

Solar Electric Propulsion for Discovery Class Missions

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This paper offers a user-centric consolidation and comparison of the full range of government and commercial solar electric propulsion (SEP) options available in the near term for primary propulsion on deep-space science missions of the class commonly proposed to NASA's Discovery program. Unlike previous papers, this work does not emphasize feasibility from a mission analysis perspective. Rather, it emphasizes requirements uniquely imposed by competitively reviewed, cost-capped mission proposals, for which system-level flight heritage can trump sheer performance and mission capture. It describes criteria that mission architects and review boards can use to select and evaluate SEP systems; provides descriptions of the viable government and commercial electric propulsion (EP) system options; describes the modifications needed to adapt commercial EP systems to deep space; discusses appropriate methods for costing commercial-based EP systems; and describes a set of standard system margins appropriate for SEP mission concepts. It concludes that the SEP systems best suited for Discovery missions have solid *system* flight heritage that can meet the requirements for deep space with minimal modifications. Commercially developed SEP systems offer significant heritage potential and, in many cases, the required changes for deep space application introduce comparatively low technical-risk and cost-risk.

I. Introduction

The dramatic success of Dawn in achieving, supporting, and then departing from its 2012 rendezvous with the asteroid Vesta demonstrated the unique value that solar electric propulsion (SEP) has for NASA planetary science, by enabling missions that are too challenging and too expensive to conduct using other methods.¹ More broadly, NASA's Principal Investigator (PI)-led Discovery Program has amply demonstrated the benefit of cost capped, competitively awarded science missions for space exploration. Yet since 2001, when Dawn was proposed, SEP has advanced on a wide variety of space missions. For science in deep space, SEP has also been used by Europe on SMART-1 and by Japan on Hayabusa.^{2,3} More impressive is that for telecommunications applications in high Earth orbit, SEP is now used on over 100 operating satellites, for primary propulsion and stationkeeping.⁴ Electric propulsion is now a mature and widely used technology.

Dawn reaches its second rendezvous target in 2015, fourteen years after it was first proposed. A key challenge for future Discovery program competitive rounds is how NASA can capitalize on the past decade's advancement of SEP to match or even exceed what Dawn has accomplished. This challenge has three components. First, Dawn's

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SEP subsystems are now obsolete, their hardware heritage mostly dissipated or superseded. Second, the commercial systems used for orbit raising and stationkeeping near Earth, while offering performance that could enable deep space missions as well as the multiple benefits of relatively high production volume (e.g., strong flight heritage, a strong industrial base, and low cost-risk), are neither developmentally adapted for, nor formally qualified for, deep space applications. Third, NASA's own developmental initiatives (e.g., the NEXT ion thruster), while advancing the state of practice in thruster lifetime and performance, do not comprise a complete SEP system. Any proposer of SEP-enabled planetary science missions must reconcile these factors.

The ongoing advancement of commercial and government systems has inspired numerous papers describing potential Discovery missions. In 2007, Oh compared NASA electric propulsion (EP) systems for Discovery missions based on several new and proposed technologies, including NEXT, to the NSTAR system used on Dawn, but did not consider commercially developed thruster options.⁵ Numerous papers have examined the benefits of the BPT-4000 Hall thruster, XIPS ion thruster, SPT-100 Hall thruster, and NEXT thruster for Discovery.^{6,7,8,9,10,11} Each study by its nature generally emphasizes a particular technology or considers only a subset of the SEP options available.

This paper provides a user-centric comparison of today's full range of government and commercial SEP options available for primary propulsion on deep-space science missions of the class commonly proposed to NASA's Discovery program. Unlike prior references, the present study does not emphasize feasibility from a wide-open mission-analysis perspective. Rather, it emphasizes the unique requirements imposed by competitively reviewed, cost-capped mission proposals, which favor strong system-level flight heritage over sheer performance and broad mission capture.

Section II describes the criteria most useful to review boards and mission architects in selecting and evaluating SEP. For deep-space missions, the SEP system accounts for most of the power consumption on a spacecraft. This couples the SEP system choice to the power system architecture. Section III begins by describing different power system architectures available for deep space. While Dawn was in development, numerous government and commercial SEP systems were qualified, or have been undergoing qualification, to fly in a variety of applications. Section III ends by listing the numerous NASA and commercial SEP options that have flown to date or undergone qualification life testing in ground facilities. Commercial systems offer strong potential benefits, but for mission architects and reviewers, it is critical to understand the scope, technical risk, cost, and cost risk of adapting these systems. Section IV lists the commercial SEP options available, and describes the changes needed to adapt them to deep space as well as methods to properly estimate their cost for cost-capped proposals. At the spacecraft level, SEP missions introduce both unique requirements and unique flexibility in establishing system-level margins for mass, power, and operational duty cycle. Establishing a standard margin philosophy is important to clarify for both designers and reviewers how to assess the adequacy of margins to address growth as the system design matures. Section V builds on previous work to recommend standard system margins for deep-space SEP missions.

II. Selection Criteria for Competitively Selected NASA Missions

The selection of missions for Discovery is a multi-step competitive process governed by an Announcement of Opportunity (AO) that lists evaluation criteria for each selection round. In 2010, the selection criteria were defined as a) the scientific merit of the proposed investigation, b) the scientific implementation merit and feasibility of the proposed investigation and c) the technical, management, and cost (TMC) feasibility of the proposed approach for mission implementation, including cost risk.¹² In Step 1, each proposal is evaluated and rated from poor to excellent against the science criteria and as low, medium, or high risk against the TMC criteria. The proposals are then ranked into four categories and presented to a selecting official, who typically selects several proposals from the top category to proceed to the next step. The maximum allowable mission cost is strictly capped (in 2010, the cost cap was \$425M FY 2010 dollars), and each proposal is required to maintain a minimum of 25% cost reserve at each Key Decision Point in the project life cycle. In addition, the use of new technology is strictly limited in the AO. Proposed missions are "generally expected" to use technologies at technology readiness level (TRL) 6 or higher,^{††} and those that use a "limited number" of less mature technologies must have a plan to mature those technologies to TRL 6 no

^{††} See NPR 7120.8 "NASA Research and Technology Program and Project Management Requirements," for definitions of technology readiness level.

later than the program confirmation meeting (KDP-C), a milestone that occurs just after the Preliminary Design Review (PDR) and culminates in completion of the mission formulation process (Phase B).

While SEP may enable a scientific investigation, it is only a weak factor in the evaluation of science merit. SEP is a strong factor in the evaluation of technical, management, and cost risk, where the best rating that can be achieved is “low risk”. This, combined with the expectation that one should use mature technology, leads mission architects to use the following ranked list of priorities when selecting SEP systems for Discovery.

1. Solid SEP *system* flight heritage – high TRL
2. Where flight heritage is not available, at least TRL 5 plus development plans that are credible and can be clearly described and costed for TMC evaluation
3. Capture of a particular scientific investigation
4. Implementation Cost

The first criterion emphasizes the heritage of the SEP system. TMC reviewers evaluate the *entire* subsystem against its *least* mature element, which recently has been the power processing unit (PPU) (see Section IV). Where flight heritage is not available, it is essential to have a technology development program which is credible and for which the cost, schedule, cost risk, and schedule risk are fully understood. Mission capture is important when developing new SEP systems because the selection process is uncertain, and those systems that capture the widest range of missions are more likely to eventually be proposed and flown. But for a mission architect working a particular mission concept, it is only necessary to capture a particular target of investigation. Additional performance beyond the target is not beneficial, unless it addresses some element of risk – mass, power, schedule, or otherwise – or it enables a new set of scientific investigations at additional targets (i.e. Dawn). Finally, while cost is less important than heritage or mission capture, it is still a very important element. It is essential that the selected system fit within the overall budget for a cost capped mission proposal.

III. Electric Propulsion and Solar Power Systems Options for Deep Space Missions

When operating, the SEP system typically constitutes the bulk of the power consumption on a science spacecraft. This means that the choice of the SEP system is tightly coupled to the architecture of the power system. In some cases, increasing the use of heritage hardware for the SEP system can decrease the use of heritage hardware in the power system, and vice versa. Mission architects and TMC reviewers are interested in understanding the implications of the SEP system for the whole spacecraft, not just one of these systems.

This section starts by describing some available architectures for deep space power systems for SEP spacecraft. It discusses the primary advantages and disadvantages of each option and describes some of the implications for the EP subsystem. The second half of the section lists the numerous NASA and commercial SEP options that have flown to date or entered qualification life testing in ground facilities.

A. Power System Architectures for Deep Space SEP Missions

A variety of different SEP systems have been implemented for use in Earth orbit or Earth vicinity. Most of these systems operate on high power communications satellites that use regulated power busses for primary power. These systems generally deliver power at a nominal voltage of either 70 V or 100 V and operate over a relatively narrow power range, typically within 3 to 5 V of the nominal voltage. A typical architecture for a 100 V regulated system operating in Earth orbit is shown in Figure 1. For simplicity, battery charge and discharge hardware are not shown in any of the figures in this section.

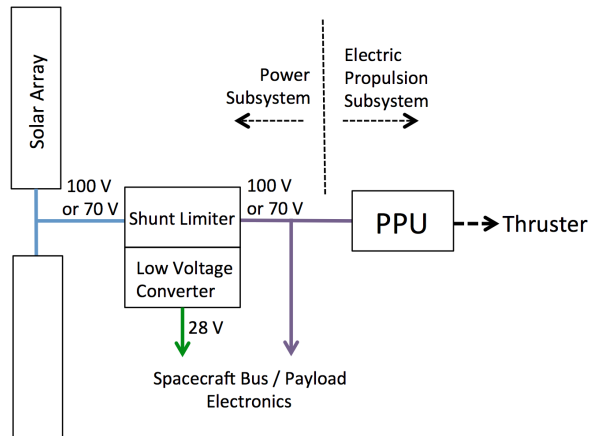


Figure 1: Typical Regulated Power Bus Architecture for Near Earth SEP Spacecraft

Regulated systems operating in Earth vicinity typically use a shunt limiter for regulation. Photovoltaic cells on the solar array are organized into strings and circuits operating in series and parallel to provide voltage and current to the spacecraft. The number of cells in series (length) determines the string voltage and the number of strings in parallel determines the operating current of each circuit on the array. The shunt limiter operates by switching circuits on and off to maintain the bus voltage within a tightly regulated range. When the spacecraft's demand for power is high, most of the circuits on the array will be connected to the main bus (switched on) to provide current. When the demand for power is low, most of the circuits will be left open circuit (switched off), and relatively little current will flow onto the bus.

Shunt limited systems work efficiently when the peak power operating voltage of the strings on the array is close to the target voltage for the bus. However, solar cell voltage depends strongly on cell temperature and is also a function of light intensity, and therefore varies with distance from the sun. On the Juno solar array, for example, the voltage generated by a single string increases 50% as the spacecraft travels away from the sun.¹³ Efficiently accommodating wide power and voltage variations from the solar array is a primary challenge for deep space solar power systems.

Figure 2 shows four possible deep space power architectures for SEP missions. Highlighted in red are elements of the system that might require significant development because they have requirements that differ significantly from elements used in near-Earth operations.

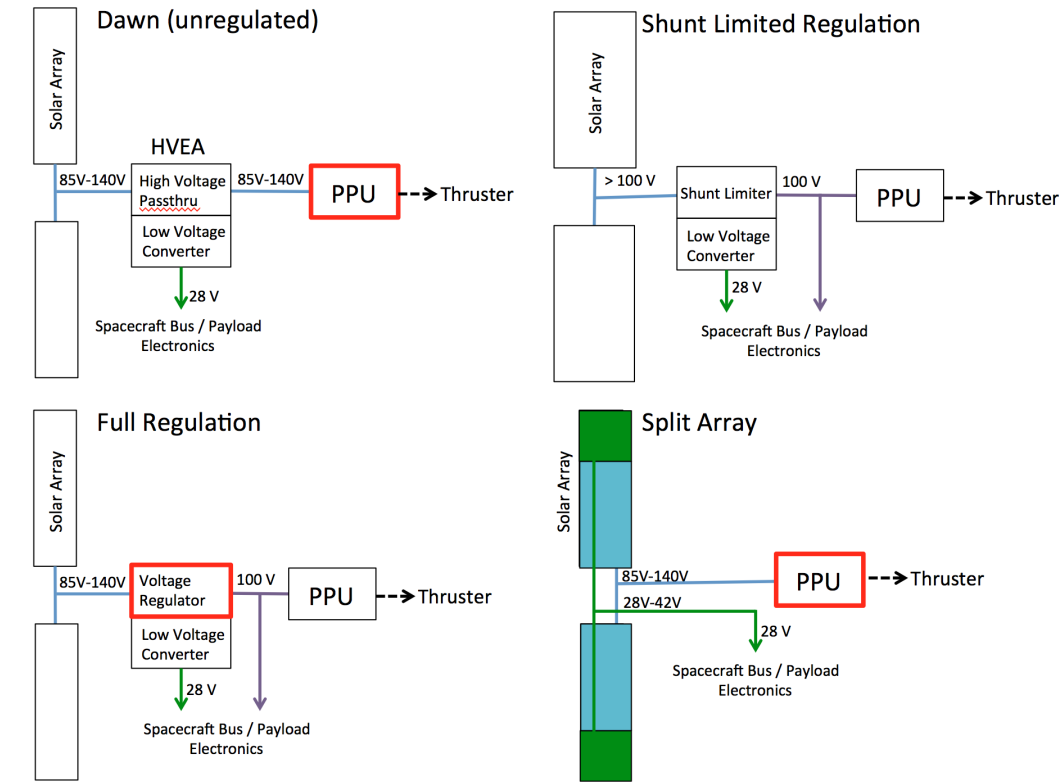


Figure 2: Power Systems Architectures for Deep Space SEP Missions.

1. Dawn Power Architecture (Unregulated)

As shown in Figure 2, the power system architecture for Dawn is a relatively elegant solution to the additional demands placed on the power system by the incorporation of electric propulsion.^{14,15} The solar array produces an output voltage of 80 to 140 V and the Dawn PPUs are designed to accept unprocessed power in this voltage range. As a result, solar array power goes directly to the EP system, avoiding the need for the complicated electronics associated with regulated power bus architectures. At the heart of the Dawn power system is the High Voltage Electronics Assembly (HVEA). The internally-redundant HVEA, designed and built by JPL for Dawn, is composed of three modules (Figure 3).¹⁴ The High Voltage Down Converter (HVDC) supplies 28 V regulated power to spacecraft loads and the battery. The High Voltage Control Electronics (HVCE) provides signals to control the HVDC and the solar array voltage (based on spacecraft loads), operates some spacecraft heaters, and monitors and controls battery charge state. The High Voltage Relay Assembly (HVRA) includes blocking diodes for the solar array, high voltage power fusing, and relays for switching power to the HVDC and the EP system PPUs. A block diagram and picture of the HVEA are shown in Figures 3 and 4.

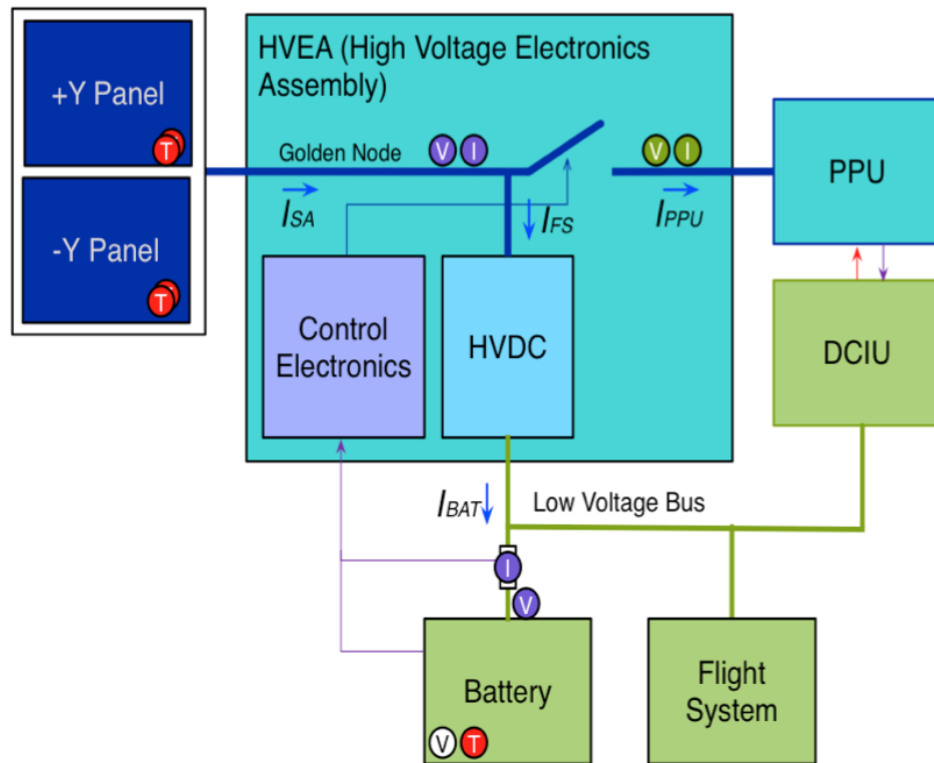


Figure 3: Simplified Block Diagram of the Dawn Spacecraft Power System showing the High-Voltage Power Domain (in blues) and the Low Voltage Power Domain (in green)¹⁵

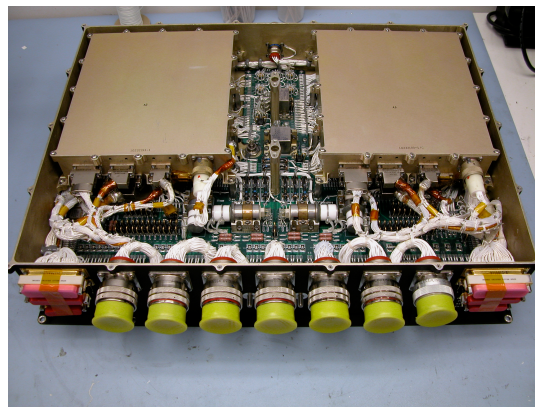


Figure 4: Photo of Dawn HVEA

The Dawn architecture, including the HVEA, is potentially an excellent model to follow for future Discovery-class power systems. The HVEA can be used for future missions with only minor design modifications. The HVDC module can be used without design modification for spacecraft with bus loads no greater than 760 W nominal and 1120 W peak. For larger bus loads the changes are simple and straightforward. The HVCE module would require at most small changes to account for differences in power system control loops.

Since the HVEA functions as little more than a pass through for power to the EP system through the HVRA module, it can accommodate EP systems with power draws much larger than the 2.6 kW used on Dawn with a few simple modifications. Changes would be limited to increasing the current carrying capability of the relays and fuses in the HVRA, and upgrading the thermal design to handle the changes. Both ion thruster and Hall thruster systems are compatible with the HVEA design. With this power system architecture design, EP systems of at least 10 to

20 kW can be accommodated with only minor design changes to the HVEA. Options exist that would increase the power density and lower the mass of the unit.

The main disadvantage of the Dawn power architecture is that the EP system's PPU is required to accommodate a relatively wide voltage range. EP systems intended for deep space use, like the NSTAR system used on Dawn, are designed and qualified to efficiently accommodate wide variations in input voltage. But EP systems intended to operate near Earth on regulated power busses use PPUs qualified to operate over only a small voltage range. On some missions, accommodations can be made to reduce the voltage range seen by the PPU. Spacecraft that remain relatively close to Earth can limit the voltage range seen by the PPU.¹⁶ In addition, commercial PPUs are often capable, with only minor or even no design changes, of operating over a wider voltage range than originally qualified for (see Section III B).¹⁶ The need to match variations in array voltage with a qualified PPU is the primary barrier to use of this architecture.

2. Fully Regulated Power Architecture

The fully regulated power architecture uses a transformer based voltage converter to increase or decrease voltage produced by the solar array to provide a regulated bus voltage to the spacecraft. The voltage converter is similar to units used to charge and discharge batteries on large communications satellites, but the variation in voltage from the array is much wider than typically seen at Earth. As a result, a substantial development or requalification effort is likely needed to develop and qualify a voltage converter for deep space applications.

The main advantage of this architecture is that commercial PPUs designed to operate on regulated power can be connected and used "off the shelf," without modification or requalification. This maximizes heritage and minimizes developmental risks associated with the PPU. However, a major disadvantage with this architecture is that the voltage converter regulates and transforms all power generated by the array. This results in power conversion losses equal to ~5% of the total power generated by the array that must be thermally accommodated by the spacecraft. This also makes the converter relatively large and heavy. This combination of lack of heritage and inherent electrical inefficiency makes it unlikely that a fully regulated power architecture would be used for a Discovery class mission concept.

3. Shunt Limited Regulation Power Architecture

This shunt regulated power architecture is a variation of the regulated bus architecture commonly used in Earth orbit. It uses the same basic topology shown in Figure 1. This architecture is simple and can be very efficient as it avoids the conversion losses associated with a voltage converter. However, unlike a converter, the limiter has no ability to "boost" voltage, so regulation is only maintained when the operating voltage of the array exceeds the regulation voltage of the bus. Efficiently maintaining high array voltage for the full duration of the mission is a challenge. Because the temperature of the array decreases as the spacecraft moves away from the sun, the voltage generated by a solar cell increases. The circuits on the array must provide voltage greater than or equal to the regulated bus voltage at all points in the mission, and are therefore sized to ensure that the voltage of the circuit exceeds the bus regulation voltage at the spacecraft's closest approach to the sun. When far away from the sun, each circuit generates excess voltage that is "clipped" by the shunt regulator. This ensures that the voltage seen by the power bus never exceeds the regulation voltage range.

Clipping the array voltage creates power losses that result in lower cell efficiency when operating far away from the sun. The power loss can be significant, and the size of the array must be increased to compensate for the loss of efficiency. For example, for a spacecraft in the main asteroid belt, a shunt regulated circuit sized to meet voltage requirements at 1 AU might generate 30% less power than an equivalent unregulated circuit. The efficiency loss would be smaller if the spacecraft remained closer to Earth.

This problem is not unique to SEP spacecraft, and the Juno mission demonstrated one possible mitigation strategy. The Juno arrays use three different solar cell string lengths, each optimized for a different phase of the mission. The array has "long" strings optimized for 0.85 AU to 1.90 AU, "medium" strings for 1.80 AU to 3.75 AU, and "short" strings optimized for 3.75 AU to 5.44 AU.¹⁷ The long strings provide sufficient power and voltage to operate the spacecraft when close to the sun. As the spacecraft moves away from the sun, the long strings generate less power and the shorter strings are switched on. Though the long strings operate at reduced efficiency when far away from the sun, the short strings operate at full efficiency, so the overall power losses to the system are reduced.

A similar strategy can be used for SEP missions. Because the full array is not used when operating near the Sun, the length of the circuits on the array can be varied to reduce power losses when operating far away from the Sun. The effectiveness of this approach is mission specific, and depends on what fraction of the array power is needed close to the sun and how much solar distance variation there is over the course of the mission. A further operational

mitigation that can be used is to “feather” the array by pointing it slightly away from the sun vector.¹⁶ This lowers the temperature of the array and boosts its voltage when close to the sun.

The advantage of this architecture is that the power regulation hardware is relatively simple and efficient and that all elements of the systems, including the shunt and the PPU, can be based on existing hardware that has already flown in Earth orbit. This means that it may be possible to construct systems that have full flight heritage for all elements of power and SEP systems. The main disadvantage of this architecture is that the solar array operates less efficiently than an unregulated array. In general, these losses will be smaller on missions that remain close to the Earth, where the voltage variations on the array are relatively small. In some cases, efficiency losses can also be mitigated by use of multiple-length strings on the array (like Juno), or by feathering the array. The penalty on array size is mission specific, a function of both thrust profile and solar distance as a function of time.

4. Split Array Architecture

The split array is a version of the unregulated architecture in which the solar array is split into two separate power domains, a high voltage domain and a low voltage domain. Power from the high voltage domain is dedicated to EP, while power from the low voltage domain is used for the balance of the spacecraft. Because the power domains are electrically isolated, no bulk voltage conversion is required within the power subsystem. However, both power busses are unregulated, so all loads on the spacecraft, including the PPU, must be able to accommodate relatively large voltage variations. In addition, because there is no power sharing between domains, there is some loss of efficiency compared to the Dawn architecture. Substantial excess power available in the high voltage domain when the EP system is turned off cannot be used to power science and telecommunications equipment connected to the low voltage domain. Instead, the low voltage domain must be sized to provide all power needs over the full mission. This results in a larger solar array for the spacecraft; how much larger depends on the mission specifics, including power-need variation by mission phase and variation in solar distance over time.

The primary advantage of this architecture is that it simplifies the power system by minimizing the number of converters and regulators required. The primary disadvantage is that the EP system’s PPU must accommodate a relatively wide voltage range compared to near-Earth operations. This requires modifications, and delta- or re-qualification of existing commercial PPUs, as described in Section III B. In addition, separating the spacecraft into separate power domains results in a larger solar array than would be required with a single bus power architecture.

B. Electric Propulsion Options for Discovery Class Missions

Among the mission architect’s highest priorities are to maximize flight heritage and minimize the cost of both power and EP subsystems. The heritage of the power processing unit (PPU) is a key part of this trade. Because NASA mission evaluators assess the *entire* subsystem against its *least* mature element, this section emphasizes three critical elements of the subsystem: thruster, PPU, and feed system. Because xenon tanks and thruster gimbals have similar heritage for all SEP options, they are not a major differentiator and can be omitted from consideration for the trade.

Tables 1, 2, and 3 summarize the performance and heritage of the thrusters, PPU, and feed systems for viable electric propulsion subsystem options available as primary propulsion for near-term Discovery class missions. This list includes the NASA-developed NEXT system and commercially developed systems from Aerojet-Rocketdyne, L-3, Fakel, and QinetiQ. All options listed have completed or entered qualification life testing. Developmental systems, such as NASA’s HivHAC Hall Thruster system, which has not started qualification life testing, are not included. Systems like the Busek BHT-600 that operate at a peak power below 1 kW are also not considered, as these systems are typically too small for use on a Dawn class missions.

For reference only, the tables also show the NSTAR ion thruster system used on Dawn. NSTAR was first flown on NASA’s Deep Space 1 mission launched in 1998. Because of low production volume it is now functionally obsolete and cannot realistically be considered viable for future applications. Further information about the NSTAR system can be found in Ref. 18.

The heritage evaluations are based on criteria defined in the Discovery-12 (2010) AO¹² and shown in the Appendix. The levels shown in the table reflect the heritage each element would likely have for a Discovery class mission, but determination of heritage is mission-dependent. In cases where mission dependencies are known, the table lists multiple heritage levels and the criteria associated with those determinations. The customization of certain elements of the feed system, like the layout of the tanks and propellant lines, is typical for space missions, and thus omitted from consideration.

Each EP system provides different mission performance relative to its power, specific impulse, and throughput capabilities and is therefore suited to a different class of mission. Rather than attempt to summarize all of the different mission types, we provide references to some of the various mission analysis studies conducted with each

type of system. Each EP system option is described below; a comparison of the different power and propulsion options follows in section III C.

In addition to the system-specific components, Table 3 includes the JPL Standard Architecture (JSA) feed system as an option for deep space missions. This simplified feed system, compatible with ion thrusters, comprises flight-heritage components and has been assembled and demonstrated on a laboratory NSTAR-class ion engine.¹⁹ The XIPS, NEXT, and T6 Ion Thruster based systems are all compatible with the JSA, making it a prime candidate for use on Discovery class missions. Elements of this system could also be used with Hall thrusters for some applications. The BPT-4000 xenon flow controller (XFC), for example, uses the same proportional flow control valve (PFCV) used in the JSA. Further details are provided in Ref. 19.

	NSTAR	NEXT	NEXT+XIPS PPU	BPT-4000	25-cm XIPS	SPT-100	SPT-140	T6
Performance Summary								
Power	Max: 2.3 kW Min: 450 W	Max: 6.9 kW Min: ~500 W	Max: 4.5 kW Min: ~500 W	Max: 4.5 kW Min: ~225 W	Max: 4.5 kW Min: ~225 W	Max: 1.5 kW Min: ~600 W	Max: 4.5 kW Min: < 2.0 kW	Max: 4.6 kW Min: 2.4 kW
Isp	3200	4200 s	3600 s	2000 s	3600 s	1600 s	1800 s	4075-4300 s
Thruster Mass	8.9 kg	13.5 kg	13.5 kg	12.5 kg	13.5 kg	3.5 kg	8.5 kg	7.5 kg
Thruster Efficiency	60% at 2.3 kW	71% at 6.9 kW	66% at 4.5 kW	57% at 4.5 kW	68% at 4.5 kW	50% at 1.35 kW	55% at 4.5 kW	66% at 4.5 kW
Throughput, demonstrated	235 kg	887 kg (Aug 13)**	see NEXT	452 kg	178 kg	182 kg	180 kg (Aug 13) [†]	78 kg
Throughput, theoretical	200-250 kg	> 887 kg	< 500 kg	> 1000 kg	~ 300 kg	not determined	521 kg ^{††}	270 - 350 kg
Total Impulse, demonstrated	7 MN-s	34.3 MN-s**	< 29.4 MN-s	8.7 MN-s	6.7 MN-s	2.7 MN-s	3 MN-s [†]	3.7 MN-s
Total Impulse, theory	10 MN-s	> 34.4 MN-s	< 19 MN-s	19 MN-s	11.4 MN-s	not determined	8.2 MN-s ^{††}	11.5 MN-s
Heritage:								
Manufacturer	L3	Aerojet-Rocketdyne	Aerojet-Rocketdyne	Aerojet-Rocketdyne	L3	Fakel	Fakel	QinetiQ
Flight Missions (previous or planned)	DS1, Dawn	None	None	AEHF (x6)	HS702 (many)	SS/L (many) European (many) Russian (many)	Future Commerical	BepiColombo (2015)
Heritage for Deep Space	Full	None***	None***	Full	Full	Full	Full*	Full/Partial
Mission Applications								
Deep Space Mission Applications (Examples)		ref 5,10		ref 8	ref 6	ref 7		
Comments	No longer manufactured	Offered with cost credit in Discovery 2010				Pole erosion begins at 43 kg		

* Full Heritage is anticipated after qualification for Earth orbiting applications is complete.

** Value current as of 6 August 2013

*** A flight-like model has passed performance & environmental testing as part of a technology development program. A full flight qualification model would need to be built and tested prior to first flight.

[†] values estimated, 3000+ hours completed as of August 2013

^{††} planned duration is for 8.2 MN-s, throughput is estimated

Table 1: Comparison of Thruster Options for Discovery

	NSTAR	NEXT	NEXT+XIPS PPU	BPT-4000	25-cm XIPS	SPT-100	SPT-140	T6
Performance Summary								
Power	Max: 2.3 kW Min: 450 W	Max: 6.9 kW Min: ~500 W	Max: 4.5 kW Min: ~500 W	Max: 4.5 kW Min: ~225 W	Max: 4.5 kW Min: ~225 W	Max: 1.5 kW Min: <500 W	Max: 4.5 kW Min: <2.0 kW	Max: 4.6 kW Min: 2.4 kW
PPU Mass	14.5 kg	33.9 kg	Same as XIPS	12.5 kg	21.3 kg	7.5 kg	15 kg	23 kg
PPU Efficiency	92% at 2.4 kW	95% at 7.1 kW	92% at 4.5 kW	92% at 4.5 kW	91%-93%	94% at 1.35 kW thruster power	not available	92%-95%
Redundancy/ Cross Strapping	1 PPU - 2 thrusters	1 PPU-2 thrusters	1 PPU-2 thrusters	1 PPU-1 thruster	1 PPU-2 thrusters	1 PPU-2 thrusters	1 PPU-4 thrusters	1 PPU-2 thrusters
Heritage:								
Manufacturer	L3	L3	L3	Aerojet-Rocketdyne	L3	SSL	SSL	Astrium Crisa
PPU Input Voltage	80V-145V	80V-160V	Same as XIPS	OTS: 68V-74V OTS*: 55V-85V	OTS: 95V-100V OTS*: 90V-110V Tested: 80V-120V	OTS: 95V-105V OTS*: 80V-120V	OTS: 95V-105V	OTS: 95-100V
Flight Missions (previous or planned)	DS 1 / Dawn	None	None	AEHF (x6)	HS702 (x many)	SS/L (many)	Future Commerical	BepiColombo (2015)
Heritage for Deep Space	Full	None	Partial	Full/Partial**	Full/Partial**	Full/Partial**	Full/Partial**,***	Full/Partial**
Comments	No longer manufactured	Offered with cost credit in Discovery 2010						

OTS = "Off-the-Shelf"

* Off-the-shelf with minimal modifications, requires delta or requal

** Heritage is application dependent. See text for description of where full and partial heritage apply.

*** Full Heritage anticipated for some applications after qualification for Earth orbiting applications is complete.

Table 2: Comparison of Power Processing Unit Options for Discovery

	NSTAR	NEXT	BPT-4000	25-cm XIPS	SPT-100	SPT-140	T6	Standard Architecture
Performance Summary								
Throttle Method	Bang-bang	PFCV and TT	PFCV	Two orifice selection plus TT	TT	TT	PFCV	PFCV
Flow Rate Throttle Range	x5	x4	x10	x2	x1***	x2****	x2	>10x**
Heritage:								
Manufacturers	custom	Aerojet-Rocketdyne	Moog	Moog	Fakel	Fakel	Moog	Multiple Commercial
Flight Missions (previous or planned)	Dawn	None	AEHF (x2)	HS702 (x many)	SS/L (many) European (many) Russian (many)	Future Commercial	BepiColombo (2015)	AEHF, Dawn, TacSat-2, Commercial
Heritage for Deep Space	Full	None	Full	Full	Partial	Partial	Partial	Full
Comments	Custom for Dawn	Offered with cost credit in Discovery 2010		Also Compatible with JPL Std Architecture			Currently in fit qualification. Also Compatible with JPL Std Architecture	Commercially-available flight-qualified parts

TT = Thermothrottle

PFCV = Proportional Flow Control Valve

* System is nominally operated at a single fixed flow rate. Further work is needed to determine the system's throttle capability.

** limited by calibrated range of pressure transducer and flow restrictor

*** commercial implementation is for single operating point. More work is required to determine the total flow range of the thermothrottle.

**** commercial implementation is for limited range. More work is required to determine the total flow range of the thermothrottle.

Table 3: Comparison of Feed System Options for Discovery

1. NEXT Ion thruster

NASA's Evolutionary Xenon Thruster (NEXT), has been under development by the NASA In-Space Propulsion Technology project since 2002, led by the NASA Glenn Research Center.^{20,21} The NEXT system development includes an ion thruster, PPU, feed system components, and gimbal. The system is highly throttleable, designed to operate over power levels of 0.6-7.2 kW to meet the needs of breakthrough science missions. The program's technology maturation goal is to attain system-level TRL 5, with significant progress toward TRL 6.²²

The NEXT thruster draws from the experience with and successful design of the NSTAR ion thruster flying on Dawn. The throttle table encompasses thrust levels from 26-236 mN, specific impulses from 1410-4310 s, and thruster efficiencies from 33-71%.²³ A flight-like thruster design has successfully passed qualification-level environmental tests,²⁴ and an engineering model has been the subject of a mission-representative long-duration wear test, having achieved 884 kg of throughput and 34.2×10^6 N-s total impulse as of July 2013.²⁵

The NEXT PPU is specifically designed to be compatible with an unregulated power bus architecture as on Dawn. The PPU supplies up to 7 kW of power to the thruster at input bus voltages between 80 and 160 V. Measured efficiencies vary from 84 to 95% over the throttle range at 100 V input voltage.²⁶ The engineering model PPU has experienced a number of failures in its development, resulting in delayed completion of full functional and environmental testing. To date, the PPU remains unqualified,^{26,27} and this current lack of maturity remains a major barrier to acceptance and infusion on near-term Discovery proposals due to development-completion schedule-risk and cost-risk ratings anticipated from TMC reviewers. In 2010, the Discovery AO offered a partial offset for risks associated with system development completion by providing a cost incentive for missions using NEXT, and by exempting the system from the requirement to provide a credible technology development plan in the proposal. Continued incentivizations would be required to make this system viable for near-term Discovery proposals; proposers may be reluctant to rely on this same approach without having a credible development plan and schedule from the technology provider.

2. NEXT Ion Thruster + XIPS Power Processing Unit

While the NEXT thruster has considerable capability, the developmental issues described above for the PPU make it useful to consider alternative PPUs that may be available for it. The XIPS ion thruster PPU has extensive flight heritage, high production rate, and proven reliability in space. Since the NEXT ion thruster has throttle table levels compatible with output voltages available from the XIPS 25-cm thruster PPU, at power levels less than 5 kW it may be desirable to use the XIPS PPU together with the NEXT engine for near-term Discovery missions.

The XIPS PPU is manufactured by L-3 Communications, Inc.; 42 units have flown to date. The flight PPU requires a regulated input bus voltage of 100 ± 5 V, but has been operated successfully during ground testing with an

input voltage range of 80 to 120 V. The XIPS PPU has been throttled from 0.4 to 5 kW, and operates at an efficiency of 91 to 93% depending on the throttle level selected. Since the XIPS PPU has a maximum output voltage of 1215 V, compared to 1800 V for the NEXT PPU, the maximum specific impulse of this combined system is lower than would be seen with the NEXT PPU, and is limited to 3600 s at a power level of 4.5-5 kW. The overall system would therefore operate at a lower peak power, and with a lower useable specific impulse, than an equivalent system using the NEXT PPU. The modifications needed to adapt the XIPS PPU for use with the NEXT ion thruster are described in section IV.

3. BPT-4000 Hall Thruster

The qualified life and throttling capabilities of Aerojet-Rocketdyne's BPT-4000 Hall thruster and its low cost relative to government systems make it a leading candidate for near-term infusion into cost-capped science missions. The BPT-4000 first flew in 2010 on the USAF's Advanced Extremely High Frequency (AEHF) geosynchronous Earth orbit (GEO) communications satellites built by Lockheed Martin Space Systems.^{28,29,30,31,32,33,34,35,36,37} A second vehicle followed in 2012 and AEHF-3 was launched September 18, 2013. A total of six vehicles is planned to complete the AEHF constellation. JPL began investigating the feasibility of using commercial EP systems in 2005, with the first BPT-4000 article published in 2006.⁸ JPL investments have included mission studies,^{8,16,38,39} hardware evaluations,^{40,41} plasma modeling investigations,^{42,43,44,45,46} technology development of the PPU,⁴⁷ and concepts and proposals to NASA under several different programs (Discovery, New Frontiers, Flagship, ISS drag makeup, Mars). Aerojet-Rocketdyne's qualification life test (QLT) for GEO applications with the Qualification Model (QM) BPT-4000 was completed in 2005^{32,33} and was subsequently extended, bringing the total operating time to 10,400 h and propellant throughput to 452 kg for discharge powers of 0.3-4.5 kW. One of these life test extensions was funded by NASA SMD's In-Space Propulsion Technology (ISPT) project in 2006.^{34,35} The total demonstrated impulse of 8.7 MN-s now exceeds the 7.2 MN-s demonstrated during qualification testing of NASA's NSTAR ion thruster flying on Dawn.^{48,49} Erosion measurements from the QLT indicate that the channel reached essentially a zero-erosion state between 5,600 h of operation and 10,400 h.³⁷ Based on erosion rates observed during the QLT, the theoretical throughput capability of a BPT-4000 thruster could far exceed 1000 kg (19 MN-s) if additional life testing were to be performed with the still-preserved QM thruster.³⁷ Models linking the detailed physics of this significant reduction of the erosion in the BPT-4000 were not available upon the conclusion of the QLT as Aerojet-Rocketdyne relied on empirical modeling and accelerated life testing to develop the magnetic field shape that achieved reduced erosion.³⁷ Subsequently, using physics-based models developed by JPL, the team demonstrated that a facility effect was not responsible; rather, the particular shape and strength of the magnetic field near the walls of the thruster (later referred to as "magnetic shielding") reduced the energy and flux of ions to the wall by orders of magnitude.^{43,44} The physics of magnetic shielding have subsequently been validated by JPL through detailed modeling and experiments on another test thruster.^{50,51,52,53}

The BPT-4000 PPU used on AEHF provides commandable power to the plasma discharge, electromagnets, cathode heater, and cathode keeper. In addition, the PPU drives two solenoid-holding valves in the Xenon Flow Controller (XFC) and utilizes closed-loop control to operate the proportional flow control valve (PFCV) to regulate the xenon flow to the thruster, which controls the discharge current to the commanded level. Commands and telemetry are communicated with the spacecraft utilizing a MIL-STD-1553B data link. Radiation hardened S-Level components provide maximum reliability. The PPU input voltage is designed to interface with a regulated 70 V spacecraft power bus; a minor modification to the PPU would enable an input voltage range of 55-85 V, with some limits on the output capability (see section IV).¹⁶ The PPU output is designed for voltages of 150-400 V, maximum current of 15 A, and maximum power of 4.5 kW. A modular PPU architecture has also been developed by the Aerojet-Rocketdyne and JPL team,⁴⁷ designed for input voltages of 70 to 140 V and output voltages of 150 to 800 V. Continued development of this modular PPU will inherit many of the design approaches now flying on AEHF.

4. XIPS Ion Thruster

The 25-cm Xenon Ion Propulsion System (XIPS) is a second-generation ion thruster manufactured by L-3 Communications, Inc.⁵⁴ capable of operating at 4.5 kW and 3600 s. The 13.7 kg thruster produces up to 167 mN of thrust at over 68% efficiency, and typically operates at 2 kW of power for station keeping on Boeing communications satellites. The heritage behind the 25-cm thruster is the 13-cm XIPS thruster flown on 15 Hughes/Boeing 601HP Communications satellites starting in 1997, and the 25-cm thruster has been used for station keeping on 21 of the Hughes/Boeing 702 class satellites flown since 2000. Thus over the last 16 years, 60 13-cm thrusters and 84 25-cm thrusters have flown successfully. The 25-cm life-test thruster successfully completed a 16,250 h life test, with 14,134 on-off cycles distributed over two power levels, and with a total throughput of over

170 kg. An extensive evaluation program, performed at JPL, has studied the use of XIPS thrusters for deep-space missions.^{55,56} The primary wear locations were determined to be the cathodes and grids. Evaluation of the grid life indicated that the XIPS thruster can process over 200 kg of xenon propellant with 50% margin, and throughputs of up to 300 kg are possible for mission trajectories with a large fraction of thrusting time at low power levels.

The PPU for the XIPS engine is also manufactured by L-3 Communications, Inc. Since Boeing satellites have redundant thrusters for north-south station keeping, and two thrusters are flown with each PPU, 42 of these 4.5 kW PPUs have flown to date. The PPU requires a regulated input bus voltage of 100 ± 5 V, but in test the PPU has operated successfully with an input voltage range of 80 to 120 V. The PPU has a mass of 21.3 kg and operates at an efficiency of 91 to 93% depending on the throttle level selected. Both the thruster and PPU have been throttled from 0.4 to 5 kW during tests for deep-space applications. The flow control system for the XIPS thrusters is manufactured in-house at Boeing, but the XIPS thrusters could run on the JSA xenon flow system developed by JPL in 2009.¹⁹

5. SPT-100 Hall Thruster

The SPT-100 is one of the most successful electric thrusters to have flown; more than 200 units have flown successfully aboard Russian, European, and American spacecraft.⁵⁷ The system implementation preferred for NASA missions would be based on that used by U.S.-based satellite manufacturer SSL (formerly Space Systems/Loral). The first SSL spacecraft with SPT-100 systems launched in 2004⁵⁸ and a total of thirteen spacecraft have been launched since then, with many more currently in production.⁵⁹ The system includes four thrusters and two PPUs designed and manufactured by SSL in Palo Alto, CA. It also includes commercially available propellant storage and feed-system components.⁶⁰

The 1.35-kW SPT-100 is manufactured by Experimental Design Bureau Fakel in Kaliningrad, Russia and produces a nominal thrust of 83 mN and specific impulse of 1500 s. Its performance has been demonstrated in three separate life tests, the longest of which had a xenon throughput of 182 kg and a total impulse of 2.71 MN-s.^{60,61,62} The SPT-100 is primarily used at its nominal maximum power for stationkeeping in GEO, but the thruster is capable of operating over a range of input powers.^{63,64} Fakel also manufactures a Xenon Flow Controller (XFC) which accepts a regulated xenon pressure and controls flow to the thruster and associated cathode via a thermosthrottle controlled by the SSL PPU.

The PPU-100 is designed to accept power from the spacecraft power bus, provide regulated power to the SPT-100, control the XFC, and communicate with the spacecraft bus computer. The efficiency is 94% at the nominal operating condition of 1350 W (300 V and 4.5 A). Since SSL spacecraft use a 100 V regulated power bus architecture, the PPU is qualified to operate over an input voltage of 95 to 105 V. The design of the unit is such, however, that it is functional over a range of 80 to 120 V with only minor losses in performance. This functionality has been demonstrated by analysis, but not yet by performance test.¹⁶

6. SPT-140 Hall Thruster

SSL has been collaborating with Fakel to develop the SPT-140 thruster, a higher-power (4.5 kW) evolution of the SPT-100. Although initial development work was performed in the late 1990s in anticipation of commercial needs,^{65,66} this was interrupted by an industry down cycle; only recently has SPT-140 work resumed in earnest.⁶⁰ Thruster performance at full power measured with an early development model unit was 289 mN at 1780 s and 55% efficiency.⁶⁷ The throttled performance has been measured down to 2 kW although stationkeeping is planned at a 3 kW operating point. A qualification-model life test is now underway with a planned total impulse of 8.17 MN-s,⁵⁹ which is equivalent to 521 kg of xenon throughput at full power assuming a life-averaged specific impulse of 1600 s. The planned commercial implementation of the system uses the same propellant feed system and storage components as the SPT-100 system.

The PPU-140 design is based on the successful PPU-100, and a qualification program is currently underway.⁶⁰ The unit is designed for a 100 V regulated bus, the same as for the PPU-100. The discharge supply is composed of three 1.5 kW converters similar to the ones used in the PPU-100, with some design improvements for increased efficiency. These converters operate at 96% efficiency when producing the 300 V, 15 A, full-power discharge. The magnet and cathode heater supplies have also been upgraded to handle the larger, more powerful thruster. An early development unit demonstrated greater than 94% efficiency at full power.⁶⁶

7. T6 Ion Thruster

The T6 ion engine is a 22-cm diameter, 4.5-kW Kaufman-type ion thruster produced by QinetiQ, Ltd. in the United Kingdom, and is baselined for both the European Space Agency BepiColombo mission to Mercury scheduled to launch in 2015, and the AlphaBus communications satellite. The T6 thruster heritage includes the 10 cm diameter T5 Kaufman ion thruster that is successfully operating on the ESA GOCE spacecraft.⁶⁸ The 7.5 kg mass T6 Kaufman-type thruster has demonstrated throttling capability from 2.5 to over 4.5 kW and produces a thrust level of 75 to 145 mN at a specific impulse of up to 4300 s and a total efficiency of nearly 66%.⁶⁹ This performance was verified in testing at JPL in 2009.⁷⁰ The unique feature of this engine is a graphite accelerator grid that has lower erosion rates than standard molybdenum grids and is largely responsible for a 200 to 300 kg xenon throughput rating depending on the throttle profile. T6 thrusters could run on the JSA xenon flow system developed by JPL in 2009.¹⁹

The PPU for the T6 thruster was also developed for the BepiColombo mission and is manufactured by EADS Astrium Crisa in Spain. This PPU provides power conditioning and control to both the QinetiQ T6 ion thruster and multiple xenon flow control units. The PPU weighs about 23 kg and requires a regulated input voltage of 95 to 100 V $\pm 2\%$. The electrical efficiency of the PPU is 92 to 95% depending on the throttle level selected, and the outputs from the PPU can be switched between two or more thrusters as needed.

C. Comparison of Power and Propulsion System Options

The previous section indicates that rapid adoption by the commercial telecommunications marketplace of SEP technology, and consequent competitive development and fielding of hardware over the past decade or so, has resulted in a variety of commercially developed SEP systems offering potential high-heritage utility for NASA's Discovery program. The changes required to adapt these systems for deep space are straightforward. Because of comparatively high production volume, these systems are also likely to be less expensive – and have lower cost risk – than dedicated NASA-only systems. Several conclusions can be reached by examining the tables against the selection criteria given in section II.

First, because of the lack of system-level heritage, the NEXT thruster, PPU, and feed system are challenging to propose successfully for competed missions unless exceptions are driven into the evaluation criteria by the AO. In 2010, the Discovery program did offer a cost-credit incentive for missions using NEXT, and exempted proposers from having to rationalize the system technology development plan. The remaining development risk for the NEXT PPU likely exceeds the adaptation risk for commercial PPU options. Given the heritage of commercial alternatives, NEXT would become most attractive to proposers if the Discovery program accepted the full cost and risk burden of delivering all elements of a NEXT flight system (i.e., provide the system as Government Furnished Equipment [GFE]), including exemption from TMC evaluation. The program would need to budget for this contingency, as selection of such a mission proposal would immediately incur the costs and cost-risk of PPU development completion no later than the end of the mission definition phase (Phase B).

If the cost and risk of developing the NEXT PPU are too rich for either the Discovery program or proposers to absorb, one alternative is to use a NEXT thruster with the XIPS 25 cm PPU. Modifications to the XIPS PPU would be required (see section IV), but this option would offer lower risk, and likely lower cost, than completing development of the NEXT PPU. The XIPS PPU is limited to a peak power of 4.5 kW, well below the 7 kW peak power capability of the thruster. However, this lower power level would be adequate for many cost capped missions, since cost considerations generally encourage mission architects to use the smallest solar array suitable to accomplish their particular mission.

Second, the unregulated Dawn power architecture has the highest intrinsic efficiency of any of the power architecture options. However, this architecture requires a PPU that can accommodate a wide input voltage range. The NEXT PPU can accommodate this voltage range by design, but is still in development. A variety of commercially developed PPUs can operate with a regulated power bus, but none have been qualified at this time to operate over a wide input voltage range. Some commercial PPUs have been tested with wider input voltage ranges, and several could be modified relatively easily to support wider operating ranges than currently available (see section IV). Once fully qualified to support an unregulated voltage range, an off-the-shelf PPU would greatly benefit deep-space missions because it would maximize both efficiency and heritage of the combined power and EP system. Lacking that, viable development plans for several commercial options could be proposed for Discovery (see section IV).

As an alternative, for missions to near Earth objects (NEOs) and other targets that remain relatively close to Earth, (perhaps as far out as the orbit of Mars), it may be possible use an unmodified PPU qualified for a regulated voltage input with a shunt regulated or split bus power architecture. Such an architecture could maintain the heritage of the PPU and power system, but would be limited in the distance it could travel from the sun so as to limit the voltage variation seen on the array. In all cases, allowing the PPU input voltage to vary, even over a limited range, increases the overall efficiency of the combined power-propulsion system. Increasing the allowable input PPU voltage range is always beneficial for deep-space missions, by providing the mission architect with greater flexibility in selecting a combined power-propulsion mission architecture that maximizes overall efficiency and flight heritage.

IV. Adaptation of Commercial Systems for Deep Space Use

Commercial SEP systems offer strong potential benefits for deep-space missions, but for mission architects and reviewers it is critical to understand the scope, technical risk, and cost of the changes needed to adapt these systems for NASA use. Section IV A describes the changes needed to adapt the commercial SEP options listed in section III for deep space. These changes are based on generic deep-space requirements; missions with particularly challenging environments, particularly for radiation and/or thermal accommodation, might require additional work. In addition, like any heritage system, a commercial SEP system on a flight program would undergo an inheritance review and analysis by mission assurance personnel that would scrutinize its parts, materials, and processes for additional items requiring changes to meet unique mission or implementation requirements.

As above, this analysis assumes that each spacecraft will use previously qualified off the shelf propellant tank(s) with a unique tank and line layout. Individualized designs of this type, typical for all space missions, are not included in the tables. The heritage definitions used in the tables are taken from the Discovery 2010 AO (see Appendix).

Cost-estimate assessments for commercial SEP systems are challenging for TMC because NASA cost databases have only a single operational mission (Dawn) for historical cost basis and analogy. Commercial industry has flown many more SEP systems than NASA, and has a much richer cost base and experience to draw on in estimating delivery cost. Section IV B describes methods that can be used to accurately estimate the cost of adapting commercial SEP systems for cost-capped deep-space proposals.

A. Summary of changes needed to adapt commercial SEP systems for Deep Space

1. BPT-4000 Hall Thruster

Table 4 updates the analysis from Ref. 8 of the changes needed to adapt a BPT-4000 Hall thruster to NASA science missions. All of the feed system components are usable without modification off-the-shelf (OTS). Analysis and laboratory experiments^{8,40,41} have shown that the thruster is OTS except for the desire to extend the throughput and possibly the need to delta qualify the thruster for the low temperature extremes that can be seen on outbound deep space missions. An alternative to delta qualification could be the addition of heaters to keep the thrusters warm in flight. A flight-model BPT-4000 is now being procured by JPL that will be used to qualify the thruster to deep-space environments. Since Ref. 8, BPT-4000 throughput has been extended to 452 kg of xenon over 10,400 h.³⁷ Many hypothetical missions could benefit from still higher throughputs; and a restart of the life test is planned in FY14 that seeks to extend the qualified throughput to 600 kg. Ultimately BPT-4000 throughput is expected to exceed 1000 kg.^{37,45}

Two paths are being pursued for development of a BPT-4000 PPU suitable for NASA missions. Analysis conducted since Ref. 8 has shown that the PPU now flying on the USAF AEHF mission can be adapted, via a resistor change, to allow 55-85 V input voltages.¹⁶ Requalification would be required, but the technical risk is low with this approach because the output power and voltage will be limited in order to retain standard part de-rating guidelines. This might, however, limit mission capture due to the smaller input voltage range.

The other path being pursued is development of a modular PPU⁴⁷ with an input voltage range of 70 to 140 V. Due to the larger input voltage range, this modular PPU would capture to all of the Decadal Survey missions needing EP, and be scalable to even high-power missions like the Asteroid Redirect Mission (ARM). A major risk associated with the modular PPU approach has already been retired through demonstration of the discharge power supply topology.⁴⁷ The design for the remaining PPU components will inherit many of the approaches already flying on the AEHF PPU, speeding development and limiting risk. The design-phase engineering plus qualification program for this second path would, however, be a significant investment; bringing the modular PPU to TRL 6

would cost approximately twice the cost of modifying the AEHF PPU to achieve the 55-85 V input voltage range. The PPU uses a MIL-STD-1553 interface.⁸

Component	Status
Thruster	Full heritage. Note: Partial heritage for missions with low temperature extremes. Thermal requalification is planned for FY15. Note: Life testing planned for FY14 would extend allowable xenon throughput from 452 to 600 kg.
PPU	Full heritage when input voltage is 68-74 V Partial heritage for input voltage is 55-85 V: resistor change required. Requalification required (low technical risk). Note: development may be required to adapt MIL-STD-1553 interface to a mission specific spacecraft avionics interface.
Feed System	Full heritage.

Table 4: Summary of changes to adapt BPT-4000 Hall thruster system for NASA science missions.

2. XIPS Ion Engine

A detailed investigation of the changes and delta-qualification steps required for flying the XIPS engine on a deep-space NASA mission was previously published.⁵⁵ In general, there are no modifications required in the design or manufacturing of the thruster. Life and throughput have been previously evaluated for deep-space mission applications based on the existing XIPS life test and analysis of the grid and cathode wear-out mechanisms. However, the lifetime of this thruster is highly dependent on the trajectory used, because of the higher erosion rates observed at higher power. A full analysis of the throughput capabilities for a given class of mission trajectories will be needed to delta-qualify the thruster life. Present environmental testing of the thruster and the PPU (vibration and thermal) generally brackets the requirements for deep-space missions, and would only require additional analysis or testing depending on the mission trajectory and launch vehicle type. There may be a need to delta qualify the thruster for the low temperature extremes that can be seen on outbound deep space missions. An alternative to delta qualification could be the addition of heaters to keep the thrusters warm in flight.

The XIPS PPU has been qualified to a regulated input bus voltage of 100 ± 5 V. A modification to the XIPS PPU will be required to accommodate variable input voltage associated with outbound deep space missions. An evaluation of the existing input inverter by L-3 showed that the PPU could operate over the input voltage range of 90-110 V and stay within the safety margins of the devices for flight. L-3 stated that the PPU had been successfully tested on the ground with input voltages from 80 to 120 V. However, it is likely that the inverter switches would have to be changed to higher rated versions to maintain the required margins for flight applications. Due to improvements in the capabilities of these switches by the manufacturer since the original design of the PPU, it was stated that the higher rated parts are available in the same package. Full testing of the upgraded input inverter with these new devices is needed for NASA mission applications, and a delta-qualification program must be completed prior to flight implementation.

Component	Status
Thruster	Full heritage for design and manufacturing. Partial heritage for operating environment. Full analysis of mission specific throughput capability needed to delta-qualify thruster life. Note: Partial heritage for missions with low temperature extremes. Additional thermal qualification may be required on outbound deep space missions.
PPU	Full heritage for input voltage 95-105 V. Partial heritage for input voltage 90-110 V. Compatibility with this range has been demonstrated in ground test. Full parts stress analysis and delta-qualification is required over this range. Partial heritage for input voltage 80-120 V. The ability of the PPU to operate over this range has been demonstrated in ground test, although higher rated input inverter switches will be required for flight to maintain margins. Full parts stress analysis and delta-qualification is required over this range. Note: development may be required to adapt serial data link to a mission-specific spacecraft avionics interface.
Feed System	Full heritage when using JSA (JPL Standard Architecture).

Table 5: Summary of changes to adapt XIPS ion thruster system for NASA science missions.

3. SPT-100 Hall Thruster

The SPT-100 system (thruster, PPU, and Xenon Flow Controller (XFC)) can be easily adapted for a limited set of missions without any changes to the hardware, provided the input voltage to the PPU is within an acceptable range and the flow throttling requirements are within the XFC limits. The PPU is qualified for a limited input voltage range of 95 to 105 V, but is operable over a much wider range of 80 to 120 V.¹⁶ Circuit modeling has been performed to demonstrate this operation, but additional testing would be required to qualify the unit for operation over this input voltage range. There are no issues with component performance or usage at input voltages as low as 80 V, but at voltages higher than 105 V some parts exceed the SSL derating requirement. This does not necessarily mean that these parts must be replaced, as operation at higher voltages is associated with larger spacecraft solar ranges and hence lower-power PPU operation. A full mission-assurance review of the circuits would be warranted. The power throttling range of the SPT-100 system is limited by the performance of the XFC thermothrottle over the range of currents supplied by the PPU. Because the commercial implementation of this system is for a single operating point, its throttleability has not been extensively characterized, so additional work is required to understand the full throttle range that could be supported by the existing SPT-100 XFC. The thruster by itself has a power throttling range of at least a factor of three.

Component	Status
Thruster	Full heritage. Note: Partial heritage for missions with low temperature extremes. Additional thermal qualification may be required on outbound deep space missions.
PPU	Full heritage when input voltage is 95-105 V. Partial heritage when input voltage is 80-120 V. Compatibility with this range has been demonstrated by analysis, but testing is required to qualify unit over this range. Note: development may be required to adapt serial data link to a mission specific spacecraft avionics interface.
Feed System	Full/partial heritage depending on throttle range. Throttling range limited by thermothrottle. Analysis and/or test required to determine the full throttle range of existing hardware.

Table 6: Summary of changes to adapt SPT-100 Hall thruster system for NASA science missions.

4. SPT-140 Hall Thruster

The SPT-140 system is still undergoing qualification for commercial use. The SPT-140 is essentially a higher power version of the SPT-100 system. Because the SPT-140 is manufactured by the same vendors, and used by the same customers, and operated in the same environments as the SPT-100, we would expect similar adaptation issues at the system level.

5. T6 Ion Engine

The T6 ion thruster and PPU are designed and manufactured by commercial companies, and are undergoing qualification at this time for use in the ESA BepiColumbo mission to Mercury. Since this is a deep-space application, the required changes for use in NASA missions would likely be small. It may be necessary to redo and/or modify the thermal qualification testing of the thruster because en route to Mercury, BepiColumbo will not experience the low temperatures expected far from the sun. The T6 PPU would likely require a delta qualification for thermal environment. Otherwise the environments and environmental testing requirements can be expected to be very similar. Deep-space missions also typically have large throttling requirements (factors of 5 to 10) over the duration of the mission. Because BepiColumbo will follow a power-rich inbound trajectory, the T6 is presently being qualified at only four throttle points. However, the thruster has been tested over a larger throttle range at QinetiQ. A full throttle table must be developed for this thruster, preferably over a 0.45 to 4.5 kW power range, and the thruster performance and life qualified over this range by a combination of analysis and test.

Component	Status
Thruster	Full heritage for design and manufacturing. Full heritage for deep space (Mercury) Note: Partial heritage for missions with low temperature extremes. Additional thermal qualification may be required on outbound deep space missions. Partial heritage for missions requiring full throttle table. Throttle table must be developed, and thruster performance and life requalified by analysis and test.
PPU	Full heritage for deep space (Mercury) Full heritage for applications where PPU input voltage range is 95 – 105 V. Partial heritage for applications where PPU input voltage range is unregulated. Note: thermal delta-qualification may be needed for low temperature environments. Note: development may be required to adapt MIL-STD-1553 interface to a mission specific spacecraft avionics interface.
Feed System	Full heritage when using JPL Standard Architecture.

Table 7: Summary of changes to adapt T6 ion engine for NASA science missions.

6. NEXT Ion Engine + XIPS PPU

In order to operate the NEXT engine, the XIPS PPU would require a modification to the output voltage and current of the heater power supply, and a new Digital Control Interface Unit (DCIU) that provides communication between the spacecraft computer and PPU. The changes to the heater supply are not critical since the PPU design utilizes relatively independent slices for each of the power supplies, and this single heater-supply module could be changed to match the NEXT heater characteristics. A new DCIU card to interface the XIPS PPU to a NASA spacecraft computer through a standard 1553 interface has been designed and built by JPL; the breadboard card is undergoing test presently, and once completed will require repackaging into a flight configuration to fit into the PPU control slice. Delta-qualification of the modified PPU with the new DCIU card and heater supply would then be required; this delta-qualification could also verify and validate the larger input voltage range (needed for deep-space missions) that was previously demonstrated in test by this PPU.

Component	Status
PPU	Partial heritage. Design modifications and delta qualification required would: • Change heater power supply output voltage and current • Allow operation over wider input voltage range (see table 5 for more information) Note: development may be required to adapt serial data link to a mission specific spacecraft avionics interface.

Table 8: Summary of changes to adapt XIPS PPU system for NEXT ion thruster.

B. Approaches to Costing the Adaptation of Commercial SEP Systems for Discovery

The Technical Management and Cost (TMC) evaluation of Discovery proposals includes Independent Cost Estimates (ICE) that use existing parametric models such as PRICE-H and SEER-H, calibrated to prior flight projects.¹² To date, these models have been heavily reliant on NASA's only SEP science mission (Dawn) for relevant model calibration. Among other issues (e.g., analytical brittleness due to the singular analogy), this “bakes” a high cost benchmark into the calibration for SEP-based proposals, because Dawn's SEP system was expensive due to its groundbreaking nature and ambitious mission requirements. Dawn's pathfinding SEP system required significant development far beyond what would be required to adapt today's state of the art SEP systems for flight on Discovery. The Dawn-unique developments included:⁷¹

- Additional development of the ion thruster and PPU beyond the Deep Space 1 technology demonstration mission, to meet the reliability requirements of a deep-space science mission
- New Digital Control and Interface Units (DCIUs)
- A new Xenon Control Assembly (XCA) to operate multiple thruster strings
- A new highly optimized xenon tank
- New thruster gimbals

Calibrating to Dawn for today's Discovery proposals would be analytically flawed, by levying technology maturation assumptions based on Dawn's first-ever, "stretch" nature, and precluding both NASA and the science community from leveraging the current SEP-system state of the art and flight experience. There has been a proliferation of similar (i.e., equivalent power levels and strings) commercial EP systems flown since Dawn,⁷² some of which have SEP system costs *an order of magnitude cheaper than* the Dawn system. Such dramatic cost avoidance is routinely achieved through:^{8,16,72,73,74}

- Efficiencies of scale (typical SEP system manufacturers fly many systems each year)
- Amortized development costs (qualification costs were paid in early development, with no significant changes as SEP systems are delivered for flight)
- Modular architectures (where thruster strings and xenon capacity can be added or subtracted in modular increments without modifying box designs such as was the case with the Dawn DCIUs and XCAs)
- Simpler technologies (Hall thrusters have fewer components and power supplies, operate at lower voltages, and require less tight machining tolerances than do ion thrusters)
- Less onerous input voltage range requirements (commercial SEP systems operate at 1 AU, while deep-space missions operate throughout a large solar range, experiencing lower solar heat flux and thus higher array output voltages).

Although the changes detailed in section IV-A would have to be made to these cost-effective, commercial SEP systems for application to NASA deep-space missions, most of the other cost-avoidance benefits listed can be leveraged for Discovery missions.

Accurate estimation and evaluation of the cost and cost-risk of adapted-commercial SEP systems in Discovery is possible within the framework of the Discovery AO Heritage Appendix,¹² where cost elements of a proposed SEP system can be broken down into constituent elements, and modifications to heritage components can be explained on a case-by-case basis.⁷⁴ For example, thruster, feed system components, and xenon tanks used in commercial SEP systems can often meet the performance, interface, and environmental requirements of deep-space missions without modification to the existing qualification heritage.⁷² Modifications to the PPUs often would be required to meet the change in input voltage requirements for deep-space missions, but these required modifications, and the costs associated with them, can be explained in further detail in a proposal.¹⁶ This information would enable the TMC to calibrate parametric cost models appropriately, in turn enabling accurate assessment and validation.

V. System Margins for Deep Space Missions using Solar Electric Propulsion

Designing low-thrust mission architectures is challenging because it requires simultaneous optimization of flight time, power, duty cycle, thrust, specific impulse, and propellant mass. Aspects of the design that are usually only weakly related, e.g., spacecraft operations and system margin management, become strongly coupled for SEP.⁷⁵ The ability to trade between parameters (e.g., power and flight time) without affecting others (i.e., delivered mass) endows SEP missions with a flexibility unattainable by conventional ballistic missions, necessitating an unconventional approach to setting and assessing the coupled technical margins. For example, SEP makes it possible to trade power margin for mass margin, and vice versa. Establishing a standard margin philosophy that is based on rigorous analysis is important so that designers and reviewers can both know when margins are adequate to accommodate growth as the system design matures.

Understanding the linkage between SEP spacecraft margins in the general case requires a sensitivity analysis that examines the influence that power, duty cycle, destination, and thruster performance including missed thrust periods have on overall performance. This involves developing and analyzing a database of trajectories in which perturbations are made of each parameter about a nominal trajectory. The missed thrust analysis determines the effect that incorporating unexpected contingencies and fault response periods into EP trajectories has on dry mass as a function of thrust-out time, power, and duty cycle. An important lesson learned from Dawn is the importance of the missed thrust analysis for operations. In 2008, Oh et al. conducted a study of the sources of uncertainty and

technical risk associated with electric propulsion mission design and showed that they can be summarized into three relatively independent parameters:⁷⁶

- 1) EP Power Margin
- 2) Propellant Margin
- 3) Duty Cycle Margin

Ref. 76 presented recommended system margin ranges for deep-space SEP missions in the early design phase. Since then, further work has clarified the appropriate division of margins, enabling specific numerical recommendations for each area. Table 9 presents a summary. Margins unique to SEP missions – accounting for thrust uncertainty, propellant flow-rate uncertainty, and unexpected thrust outages (missed thrust) due to fault protection-induced safing or other unexpected operational events – are EP power, duty cycle, and EP propellant. Standard margins for the other design parameters apply as usual.

EP power margin is only applied to the power available for SEP (i.e., the difference between array output power [degraded from beginning of life] and power available to the spacecraft). Spacecraft (non-EP) power should hold standard margins. The trajectory should be designed so that the sum of margined spacecraft and SEP power never exceeds array output at any point in the trajectory. The total EP power margin of 15% includes 5% uncertainties in PPU efficiency, thruster performance and control system performance plus another 10% margin to cover missed thrust periods and solar array uncertainty. A duty cycle is also applied to the trajectory to model intermittent coast periods. For example, a duty cycle of 90% (10% duty cycle margin) models a thruster that can operate at full capacity for nine days and is off for one day. The 10% duty cycle margin covers missed thrust, communications and tracking (e.g. sending navigation signals and receiving guidance commands), and uploads and maintenance.

A 10% margin is also applied to EP propellant to cover commissioning and startups, leakage, fill errors, propellant use uncertainties, and residuals.⁷⁶ This margin is only applied to the propellant for the EP system; any chemical propellant should carry standard margins. The 10% applies to a trajectory designed for a fully margined spacecraft mass and power system.

For trajectory design, thruster performance is typically modeled using thrust and propellant flow rate as functions of PPU input power. Uncertainties in delivering the minimum predicted thrust is accounted for in the EP power and propellant margin; thus, the mission design should assume that thruster performance is exact.⁷⁶

Margin to allow accommodation of unexpected thrust outages makes up a significant component of the power, duty cycle, and propellant margins (typically 5% in each). In cases where additional precision is desired, this component of the margin can be replaced by a missed thrust analysis that computes mission performance when thrust is cut out for some required number of days (typically 14 or 28 days) throughout the baseline trajectory. In this case, the additional power, duty cycle, and propellant necessary to accommodate missed thrust periods would be incorporated into the nominal trajectory, and margins to cover these contingencies would be reduced accordingly.⁷⁶

It is noteworthy that the SEP margins are largely interchangeable: reducing one margin (e.g., additional power for SEP) can increase another (e.g., days tolerant to missed thrust). For initial design purposes, we recommend that the SEP mission margin be distributed among power, duty cycle, and propellant as shown in Table 9. This eliminates the need to explicitly model thrust uncertainty, propellant flow rate uncertainty, and unexpected thrust outages.

Neutral mass is defined as dry mass plus any chemical propellant carried by the spacecraft. The standard spacecraft mass margin for high-thrust missions is carried in the neutral mass allocation. The mass margin is held in neutral mass because the SEP and trajectory combination must demonstrate capability to deliver the margined mass to the target. If the margin were instead held in the launch mass, then the trajectory design would use the unmargined (maximum expected) spacecraft mass, which provides the spacecraft with higher acceleration (due to lower mass) than a design that makes full use of launch capability and holds all of the requisite margin in neutral mass. This higher acceleration leads to a lower ΔV with low-thrust trajectories, resulting in an ostensibly higher mass margin and a trajectory incompatible with a fully margined dry or neutral mass. Unlike chemical missions, the launch (wet) mass margin can always be greater than the neutral (dry) mass margin, thus the neutral mass margin is chosen as the more conservative approach.

The margins presented in Table 9 are based on the analysis presented in Ref. 76 and have been updated as follows. The margins in Table 9 specify values rather than ranges. This simplifies both the design and review of SEP missions. The EP power margin in Table 4 is 5% larger than specified in Ref. 76. This value now includes array output uncertainty, and the additional margin allows the use of specific values rather than ranges for duty cycle and propellant uncertainty. As in previous analysis, thruster uncertainty is accounted for in power and propellant margin, and missed-thrust margin is initially held within the power, duty cycle, and propellant margins.

SEP Technical Parameters	Margins	Comments
EP Power	15%	EP power margin should be maintained <i>separately</i> from non-SEP power margin. Includes missed-thrust, PPU & thruster uncertainty, and uncertainty in array output. Does not include beginning versus end of life degradation.
Duty Cycle	10%	Includes missed-thrust, uploads & maintenance, and communications & tracking.
Propellant	10%	Includes missed-thrust, PPU & thruster uncertainty, and residuals. This margin is in addition to the propellant required to deliver the allocated (margined, as opposed to the maximum expected) neutral mass.
Thrust	0%	Thrust uncertainty accounted for by increased power and propellant margins.
Propellant flow rate	0%	Flow rate uncertainty accounted for by increased power and propellant margins.
Missed thrust tolerance	0 days	Accommodation of missed-thrust periods accounted for by increased power, duty-cycle, and propellant margins.
Neutral Mass Allocation	Standard Margins	Mass budget should carry standard margins against this allocation. Trajectory should be designed to deliver allocated (current best estimate + contingency + margin) mass.
Launch Wet Mass	-	Not as relevant as neutral mass margin. Additional launch margin does not necessarily translate to additional neutral mass margin with SEP.
Flight Time	-	Typically not explicitly margined, but time margin could be traded against power and mass margins

Table 9: Recommended System Margins for Preliminary SEP Mission Designs

VI. Conclusions

SEP is no longer the novelty it was when Dawn was proposed and selected. ESA and JAXA have now joined NASA in using SEP for the scientific exploration of deep space. Literally hundreds of commercial SEP units have flown in long-duration applications in space over the intervening decade and a half, and today's telecommunications infrastructure depends on this technology – both for performance and for business success. In this context, the NASA science-mission community faces a future with two distinct sets of needs: (1) strategic technology advancement to “capture” a broad range of hypothetical missions in the coming decades; (2) pragmatic decisions in designing and evaluating specific missions proposed in the next few years.

We offer a user-centric consolidation and comparison of the full range of government and commercial SEP options available in the near term for primary propulsion on deep-space science missions of the class commonly proposed to NASA's Discovery program. Unlike previous papers, we do not emphasize feasibility from a mission analysis perspective. Rather, we emphasize requirements uniquely imposed by competitively reviewed, cost-capped mission proposals, for which system-level flight heritage can trump sheer performance and mission capture. We describe criteria that mission architects and review boards can use to select and evaluate SEP systems; provide descriptions of the viable government and commercial EP system options; describe the modifications needed to adapt commercial EP systems to deep space; discuss appropriate methods for costing commercial-based EP systems; and describe a set of standard system margins appropriate for SEP mission concepts.

The principal conclusion is that the SEP systems best suited for Discovery missions have solid *system* flight heritage that can meet the requirements for deep space with minimal modifications. Commercially developed SEP systems offer significant heritage potential and, in many cases, the required changes introduce comparatively low technical-risk and cost-risk. Because of high production volume, such systems are also likely to be less expensive than “optimized” NASA systems.

Several secondary conclusions are:

1) The choice of SEP system hardware is tightly coupled to the architecture of the power system. In some cases, increasing the use of heritage hardware for the SEP system can decrease the use of heritage hardware in the power system, and vice versa. Mission architects and TMC reviewers are interested in understanding the implications of the SEP system for the whole spacecraft, not just one of these systems. The heritage and input voltage capability of the PPU is a key element in this choice.

2) An off-the-shelf PPU fully qualified to support an unregulated voltage range would greatly benefit deep-space missions because it would maximize both efficiency and heritage of the combined power and EP system. A viable fallback would be straightforward modification of existing PPUs to operate across wider voltage ranges (see section IV). Increasing the PPU’s allowable input voltage across even a small range would benefit deep-space mission applications.

3) Because of the dominating value of flight heritage for competitively selected missions, the NEXT thruster is likely to be acceptable only if the Discovery program itself shoulders the full cost and risk associated with development and delivery of the entire system, including the PPU. This means both offering the entire system as GFE, and exempting it from TMC evaluation. If the program cannot take this path, proposers must absorb the challenge of completing the PPU development within the schedule, risk, and cost framework of their proposed Preliminary Design and Project Completion phase (Phase B – prior to PDR). This is unlikely to result in successful NEXT infusion.

4) An alternative, lower-risk and likely lower-cost infusion path for the NEXT thruster would pair it with the XIPS 25 cm PPU. Some modifications to the XIPS PPU would be required (see section IV), and the thruster performance would be derated.

5) The calculation and evaluation of costs for the adaptation of commercial SEP systems to deep space should be based on the cost actuals incurred in the manufacturing of the commercial systems, plus element-by-element estimates of the costs to modify those systems, rationalized in a Heritage Appendix. Using Dawn’s SEP system as a basis for analogy comparison or model calibration would be inappropriate because Dawn required first-time development far beyond what would be required to adapt today’s state of the art SEP systems to Discovery missions (see section V).

6) Margins appropriate for SEP mission concepts are different from those typical for chemical propulsion missions because of strong interdependencies between mass margin, power margin, and operations strategy for SEP. Section V captures appropriate standards for EP system power, propellant, duty cycle, and mass margins.

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Appendix

The following criteria for definition of “Heritage” were provided in the Discovery 2010 Announcement of Opportunity.¹² These criteria were used to evaluate heritage throughout this paper.

	Full heritage	Partial heritage	No heritage
Design	Identical	Minimal modifications	Major modifications
Manufacture	Identical	Limited update of parts and processes necessary	Many updates of parts or processes necessary
Software	Identical	Identical functionality with limited update of software modules (<50%)	Major modifications (>=50%)
Provider	Identical provider and development team	Different however with substantial involvement of original team	Different and minimal or no involvement of original team
Use	Identical	Same interfaces and similar use within a novel overall context	Significantly different from original
Operating Environment	Identical	Within margins of original	Significantly different from original
Referenced Prior Use	In operation	Built and successfully ground tested	Not yet successfully ground tested

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