Qualification of Commercial Electric Propulsion Systems for Deep Space Missions

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Solar electric propulsion (EP) enables or enhances several future exciting and challenging solar system exploration robotic science missions. While NASA is about to fly DAWN, the first science mission to use electric propulsion, more than two dozen commercial geosynchronous communications (GEO-COM) satellites presently use commercially manufactured electric propulsion systems. The commercial electric propulsion systems are much less expensive than those developed solely for use by NASA, and second generation commercial systems have proven to be extremely reliable on-orbit. NASA solar system explorations missions require EP systems with a somewhat longer life and a significantly greater operating power range than is required for GEO-COM satellites. Other than the life time and operating power range difference, the requirements for commercial and NASA missions are roughly comparable and in many cases, the commercial requirements are more demanding. A methodology is proposed for the application and qualification of these low cost commercial electric propulsion systems to NASA science missions. A standard architecture is described that will provide maximum commonality with commercially qualified components and minimum changes for different NASA mission applications. A qualification methodology consisting of limited delta qualification testing and lifetime validation by test and analysis is proposed to validate these commercial systems for NASA science missions. Finally, the performance and cost benefits of the use of such commercial Electric Propulsion systems on NASA missions will also be described.

I. Introduction

SINCE the mid-1990s, solar electric propulsion (SEP) technology has enjoyed a string of successes on-board commercial, military, and science spacecraft[1-10]. 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 xenon Hall and ion thrusters in 1997, and now more than 30 commercial geosynchronous earth orbiting (GEO) communication satellites of US origin have flown using these technologies. Commercial programs in Europe, Russia, and Japan have 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.

Although government supported technology demonstration and science missions dominated the early flight application of EP, commercial systems now comprise the vast majority of flight systems. Figure 1 illustrates the distribution of xenon Hall and ion thruster systems, the technology of most interest to NASA deep space missions,

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among different government and commercial users. Russian Hall thrusters systems make up the largest share of this EP systems market. Since the first flight in 1972 of the Fakel Design Bureau built Stationary Plasma Thruster (SPT), 42 Russian systems have flown with 214 thrusters and no failures[3,7]. For the first two decades, these systems consisted primarily of short lifetime sub-kW level thrusters applied to East-West station keeping for solar power communication satellites and nuclear power experimental satellites. In 1994, the first SPT-100 thruster was flown for North-South station keeping and limited orbit-raising on a new type of Russian spacecraft designed to compete internationally with other long life communication satellites. The development of this higher power (1.35 kW) longer life (> 2 million Ns total impulse) thruster prompted interest in non-Russian users resulting in the flight of hybrid systems with Fakel thrusters and western built power electronics[3,5]. So far, six of these systems have been flown on geostationary communication satellites, three by Space Systems/Loral and three by Astrium. One additional hybrid system, consisting of a Russian TsNIIMASH Hall thruster and an Aerojet power processing unit (PPU), was flown on the technology demonstration mission STEX[8]. The development of this PPU by Aerojet ultimately led to the Aerojet/Busek BPT-4000 thruster currently integrated and ready for a 2008 flight[8]. The two all-western flight Hall systems include the SNECMA PPS-1350, derived from the Fakel SPT-100, flown on the SMART-1 technology demonstration mission and the Busek BHT-200 flown on the TacSat-2 mission with a Northrup Grumman Space Technology (NGST) power processing unit[5,10].

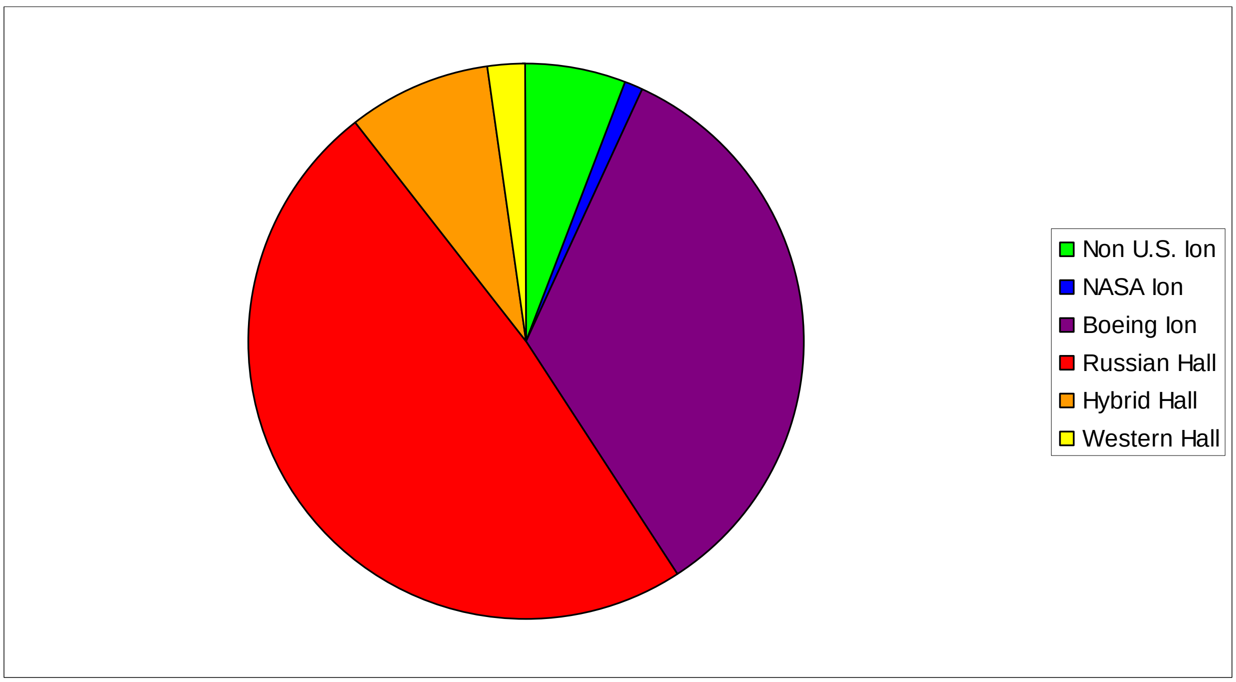

Figure 1: Expanding flight applications of xenon Hall and ion thruster systems.

Flight application of ion thrusters began with the U. S. Air Force suborbital cesium ion thruster mission in 1962[11]. Difficulties in handling cesium and mercury propellant prompted further development of xenon ion thrusters at Hughes (later Boeing and then L-3) and the NASA Glenn Research center (GRC). The first xenon ion thruster flight was made with the Boeing 13 cm Xenon Ion Propulsion System (XIPS) for North-South station keeping on the Panamsat 5 geostationary communication satellite[12]. Increasing power and size of geostationary satellites prompted Boeing to fly a higher power (4.5 kW) 25 cm XIPS system in 1999 that also could perform limited orbit-raising. As shown in Figure 1, the XIPS system makes up by far the bulk of flight ion thruster systems, and although there were problems on the first generation 13 cm system, no failures have occurred on the 14 flight 25 cm systems. The only NASA flight of a xenon ion thruster system to date has been the NSTAR system flown Deep Space 1 (DS1)[4]. This system consisted of thrusters and PPUs developed by NASA GRC, built by Boeing, and integrated by JPL with gimbals, tanks, and feed system components. Three NSTAR thrusters will fly again on the JPL spacecraft DAWN before the end of 2007[13]. Substantial progress has also been made in Japan for flight application of ion thruster systems. This includes the Mitsubishi electric company ion thrusters on ETS III, VI, and VIII; and also the use of microwave discharge ion thrusters on the Hyabusa asteroid sample return mission[2]. Finally, two British and two German ion thrusters were flown on the Artemis technology demonstration mission[6].

The high-specific impulse (1000-4000 s) of SEP systems primarily benefits 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.

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.

II. Commercial Options

A. System Architecture

In developing a SEP system 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 costs 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 EP system.

The results of these trade studies are shown in Figure 2. 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 change, 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.

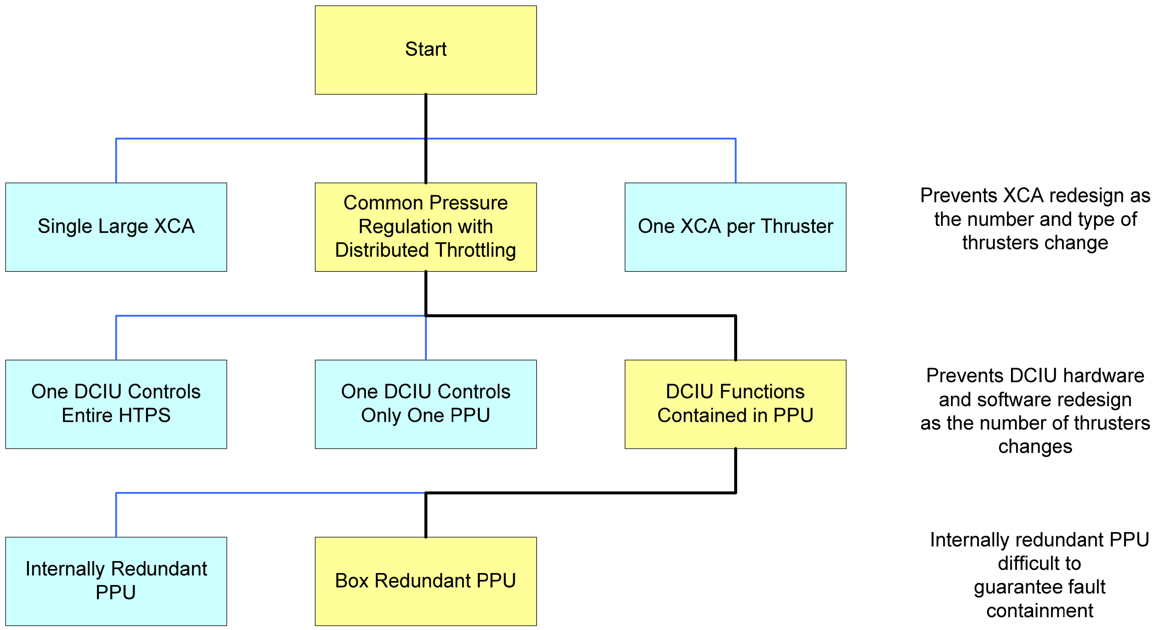

Figure 2. Solar Electric Propulsion (SEP) standard architecture trade studies.

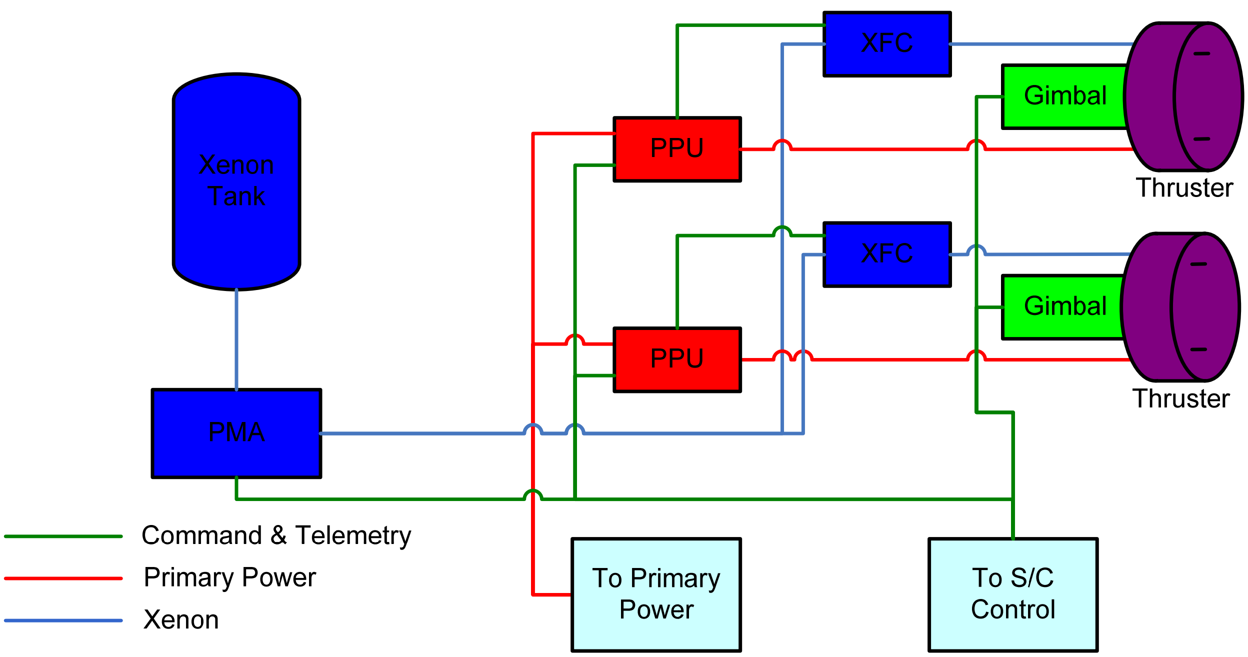

Figure 3. Solar Electric Propulsion System (SEP) block diagram.

The implementation of the trade study results is shown in Figure 3. 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 both commercial Hall and ion thruster systems; thus allowing the possible integration of commercial off-the-shelf components into a NASA SEP system. The only deviation from some commercial systems is the use of a dedicated PPU 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. Thrusters

A wide variety of both commercial Hall and ion thrusters exist for application to NASA missions. Hall thruster options include thrusters with flight heritage such as the Fakel SPT-70 and SPT-100, TsNIIMASH D-55, SNECMA PPS-1350, and the Busek BHT-200[8]. Additional options have made substantial progress in development including the Aerojet BPT-4000, Fakel SPT-140, Busek BHT-600, and SNECMA PPS-5000. From these Hall thrusters, only the BPT-4000 was chosen for this study, as it was the most mature design that could fit the throttling range and lifetime of most interest to near-term NASA missions (Figures 4 and 5). By selecting the BPT-4000 in combination with the SEP system standard architecture described above, the amount of system component development and qualification effort is substantially reduced. No design changes are required for any system component; however certain delta qualification tests are required to meet NASA science mission requirements and will be discussed later.

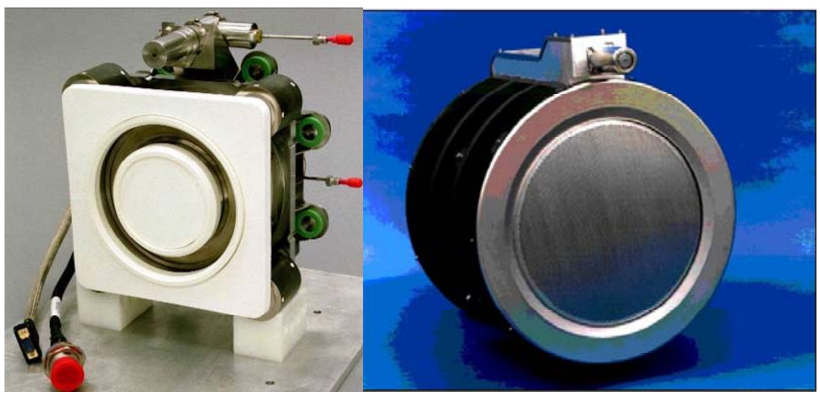

Figure 4. Aerojet BPT-4000 thruster (left), L-3 25 cm XIPS thruster (right)

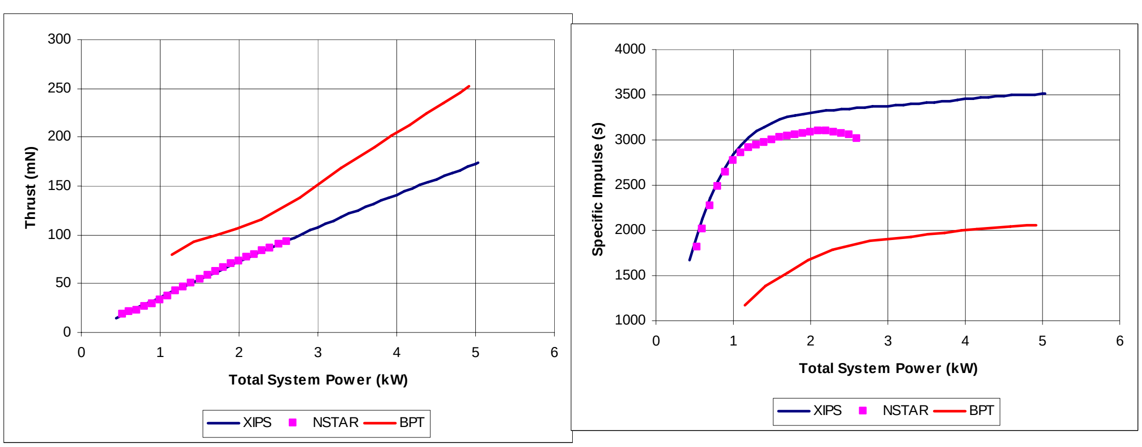

Figure 5: 25 cm XIPS, NSTAR, and BPT-4000 thrust and specific impulse versus PPU input power.

Ion thruster options are also extensive and include thrusters with flight heritage such as the L-3 ETI 13 and 25 cm XIPS© thrusters, the MELCO 20 mN class engine, the QinetiQ thruster, and the RIT thruster[2,6,9,12,14]. From these ion thrusters, only 25 cm XIPS was chosen for comparison with the NSTAR system, as it was the most mature design that could fit the throttling range and lifetime of most interest to near-term NASA missions (Figures 4 and 5). Implementing the XIPS into the NASA SEP standard architecture requires no changes to the thrusters, xenon tanks, PMA, or gimbals; but minor modifications are required to the XFC and PPU to accommodate greater throttling ranges required for NASA missions. Additionally, certain delta qualification tests are required to meet NASA science mission requirements and will be discussed later.

C. Power Processing Unit (PPU)

For the BPT-4000 thruster, the Aerojet BPT-4000 PPU has also recently completed qualification testing for GEO applications[8]. The PPU design provides commandable power to the thruster plasma discharge, thruster electromagnets, cathode heater, and cathode keeper (Figure 6). 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 units delivered in 2006 for the first flight in 2008. Radiation hardened S-Level components are utilized to provide maximum reliability.

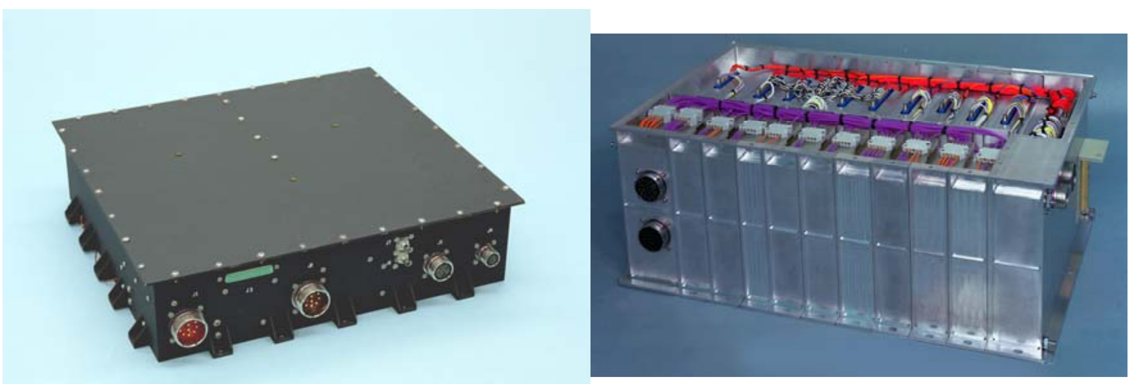

Figure 6. Aerojet BPT-4000 PPU (left), L-3 25 cm XIPS XPC (right)

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 and total system efficiency over this range is shown in Figure 7. The difference between the qualification and design performance envelope will only require protoflight testing on the first unit produced for NASA.

The XIPS power controller (XPC in place of PPU) contains the electrical control and power interface system for the 25 cm XIPS thruster[12]. The XPC provides the required voltages for operation of the thruster, interfaces with the spacecraft computer, and performs timed functions for the thruster start-up. A single XPC is capable of operating 2 thrusters, one at a time; but one output can be terminated for single string operation. It is powered from a 100-Vdc spacecraft bus, and consists of seven separate power supplies. One screen, one accelerator, one discharge, two cathode keeper and two cathode heater supplies control all of the thruster functions. Assembly of the XPC, as with the thruster, is in a dedicated clean room by a trained assembly team under the supervision of a full time manufacturing engineer and Quality Assurance personnel. Parts selection and traceability, assembly and slice-level testing are highly regimented processes designed to ensure quality assurance. A photograph of the XPC with the top removed is presented in Figure 6. Acceptance testing of the XPC involves evaluation of individual components, and bench testing of each power supply and control unit. The final level of testing involves integration testing with a flight thruster where the XPC must turn on and drive a 25-cm thruster in the L-3 test facilities.

For NASA missions, additional throttling capability is required. Modification of the control system in the XPC for adjustable discharge current and the screen voltage set points has been demonstrated in breadboard units (Figure 7). An initial review of the XPC layout indicated that a computer interface and gas flow control slice similar to that used on DS1 or DAWN in a separate box could be readily added inside the existing XPC, replacing the existing interface slice to the Boeing spacecraft computer.

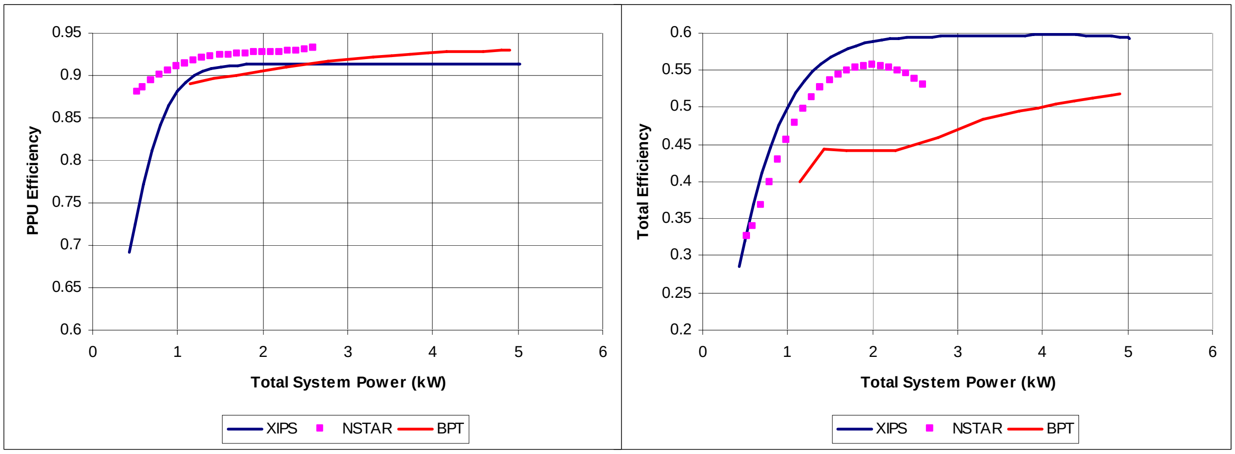

Figure 7. 25 cm XIPS, NSTAR, and BPT-4000 PPU and total system efficiency versus PPU input power.

D. Xenon Feed System (XFS)

For a Hall thruster system, to maximize commonality with existing commercial off the shelf hardware, the Moog Model 50E947 was chosen for the XFC (Figure 8). 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[8]. 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.

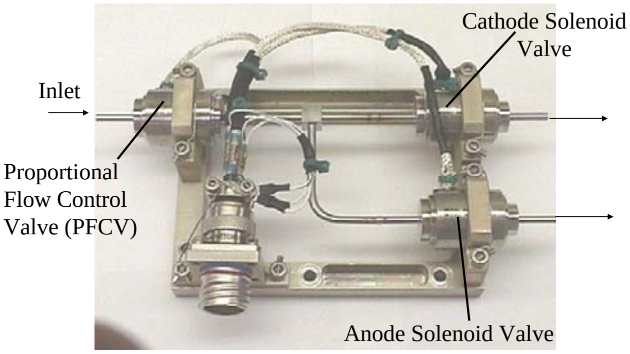

Figure 8. Moog xenon flow controller (XFC) and propellant management assembly (PMA).

Options for throttleable XIPS XFCs include modified versions of the Moog XFC, similar to the NEXT feed system, or modified versions of the Vacco flow control modules flow for the MELCO ion thruster on ETS-VIII[2,15]. Note in either case, no new component qualifications are required, only delta qualification testing of the new XIPS module with the XIPS XPC gas flow control slice.

For the propellant management assembly, flight proven Moog options (with a Moog regulator, see Figure 8) and flight proven Vacco options (with a Stanford Mu regulator) exist[8]. For the purposes of this study, the Moog option was chosen based on the flight heritage and ease of integration into the SEP 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 1, a wide variety of qualified xenon tanks already exist and most have flight heritage[2,8]. 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 9, this tank is a skirt mounted tank with a titanium liner.

Table 1. 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	Yes
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 9. 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[8]. This gimbal successfully completed qualification in 2006 for GEO applications (Figure 10). 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. Boeing has also produced a flight qualified gimbal for the XIPS system[16].

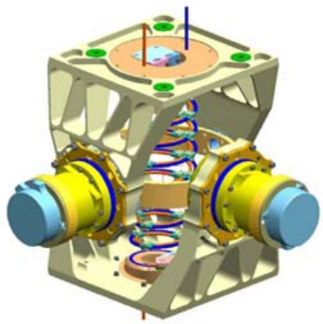

Figure 10. Moog thruster gimbal.

III. Qualification Methodology

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 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[8]. 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 and XIPS subsystem thruster, gimbal, XFC, and PPU have completed qualification for GEO applications[8,12]. 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 NASA's 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. 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 2 compares the random vibration requirements for the Dawn IPS to the test programs for the commercial components discussed here. The test programs for all Hall thruster subsystem components exceed the Dawn requirements at all frequencies. Ion thruster system components are roughly comparable; but the levels for the thruster, PPU, and gimbal are currently not available for public release. 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 2 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 2. Random vibration levels for EP system components.

Component	Dawn IPS Requirement (Grms)	Existing Hall thruster components (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 11. The Hall system 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. Ion thruster system components are roughly comparable; but the levels for the thruster, PPU, and gimbal are currently not available for public release.

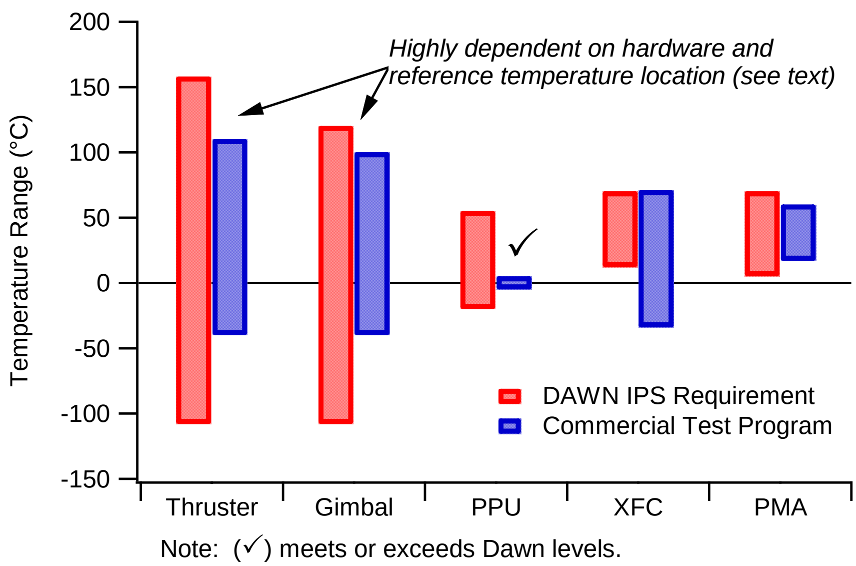

Figure 11. Temperature range requirements for Hall thruster subsystem components

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 and XIPS thruster is at the rear interface 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 and XIPS 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[8]. Thrusters can be evaluated for emissions at specific spacecraft communications frequencies to aid in spacecraft design[8]. Whether a NASA science mission uses a commercial thruster or a government-developed thruster the process is similar. The BPT-4000 and XIPS thrusters have undergone an extensive array of tests and analyses to demonstrate compatibility with GEO communications spacecraft[8].

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 and ion thruster plumes and also for EP missions beyond GEO[8]. The BPT-4000 and XIPS thruster plumes have been characterized through detailed diagnostic experiments, materials sputter testing and modeling, and spacecraft-level plume effects modeling[8].

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. 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 and XIPS 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 characteristics of NSTAR, BPT-4000, and XIPS are compared in Table 3[8,9]. The table shows the results from qualification testing and the resulting usable flight limits after applying standard 50% margins. Because of the variation in mission throttle profiles, NSTAR lifetime was validated using wear models validated by test. Commercial electric propulsion systems are being qualified for NASA mission requirements by extending the commercial qualification program results through a combination of testing and analysis. In addition to the commercial life test results, thruster life qualification will use validated models of electric propulsion thruster life for the variable operating power throttling ranges required by NASA missions. This approach is a rigorous process, documented in peer reviewed journal articles and conference presentations, and is based on a fundamental understanding of electric propulsion physics codified into computer models. This approach of qualifying ion and Hall thrusters for the extended throttling conditions required by NASA science missions based on the extension of limited life test data using benchmarked computer codes was used to establish and qualify the throughput rating of the NSTAR thrusters for DAWN. These and more advanced techniques will be used to qualify the commercial thrusters for future NASA science missions. This systematic approach to qualifying commercial electric thrusters reduces the risk associated with applying them to NASA science missions, and gives mission planners the dual advantages of higher performance and potential lower cost EP systems compared with the NASA only system used on Dawn.

Table 3. Throughput capabilities of NSTAR, BPT-4000, and XIPS. Rated throughput is relative to total throughput less 50% margin.

Thruster	Rated Throughput (kg)	Total Throughput (kg)	Notes
NSTAR	157	235	Test & analysis completed
XIPS	200	300	Predicted capability not yet demonstrated. Test and analysis ongoing
BPT-4000	387	580	Predicted capability not yet demonstrated. Test and analysis ongoing

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.

IV. Mission Analysis

Discovery missions are selected competitively and cover a wide range of scientific goals and destinations. Three reference missions are used for performance evaluations in this study[8,9]. The destinations are generic, but are similar to current and proposed Discovery class missions that utilize electric propulsion. This section discusses each of the three concepts and provides analysis results for each mission.

A. Near Earth Asteroid Sample Return

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 4.

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
Thruster Max Duty Cycle	90%
Launch and Rendezvous Dates	Selected by Optimizer
Optimization Method	SEPTOP

Table 4: Near Earth Asteroid Sample Return Mission Characteristics

The SEPTOP low thrust optimization tool was used to generate optimized trajectories for four different operating scenarios: single BPT-4000, single XIPS, and dual NSTAR. 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 due to the extra thruster necessary for redundancy. 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 EP mission. 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 total burnout mass for the XIPS and NSTAR options are shown in Figure 12. 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. All three systems offer reasonably comparable performance; however, the NSTAR system requires two operating thrusters while the BPT and XIPS systems require only one. This significantly reduces the cost and complexity of the commercial BPT and XIPS systems with respect to NSTAR. The XIPS system offers superior performance relative to the BPT system; however the thrusting time is substantially lower for the BPT system.

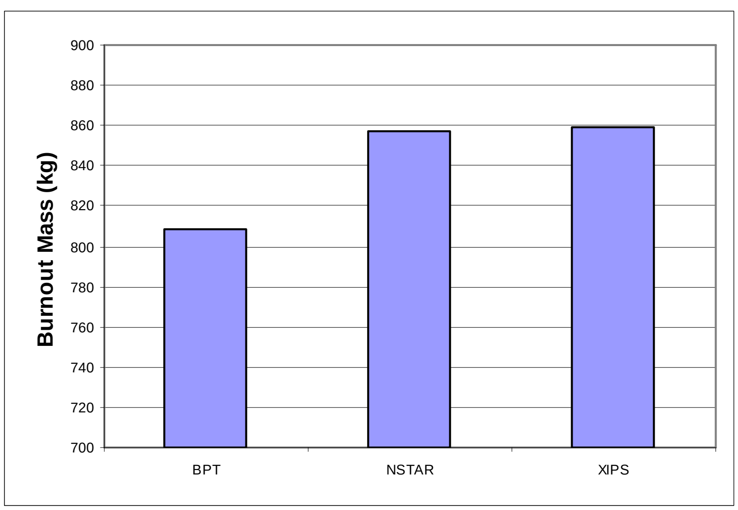

Figure 12: Burnout Mass, Near Earth Asteroid

G. Comet Rendezvous

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 5.

Comet Rendezvous	
Target Body	Kopff
Launch Vehicle	Delta 2925-9.5
Power System	Triple junction GaAs solar array, 6 kW solar array at 1 AU
Bus Power	250 W
Duration	3.8 years
ΔV	7.1 to 7.7 km/s
Launch Year	2006
Ion Thruster Duty Cycle	90%
Launch and Rendezvous Dates	Selected by Optimizer
Optimization Tool	SEPTOP

Table 5: Comet Rendezvous Mission Characteristics

A separate optimized trajectory is generated for each scenario using SEPTOP. All trajectories assume a nominal array power of 6 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 13, which shows total delivered mass for the BPT, XIPS, and NSTAR options. The XIPS and NSTAR systems are roughly comparable; however the BPT system performance is noticeably lower due to the higher mission delta-v.

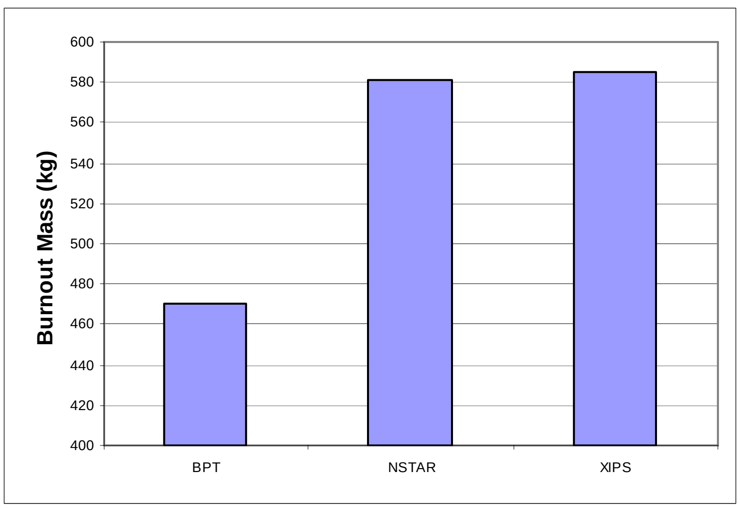

Figure 13: Burnout Mass for Comet Rendezvous

The relative cost benefits of XIPS and BPT-4000 systems vs. the NSTAR system is shown in Figure 14. Both the BPT and XIPS systems have two PPUs, sufficient to operate one thruster at a time, with a second PPU for redundancy. The NSTAR system has three thrusters and one PPU per thruster. The NSTAR cost estimate is based on actuals from the DAWN project. The XIPS and BPT-4000 cost estimates are based on DAWN costs for the feed system, tank and gimbal, and estimates of the thruster and PPU costs from the vendor.

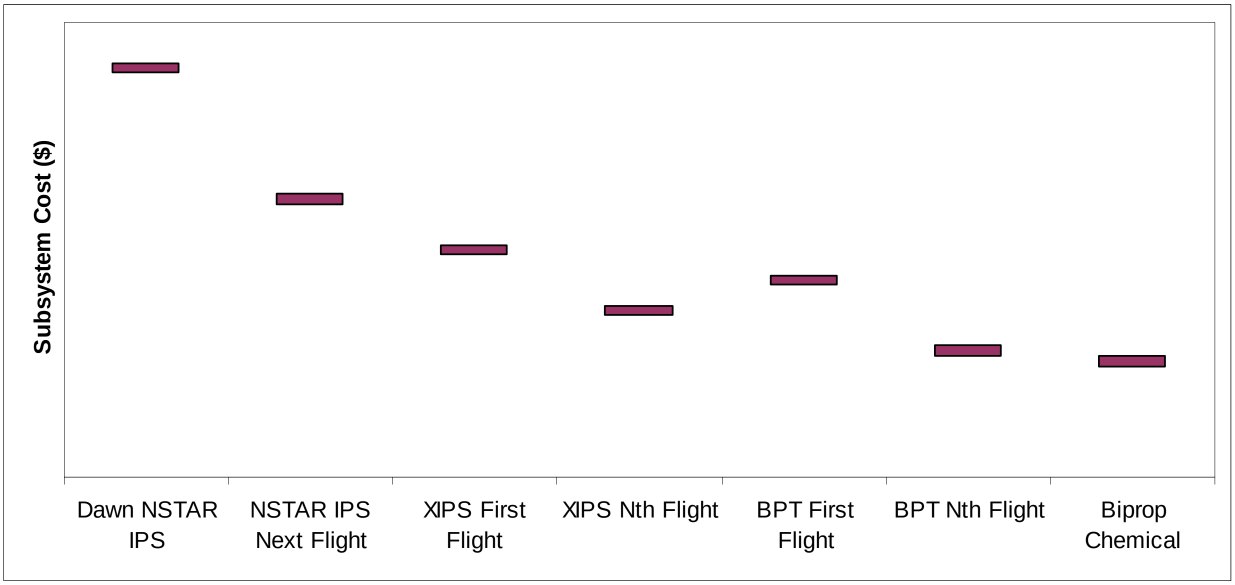

Figure 14: Cost Comparison of NSTAR, XIPS, BPT, and Chemical propulsion systems (Note: this chart does not include the cost of the solar array)

Figure 14 clearly shows that even for the first deep space use of the XIPS or BPT-4000, the cost of the system is substantially less than the cost of an equivalent NSTAR system. For subsequent flights, the cost of a single XIPS system is expected to be slightly more than half that of a single-operating NSTAR system and a little less for the BPT-4000 system with equivalent xenon throughput capability. After first flight, these commercial EP systems will become comparable in cost to chemical biprop systems.

V. Conclusion

Our assessment of the commercial XIPS and BPT-4000 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 and ion 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 and XIPS commercial systems offer mass performance competitive with NSTAR at much lower cost.

A review of the qualification status of the BPT-4000 and XIPS systems show 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 additional lifetime validation of the BPT-4000 and XIPS thrusters and the redesign of the XIPS control card and XFC.

Once the delta qualification costs are absorbed on the first mission, the resulting commercial SEP system cost reduction will lead to wider application into missions where the technology is not enabling but merely performance enhancing. Overall, a commercial SEP system is a viable and important cost saving alternative to cost capped deep space missions.

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