. In cost-capped missions, the capability to step down in launch vehicle class can be enabling since such a reduction represents several tens of millions of dollars in program costs. Assuming that a less expensive chemical propulsion system could complete the mission if a larger launch vehicle is used, the argument for using SEP is weakened if the cost of the SEP system approaches the cost differential of the launch vehicles. Solar array costs are continuing to decrease and so to must the EP system cost in order for SEP to broaden its application on cost-capped missions. It is therefore imperative that efforts to decrease the cost of EP systems be aggressively pursued.

Government-funded technologies such as the NSTAR IPS used on Dawn are inherently more expensive than commercial technologies since government systems are specialized and infrequently manufactured (less than twice a decade), making each build a unique activity. This makes government systems susceptible to cost and schedule risks for a variety of reasons that have motivated a reconsideration of the basic approach to implementing EP on NASA science missions. One possible solution is the implementation of a standard architecture for EP that aims to reduce the complexity and improve the manufacturability of government systems.[8,9]

The focus of this paper is to further the potential cost and schedule benefits of a standard architecture approach by the implementation of commercial technologies on cost-capped NASA science missions. This approach is enabled by the now relatively-widespread availability of commercial technologies worldwide since the launch of DS1 in 1998. Through procurements off existing product lines, it may be possible to significantly reduce system cost. In some cases, commercial EP systems may be cost competitive with chemical bipropellant systems. The success of this approach relies on our ability to identify the qualification gaps that exist between commercial systems and the requirements of science missions. A similar approach has been used with chemical propulsion systems for decades.

The current standard for deep space missions is the gridded ion thruster, such as NSTAR. This is mainly due to the ability of ion thrusters to operate at specific impulses (>3000 s) commonly thought to be optimum for interplanetary maneuvers. While this general principle is first-order accurate, there are classes of missions where other EP technologies can compete with ion thrusters, especially on cost-capped missions where the cost and performance of the EP system must be carefully considered during selection. Several deep space mission studies have considered the cost and performance benefits of Hall thrusters,[10,11,12,13,14] which due to their unique combination of simplicity (low-cost), robustness (low-risk), and performance (1200-2000 s for commercial models, up to 3000 s for advanced models), are now being implemented worldwide by nearly every major GEO spacecraft manufacturer.

In this paper, we evaluate the potential of a 4.5 kW commercial Hall thruster system for near-term use on competitively-award, cost-capped science missions like the NASA Discovery program. The present status of Hall thruster technology and flight heritage are first reviewed. A standard architecture envisioned for a NASA Hall thruster system is then presented and compared with commercial systems. This is followed by the identification of differences in qualification requirements between commercial GEO and NASA missions. Next, mission analysis is discussed for Discovery-class reference missions demonstrating how commercial Hall thruster technology can benefit missions of near-term interest. We conclude by offering suggestions for the near- and far-term implementation of commercial Hall thrusters on NASA science missions.

II. Overview of Hall Thruster Technology

A. Ion and Hall Thruster Comparison

The state-of-the-art 2.3 kW NSTAR ion thruster and the commercial 4.5 kW BPT-4000 from Aerojet are shown in Figure 1. Several important similarities and differences exist for ion and Hall thrusters. The major similarities between Hall and ion thrusters are the use of xenon propellant and hollow cathodes. The shared use of xenon allows both technologies to potentially use common propellant tanks, high-pressure propellant management assemblies and other feed system components (differences in mass flow control requirements lead to some important differences that are discussed below).

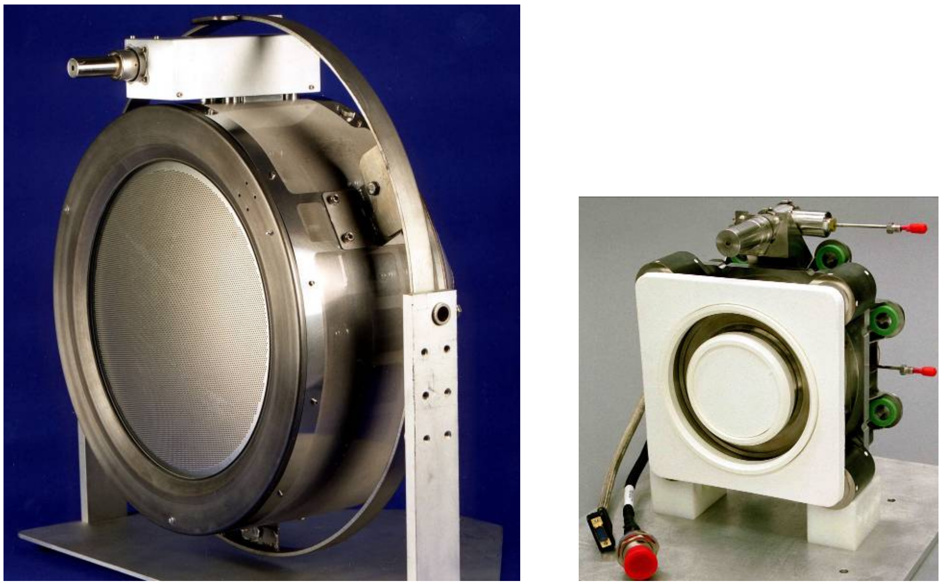


Figure 1. The 2.3 kW NSTAR ion thruster and the 4.5 kW BPT-4000.

Performance and life characteristics of typical Hall and ion thrusters are compared in Table 1. At constant power, Hall thrusters generally have lower specific impulse, efficiency, and total impulse capability (lifetime) than ion thrusters, but have higher thrust-power ratios. If a de-rating approach is taken, Hall thruster lifetime can approach that of an ion thruster. For instance, the lifetime of the nominally 4.5 kW BPT-4000 can compete with the nominally 2.3 kW NSTAR (see Table 5 in section IV). By using an overpowered Hall thruster for NSTAR-class applications, additional total impulse capability is gained that would not be possible if a Hall thruster of equivalent power were instead used. This approach trades the mass advantage Hall thruster systems have over ion thruster systems for additional life.

Table 1. Comparison of typical Hall and ion thruster performance and life characteristics.

	Hall Thruster	Ion Thruster
Specific Impulse	1000-3000 s	2000-4000 s
Thrust/Power	40-80 mN/kW	20-40 mN/kW
Efficiency	50-60%	60-70%
Impulse Capability	5-8 MN-s	>7 MN-s

From a systems perspective, Hall thruster systems are generally less complex than ion thruster systems, which can translate in to mass and cost reductions. The relative simplicity and reliability of Hall thrusters compared to ion thrusters like NSTAR is due to several differences, including:

- System Complexity
    - 1 less cathode (ion thrusters have two cathodes)
    - Power Processing Unit (PPU): No recycle circuit, Digital Control Interface Unit (DCIU) functionality built-in, fewer power supplies
    - Mass flow rate: Determined through closed-loop current control (no direct measurement required)
    - Reduced complexity leads to mass and cost savings
- Reliability
    - Lower parts count
    - No grids (reduced alignment issues, higher particulate contamination tolerance)
    - Lower operating voltage (200-400 V in Hall vs. 600-1100 V in ion) – No small high-voltage gaps

Important differences between the plumes from Hall and ion thrusters also exist. Hall thrusters with ceramic discharge channels have an advantage over ion thrusters in that their contaminant efflux is mostly ceramic, which has a much more benign effect on spacecraft optical surfaces compared to the metals sputtered from ion thruster grids. While this eases spacecraft integration, Hall thrusters have higher plume divergence than ion thrusters that somewhat offset these benefits.

One additional consideration is the long-term availability of commercial flight systems. In the US and abroad, Hall thrusters are being widely adopted by nearly every major commercial satellite manufacturer. This trend significantly increases the probability that commercial Hall thrusters will be available in the long-term for procurements from existing product lines.

In sum, the choice between a Hall or ion thruster for a particular deep space mission must trade the high-performance, higher complexity, and higher cost of an ion thruster against the lower performance, simplicity, and low-cost of a Hall thruster. These differences tend to create a natural divide in the mission applications for each system: low-cost Hall thrusters for moderate ΔV missions and high-performance ion thrusters for high ΔV missions.

B. Hall Thruster Flight Heritage

Commercially developed Hall thrusters typically operate between 50–60% efficiency, thrust densities on the order of 1 mN/cm^2, and specific impulses of 1200–2000 s. While Hall thrusters have historically been flown as single operating point devices, additional mission benefits can be realized for Hall thrusters capable of multi-mode operation.[10,15,16] Sometimes referred to as variable specific impulse operation, a multi-mode Hall thruster operates over a range of specific impulses either at constant or variable power. Efforts to expand the specific impulse range have shown that Hall thruster technology can be expanded in the near-term to specific impulses of 1000-3000 s with only minor design modifications.[17] Two examples of multi-mode Hall thrusters are the Aerojet BPT-4000 and Snecma PPS-1350-G. The BPT-4000 is qualified for GEO applications over a specific impulse range of 1700-2000 s and discharge power levels of 3.0-4.5 kW. A NASA funded effort to extend the life test of the BPT-4000 has recently demonstrated power throttling of 1-4.5 kW.[1] The Snecma PPS-1350-G, which is also being qualified for GEO applications at 1.5 kW,[18] was successfully modified for variable power operation on the SMART-1 mission.[7]

Major Hall thruster spaceflight milestones, with an emphasis on domestic achievements since the 1990s, are shown in Figure 2. Over 238 Hall thrusters have been operated in space on 48 spacecraft since 1971 when the Soviets flew a pair of SPT-60s on the Meteor satellite.[19,38] The 1998 launch of a TsNIIMASH D-55 on the National Reconnaissance Office's Space Technology Experiment Satellite (STEX) marked the first Hall thruster flight on a US spacecraft.[20] Commercial Hall thrusters are now available in the US from Aerojet, Busek, and International Space Technologies Incorporated (ISTI). The first commercial use of Hall thrusters by a US spacecraft manufacturer was in 2004 on Space Systems/Loral's MBSAT, which used Fakel SPT-100s provided through ISTI.[38] Busek will likely first demonstrate American Hall thruster technology when the 200 W BHT-200 flies on-board the Air Force TacSat-2 spacecraft in late 2006.[21,22] Domestically-developed Aerojet BPT-4000 Hall thrusters are scheduled to begin flying on Lockheed Martin Space Systems Company GEO satellites in 2008.[1,23,24] ESA recently demonstrated Hall thrusters for primary propulsion on the lunar orbiting SMART-1 technology demonstration mission. This spacecraft made use of the commercial PPS-1350-G and has set several new endurance records for the in-space use of Hall thruster technology.[7,18]

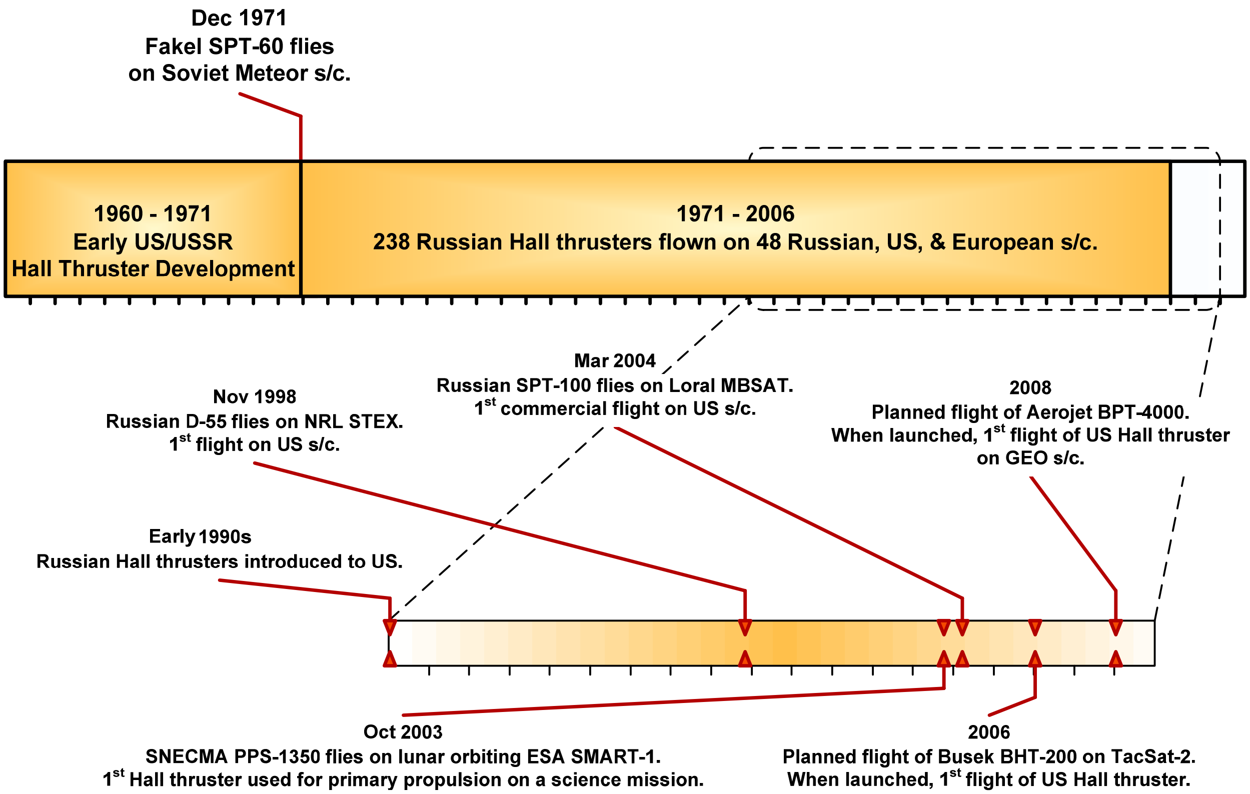


Figure 2. Hall thruster spaceflight milestones.

III. BPT-4000 System for Science Missions

A. System Architecture

In developing a Hall thruster propulsion system (HTPS) for a wide range of NASA missions, the system architecture must both meet the mission requirements and be reliably produced at low cost. High EP system cost and schedule delays have been negatively impacting potential flight implementations due to the cost constrained nature of most recent NASA proposal activities. These high EP system cost and schedule delays arise from a variety of reasons. First, because components are procured only a few times a decade, vendor management costs, engineering cost to resolve parts obsolescence issues, and minimum lot buys can not be spread out over a large number of units as they are in a product line. Second, the current NASA EP system architecture is extremely mission specific and requires substantial component redesign and requalification costs and schedule delays for each new mission. To mitigate the problem of high EP system costs for NASA missions, a study was performed to determine a standard architecture for NASA ion thruster systems.[8] Using these lessons learned, trade studies were performed to determine the optimal architecture for a standard NASA HTPS.

The results of these trade studies are shown in Figure 3. For the feed system, instead of an integrated xenon control assembly (XCA), a common high pressure regulation module combined with a distributed low pressure throttling module was chosen; because additional thrusters can be added with the addition of only one low pressure throttling module. To reduce the cost of power electronics and software redesign as the number of thrusters changes, the power and flow throttling command implementation capability was included in the PPU instead of a separate DCIU. This also significantly reduces programmatic costs by eliminating an additional system component which must be designed, managed, and qualified separately from the PPU. To insure proper fault containment a single string redundant PPU was chosen.

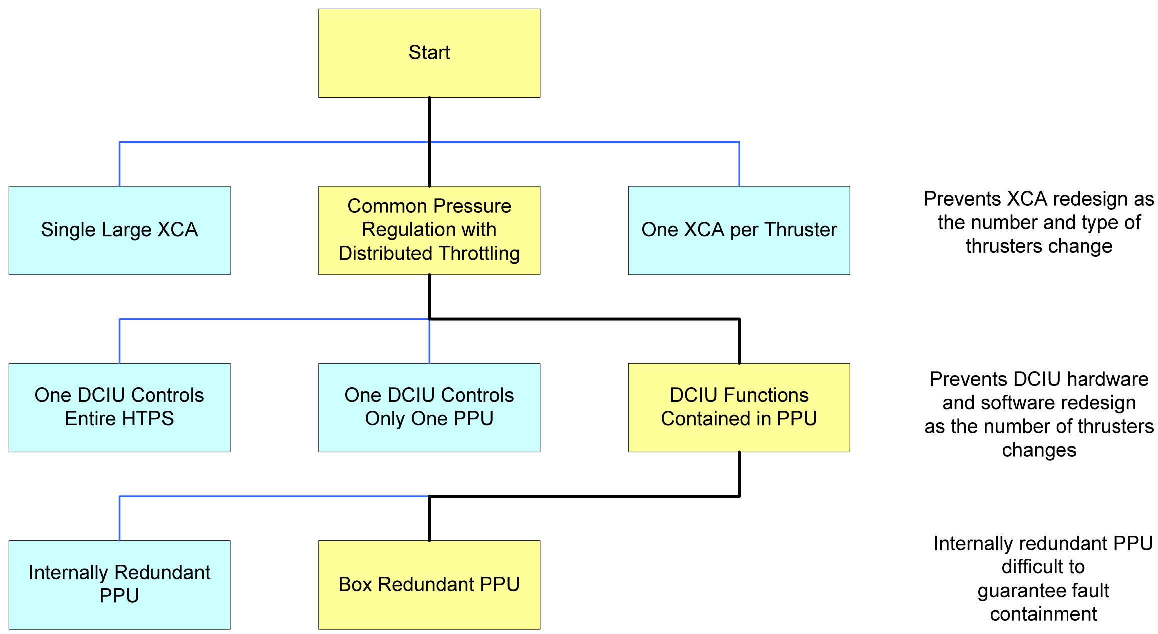


Figure 3. Hall thruster propulsion system (HTPS) standard architecture trade studies.

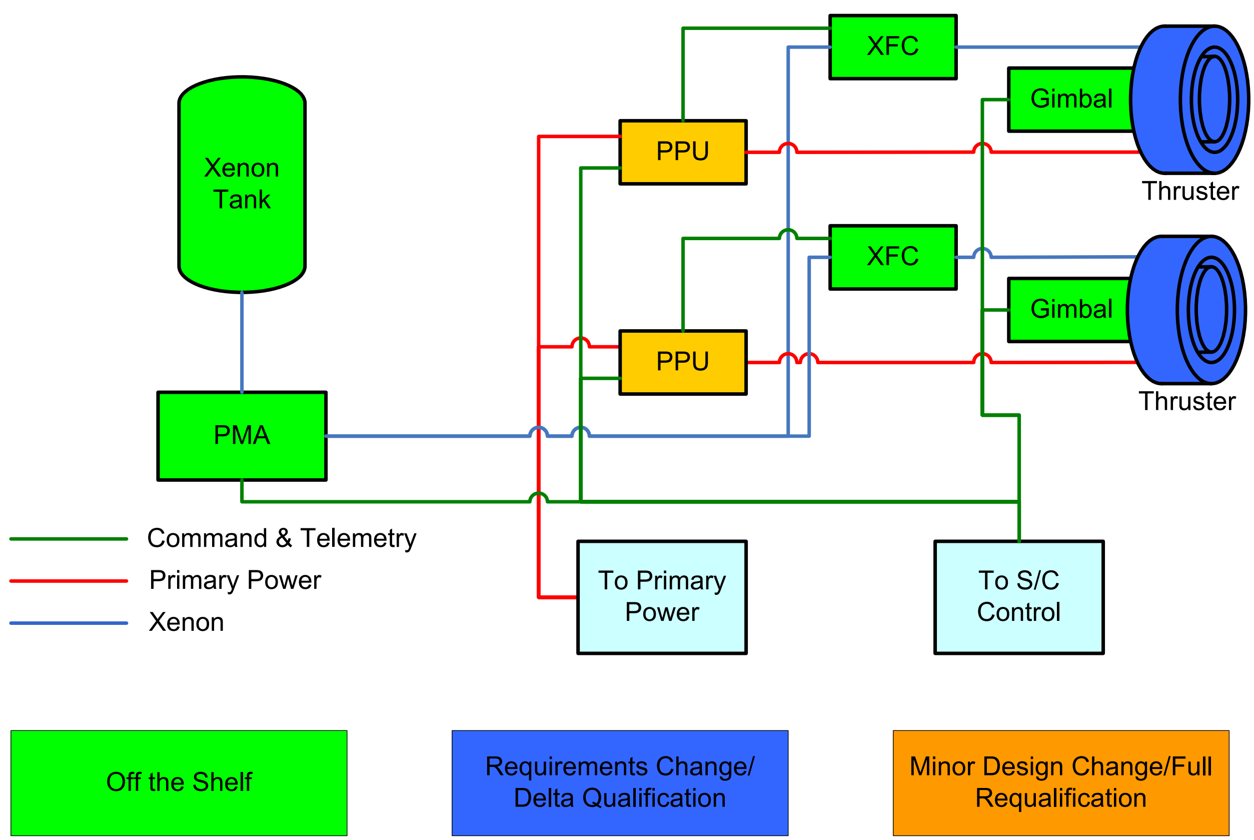


Figure 4. Hall thruster propulsion system (HTPS) block diagram.

The implementation of the trade study results is shown in Figure 4. This system architecture provides single string thruster, gimbal, xenon flow controller (XFC), and PPU combinations. Although it does not allow for cross strapping, it prevents fault propagation between strings. A single upstream propellant management assembly (PMA) allows both the distribution of propellant at low pressures and a simplified interface with the spacecraft electronics. Xenon tanks can be manifolded together depending on the required propellant load for the mission. Additional thruster strings can be added to the system with minimal impact.

Another benefit to this architecture is that it allows maximum commonality with commercial systems; thus allowing the possible integration of commercial off-the-shelf components into a NASA HTPS. The only deviation from some commercial systems is the use of dedicated PPUs for each thruster instead of cross strapped PPUs. Such cross strapping benefits station keeping systems which need two thrusters pointing in different directions; but only one thruster is required to fire at a time. For NASA missions, which primarily require impulse in one direction, the benefits of such cross strapping are reduced.

B. Commercial Hall Thruster Options

Given the wide range of applicability to NASA science missions for Hall thrusters with throttling ranges from a few hundred watts to several kilowatts, several commercial options exist to fulfill these requirements. These options include thrusters with flight heritage such as the Fakel SPT-70 and SPT-100, TsNIIMASH D-55, and the SNECMA PPS-1350.[7,19,20,25,38] Additional options have made substantial progress in development including the Aerojet BPT-4000, Fakel SPT-140, Busek BHT-200 and BHT-600, and SNECMA PPS-5000.[22,23,26,27,28] For this study, the BPT-4000 was chosen as it was the most mature design that could fit the throttling range and lifetime of most interest to near-term NASA missions (Figure 5). By selecting the BPT-4000 in combination with the HTPS architecture described above, the amount of system component development and qualification effort is substantially reduced. As shown in Figure 4, no design changes are required for the thruster, gimbal, XFC, PMA and xenon tanks. The PPU requires only a change to input bus voltage requirements (see section III.C below).

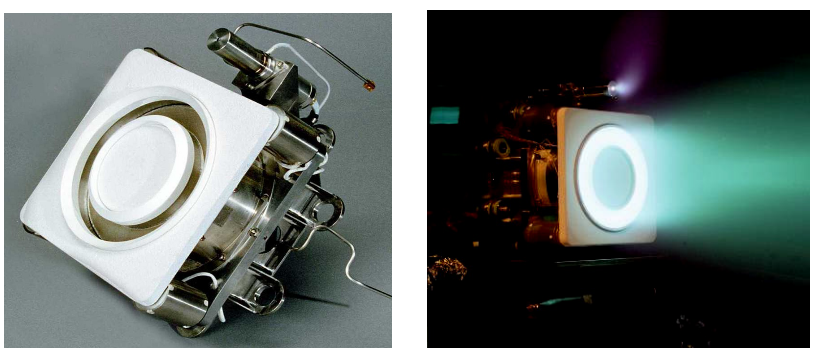
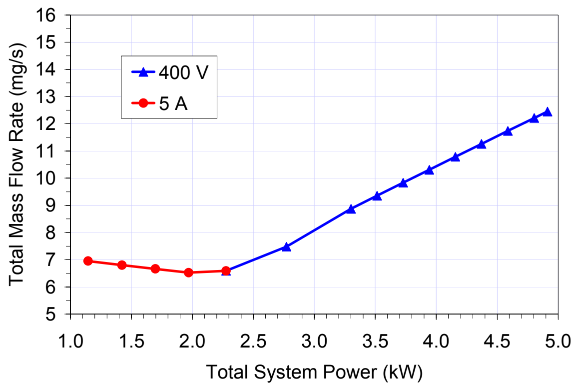


Figure 5. Aerojet BPT-4000 Hall thruster.

The Aerojet BPT-4000 thruster has recently completed qualification testing for GEO applications. This qualification included operation in the 3.0 to 4.5 kW input power range and the 300 to 400 V input voltage range. Life testing included 5800 hours and 6700 cycles of operation with a demonstrated throughput of 250 kg xenon and 4.9 MN-s total impulse (see Table 5).[1] Qualification of the thruster was completed in 2005 with the first flight deliveries planned in 2006 for the first flight in 2008.

The BPT-4000 qualified 3.0-4.5 kW input throttling range is inadequate for the wider range low power operation required for most NASA missions; however, the thruster has demonstrated outstanding performance at low power (down to 1 kW) during the recent extended life test funded by NASA.[1] Using time-averaged data from both the GEO qualification life test and the current extended life test, a throttling curve extended to the low power regime was constructed for NASA missions (see Figure 6). By throttling current and voltage, it is possible to construct separate "high thrust" and "high specific impulse" throttling curves depending on which parameter is more important to the mission; however, nearly all trajectory analysis performed to date has shown superior mission performance for the throttling curve with the highest specific impulse. The performance curves in Figure 6 reflect mission need for high specific impulse by always operating at the highest voltage possible in the current PPU (400 V). This 400 V "high specific impulse" operation is limited by the minimum discharge current for stable thruster operation, which is approximately 5 A (2 kW). Below 2 kW discharge power, voltage is throttled from 400 to 200 V at constant current.



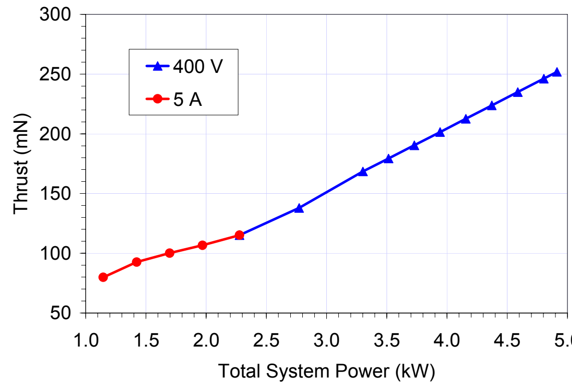


Figure 6. Total mass flow rate (thruster plus cathode) and thrust as a function of input power to the PPU. Above 2.3 kW, the voltage is fixed at 400 V and the current is allowed to vary. Below 2.3 kW, the current is fixed at 5 A and the voltage is allowed to vary.

C. Power Processing Unit (PPU)

In concert with the BPT-4000 thruster, the Aerojet BPT-4000 PPU has also recently completed qualification testing for GEO applications.[24] The PPU design provides commandable power to the thruster plasma discharge, thruster electromagnets, cathode heater, and cathode keeper (Figure 7). In addition, the PPU drives two solenoid-holding valves in the XFC and utilizes closed loop control to operate the proportional flow control valve (PFCV) to regulate the xenon flow provided to the thruster, which controls the discharge to the commanded level. The qualification operational range and lifetime requirements are the same as the thruster. Commands and telemetry are communicated with the spacecraft utilizing a MIL-STD-1553B data link. The PPU input power is designed to interface with a regulated 70 V spacecraft power bus. Qualification of the PPU was completed in 2005 with the first flight deliveries planned in 2006 for the first flight in 2008. Radiation hardened S-Level components are utilized to provide maximum reliability.

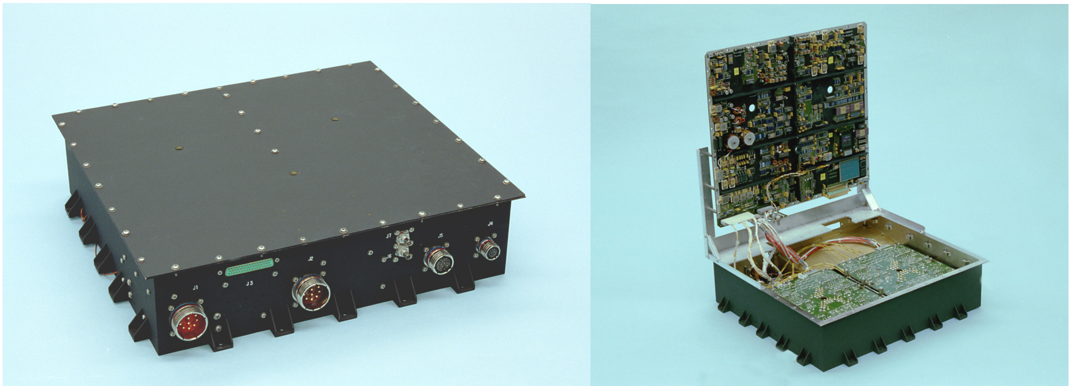


Figure 7. Aerojet BPT-4000 Power Processing Unit (PPU).

As with the thruster, the qualified PPU 3.0 to 4.5 kW throttling range is inadequate for the wider range low power operation required for most NASA missions; however, the PPU is designed to allow operation over a wider throttling range between 0.6 to 4.5 kW at voltages between 150 and 400 V. PPU efficiency over this range is shown in Figure 8. The difference between the qualification and design performance envelope would only require a delta qualification; however, the input power bus power voltage range is incompatible with NASA interplanetary spacecraft. Instead of a regulated 70 V spacecraft bus input, the PPU power interface would need to be changed to accommodate a variable 70 to 130 V input voltage range. The risk to such a change is relatively low; however it will still require a new design and a dedicated engineering model to be subjected to full qualification testing.

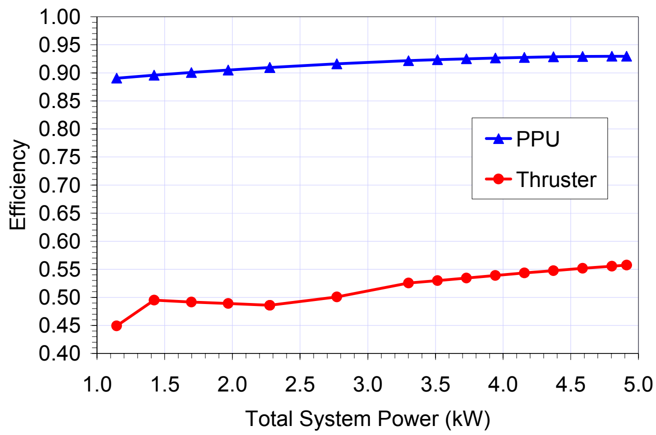


Figure 8. PPU and thruster efficiencies versus input power to the PPU.

D. Xenon Feed System (XFS)

To maximize commonality with existing commercial off the shelf hardware, the Moog Model 50E947 was chosen for the XFC (Figure 9). This XFC successfully completed qualification at the component level in 2003 and system level qualification with the BPT-4000 thruster and PPU in 2006. The XFC, which includes two solenoid valves for anode and cathode isolation and a proportional flow control valve (PFCV) for throttling, provides the appropriate flow over the range of operating conditions.[29] Maximum expected operating pressure is 2700 psi while the normal operating pressure at the inlet is 34 to 40 psi. The total controlled flow rate range is 6 to 20 mg/s with a 5 to 9 % cathode to anode flow split provided by solenoid valve orifices. All the electrical interfaces are with the BPT-4000 PPU.

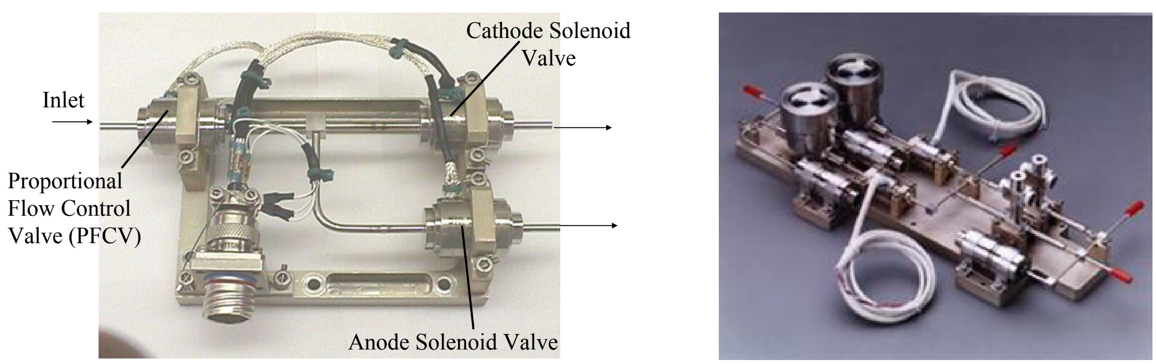


Figure 9. Moog xenon flow controller (XFC) and propellant management assembly (PMA).

For the propellant management assembly, both a Moog option (with a Moog regulator, see Figure 9) and a Vacco option (with a Stanford Mu regulator) exist.[30,31] For the purposes of this study, the Moog option was chosen based on the flight heritage and ease of integration into the HTPS architecture. The PMA, consisting of parallel redundant regulator/isolation latch valve legs with additional fill drain valves and pressure transducers, provides propellant isolation and pressure regulation. The inlet pressure range is 100 to 2700 psi. Over a flow rate range of 4 to 60 mg/s, the regulated outlet pressure is 35.5 to 38.5 psi. All the PMA electrical interfaces are with the spacecraft computer.

E. Xenon Propellant Tank

Selection of a propellant tank is strongly mission dependent on the xenon load specified by the trajectory and other reserves. Fortunately, as shown in Table 2, a wide variety of qualified xenon tanks already exist and most have flight heritage.[32,33,34,35,36] All of these tanks have been flown in ion and Hall thruster applications except where indicated. For simplicity in this study, the Carleton Technologies tank used for the Dawn mission was baselined. Shown in Figure 10, this tank is a skirt mounted tank with a titanium liner.

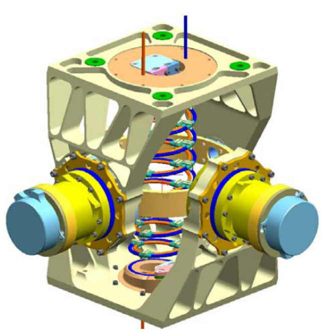
Table 2. Qualified composite overwrapped pressure vessels qualified for xenon application. * = largest diameter of a conical tank

Manufacturer	Part #	Dia	Len	Vol	Mass	Xe	MEOP	Flown
		(in)	(in)	(lit)	(kg)	(kg)	(psi)	
ARDE	D4790	28.3	28.3	178.4	39.5	300.0	2650	Yes
Carleton	7169	35.5	26.5	267.9	22.2	450.0	1750	No
General Dynamics	220142-1	12.9	37.9	65.5	9.1	112.7	2700	Yes
General Dynamics	220145-1	13.0	37.0	65.5	10.8	115.0	3500	Yes
General Dynamics	220615-1	12.9	50.6	81.9	11.1	140.9	2700	Yes
PSI	80386-101	13.2*	29.6	32.1	6.4	56.0	2500	Yes
PSI	80412-1	12.8	27.5	50.0	6.8	90.0	2176	No
PSI	80458-1	16.5	44.3	132.7	20.4	228.6	2700	No
PSI	80458-201	16.5	27.0	54.1	12.2	93.1	2700	No

Figure 10. Dawn xenon propellant tank manufactured by Carleton Technologies.

F. Thruster Gimbal

To maximize compatibility with the BPT-4000 thruster, the Moog Hall thruster gimbal was chosen. This gimbal successfully completed qualification in 2006 for GEO applications (Figure 11). The Moog gimbal has dual axis rotary actuators with stepper motors and helical accommodations for thruster propellant lines and harnesses. The performance easily meets the typical requirements of NASA spacecraft with a range of +/- 36.5 degrees in both axes, an angular accuracy of better than 0.02 degrees, and an angular velocity of 1 degree per second nominally.[37]



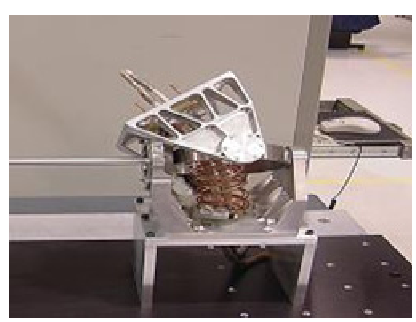


Figure 11. Moog thruster gimbal.

G. System Mass and Cost Estimates

Mass estimates for Hall and ion thruster systems are provided in Table 3 for 1+1 and 2+1 single-string configurations (i.e., primary plus redundant strings). The ion thruster system mass is based on NSTAR flight hardware from the Dawn mission. The Hall thruster system mass does not include valve drivers for the PMA latch valves and the gimbal drive electronics, as the ion thruster system mass estimate does; however this should only affect the totals by a few kilograms. Also, as discussed in the mission applications section, for an equivalent mission, Hall thruster systems will tend to require larger xenon propellant mass loads; thus it may be possible to save additional mass by using lighter xenon tanks for the ion thruster system. Even with these factors taken into account, the mass benefits of a Hall thruster system over an ion thruster system are substantial.

The cost benefits of a 1+1 ion thruster system versus a 1+1 Hall thruster system is shown in Figure 12, which compares in a relative sense the estimated cost of the next flight of the NSTAR IPS versus the estimated cost of the first flight and of follow-on flights of the BPT-4000 HTPS. The 1st flight cost includes the cost of the PPU redesign and delta qualification, first flight feed system design documentation changes, and the delta qualification of the thruster. Figure 12 shows that even for the first deep space use of the BPT-4000, the cost of the system is substantially less than the cost of an equivalent NSTAR system. In addition, Figure 12 shows that once the non-recurring costs associated with the first flight have been retired, the cost of the commercial EP system hardware is expected to be competitive with the cost of a bipropellant chemical propulsion system. This offers the potential to make EP cost competitive for missions currently conducted using aerobraking (see section V.D below).

Table 3. Current best mass estimate for Hall and ion systems. Both 1+1 and 2+1 systems include the additional mass associated with redundant feed system, thruster, PPU, and gimbal components. Mass estimates include contingencies of at least 5%.

Components	1+1 Hall Mass (kg)	2+1 Hall Mass (kg)	1+1 Ion Mass (kg)	2+1 Ion Mass (kg)
Thrusters	26.78	40.17	18.69	28.04
PPUs/DCIUs	26.25	39.38	42.28	57.65
Feed System	6.06	7.37	20.25	21.16
Tanks	22.20	22.20	22.20	22.20
Gimbal	10.50	15.75	9.76	14.64
Misc	2.00	3.00	4.00	6.00
Total	93.8	127.9	117.2	149.7

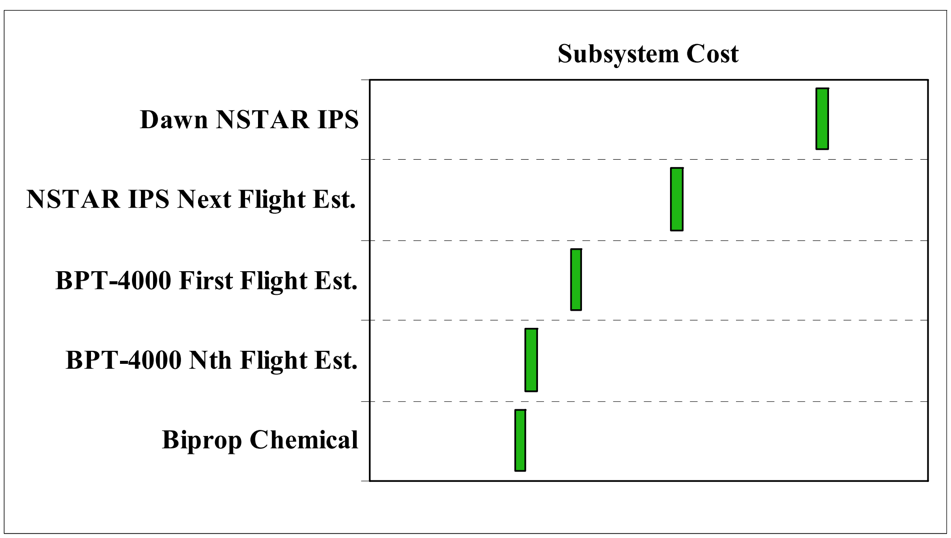


Figure 12. Cost comparison of 1+1 NSTAR, 1+1 BPT-4000 and chemical propulsion systems. Dawn used a cross-strapped PPU 2+1 system. (Note: this chart does not include the cost of the solar array).

IV. Mission Assurance

A common misconception regarding the application of commercially-developed hardware to NASA science missions is that the different applications, e.g. deep-space vs. geosynchronous orbit, mandate vastly different requirements. In practice, many of the requirements can be quite similar: both types of spacecraft must be qualified for the launch vehicle ride out of the Earth's atmosphere and possibly an additional chemical propulsion stage to reach their final orbit or trajectory; thermal environments inside the spacecraft structure are controlled within set limits; electromagnetic compatibility is assessed according to standard guidelines; and mission assurance analyses and documentation must be completed. The complete set of mission assurance requirements should be evaluated on a case-by-case basis to determine the adequacy of commercial hardware for NASA science missions. The use of commercial Hall thrusters for use on a science mission is not without precedent. ESA has successfully adapted the commercial PPS-1350-G Hall thruster for the lunar technology demonstration mission SMART-1.[7] Here, we will compare some of the typical requirements for NASA missions to the work that has been performed on commercial hardware.

The BPT-4000 subsystem thruster, gimbal, XFC, and PPU have completed qualification for GEO applications.[1] The qualification program and accompanying mission assurance analyses and documentation can be compared to those required for typical NASA science missions to understand the similarities and differences, and ultimately what additional work must be done to implement such a system on NASA missions. For comparison, the commercial components considered here will be evaluated against requirements for the Dawn mission, selected under NASAs Discovery Program to visit two major main-belt asteroids, which will be the first use of an ion propulsion system on a full-up NASA science mission.[5] The Dawn IPS includes three NSTAR ion thrusters and accompanying gimbals mounted on the exterior of the spacecraft, two PPUs, a xenon tank, and propellant management hardware inside of the spacecraft. The use of Dawn as a benchmark for the initial evaluation of commercial components is appropriate because (1) it is an active mission in a cost-capped program, (2) it is using an EP subsystem for primary propulsion, and (3) the mission assurance requirements are mature. Note, however, that different NASA science missions can have different requirements and there are also differences in hardware environmental requirements that are unique to their design (e.g. an ion thruster vs. a Hall thruster).

Requirements for dynamic environments, e.g. vibration and shock, are largely dependant on launch vehicle and spacecraft configurations and they are among the easiest mission assurance items to compare. Table 4 compares the random vibration requirements for the Dawn IPS to the test programs for the commercial components discussed here. The test programs for all components exceed the Dawn requirements by large margins at all frequencies except for the PMA which meets the requirements but has lesser margin. Pyroshock requirements are more dependent on the spacecraft configuration and specifically the presence of pryo-released mechanisms. All of the commercial components shown in Table 4 exceed the Dawn requirements for shock. Many qualified and flight-proven xenon tanks exist as options for a Hall thruster subsystem; they were not specifically addressed as a part of this initial mission assurance evaluation.

Table 4. Random vibration levels for EP system components.

Component	Dawn IPS Requirement (Grms)	Commercial Hardware Test Program (Grms)
Thruster	8.4	18.0
Gimbal	8.4	18.0
PPU	8.4	✓
XFC	13.0	18.6[a], 23.6[b]
PMA	13.0	14.9

Notes: (a) in-plane, (b) normal plane, (✓) exceeds Dawn levels

Thermal environments for EP subsystem components are relatively simple to evaluate for components which are inside the spacecraft bus such as the PPU and PMA, or for those that can be outside of the bus but protected with thermal blanketing such as an XFC. In these cases the thermal environments are designed and controlled by the thermal systems. Slight differences between NASA requirements and commercial test programs can be resolved, for example, by adding additional heaters, blanketing, or increasing radiator area. More consideration must be given to requirements for externally-mounted hardware (i.e., thruster and gimbal) because there can be appreciable differences in the thermal requirements due to mission type. For example, equipment mounted outside a GEO communications satellite structure must be able to survive the unique thermal conditions during the eclipse seasons, and NASA deep-space missions can expect much less solar heating and thermal cycling.

With these caveats, thermal requirements for the Dawn IPS and commercial components can be compared as shown in Figure 13. The PPU and XFC meet or exceed the Dawn IPS component requirements, and the commercial PMA temperature limits are different from the DAWN requirement by 10 °C on both the hot and cold sides. These differences for the PMA could be resolved by re-testing the PMA to the larger temperature range, or by the aforementioned addition of thermal control design on the spacecraft.

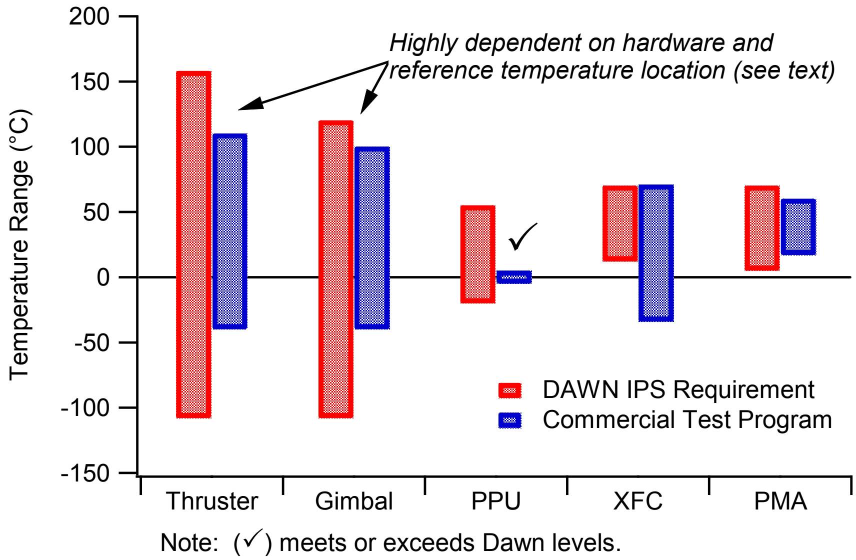


Figure 13. Temperature range requirements for electric propulsion subsystem components.

The requirements for the externally-mounted thruster and gimbal are more difficult to compare directly. Some of these differences are due to the different mission classes (GEO vs. main asteroid belt) but a significant part of the difference in the thruster requirements is also due to the location of the temperature reference point for qualification. The Dawn ion thruster temperature reference is on the front mask of the thruster which has a full view to deep space whereas the reference location for the BPT-4000 thruster is at the mounting foot which does not have such a view. Therefore, the cold-side temperature requirements can be significantly different. On the hot side, the Dawn thruster front mask is adjacent to the front magnet ring which is a relatively hot portion of the thruster, whereas the BPT-4000 reference is not adjacent to a hot portion of the thruster. The gimbal temperature requirements are also highly dependent on their view to space, temperature reference location, and surrounding thermal environments. The lesson to be drawn from this comparison is that externally-mounted components must be evaluated on an individual basis within the framework of the thermal configuration of the spacecraft. Where necessary, additional analysis or test can be performed if the thermal test program for a commercial component does not match the requirements of a NASA science mission. Often, components and their materials are perfectly capable of meeting tougher requirements but their test programs were designed only to meet the easier requirements of a different mission.

In addition to dynamics and thermal environments, electromagnetic compatibility (EMC) requirements form a large part of the qualification test program for spacecraft hardware. Electronics boxes such as the PPU and hardware which use power and instrumentation such as the XFC and PMA typically are required to meet the requirements detailed in the military standard MIL-STD-461, "Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference." These requirements are typically modified for specific programs based on known differences, e.g. radiated susceptibility requirements for NASA spacecraft may be modified for launch base range radar frequencies, launch vehicle transmitters, and spacecraft communications bands. Commercial hardware is tested against EMC requirements in the same way and the results can be evaluated against NASA science mission requirements similar to the dynamic and thermal environments.

Radiated emissions from electric thrusters are often cited as a source of concern, but the experience from Hall and ion thrusters on commercial and civil spacecraft across the world is that with good design there are no significant problems.[38] Thrusters can be evaluated for emissions at specific spacecraft communications frequencies to aid in spacecraft design.[39,40] Whether a NASA science mission uses a commercial thruster or a government-developed thruster the process is similar. The BPT-4000 thruster has undergone an extensive array of tests and analyses to demonstrate compatibility with GEO communications spacecraft.[41,42]

There is the possibility for significant differences in commercial hardware test programs and the requirements of NASA science missions in the areas of radiation tolerance and other space effects. These can depend on the mission duration and profile, and should be evaluated on a case-by-case basis. Commercial components are typically not rated for heavy radiation environments such as those found near Jupiter. However, note that the PPU for the BPT-4000 system includes radiation-hardened S-level components.

The effects of the thruster plume on spacecraft surfaces is a potential concern, but one that has been extensively characterized over recent years and can be mitigated with good design. The main areas of concern include sputtering of spacecraft surfaces due to high-energy particles in the plume, the forces imparted on spacecraft surfaces due to the momentum flux in the plume, and the deposition on the spacecraft of contaminant efflux from the thruster. Many experimental and modeling studies have been performed for Hall thruster plumes[43,44,45,46] and also for EP missions beyond GEO.[47,48] The plume of the BPT-4000 thruster has been characterized through detailed diagnostic experiments,[49] materials sputter testing and modeling,[50] and spacecraft-level plume effects modeling.[51]

Plume effects depend in part on the specific thruster used but also importantly on the spacecraft configuration. The configuration of GEO communication spacecraft typically makes a trade between adverse plume effects on solar arrays and thrust loss due to cant angle, whereas deep-space spacecraft using electric thrusters as primary propulsion can be configured to minimize or eliminate direct plume impingement on spacecraft surfaces. As an example of a thruster-specific plume effect, it is noted that Hall thrusters with ceramic discharge channels like the BPT-4000 have an advantage over ion thrusters in that their contaminant efflux is mostly ceramic, which has a much more benign effect on spacecraft optical surfaces compared to the metals sputtered from ion thruster grids. Regardless of thruster type or spacecraft configuration, however, the tools that have been developed to-date to model plume effects on spacecraft can be used to design and analyze NASA science spacecraft with confidence. There is little reason to expect major differences in plume effects on spacecraft due to the differences between the GEO and the deep-space environments. Additional measurements and modeling at BPT-4000 low-power operating points are warranted before application to NASA science missions, but are not expected to reveal any new or significant issues.

The throughput and total impulse characteristics of NSTAR and BPT-4000 are compared in Table 5. The table shows the results from qualification testing and the resulting usable flight limits after applying standard 50% margins. Although throughput is often used to quantify the lifetime of ion thrusters, comparing the total impulse of NSTAR and BPT-4000 is more appropriate due to the difference in specific impulse of the thrusters. For example, without applying margins, the NSTAR life test processed 235 kg of xenon or 7.2 MN-s of total impulse in just over 30,000 h while the BPT-4000, if tested through a Phase Two life test extension (discussed below), will process approximately 426 kg of xenon or 7.7 MN-s of total impulse in just over 13,000 h.[52] Hence, after completion of the Phase Two life test extension, the BPT-4000 will have a greater lifetime than NSTAR as measured by total impulse delivered to a spacecraft.

Table 5. Total impulse capabilities of NSTAR and BPT-4000.

	Life Test Results (or Plans)		Usable Flight Limits (50% Margins)		
Test Description	**Throughput (kg)**	**Total Impulse (MN-s)**	**Throughput (kg)**	**Total Impulse (MN-s)**	**Notes**
NSTAR	235	7.2	157	4.8	Completed
BPT-4000 - 5,800 h Life Test	250	4.9	167	3.3	Completed FY05
BPT-4000 - Phase One Extension	274	5.3	183	3.5	Funded FY06
BPT-4000 - Phase Two Extension	426	7.7	284	5.1	Funding requested for FY07/08

Although exceptions exist, Discovery-class missions typically have total impulse requirements similar to or exceeding the capabilities of NSTAR.[12] This has prompted NASA to fund a low-power life test extension of the BPT-4000,[1] as well as an effort to develop predictive life models of Hall thrusters.[53,54] Figure 14 compares the usable total impulse, or flight total impulse limits, for the BPT-4000 after the previously completed 5,800 h life test and the NASA funded Phase One (on-going) and Phase Two (proposed) life test extensions. Results from an Aerojet erosion model of the BPT-4000 indicate that the thruster is capable of completing Phase Two without exposing the magnetic pole pieces. The Phase One extension is nearly complete and aims to demonstrate 1,000 h of 1 to 2 kW operation with the same qualification unit from the 5,800 h test. When Phase One is completed, the total impulse capability of the BPT-4000 will be extended to approximately 3.5 MN-s, but more importantly, will demonstrate long duration firing of this thruster at low-power. If funded, Phase Two of this life test extension aims to increase the total impulse capability of the BPT-4000 to approximately 5.1 MN-s. Completing Phase Two enables the BPT-4000 to deliver NSTAR impulse capabilities for NASA science missions, which will qualify the thruster for the largest number of near-term applications.

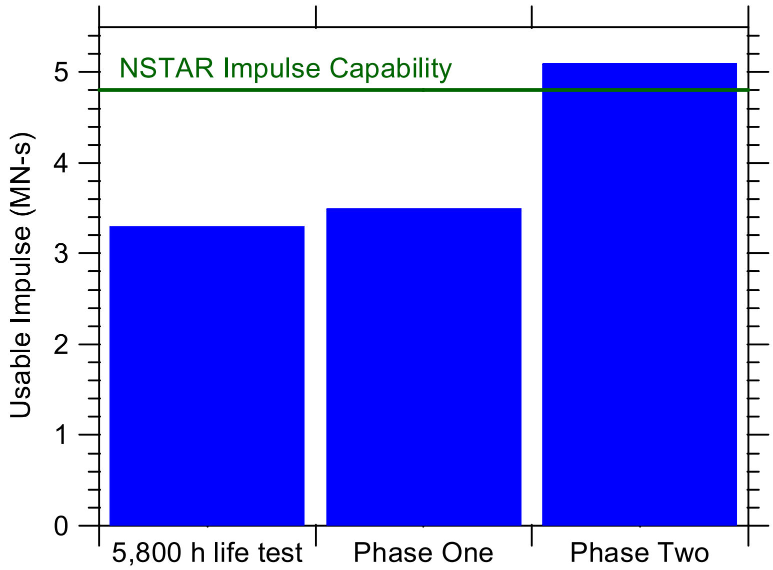


Figure 14. Usable total impulse, or flight total impulse limits, for the BPT-4000 after the previously completed 5,800 h life test and the NASA funded Phase One (on-going) and Phase Two (proposed) life test extensions. Completing Phase Two enables the BPT-4000 to deliver NSTAR impulse capabilities for NASA science missions.

A final concern for mission assurance is in the analyses and documentation which are produced by commercial product vendors as a part of the hardware build and test program. The types of documents which are required are similar across the commercial and civil spacecraft industries, although various parts can be unique to each institution. These include Contamination Control plans; Parts, Materials, and Processes Lists; Derating Analyses; Failure Modes, Effects, and Criticality Analyses; Project Safety; and Reliability Predictions and Analyses. When commercial components come under serious consideration for use, NASA Quality Assurance personnel would review these documents thoroughly as they would for any vendor.

To summarize, there is no fundamental reason why commercial electric propulsion system components can not be readily implemented on NASA science missions. With few exceptions, the environmental requirements for which commercial hardware is designed are similar to those for NASA missions. Many of the cases for which differences exist can be resolved through additional design, test, or analysis (e.g. PMA temperature requirements) at much less cost than for design and fabrication of a new component specific to a single government mission. In those cases where the requirements may be truly unique to a NASA science mission, such as the radiation environment near Jupiter, additional work or a new design may be necessary.

V. Mission Applications

The NASA Discovery program gives scientists the opportunity to address questions in solar system science with lower-cost, highly focused planetary science missions. The Dawn project is the ninth Discovery program project and is designed to increase our understanding of the early solar system by investigating two main belt asteroids, Vesta and Ceres.[55] Dawn utilizes an NSTAR ion thruster based EP system for primary propulsion.[5] This system represents the current state of the art for NASA deep space SEP systems. This section examines the benefit that a SEP system based on the 4.5 kW BPT-4000 commercial Hall thruster could have for Discovery class deep space missions. It compares the relative cost and performance of a BPT-4000 based system to an NSTAR ion thruster based system and discusses the possible range of application for this system.

Discovery missions are selected competitively and cover a wide range of scientific goals and destinations. Previous studies examining the benefits of SEP technologies for Discovery class missions have identified a set of reference missions to generic destinations that are generally representative of current and proposed missions that utilize EP.[10] In this section, the performance of the BPT-4000 to NSTAR is compared for two reference missions defined in Ref. 10:

- Near-Earth Asteroid Sample Return Mission
- Comet Rendezvous Mission

The use of SEP as an alternative to aerobraking for small planetary science orbiter missions is also considered.

A. Thruster Performance Model

A BPT-4000 system performance model was created from the throttle table shown in Figure 6 by fitting polynomials to the throttle table to generate expressions for thrust and propellant flow rate as a function of input power to the PPU.

Table 6 shows the resulting polynomial coefficients for the BPT-4000 as well as coefficients for the NSTAR thruster. The NSTAR curve fit is derived from NSTAR throttle table Q-mod (or modification to Table Q) which is based on DS1 flight data and is the throttle table used for mission planning in the initial design phases of the Dawn program.[56] Figure 15 shows the resulting thrust curve for both thrusters. The overall system specific impulse and efficiency can be derived from these expressions and are shown in Figure 16 and Figure 17, respectively. Table 7 shows the available operating range and throughput capability for each device. The impulse capability of each thruster was shown previously in Table 5.

Table 6. Throttle curve coefficients for NSTAR and BPT-4000.

		Mass Flow Coefficients				
System	Table Name	A	B	C	D	E
NSTAR	Q-Mod	0.36985	-2.5372	6.2539	-5.3568	2.5060
BPT-4000	Table 2	0.0025	-0.2629	2.712	-7.232	12.204
		Thrust Coefficients				
System	Table Name	A	B	C	D	E
NSTAR	Q-Mod	5.145602	-36.720293	90.486509	-51.694393	26.337459
BPT-4000	Table 2	-0.6574	6.2683	-14.282	37.751	49.466

$$\dot{m}[mg/s] = A\left(P[kW]\right)^4 + BP^3 + CP^2 + DP + E$$

$$T[mN] = A\left(P[kW]\right)^4 + BP^3 + CP^2 + DP + E$$

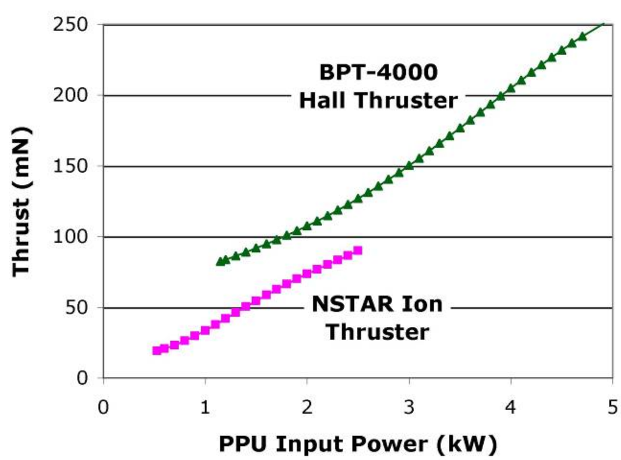


Figure 15. Thrust vs. PPU input power from the throttle curves of Table 6.

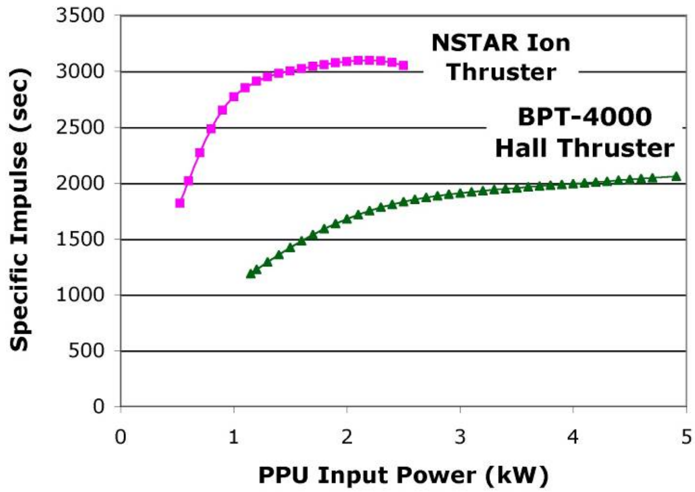


Figure 16. Specific impulse vs. PPU input power from the throttle curves of Table 6.

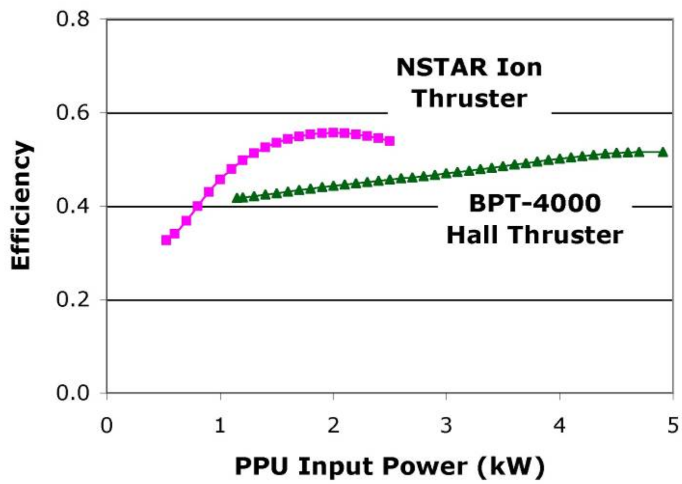


Figure 17. Propulsion system efficiency vs. PPU input power from the throttle curves of Table 6.

Table 7. Thruster characteristics summary.

	TRL	Minimum PPU Input Power	Maximum PPU Input Power
NSTAR	9	0.53 kW	2.60 kW
BPT-4000	6	1.15 kW	4.91 kW

When compared to NSTAR at equivalent power levels, the BPT-4000 generally operates with higher thrust, lower specific impulse and lower efficiency. However, the BPT-4000 has higher maximum power and wider absolute throttle range, although its minimum operating power is twice that of NSTAR. As will be shown, the higher thrust and high maximum power generally enhance mission performance when operating relatively close to the sun at high power levels while the higher minimum power decreases mission performance when operating far away from the sun or at low power levels.

B. Near Earth Asteroid Sample Return Mission

The first reference mission examined was a near Earth asteroid sample return mission. The spacecraft launches on a Delta II 2925 directly to an Earth escape trajectory and uses SEP to rendezvous with the asteroid Nereus. The spacecraft remains in the asteroid's vicinity for 90 days before using SEP to return to Earth and conducts a flyby as it releases the sample for direct entry. The basic characteristics of this mission are shown in Table 8.

Table 8. Near earth asteroid sample return mission characteristics.

Near Earth Asteroid Sample Return	
Target Body	Nereus
Launch Vehicle	Delta 2925
Power System	Triple junction GaAs solar array, 6 kW at 1 AU
Bus Power	300 W
Duration	3.3 years
Onboard ΔV	4.2 to 6.5 km/s
Launch Year	2007/08
Ion/Hall Thruster Duty Cycle (for a given burn period)	90%
Launch and Rendezvous Dates	Selected by Optimizer
Optimization Method	SEPTOP (Ref. 57)

The SEPTOP low thrust optimization tool was used to generate optimized trajectories for three different operating scenarios: single BPT-4000, single NSTAR, and dual NSTAR.[57] The terms "single" and "dual" refer to the maximum number of thrusters operating *simultaneously* at any point in the mission. The *total* number of thrusters mounted on the spacecraft is higher when spares and throughput limitations are considered (see Figure 22). The single and dual NSTAR options both represent state of the art systems. Single NSTAR operation has been flight demonstrated on DS1 and is the baseline for Dawn. Simultaneous operation of multiple thrusters has been flight demonstrated on commercial missions, but has not been demonstrated with NSTAR. All trajectories assume a nominal array power of 6 kW at 1 AU distance from the Sun and include no power margin or allowance for array degradation. The array sizing is typical for a cost-capped Discovery mission using SEP. Power available from the array varies with distance from the sun and is modeled using a high efficiency gallium arsenide array model. The entry velocity at Earth return is not constrained and is optimized for maximum total delivered mass. The entry velocity is approximately 13.4 km/s for the BPT-4000, and 14.9 km/s and 13.9 km/s for the single and dual NSTAR scenarios, respectively. By comparison, the entry velocity for the Stardust mission was approximately 12.6 km/s. Higher entry velocities require a heavier and more expensive thermal protection system (TPS). Variations in the mass and cost of the TPS are not accounted for in this analysis, but would tend to improve the performance of the BPT-4000 relative to the NSTAR systems.

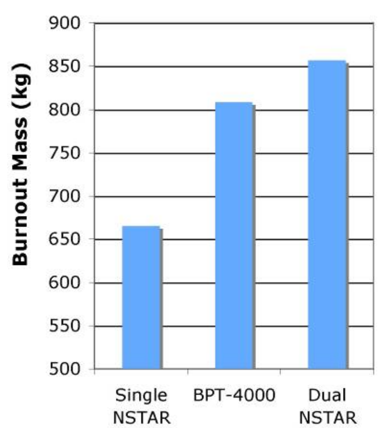


Figure 18. Burnout mass at Earth return for the near Earth asteroid mission.

The total burnout mass for the BPT-4000 and NSTAR options are shown in Figure 18. Burnout mass is defined as the total mass of the spacecraft when it reaches its final destination (in this case, returns to Earth) including the payload, propulsion system, and residual propellant.

The BPT-4000 system offers considerably better performance than a single NSTAR system for this trajectory, delivering approximately 150 kg of additional mass to the final destination with the same launch vehicle and power level. The reason for this performance improvement can be seen in Figure 19 which shows the location and length of thrust and coast periods for the single NSTAR and single BPT-4000 options. Although the overall mission duration is the same, NSTAR uses a long, nearly continuous thrust arc to accomplish the rendezvous with the asteroid nearly two years after launch followed by a five month thrust arc on the trip back to Earth. These long thrust arcs are relatively inefficient, as the thruster is forced to operate at non-ideal points in the orbit, resulting in a total on-board ΔV requirement of 6.5 km/s for single NSTAR (the dual NSTAR ΔV is 5.2 km/s). The BPT-4000 uses much shorter thrust arcs that are optimally located in the trajectory to minimize ΔV. The total on-board ΔV is therefore much lower, only 4.2 km/s, which more than compensates for the lower specific impulse of the Hall thruster.

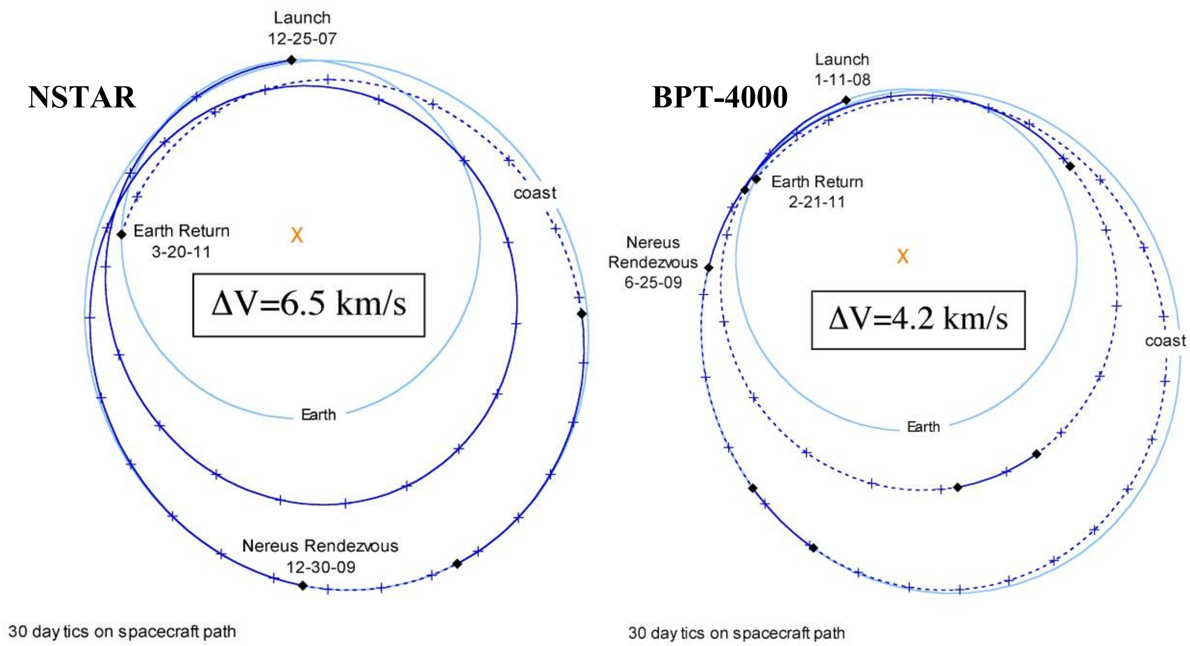
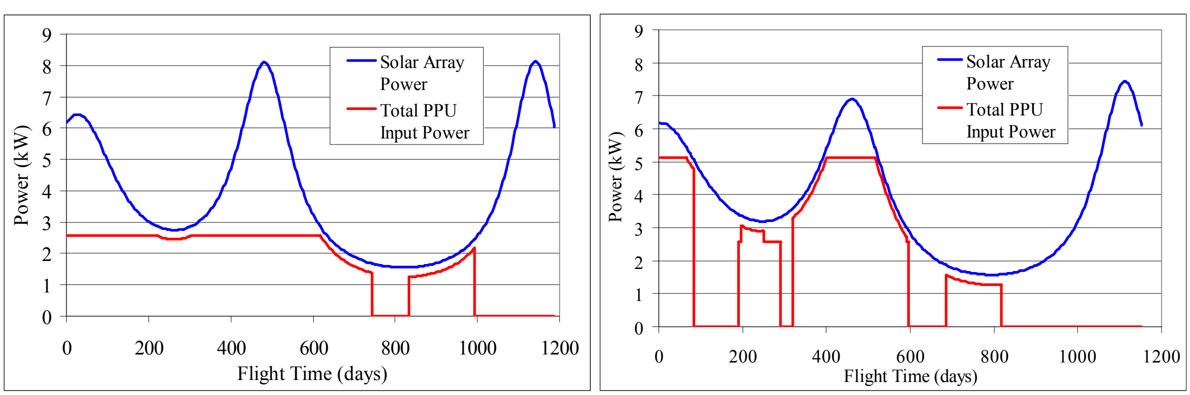


Figure 19. Single NSTAR (left) and single BPT-4000 (right) trajectories for the near Earth asteroid mission (launch C_3 = 10.1 km^2/s^2 for NSTAR, 2.4 km^2/s^2 for BPT-4000). Solid lines indicate thrust periods, dashed lines indicate coast periods.



Single NSTAR Thruster **Dual NSTAR Thruster**

Figure 20. Single and dual NSTAR power profiles for the near Earth asteroid mission.

Figure 20 and Figure 21 show power profiles for the near Earth asteroid sample return mission. While the single NSTAR scenario requires extended operation below 2 kW, the BPT-4000 primarily operates above 3 kW, a relatively efficient and high specific impulse portion of its throttle table. This also contributes to the relatively good performance of the system relative to NSTAR.

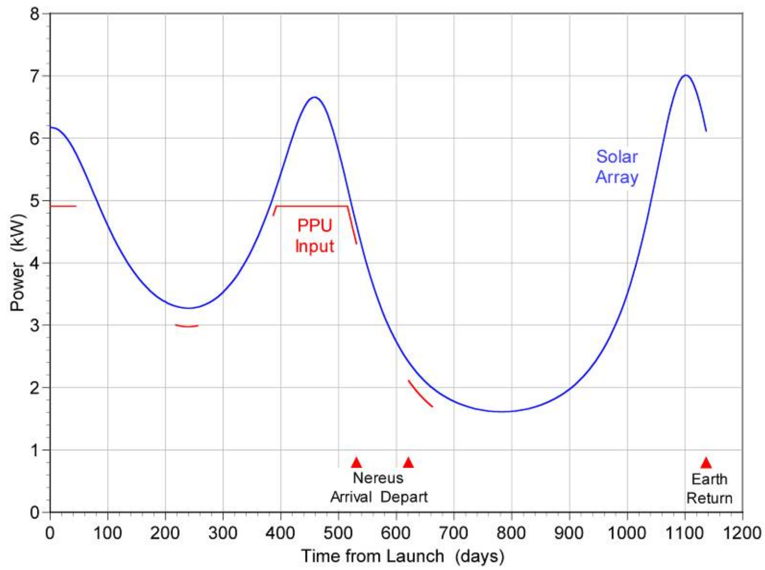


Figure 21. Single BPT-4000 power profile for the near Earth asteroid mission.

In terms of burnout mass, the performance of the single BPT-4000 system is only 50 kg (6%) less than the performance of a dual NSTAR system, but this mass improvement would be offset by the 56 kg higher mass of the NSTAR subsystem (see Table 3). Also, the dual NSTAR system (which is 2+1, see Figure 22) is nearly twice as expensive as the second flight cost of the 1+1 BPT-4000 system, resulting in an unfavorable cost-performance trade for cost-capped mission applications.

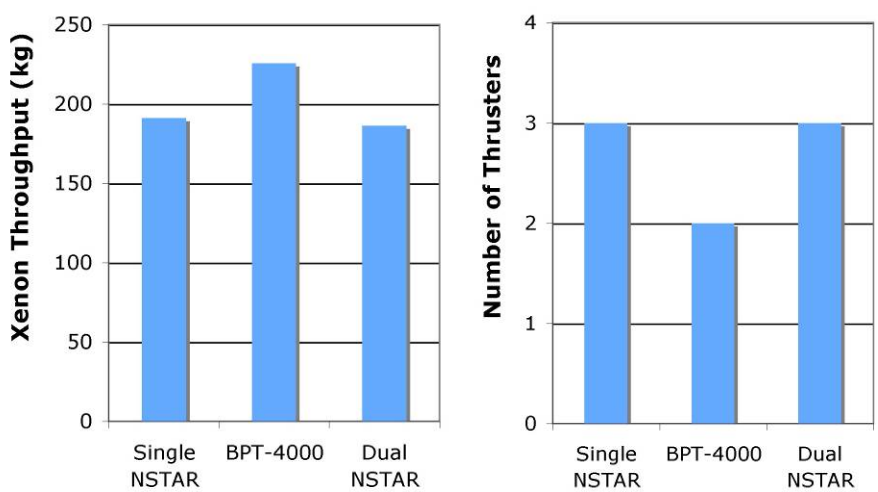


Figure 22. Xenon propellant throughput and total thrusters for the near Earth asteroid mission.

Figure 22 shows the xenon throughput and total number of thrusters required for each option. The total number of thrusters is calculated by dividing the total xenon throughput by the thruster's throughput capability and adding an extra engine for redundancy. The BPT-4000 is assumed to have a throughput capability of 284 kg, which assumes that the Phase Two low-power life test extension is completed (see Table 5). Although the 1+1 BPT-4000 system requires less on-board ΔV, because of its lower specific impulse, it ends up processing more xenon than the

2+1 NSTAR systems. Nevertheless, because the BPT-4000 throughput capability is higher than NSTAR (see Table 5), fewer BPT-4000 thrusters are required for the mission than in either of the NSTAR scenarios. This results in additional cost and mass savings relative to NSTAR.

Overall, the commercial Hall system offers better mass performance (after factoring in the difference in system mass) at much lower cost than NSTAR for the reference near Earth asteroid sample return mission. It should be noted that the relative mass benefit can be highly dependent on target and launch date and may be much less than shown here in some cases. However, based on these results, the following general conclusion can be reached:

• On the near Earth asteroid sample return mission, the BPT-4000 commercial Hall systems offers mass performance competitive with NSTAR at much lower cost.

C. Comet Rendezvous Mission

The second reference mission considered in this study is a rendezvous mission with an active short period comet. The spacecraft launches directly to an Earth escape trajectory and uses SEP to rendezvous and orbit the comet Kopff. The basic characteristics of this mission are shown in Table 9.

Table 9. Comet rendezvous mission characteristics.

Comet Rendezvous	
Target Body	Kopff
Launch Vehicle	Delta 2925-9.5
Power System	Triple junction GaAs solar array, 6 kW or 7.5 kW at 1 AU
Bus Power	250 W
Duration	3.8 years
ΔV	7.1 to 8.8 km/s
Launch Year	2006
Ion/Hall Thruster Duty Cycle (for a given burn period)	90%
Launch and Rendezvous Dates	Selected by Optimizer
Optimization Tool	SEPTOP

A separate optimized trajectory is generated for each scenario using SEPTOP. All trajectories assume a nominal array power of 6 or 7.5 kW at 1 AU and include no power margin or allowance for array degradation. The array model used is a triple junction GaAs array model. The overall results are summarized in Figure 23, which shows total delivered mass and propellant mass at 6 kW.

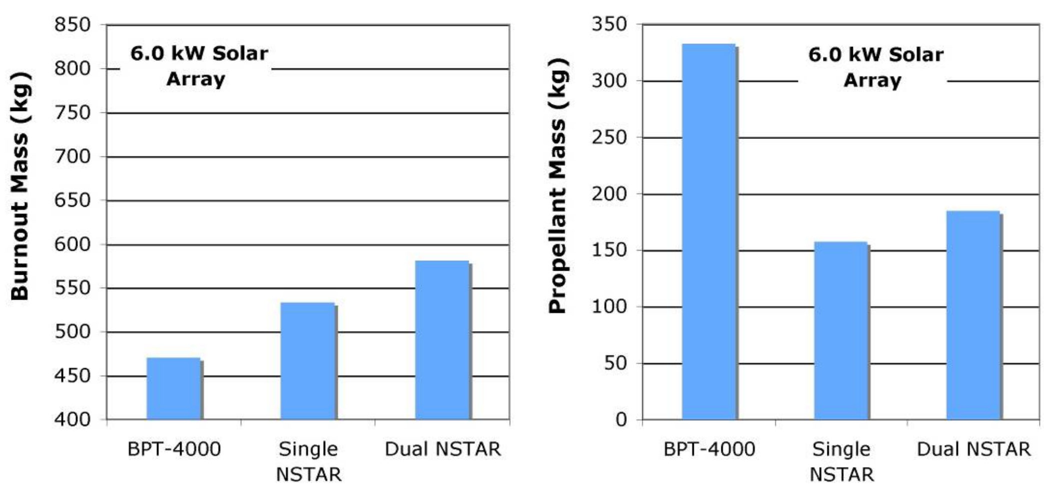


Figure 23. Burnout mass (left) and propellant mass (right) for the comet rendezvous mission (6 kW array).

The BPT-4000 does not perform as well as NSTAR on the reference comet rendezvous mission, and requires much more propellant. The reason for the relatively low performance of the BPT-4000 can be seen in the power profile shown in Figure 24. Because the comet is located relatively far away from the sun, this mission requires extended operation at moderately low power (< 1.5 kW). This is a range in which the BPT-4000 operates at relatively low specific impulse and efficiency and therefore has lower overall performance.

Figure 25 shows overall performance results with a larger 7.5 kW solar array. Although the performance of the BPT-4000 system improves significantly, the performance of the NSTAR options also improves, again resulting in relatively lower performance for the BPT-4000. However, it is interesting to note that the burnout mass of the BPT-4000 option at 7.5 kW is higher than the burnout mass of the NSTAR options at 6 kW. Given the substantial cost differential between the BPT-4000 and NSTAR systems, a BPT-4000 system with a larger solar array may be more cost effective than a single or dual NSTAR system with a smaller solar array.

Overall, the commercial Hall system does not perform as well as NSTAR for missions requiring high throughput or extended low power operation. Mission analysis of main belt asteroid missions (not shown) has shown that as one moves farther away from the sun, the performance of the BPT-4000 continues to fall relative to NSTAR systems. Based on these trends, we conclude that:

• The BPT-4000 commercial Hall system seems best suited to destinations located relatively close to the sun, inside approximately 2 AU. The estimated range of applicability is approximate, and further work is needed to establish the exact limit.

• The BPT-4000's range of applicability will tend to increase as the cost of solar arrays continues to drop and it becomes less expensive to obtain power at distances far away from the sun.

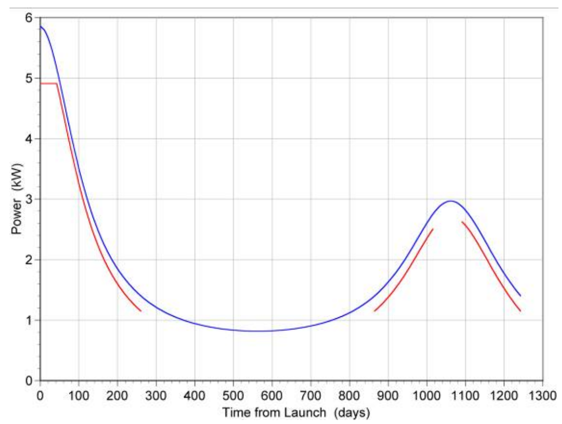


Figure 24. BPT-4000 power profile for the comet rendezvous mission (2006 launch date).

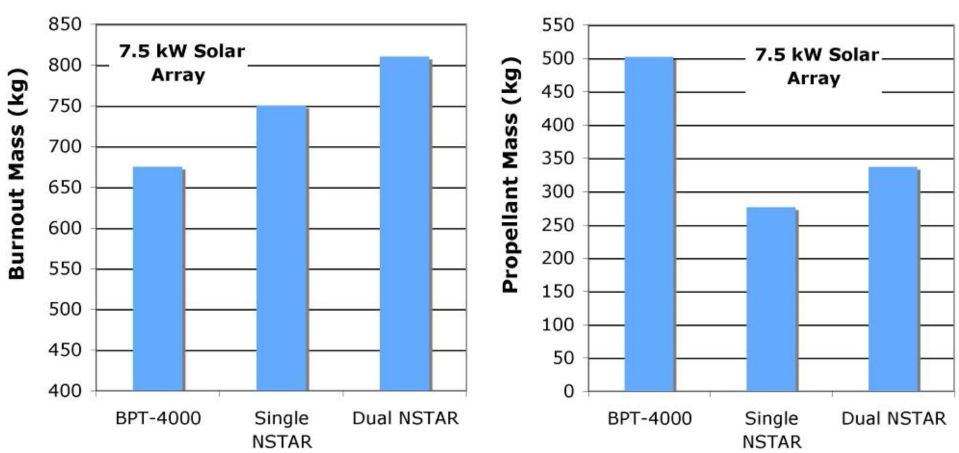


Figure 25: Burnout mass (left) and propellant mass (right) for the comet rendezvous mission (7.5 kW array).

D. Electric Propulsion as an Alternative to Aerobraking

The major advantage associated with the use of commercial EP systems for deep space missions is the cost savings relative to the custom built NSTAR system. As the cost of EP system hardware approaches the cost of chemical propulsion, SEP could become a cost-effective alternative to aerobraking. Currently, small planetary orbiters like Mars Odyssey and the Mars Reconnaissance Orbiter (MRO) use a combination of chemical propulsion and aerobraking for orbit insertion around planetary destinations with atmospheres. The orbit insertion is accomplished in two steps. First, a fairly large chemical propulsion maneuver is used to enter a highly elliptical capture orbit around the destination. Second, the periapsis of the orbit is carefully lowered until the spacecraft experiences sufficient atmospheric drag to slowly lower the apoapsis with each atmospheric pass. The allowable drag is limited by the thermal design of the spacecraft, and it typically takes several months to transition from the capture orbit to the final, low science orbit. During aerobraking, there is a risk that unexpected atmospheric density variations will severely perturb the orbit, resulting in damage to or loss of the spacecraft. To mitigate this risk, aerobraking operations are currently conducted with near continuous monitoring of the spacecraft as it orbits the planet.

The cost of aerobraking is dominated by ground operations costs. Figure 26 shows the estimated cost of 6 months of continuous Deep Space Network (DSN) coverage for an aerobraking campaign, as calculated using an aperture feed costing tool available from the DSN web site.[58] The cost is significant, over \$15M, and does not include the cost of the operations, navigation, or management teams.

An alternative to aerobraking is to remove the chemical propulsion system and replace it with a SEP system used for the interplanetary transit, orbit insertion, and to spiral down to the final orbit. SEP operations are much less risky than aerobraking maneuvers and do not typically require continuous monitoring. Figure 26 shows the cost of DSN coverage for a six month spiral orbit lowering campaign using SEP assuming that four, eight hour DSN passes are required per week for navigation and control of the spacecraft. The cost of DSN coverage for the SEP option is an order of magnitude lower than the cost of the aerobraking option, saving more than \$14M over the duration of the campaign. Given the cost savings, it follows that the combination of a SEP transit plus a SEP spiral down to the science orbit may be a cost effective alternative to aerobraking if the cost of the EP system plus the power system is less than the cost of the equivalent chemical propulsion system plus aerobraking. For the BPT-4000 n^{th} flight system shown in Figure 12, the cost of the chemical and EP propulsion system hardware is roughly equivalent. Therefore, if the cost of the power system needed to support SEP can be kept to less than \$14M and the duration of the SEP spiral down is equivalent to the aerobraking duration, SEP can be a cost effective and relatively low risk alternative to aerobraking.

Overall, we conclude that a low cost, mid-power commercial Hall thruster EP system may be a viable alternative to aerobraking for some missions. Further work is needed to develop this mission concept and determine if the overall cost savings is offset by the cost and complexity associated with the use of SEP for the interplanetary transit, orbit insertion and orbit lowering.

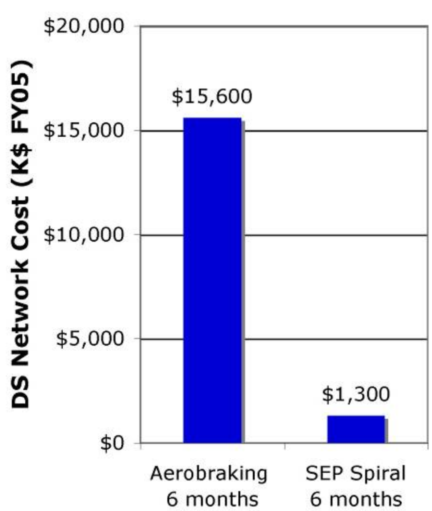


Figure 26. Estimated cost of Deep Space Network operations time for aerobraking and SEP spiral operations.

VI. Conclusion

Our assessment of a 4.5 kW commercial Hall thruster system is that the system is a viable option for near-term missions that are cost-capped like the Discovery program. The success of recent commercial Hall thruster flights provides an opportunity for NASA to capitalize on commercial technology investment to substantially improve science mission capability. By buying off commercial product lines, cost and schedule risks can be significantly reduced when compared to specialized, infrequently manufactured government systems.

Mission analysis indicated that on the reference near Earth asteroid sample return mission, the BPT-4000 commercial Hall thruster system offers mass performance competitive with or superior to NSTAR at much lower cost. The BPT-4000 seems best suited to destinations located relatively close to the sun, inside approximately 2 AU. The estimated range of applicability is approximate, and further work is needed to establish the exact limit. In the far-term, this range could be expanded through investments in a PPU capable of higher output voltage and thrusters capable of operating with sufficient life at high specific impulse. Also, a low-cost, mid-power commercial Hall EP system may be a viable alternative to aerobraking for some missions. Further work is needed to develop this mission concept and determine if the overall cost savings is offset by the cost and complexity associated with the use of SEP for the interplanetary transit, orbit insertion and orbit lowering.

A review of the qualification status of the BPT-4000 Hall thruster system shows no substantial risk items associated with a delta qualification for NASA science missions. In most cases, the completed qualification programs for commercial programs equals or exceeds the requirements for NASA science missions. For those requirements currently not met by commercial components, a low risk delta qualification has been planned and the cost and risks are manageable. Of primary importance is the extended life test of the BPT-4000 thruster and the redesign and qualification of the PPU to accept input bus voltages common to NASA science missions.

Once the delta qualification costs are absorbed on the first mission, the resulting Hall thruster system cost reduction will lead to wider application into missions where the technology is not enabling but merely performance enhancing. Future Hall thruster technology development programs, such as the HIVHAC thruster,[9] can expand the range of missions to missions beyond 2 AU from the sun as long as the technology can be integrated simply within the proposed system architecture.

Overall, a commercial Hall thruster system based on the BPT-4000 is a viable and important cost saving alternative to cost capped deep space missions.

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