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Evaluation of Solar Electric Propulsion Technologies for Discovery-Class Missions

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A detailed study examines the potential benefits that advanced electric propulsion technologies offer to cost-capped missions in NASA's Discovery program. The study looks at potential cost and performance benefits provided by three electric propulsion technologies that are currently in development: NASA's evolutionary xenon thruster, an enhanced NSTAR system, and a low-power Hall effect thruster. These systems are analyzed on three potential Discovery-class missions and their performance is compared with a state-of-the-art system using the NSTAR ion thruster. An electric propulsion subsystem cost model is used to conduct a cost–benefit analysis for each option. The results show that each proposed technology offers a different degree of performance and/or cost benefit for Discovery-class missions. However, lower subsystem costs (particularly, power processing and digital control interface unit costs) are needed for ion thruster systems, to make them more competitive for cost-capped missions. It is observed that the best mass performance generally comes from electric propulsion systems that best use available solar array power during the mission. Finally, first-flight qualification costs are identified as a significant barrier to the implementation of new electric propulsion technologies on cost-capped missions.

I. Introduction

THE NASA Discovery program gives scientists the opportunity to address questions in solar system science with lower-cost, highly focused planetary science missions. Dawn is the ninth Discovery program project and is designed to increase our understanding of the early solar system by investigating two main belt asteroids, Vesta and Ceres, using solar electric propulsion (SEP) for its primary propulsion. When launched in 2007, this program will demonstrate the benefit that SEP can bring to the competitively awarded, cost-capped missions in NASA's Discovery program [1]. Dawn's primary propulsion system is based on the NSTAR thruster, a 30-cm-diam ion thruster that was flight-demonstrated on Deep Space 1 (DS1). This paper examines the benefits that several next generation EP technologies could have for Discovery-class missions and is the first comprehensive study of electric propulsion systems to combine detailed cost and performance models, to look at the cost benefit that different systems provide for deep space missions. The new technologies considered in this study are all currently proposed or in development through NASA's In-Space Propulsion (ISP) program. They are 1) NSTAR, a 30-cm ion thruster subsystem, previously flown on Deep Space 1, representing the current state of the art (SOA) in electric propulsion for deep space missions [2]; 2) NASA's Evolutionary Xenon Thruster (NEXT), a 6-kW ion thruster subsystem that is currently under development by a joint government–industry–academia team [3,4]; 3) enhanced NSTAR, a proposed improvement to the existing NSTAR subsystem using a combination of carbon-based ion optics (CBIO) that are currently under development and a high-power processing unit (PPU) that was developed through the NEXT program [5]; and 4) low-power Hall, a proposed Hall thruster subsystem that operates efficiently at low power with enhanced throughput capability that is currently under development [6].

Discovery missions are selected competitively and cover a wide range of scientific goals and destinations. For this study, three reference missions are selected and evaluated using each SEP technology. Although the destinations are generic, they are similar to current and proposed Discovery-class missions that use electric propulsion [1,7–12]. The primary focus of this study is on the improvements that advanced technologies bring compared with a state-of-the-art NSTAR SEP system. The reference missions selected for this study are the 1) Vesta–Ceres rendezvous mission (based on Dawn), 2) near-Earth asteroid sample return mission, and 3) comet rendezvous mission.

Because cost is a major driver for competed missions, a detailed electric propulsion subsystem cost model is used to conduct a cost–benefit analysis for each option. The model generates a full cost that includes component costs, labor and integration, subsystem engineering, fabrication, assembly, testing, ground support, and launch operations. Development, qualification, and recurring costs are all considered in the study.

The technology/mission matrix in Table 1 shows the different propulsion technologies/mission scenarios considered in this study.

For each scenario, a launch date is chosen and a low-thrust trajectory optimizer is used to calculate the flight time, xenon propellant consumed, and the total mass delivered to the final destination. The size of the propulsion system is determined from the xenon throughput and a *net mass delivered* is calculated by subtracting the mass of the propulsion system from the total delivered mass. The electric propulsion cost model is used to calculate the total cost of the propulsion system. The cost and mass results are compared, to show the relative cost/benefits for each technology–mission combination, and an application matrix is generated showing valid mission options. Finally, the technology readiness level (TRL) and development costs are considered, to reach several broad conclusions on the relative benefit of each technology for Discovery missions.

The thruster performance, mass, and cost models used in this study are described in Sec. II. Trajectory and mission analysis assumptions and results are described in Sec. III, and overall conclusions are described in Sec. IV.

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Table 1 Study mission/technology matrix

	Near-Earth asteroid	Comet rendezvous	Dawn (w/gravity assist)
NSTAR, one thruster	×		×
NSTAR, two thrusters	×	×	
NEXT	×	×	×
Enhanced NSTAR w/NEXT components	×	×	×
Low-power Hall thruster	×	×	×

II. Methodology and Assumptions

A. Thruster Performance Model

For each propulsion technology, a performance model was created and used to calculate thrust and propellant mass flow rate as a function of input power to the PPU. The model is derived from throttle tables provided by technologists and the source for the throttle table varies, depending on the maturity of the technology. For the NSTAR thruster, the throttle table (titled Q-mod, or modification to Throttle Table Q) is based on Deep Space 1 flight data and is the same throttle table used for mission planning in the design phases of the Dawn program [13]. For NEXT, two throttle tables, a high-thrust option and a high-specific-impulse option, are derived from laboratory data generated using an engineering model thruster [4]. For enhanced NSTAR, the throttle table is derived from laboratory data taken using carbon-based optics with a laboratory thruster-cathode model [5]. The throttle table is a high-thrust table, maximizing the thrust-to-power ratio at each throttle point. The low-power Hall throttle table is derived from theory and not based on laboratory data. Polynomials are fitted to each table to generate expressions for thrust and mass flow as a function of power. Table 2 shows the resulting polynomial coefficients. The overall system efficiency and specific impulse I_{sp} can be derived from these expressions and is shown in Figs. 1 and 2. Table 3 shows the allowable operating range and xenon throughput capability assumed for each option.

NSTAR has demonstrated the flight throughput shown in Table 3, with 50% margin in an extended-life test [14]. The other proposed technologies have not demonstrated these capabilities in life tests, so the analytic capabilities are assumptions based on analytical forecasts. The enhanced NSTAR throughput assumes the use of carbon-based ion optics to increase grid life [5], and the low-power Hall thruster throughput assumes use of an innovative actuated ceramic discharge channel to increase the operational lifetime [15].

The three ion thruster options have similar efficiency curves at low power, but very different operating ranges. The NEXT thruster has the widest power range, and at high power, it operates at the highest efficiency of any of the devices. The high-thrust and high-I_{sp}-throttle curves for NEXT have the same operating range. Both NEXT throttle curves are evaluated for each mission, but only the best performing option is reported in this paper. The enhanced NSTAR operates at input powers as low as 450 W, but with relatively low efficiency. The Hall thruster operates very efficiently at low power, but does not operate as efficiently as ion thrusters at higher power. This behavior is typical for Hall thrusters and will be shown to significantly benefit power-limited missions.

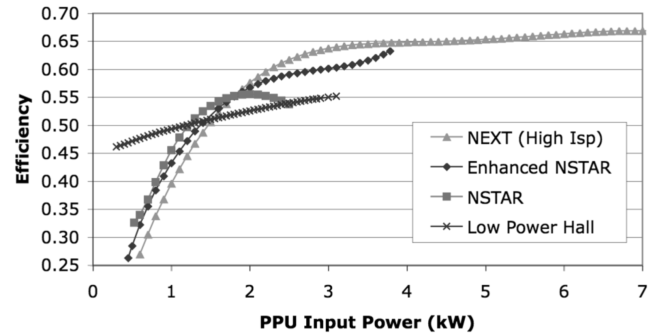

Fig. 1 Propulsion subsystem efficiency vs PPU input power.

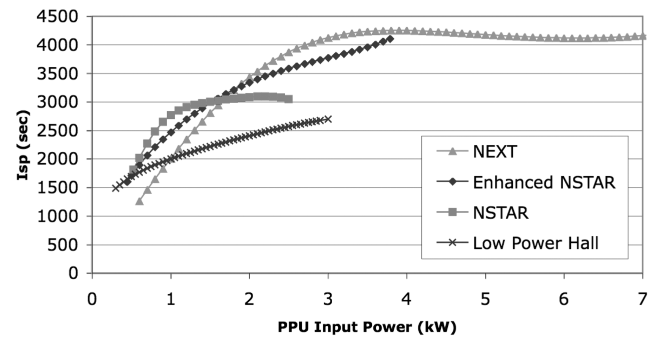

Fig. 2 Specific impulse vs PPU input power.

B. Mass Assumptions

Hardware mass is technology-dependent and is a function of xenon throughput, the number of thrusters, and the number of PPUs

Table 3 Thruster characteristics summary

	TRL	Minimum power	Maximum power	Throughput capability assumption
NSTAR	9	525 W	2.6 kW	150 kg
NEXT	5	620 W	7.3 kW	>400 kg
Enhanced NSTAR	4	450 W	3.8 kW	>300 kg
Low-power Hall thruster	3	330 W	3.0 kW	>300 kg

Table 2 Throttle curve coefficients

System	Table name	A	B	C	D	E
Mass flow coefficients, $\dot{m}$[mg/s]= $A(P[\mathrm{kW}])^4 + BP^3 + CP^2 + DP + E$						
NSTAR	Q-Mod	0.36985	−2.5372	6.2539	−5.3568	2.5060
NEXT	Table 9A high thrust	0.0153	−0.2832	1.660	−2.621	3.131
NEXT	Table 9A high I_{sp}	−0.00602	0.07637	-0.1898	0.1768	1.944
Enhanced NSTAR	Table 1d.1	−0.01647	0.15774	−0.5941	1.5665	0.3594
Low-power Hall	Table 1	0.00000	0.07988	−0.61355	2.4888	0.6131
Thrust coefficients, $T[\mathrm{mN}] = A(P[\mathrm{kW}])^4 + BP^3 + CP^2 + DP + E$						
NSTAR	Q-Mod	5.145602	−36.720293	90.486509	−51.694393	26.337459
NEXT	Table 9A high thrust	0.1102	−2.234	12.93	11.88	16.8
NEXT	Table 9A high I_{sp}	−0.1889	2.926	−14.02	52.32	−0.08954
Enhanced NSTAR	Table 1d.1	0.2340	−1.5554	0.8267	38.4804	−2.2822
Low-power Hall	Table 1	0.0000	0.73434	−5.9528	51.1473	4.0924

Table 4 Electric propulsion unit mass assumptions

Thruster	NSTAR baseline	Enhanced NSTAR	NEXT	Low-power Hall
Inputs				
Number of engines	**3**	**2**	**2**	**2**
Number of PPUs	**3**	**2**	**2**	**2**
Number of DCIUs	**2**	**2**	**2**	**0**
Total system xenon throughput, kg	**200**	**200**	**200**	**300**
Xenon contingency				
Navigation and trajectory errors	5%	5%	5%	5%
Residuals	5.0%	5.0%	3.6%	5.0%
Assumptions				
Mass/thruster, kg	8.2	7.39	12.4	3.6
Mass/PPU, kg	13.9	17.6	26	8.4
Mass/DCIU, kg	5.7	5.7	5.7	0.0
Mass/gimbal, kg	4.6	4.6	5.0	4.6
Gimbal drive electronics, kg	Included	Included	Included	2.0
Feed system, kg				
Fixed mass	8.1	8.1	2.2	4.0
Additional mass/engine	3.3	3.3	4.1	1.0
Xenon tank mass fraction	4.5%	4.5%	4.5%	4.5%
System contingency	10%	18%	30%	30%
Calculations				
Thrusters, kg	24.6	14.8	24.8	7.2
PPUs, kg	41.7	35.2	52.0	16.8
DCIUs, kg	11.3	11.3	11.3	0.0
Xenon feed system, kg	18.1	14.7	10.4	6.0
Xenon tanks, kg	9.9	9.9	9.8	14.9
Gimbals, kg	13.9	9.3	10.0	11.3
Subsystem dry mass	120	95	118	56
Xenon residuals, kg	20.0	20.0	17.2	30.0
Hardware mass contingency, kg	12.0	17.1	35.5	16.8
Propulsion system mass (w/contingency), kg	151	132	171	103
Mass delta	Baseline	−13%	+13%	−32%

required for each mission. A subsystem mass model is used to calculate propulsion dry mass based on mission requirements. The unit level mass estimates for NSTAR are based on Dawn actuals [2], whereas the unit level mass estimates for the advanced technologies were provided directly by technologists. A contingency margin is added, with a mass margin of 10% for inherited hardware and 30% for new designs. The result is an overall subsystem margin that varies from 10% for NSTAR to 30% for NEXT and low-power Hall. The xenon tank mass is calculated as a fixed fraction of xenon propellant mass, to allow direct comparison of propulsion hardware. Five percent fuel mass margin is added to account for navigation and trajectory errors. Additional fuel is added to account for residuals in the tank and feed system.

The subsystem mass breakdown is shown in Table 4 (masses shown are the current best estimates, as of September 2004). The inputs selected for this table (shown in bold) are generic, but are typical of trends observed in the detailed analysis. The advanced thrusters have greater throughput capability than NSTAR and typically require fewer thrusters to meet mission requirements. The NEXT feed system is unique and is modeled as having somewhat smaller system residuals. The ion systems use redundant digital control and interface units (DCIUs) to provide command and telemetry interfaces, control the gimbals, and actively regulate xenon flow rates. The Hall system requires no DCIU and incorporates flow control functions into the PPU and gimbal control into the spacecraft command system. This distribution of functions is typical for commercial Hall systems and improves the system's mass and cost. The difference in control architecture is somewhat arbitrary, but partially reflects differences in the operation of Hall and ion thrusters. With ion thrusters, the flow controller regulates separate discharge and neutralizer cathodes and typically closes the loop around temperature and pressure sensors associated with the feed system. This requires the use of redundant pressure sensors and makes the control logic relatively complex. With current flight Hall thrusters, the flow controller typically regulates the flow to a single cathode/anode pair (a flow splitter divides the flow between the two devices) and closes the cathode/anode flow control loop around the discharge current, making it relatively easy to incorporate the control logic into the PPU. The gimbal control electronics increase the mass and cost of the spacecraft command distribution system, a penalty that is included in this analysis. It may be possible to improve the mass and cost of ion systems by incorporating their flow control functions into the PPU.

C. Costing Methodology

Cost is a major consideration in the design of Discovery-class missions and is a major driver in the decision whether to use electric propulsion (EP) on a given mission. A detailed EP subsystem cost model is used to estimate the relative cost of each option. The model is based on a combination of flight mission actual costs, flight mission estimates, and cost estimates provided by technologists. Four types of cost estimates were generated for this study: 1) development cost [cost to develop the subsystem from its current development status (as of August 2004) to TRL 6, in which the subsystem has been demonstrated in relevant environments]; 2) qualification cost (cost to develop the subsystem from TRL 6 to a fully flight qualified design); 3) nonrecurring cost (cost of postqualification engineering, data, and drawings for the first-flight units); and 4) recurring cost (cost of first/second/third/fourth flight units).

Development and qualification costs were estimated at a subsystem level, whereas nonrecurring and recurring costs were estimated at a unit level. The unit costs serve as inputs to a cost model that is derived from actual costs incurred and projected on the Deep Space 1 and Dawn missions. This model is commonly used for the costing of space missions in the Team X concurrent design environment [16]. The model accounts for component costs, spare

Table 5 Recurring cost assumptions ($FY05 K)

Thruster	NSTAR baseline	Enhanced NSTAR	NEXT	Low-power Hall
Inputs				
Xenon, kg	**300**	**300**	**300**	**400**
Number of engines	**3**	**2**	**2**	**2**
Number of PPUs	**2**	**2**	**2**	**2**
Number of DCIUs	**2**	**2**	**2**	**0**
Component cost assumptions				
Recurring cost/thruster	580	580	800	250
Recurring cost/PPU	1520	1900	1976	800
Recurring cost/DCIU	1310	1310	1310	0
Recurring cost/gimbal	325	325	325	325
Recurring cost/xenon tank	450	450	450	450
Fixed cost/feed system	1610	1610	1610	1610
Per thruster cost/feed system	200	200	200	200
Nonrecurring/thruster	130	250	250	50
Nonrecurring/PPU	200	200	200	200
Nonrecurring/DCIU	0	0	0	0
Nonrecurring/xenon tank	0	0	0	0
Nonrecurring/gimbal	510	510	510	1010
Design and qual. new xenon tank	0	0	0	0

Table 6 Technology development cost assumptions ($FY05 K), single string only

Thruster	SOA NSTAR	Enhanced NSTAR	NEXT	Low-power Hall
Component cost assumptions				
Development (to TRL 6)/thruster	0	3000	7050	7000
Qualification/thruster	0	1200	2200	2200
Development (to TRL 6)/PPU	0	3800	2280	3500
Qualification/PPU	0	1500	1600	1500
Development (to TRL 6)/DCIU	0	0	1000	0
Qualification/DCIU	0	0	400	0
Development (to TRL 6)/gimbal	0	0	400	400
Qualification/gimbal	0	0	800	800
Development (to TRL 6)/PMS	0	0	660	2600
Qualification/PMS	0	0	1000	1000
System integration tests (1)	0	500	500	500
Development cost (to TRL 6)	0	7300	11,890	14,000
Qualification cost (after TRL 6)	0	2700	6000	5500
Total EP subsystem cost	0	10,000	17,890	19,500

parts, labor and integration, subsystem engineering, design and analysis, engineering procurement support, fabrication assembly and test, ground support and launch operations, xenon tanks and gas, and harness manufacturing. The output from the model is combined with the development and qualification cost estimates to provide three outputs: 1) development cost (to develop the subsystem from its current development status to TRL 6); 2) first-flight cost (the sum of subsystem qualification, nonrecurring, and recurring costs); and 3) nth flight cost (the sum of subsystem nonrecurring and recurring costs).

The unit recurring and nonrecurring cost estimates used in this study are shown in Table 5. The NSTAR values are derived from, but not the same as, Deep Space 1 and Dawn actual costs, as of May 2004. They represent the projected cost of these units for the next flight mission build. The NEXT, enhanced NSTAR, and Hall values are estimates provided by technologists, based on vendor data when available. The inputs selected for this table (shown in bold) are generic, but are typical of trends observed in the detailed analysis.

The costs shown in Table 5 comprise a substantial fraction of the total cost of a flight electric propulsion system. Several general observations can be made based on these values. In general, units associated with the Hall system cost considerably less than equivalent units associated with the ion systems. Some of this cost difference may be due to the relative lack of maturity of the Hall estimates, which carry greater uncertainty, because the design is conceptual and the hardware is at a relatively low TRL (see Table 3). It should be noted that the Hall units costs reported in Table 5 are consistent with the costs of existing commercial Hall systems. Because of the uncertainty in the cost estimates, a cost sensitivity analysis was conducted for the Hall system on the near-Earth asteroid mission and is shown in Sec. III.

PPU costs vary widely and are a key differentiator when comparing EP systems. For purposes of this analysis, we are assuming that the recurring cost of the next NSTAR flight PPU will be about half of the cost of the Dawn units, but there is considerable uncertainty in this value. The DCIU costs over $1.3 million per flight unit and also has a significant impact on overall subsystem cost. This is one reason that the proposed Hall systems are generally less expensive than the ion options. As discussed in the previous section, the Hall system has no DCIU. The cost to incorporate DCIU functions into other units is incorporated into the Hall PPU and the gimbal nonrecurring cost. The recurring cost of the feed system is difficult to estimate, because it includes component costs incurred at the subsystem level and assembly/integration costs incurred at the spacecraft level. For this analysis, the cost of a fully integrated NSTAR feed system is used in every case. This assumption is conservative, because the NEXT and Hall feed systems incorporate design features that are expected to make them less expensive than the NSTAR feed system. Thruster costs also vary widely, but because the cost of the thrusters is relatively small compared with other units, the nth flight cost is largely driven by PPU and DCIU costs. Based on this observation, we draw the following conclusion: the PPU and DCIU unit costs are major cost drivers for electric propulsion systems.

Table 7 Near-Earth asteroid sample return mission characteristics

Near-Earth asteroid sample return	
Target body	Nereus
Launch vehicle	Delta 2925
Power system	6 kW solar array at 1 AU
Bus power	300 W
Duration	3.3 yr
ΔV	4.5 to 6.5 km/s
Launch date	2007/08
Ion/Hall thruster duty cycle	90%
Launch and rendezvous dates	Selected by optimizer
Optimization tool	SEPTOP

Feed system costs may also be a major cost driver, but are not evaluated in this study. The development cost estimated for each subsystem is shown in Table 6. These costs are for single-string systems and do not include the cost of multithruster testing. The development costs were provided by technologists and contain considerable schedule and cost uncertainty. They are reported in fiscal year 2005 dollars ($FY05) and represent the cost to complete development, as of September 2004. Previously expended funds are not reported. These estimates are of varying fidelity, have not been independently reviewed or compared with historical data, and should be treated as rough order-of-magnitude values. It is assumed that qualification costs are to be paid by the first-flight program, whereas development costs are paid by the NASA In-Space Propulsion program. This is consistent with the current financial structure of these programs.

This study focuses on the cost of the electric propulsion system. In most cases, the missions compared have the same flight time and solar array power, and so changing the propulsion system has relatively little impact on the overall cost of the mission. There are also costs associated with the accommodation of multiple thrusters into the spacecraft configuration. These costs can vary greatly, depending on the specific characteristics of the spacecraft (i.e., its size, the mounting location of the thrusters, and their relative proximity to the solar arrays and other appendages). Because these costs are spacecraft-specific and difficult to estimate, they are not included in our analysis.

III. System Analysis Results

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. The destinations are generic, but are similar to current and proposed Discovery-class missions that use electric propulsion. This section discusses the three concepts and provides analysis results for each mission. General implications of the results are discussed, and some mission-specific findings are provided. Additional general findings are given in the final section of this paper.

A. Near-Earth Asteroid Mission

The first mission considered is a near-Earth asteroid sample return mission. The spacecraft launches directly to an Earth escape trajectory and uses SEP to rendezvous with the asteroid Nereus. It 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 7.

A separate optimized trajectory is generated for each scenario using the SEPTOP low-thrust optimization tool [17]. All trajectories assume a nominal array power of 6 kW at 1 AU (astronomical unit) 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 the distance from the sun and is modeled using a high-efficiency gallium arsenide array model. The spacecraft is launched directly to an Earth escape trajectory with a net positive velocity at infinity, or positive C_3. C_3 is defined as the square of the spacecraft's theoretical orbital velocity at infinity. The entry velocity at Earth return is not constrained and is optimized for maximum total delivered mass. As shown in Table 8, the entry velocity varies from 13.6 to 14.9 km/s. By comparison, the entry velocity for the Stardust mission will be approximately 12.6 km/s. Higher entry velocities require a heavier and more expensive thermal protection system (TPS). Variations in the mass

Table 8 Summary of results of the near-Earth asteroid mission

	Single NSTAR	Single NSTAR enhanced	Single low-power Hall	Single NEXT (high I_{sp})	Dual NSTAR	Dual NSTAR enhanced	Dual Hall
Mission							
Launch vehicle	Delta 2925	Delta 2925	Delta 2925	Delta 2925	Delta 2925	Delta 2925	Delta 2925
Trip time, yr	3.25	3.20	3.15	3.18	3.16	3.18	3.14
C3, km^2/s^2	11.8	7.1	4.2	1.8	2.2	1.9	1.8
Power, 1 AU, BOL	Baseline 6 kW	Same	Same	Same	Same	Same	Same
Net payload mass delivered, kg	542	674	712	772	746	766	793
Launch mass, kg	1010	1115	1183	1245	1235	1243	1245
Xenon propellant throughput, kg	194	174	238	167	198	175	203
Net trajectory mass capability, kg	677	801	805	938	897	929	903
Propulsion subsystem mass, kg	135	127	93	166	151	163	110
Free space delta-V, km/s	6.5	6.3	5.8	5.6	5.3	5.5	4.6
Earth entry velocity, km/s, approx.	14.9	14.9	14.3	14.2	14.0	14.1	13.6
Configuration							
Simultaneous operating thrusters	1	1	1	1	2	2	2
Total thrusters (primary and redundant)	3	2	2	2	3	3	3
Total PPUs (primary and redundant)	2	2	2	2	3	3	3
Nth mission cost, $K	Baseline	−2283	−5277	−1770	+1529	−297	−3631
First mission cost, $K	Baseline	+417	+223	+4230	+1529	+2403	+1869
Propulsion subsystem costs							
Recurring cost, $K	Baseline	−2283	−5277	−1770	+1529	−297	−3631
Qualification cost, $K	Baseline	+2700	+5500	+6000	+0	+2700	+5500
Development							
Maturity (TRL)	9	4	3	5	6	4	3
Development schedule (TRL 6, no life test)	None	~1.5 yr	~4 yr	~1.5 yr	None	~1.5 yr	>4 yr
Development schedule (w/life test)	None	~3 yr	>5.5 yr	>2.5 yr	None	~3 yr	>5.5 yr
Development cost, $M	None	9.5	19	21	Small	9.5	19

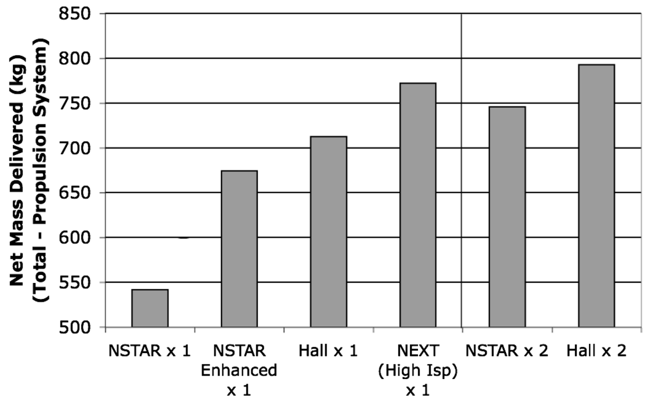

Fig. 3 Net mass delivered for a near-Earth asteroid mission.

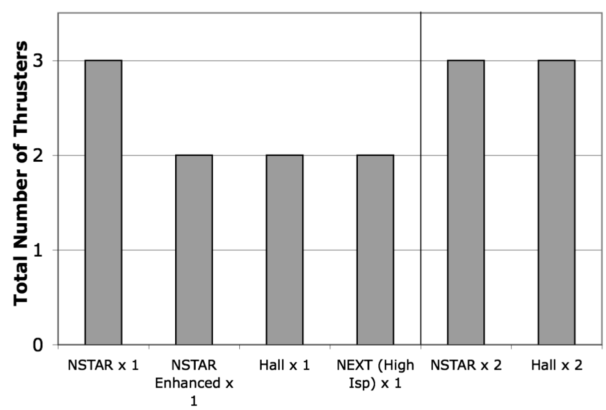

Fig. 4 Total thrusters required for a near-Earth asteroid mission.

and cost of the TPS are not accounted for in this analysis. The overall results are summarized in Table 8, and net delivered mass is shown in Fig. 3. Net delivered mass is defined as the total end-of-mission delivered mass minus the mass of the electric propulsion system. It includes the payload, solar arrays, Earth return vehicle, and main spacecraft bus. Both single and multithruster operation are considered. All options carry an extra thruster and power processing unit for redundancy. In some cases, an extra thruster is also required to meet xenon throughput requirements.

The single and dual NSTAR options both represent state-of-the-art systems. Single NSTAR operation has been flight-demonstrated on DS1 and is the baseline for Dawn. Simultaneous operation of multiple thrusters has been flight-demonstrated on commercial missions, but has not been demonstrated with the NSTAR thruster. Systems allowing multithruster operation are more complex and therefore have higher cost and propulsion system mass than single thruster equivalents. Figure 3 shows that the advanced EP technologies generally offer a considerable mass performance

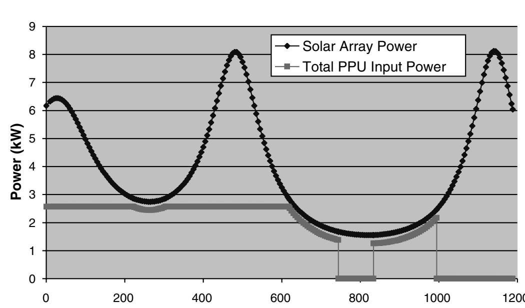

Fig. 5 NEAR power vs time for single NSTAR thruster; delivered mass is 540 kg.

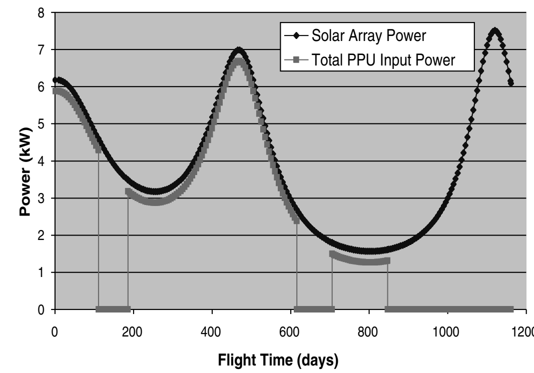

Fig. 6 NEAR power vs time for single NEXT thruster; delivered mass is 770 kg.

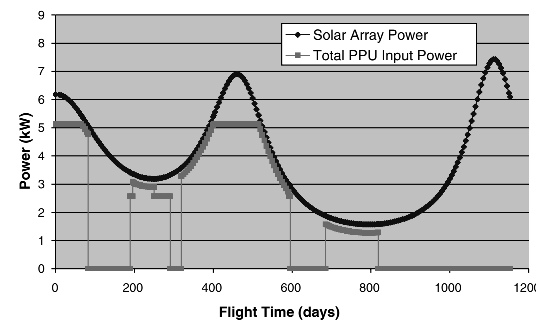

Fig. 7 NEAR power vs time for dual NSTAR thruster; delivered mass is 740 kg.

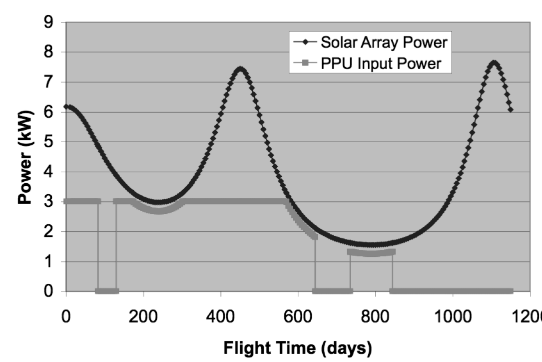

Fig. 8 NEAR power vs time for single low-power Hall thruster; delivered mass is 712 kg.

advantage over single NSTAR, but do not necessarily offer an advantage over dual NSTAR. This result is explained in the following discussion of the power vs time histories. Figure 4 shows the total number of thrusters required for each option. All options carry an extra engine for redundancy. The single NSTAR

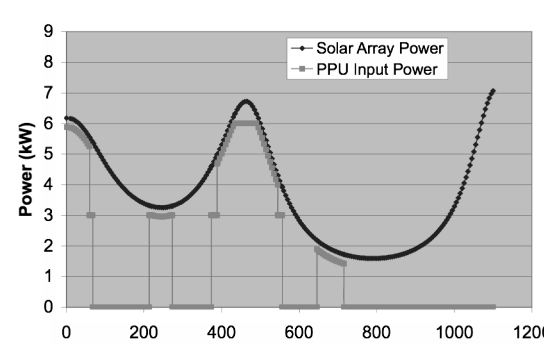

Fig. 9 NEAR power vs time for dual low-power Hall thruster; delivered mass is 793 kg.

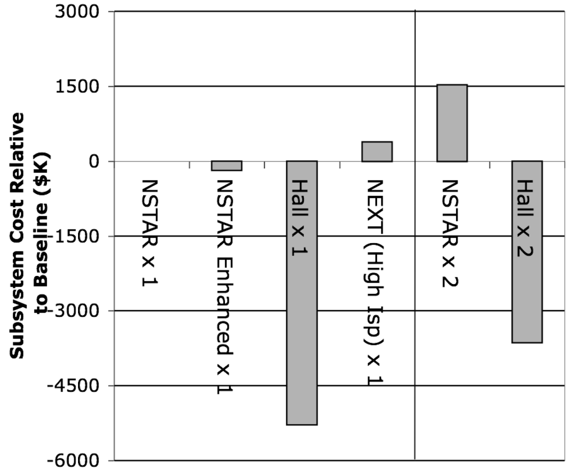

Fig. 10 Nth flight cost comparison for a near-Earth asteroid ($FY05).

configuration requires the same number of thrusters as the dual NSTAR configuration, because the mission's xenon throughput requirements exceed the capability of a single NSTAR thruster. The second thruster provides additional throughput, and the third thruster provides redundancy. All of the proposed technologies require fewer thrusters in their single active-thruster configuration.

Figure 5 shows the power vs time history for both the array and the PPU for the single NSTAR option. The power generated by the array varies with the distance from the sun, starting at 6 kW, dropping as low as 1.4 kW, and peaking near 8 kW at the middle and end of the mission. The PPU's input power is limited by NSTAR's power-handling capacity, and a significant fraction of the array's capability is unused for much of the mission. Figure 6 shows the power vs time history for the NEXT option. NEXT has a much higher peak input power, better uses the array near perihelion, and generates more thrust than NSTAR near perihelion. The higher thrust produces lower burn durations and more efficient orbital maneuvers. This, in turn, results in a lower C_3, higher separated mass, and less onboard ΔV than the single NSTAR option. The dual NSTAR option, shown in Fig. 7, also uses the array more efficiently than single NSTAR and therefore delivers more mass. Enhanced NSTAR uses the array more efficiently than single NSTAR, but less efficiently than dual NSTAR, and its mass performance lies between these two cases. Based on these results, the following conclusion is reached. In general, the best mass performance comes from the EP systems that best use available solar array power.

This generalization that can be used by EP system designers as a guideline to optimize mass performance on SEP missions. However, the system's efficiency as a function of power and the choice of specific impulse will also impact delivered mass capability. The influence of these factors is illustrated in Figs. 8 and 9. The single Hall system has roughly the same peak power as enhanced NSTAR, but delivers somewhat more mass. This occurs because the Hall system operates at lower I_{sp} and generates higher thrust at aphelion and perihelion, lowering the total ΔV required for the mission. The net result is that the Hall system operates closer to this mission's mass optimum I_{sp}. The dual Hall system, shown in Fig. 9, delivers somewhat more mass than NEXT, for the same reasons.

Figure 10 shows nth flight costs calculated for the near-Earth asteroid sample return mission. Although all systems offer a significant mass performance advantage over single NSTAR, only the Hall system simultaneously offers a considerable cost advantage. The cost savings is quite significant for a mission of this size, equivalent to $\sim$25% of the total cost of the subsystem. All of the systems offer considerable cost advantages over dual NSTAR and most require fewer thrusters to achieve similar mass performance. Comparing Figs. 3 and 10, we reach the following conclusions concerning the relative cost and benefits of each option.

1) All proposed systems offer a considerable mass advantage over a single NSTAR system.

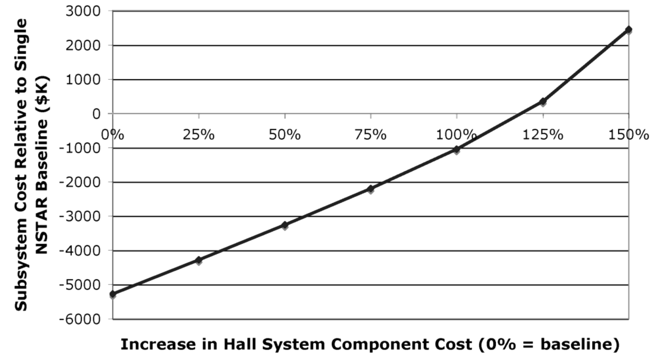

Fig. 11 Sensitivity of single Hall system cost to increases in Hall component costs (thrusters/gimbals/PPU).

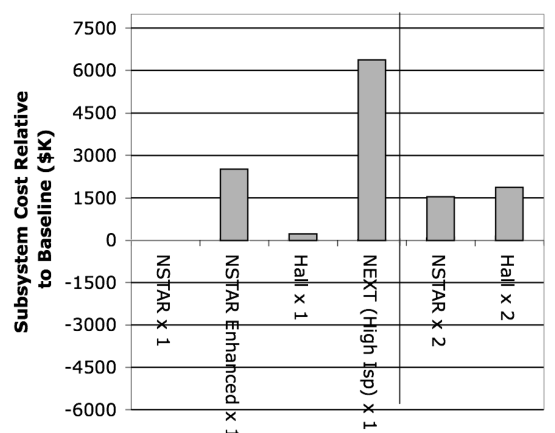

Fig. 12 First-flight cost estimate for a near-Earth asteroid mission.

2) Single low-power Hall offers both significant cost and mass advantages over single NSTAR. It also offers superior mass performance over dual NSTAR, at much lower cost.

3) Dual low-power Hall offers the highest mass performance of any option and still costs considerably less than a single NSTAR system.

4) Enhanced NSTAR offers a significant mass advantage over single NSTAR and may offer a slight cost advantage.

5) NEXT offers superior cost and mass performance to dual NSTAR. It has superior mass performance to single NSTAR, but at somewhat higher cost.

As mentioned previously, the cost estimates provided for the Hall system components carry greater uncertainty than the cost estimates for the other systems, because the design is conceptual and the hardware is at a relatively low TRL (see Table 3). An analysis was conducted to determine the sensitivity of the Hall system cost savings to increases in the cost estimates for the Hall system components. Figure 11 shows that the cost of a single low-power Hall system configured for the near-Earth asteroid mission increases as a function of increases in the cost of the Hall thruster, gimbal, and PPU. A 50% cost increase reduces the cost saving from \$5 to \$3 million, compared with NSTAR. A 100% increase results in only a \$1 million cost savings.

Figure 12 shows first-flight costs for this mission. This includes the cost of qualifying the first set of flight hardware after it has been developed to TRL 6.

The qualification costs considerably increase the cost of the first-flight system. Although the newer systems still offer a considerable mass advantage over single NSTAR, from the flight project's point of view, it is often less expensive to use a dual NSTAR system than it is to adopt a newer technology. The only exception to this is the single low-power Hall system, which still offers comparable performance for less cost than dual NSTAR, even including qualification costs. From these results, it is concluded that qualification costs are a significant barrier to first flight of new SEP technologies on Discovery missions.

Table 9 Vesta–Ceres rendezvous mission characteristics

Vesta–Ceres rendezvous	
Target bodies	Vesta and Ceres
Launch vehicle	Delta 2925H
Power system	8.67 kW solar array at 1 AU
Bus power	540 W
Duration	9 yr
ΔV	10.2 to 11.6 km/s
Launch date	7 July 2006
Ion/Hall thruster duty cycle	90%
Launch and rendezvous dates	Fixed
Optimization tool	Mystic
Gravity assist	Mars gravity assist
Redundancy	Spare thruster required at Vesta. No spare required for Ceres.

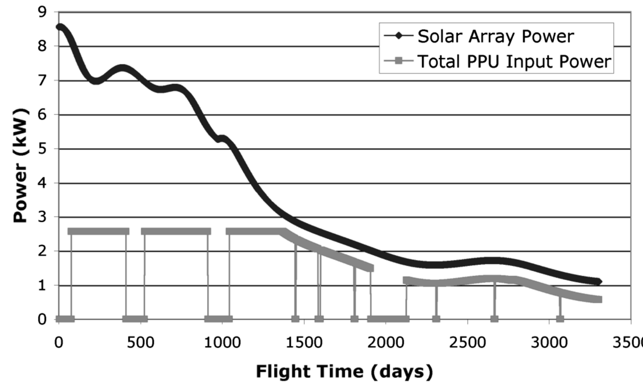

Fig. 13 NSTAR power profile for a Vesta–Ceres rendezvous mission.

Once the first system is flown, future SEP flights will see the full cost benefit from the use of new technology. However, the first project has little incentive to adopt a technology that primarily benefits future missions. This problem could be partially addressed by incorporating additional qualification testing into the technology development programs, incorporating full-length life tests into the development programs, or by procuring flight qualification units as part of the development. It could also be addressed programmatically though cost-sharing arrangements that fund qualification efforts outside of the traditional mission cost-cap structure.

B. Vesta–Ceres Rendezvous Mission

The Vesta–Ceres rendezvous mission is a main belt asteroid science mission that is modeled closely on Dawn. The spacecraft launches directly to an Earth escape trajectory and uses a combination of SEP and a Mars gravity assist to rendezvous with the asteroid Vesta. SEP is used to maneuver in the asteroid's vicinity for 200 days before continuing on to a second rendezvous with the asteroid Ceres. The basic characteristics of this mission are shown in Table 9.

This mission is based on Dawn and uses the same launch vehicle, solar array, launch date, and arrival date. The solar array is modeled using a triple junction InGaP/InGaAs/Ge array model. The power shown in Table 9 is usable power from the array and does not include margins and degradation factors that are included in the flight-power budget. Consistent with Dawn, no thruster redundancy is required after the first rendezvous. A separate optimized trajectory is generated for each scenario using the low-thrust optimization tool Mystic [18]. Figure 13 shows the power vs time profile operating a single NSTAR thruster at a time (the Dawn mission baseline). Three NSTAR thrusters are fired in series, to accommodate the 450 kg of xenon that is required to complete the mission. One challenging aspect of this mission is the need to maneuver for extended periods far away from the sun. Although significant excess power is available at the beginning of the mission, only 570 W is available to the PPU at the end of the mission, which is only 50 W greater than the thruster's minimum operating power. The overall analysis results are summarized in Table 10, with net mass performance shown in Fig. 14, and the total number of thrusters required shown in Fig. 15.

Figure 15 shows the primary advantages that the proposed EP devices have over NSTAR: significantly greater throughput capability. All of the proposed devices require fewer thrusters to complete the mission, compared with the SOA NSTAR system. Because the same conclusion was reached for the near-Earth asteroid sample return mission, this leads to the following finding: Discovery-

Table 10 Vesta–Ceres rendezvous summary performance matrix

	Single NSTAR	Single NSTAR enhanced	NEXT (high thrust)	Low-power Hall
Mission				
Launch vehicle	Delta 2925H	Delta 2925H	Delta 2925H	Delta 2925H
Trip time, yr	9.00	9.00	9.00	9.00
C3, km^2/s^2	7.3	6.8	0.0	0.0
Power, 1 AU, BOL, kW	8.67	8.67	8.67	8.67
Payload mass performance, kg	702	726	738	800
Launch mass, kg	1294	1307	1509	1509
Xenon propellant throughput, kg	385	395	527	537
Net trajectory mass capability, kg	882	888	958	942
Propulsion subsystem mass, kg	180	162	220	142
Free space delta-V, km/s	10.55	NA	10.70	10.59
Configuration				
Simultaneous operating thrusters	1	1	1	1
Total thrusters (primary and redundant)	3	2	2	2
Total PPUs (primary and redundant)	2	2	2	2
Nth mission cost, \$K	Baseline	−136	+653	−5100
First mission cost, \$K	Baseline	+2564	+6653	+400
Propulsion subsystem costs				
Recurring cost, \$K	Baseline	−136	+653	−5100
Qualification cost, \$K	Baseline	+2700	+6000	+5500
System costs (power)	Baseline	Baseline	Baseline	Baseline
Development				
Maturity (TRL)	9	4	5	3
Development schedule (TRL 6, no life test)	None	∼1.5 yr	∼1.5 yr	∼4 yr
Develoment schedule (w/life test)	None	∼3 yr	>2.5 yr	>5.5 yr
Development cost, \$M	None	10	17.9	19.5

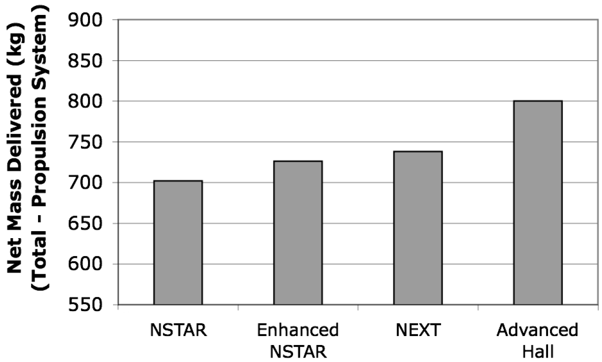

Fig. 14 Delivered mass for a Vesta–Ceres rendezvous mission.

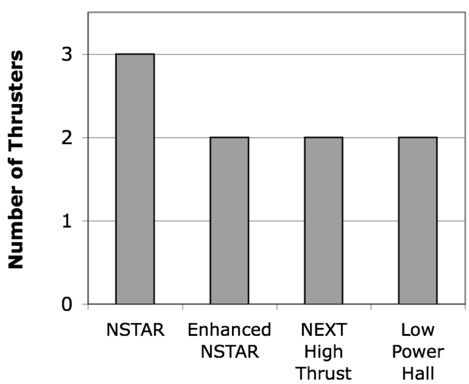

Fig. 15 Number of thrusters required for a Vesta–Ceres rendezvous mission.

class missions will generally benefit from the increased xenon throughput capability offered by the proposed EP technologies.

Figure 14 shows that enhanced NSTAR and NEXT deliver relatively small mass advantages over NSTAR, whereas low-power Hall delivers almost 100 kg of extra payload. The Hall system performs well, because it operates at relatively high efficiency at low power (see Fig. 1), giving it a significant performance advantage for the last third of the mission. Enhanced NSTAR and NEXT better use the array at the beginning of the mission, but gain relatively little mass advantage, because a Mars gravity assist is used in combination with fixed launch and arrival dates. Most of the extra array power is only available on the initial Earth–Mars leg, so the extra thrust capability is useful for the Mars flyby, but has little benefit for the rest of the mission. The size of the Dawn array is determined by the need to provide enough power to operate the propulsion system at the final destination. Examination of lower power levels showed that none of the ion thrusters were able to complete the mission with an array smaller than 8.7 kW. Because the Hall thruster operates relatively efficiently at low power, it is able to complete the mission at lower power levels, giving additional savings in solar array cost. At a power level of 7.1 kW, the Hall system delivers the same payload as the baseline case. This result clearly shows the benefit that efficient low-power operation has for this type of Discovery-class mission and leads to the finding that EP systems that operate with high efficiency at low power can greatly benefit Discovery-class missions.

This finding is specific to Discovery, because these missions are strongly cost- and power-limited. As a result, the EP system ends up driving the size of the power system, which, in turn, represents a substantial fraction of the total spacecraft cost. Note that for this mission scenario, because the low-power segment dominates the performance of the trajectory, configurations that allow simultaneous operation of multiple thrusters would provide little benefit for this mission. For the critical portions of the mission, the systems would be limited to single thruster operation by the power available from the array.

Figure 16 shows nth flight cost calculated for the Vesta–Ceres mission. The Hall system offers a considerable cost and performance

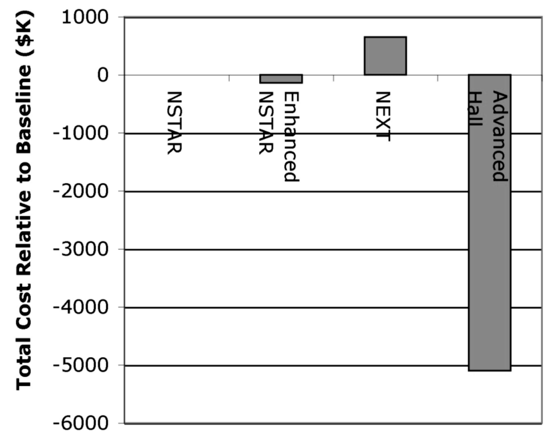

Fig. 16 Nth flight cost for a Vesta–Ceres rendezvous mission.

advantage on this mission, saving almost \$5 million, compared with the SOA case. In addition, because it operates relatively efficiently at low power, the Hall system could instead be used with a smaller array, resulting in additional cost savings. The ion systems, on the other hand, offer a moderate performance advantage, but little to no cost advantage over the SOA NSTAR. This occurs despite the use of fewer thrusters, because low thruster costs are offset by higher costs associated with the PPU. Like many Discovery missions, Dawn is strongly cost-constrained, and so these higher costs represent a significant barrier to the adoption of these technologies. Lower subsystem costs (particularly, PPU/DCIU costs) are needed for ion thruster options. The costs reported do not account for the system benefits of using fewer thrusters, which include simpler spacecraft configuration and integration testing.

Based on this analysis, we reach the following conclusions for the Vesta–Ceres rendezvous mission.

1) For all options, greater throughput allows use of fewer thrusters than NSTAR. This will simplify the spacecraft's configuration and reduce integration testing.

2) Hall has a considerable performance and cost advantage over the state of the art. In particular, its high efficiency at low power allows the use of a smaller solar array on this mission, resulting in substantial cost savings at the system level.

3) Lower subsystem costs, particularly PPU/DCIU costs, are needed for ion thruster options.

C. Comet Rendezvous Mission

The last mission considered 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 11.

A separate optimized trajectory is generated for each scenario using SEPTOP. All trajectories assume a nominal array power of 9 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. Although this mission is not as power-limited as the

Table 11 Comet rendezvous mission characteristics

Comet rendezvous	
Target body	Kopff
Launch vehicle	Delta 2925
Power system	9 kW solar array at 1 AU
Bus power	250 W
Duration	3.8 yr
ΔV	8.1 to 8.9 km/s
Launch year	2006
Ion/hall thruster duty cycle	90%
Launch and rendezvous dates	Selected by optimizer
Optimization tool	SEPTOP

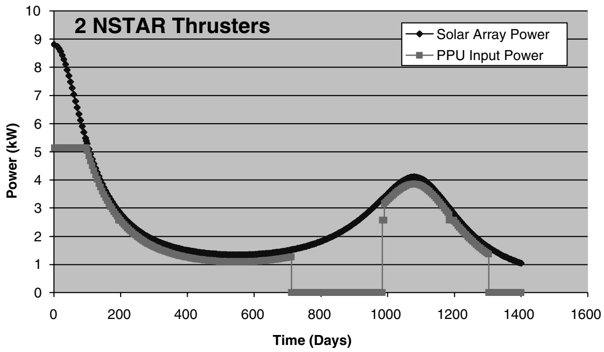

Fig. 17 Power profile for a comet rendezvous mission.

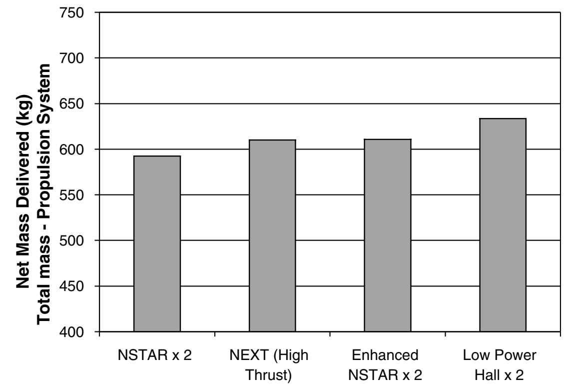

Fig. 18 Delivered mass for a comet rendezvous mission.

Vesta–Ceres rendezvous, it does require extended operation at moderately low power (less than 1.5 kW). The baseline SOA NSTAR mission uses two NSTAR thrusters. Its power profile is shown in Fig. 17. The overall results are summarized in Table 12 and net delivered mass and relative cost are shown in Figs. 18 and 19.

All proposed SEP systems offer a relatively small mass advantage over the SOA NSTAR. The Hall system also offers considerably lower nth flight cost, whereas NEXT offers a moderate cost savings and requires only one thruster rather than the two thrusters used in the other options. The enhanced NSTAR offers only a slight mass advantage and costs more than the SOA NSTAR system, so it is not considered applicable for this mission. As with the previous two missions, qualification costs exceed the nth flight cost savings shown earlier, resulting in unattractively high first-flight costs on this mission. This represents a significant barrier to the implementation of these technologies on Discovery missions.

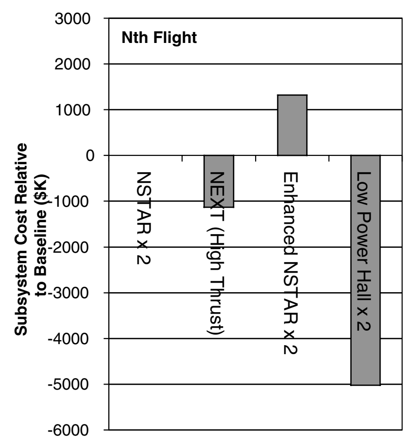

Fig. 19 Nth flight cost comparison for a comet rendezvous mission ($FY05).

D. General Observations

In the previous section, two general observations were made about the overall performance of EP systems for SEP missions:

1) In general, the best mass performance comes from EP systems that best use available solar array power.

2) EP systems that operate with high efficiency at low power can greatly benefit Discovery-class missions.

These qualitative findings were made in the context of specific missions (the near-Earth asteroid and Vesta–Ceres rendezvous). To better understand the relationship between system performance and array power, a series of optimized trajectories were run at different

Table 12 Comet rendezvous summary performance matrix

	Dual NSTAR	NEXT (high thrust)	Dual enhanced NSTAR	Dual Hall	NEXT (high I_{sp})	Triple NSTAR
Mission						
Launch vehicle	Delta 2925	Delta 2925	Delta 2925	Delta 2925	Delta 2925	Delta 2925
Trip time, yr	3.83	3.84	3.81	3.86	3.80	3.83
C3, km^2/s^2	15.8	13.7	15.3	11.0	15.8	12.9
Power, 1 AU, BOL	Baseline 9 kW	Same	Same	Same	Same	Same
Payload mass performance, kg	592	610	611	633	609	592
Launch mass, kg	987	1030	997	1090	987	1047
Xenon propellant throughput, kg	238	246	213	327	204	261
Net trajectory mass capability, kg	749	784	785	763	783	786
Propulsion subsystem mass, kg	157	174	174	130	174	194
Free space delta-v, km/s)	8.15	8.41	8.17	8.88	8.10	8.50
Configuration						
Simultaneous operating thrusters	2	1	2	2	1	3
Total thrusters (primary and redundant)	3	2	3	3	2	4
Total PPUs (primary and redundant)	3	2	3	3	2	4
Nth mission cost, $K	Baseline	−1128	+1321	−5020	−1128	+2693
First mission cost, $K	Baseline	+4872	+4021	+480	+4872	+2693
Propulsion subsystem costs						
Recurring cost, $K	Baseline	−1128	+1321	−5020	−1128	+2693
Qualification cost, $K	Baseline	+6000	+2700	+5500	+6000	+0
Development						
Maturity (TRL)	6	5	4	3	5	6
Development schedule (TRL 6, no life test)	None	∼1.5 yr	∼1.5 yr	>4 yr	∼1.5 yr	None
Development schedule (w/life test)	None	>2.5 yr	∼3 yr	>5.5 yr	>2.5 yr	None
Development cost, $M	Small	17	10	19.5	17.9	Small

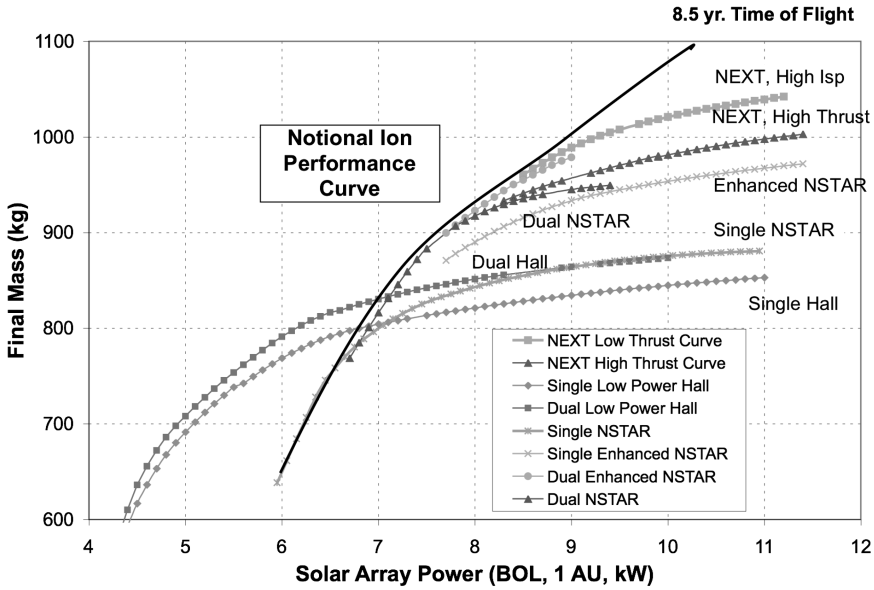

Fig. 20 Final mass vs solar array power for a notional main belt asteroid mission.

powers on a separate model main belt asteroid mission. The model mission is also a Vesta–Ceres rendezvous, but to simplify the analysis, the Mars gravity assist is removed and the optimizer is allowed to select the launch and arrival dates. The resulting mission is notional and is intended to clarify governing relationships. The trajectories were optimized using SEPTOP, were constrained to an 8.5-yr flight time, and use a Delta 2925H launch vehicle. Figure 20 shows total delivered mass as a function of array power for a series of different EP options. The *final mass* in this case includes the mass of the EP system. Also shown as a solid line is a notional ion performance curve that is created empirically by hand-drawing a line that envelopes the ion performance data. The curve is not generated numerically and is therefore entirely notional.

Note that the terms "NEXT, high I_{sp}" and "NEXT, high thrust" refer to the high-thrust and high-specific-impulse curves shown in Table 2. In general, for low-power operation, delivered mass increases rapidly with array power, then transitions to a flatter region at high power. The location of the transition varies directly with the maximum allowable PPU input power. As array power increases, the delivered mass increases until it reaches a point at which the PPU input is saturated and the EP system is no longer able to take advantage of extra power from the array. The ion systems have similar low- and midpower efficiency curves, and so their performance overlaps until the PPU input is saturated. Beyond this point, the performance curves flatten and the delivered mass is primarily a function of maximum PPU input power. The black line in Fig. 20 corresponds to an empirical "ion performance curve." This curve is a hand-drawn enveloping curve that appears to correspond to the mass delivered by an ion thruster system that has no maximum power limit. The ion thruster systems follow this curve until the PPU input is saturated and then transition to a plateau region. These results show how EP systems that best use available solar array power deliver the most mass at higher array powers. The choice of specific impulse also influences delivered mass, but this influence is secondary compared with the thruster's min/max power range.

The mass vs power performance curves for the low-power Hall system shows the influence that efficient low-power operation has on delivered mass. The ion systems deliver more mass above 7 kW, but their performance falls off rapidly with reduced array power, whereas the Hall system continues to offer reasonable performance below 5 kW. The higher performance is due to the Hall system's relatively high efficiency when operating far away from the sun. At higher powers, Hall systems operate less efficiently than ion, so their performance is not as good. With heavier spacecraft, it is necessary to use a larger solar array and an EP system with a higher maximum power to meet mission requirements. However, the Hall system offers better performance with smaller spacecraft and small solar arrays, which is highly desirable for cost-capped missions. This result shows that EP systems, which operate with high efficiency at low power, benefit small missions by enabling the use of smaller solar arrays, significantly lowering the cost of these missions.

Table 13 provides a matrix showing the applicability of each proposed technology to the Discovery missions analyzed in this study. Applicability is defined as a combination of performance and nth flight cost, and the mission is marked as applicable (×) if the technology either provides a performance benefit or a cost benefit compared with the SOA. As discussed previously, adding qualification costs to the nth flight cost significantly reduces the range of application for each technology. The low-power Hall system is marked as possibly cost-enabling (××), because the magnitude of the cost savings may be large enough to enable use of electric propulsion on some missions that would otherwise be cost-limited to chemical propulsion. This study is part of a larger effort that looked at both Discovery and New Frontiers class missions. A complete matrix, including the larger missions considered in other studies, is available in [19].

Table 13 General application matrix

	Discovery, less than 10 kW		
	Vesta–Ceres RV	Near-Earth asteroid	Comet RV
BOL array power	9 kW	6 kW	9 kW
NSTAR, one thruster	×	×	——
NSTAR, multithruster	——	×	×
Low-power Hall thruster	××	××	××
Enhanced NSTAR	×	×	——
NEXT	×	×	×

IV. Conclusions

A detailed study has been conducted to examine the potential benefits that advanced electric propulsion technologies offer to the competitively awarded, cost-capped missions in NASA's Discovery program. The study looked at the cost and performance benefits provided by three EP technologies (currently proposed or in development through NASA's In-Space Propulsion program) on three potential Discovery-class missions and compares their performance to a state-of-the-art system using the NSTAR ion thruster. The different propulsion technologies/mission scenarios considered are shown in Table 1. Because cost is a major driver for competed missions, a detailed electric propulsion subsystem cost model was used to conduct a cost–benefit analysis for each option. Development, qualification, and recurring costs were all considered in the study.

For each scenario, a launch date was selected and a low-thrust trajectory optimizer was used to calculate the flight time and total mass delivered to the final destination. The electric propulsion cost model was used to calculate the total cost of the propulsion system. The following general conclusions have been reached about each of the proposed EP technologies:

1) Low-power Hall offers very significant performance and cost benefits to Discovery-class missions. However, this technology is relatively immature (TRL 3), and further investment is required to determine if the technology can achieve its performance and cost goals.

2) The NEXT thruster offers moderate performance and noticeable cost benefits to Discovery-class missions vs dual NSTAR systems. It also offers strong performance benefits vs single NSTAR systems.

3) An enhanced NSTAR thruster using carbon-carbon-based ion optics and components from the NEXT subsystem development offers moderate performance and cost benefits to Discovery-class missions with relatively moderate total subsystem development costs.

It has also been found that for many missions, the increased throughput capability for the proposed EP technologies reduces the number of thrusters required to conduct the mission, leading to the finding that Discovery-class missions will generally benefit from increased xenon throughput capability offered by the proposed EP technologies.

Two conclusions have been reached about the cost of EP subsystems and the improvements needed to increase their application to cost-capped missions:

1) Lower subsystem costs (particularly, PPU and DCIU costs) are needed for ion thruster systems.

The proposed low-power Hall system carries a significantly lower nth flight cost estimate than the ion thruster systems, primarily because of lower PPU and DCIU costs associated with the subsystem. The NSTAR DCIU costs over $1.3 million per flight unit, and both primary and redundant units are required for a flight system, whereas the Hall system requires no DCIU. Future ISP development efforts should consider incorporating DCIU functions into other units to lower the cost of ion thruster systems.

2) First-flight qualifications costs are a significant barrier to implementation of new SEP technologies on Discovery missions.

With the current structure of in-space propulsion's development programs, significant qualification costs are borne by the first-flight mission using a new EP technology. The qualification cost generally exceeds the hardware and subsystem cost benefit brought by the new technology, leaving the first program no financial incentive to adopt a technology that will benefit future missions. This problem can be addressed programmatically by lowering qualification costs, perhaps by incorporating qualification testing into the subsystem technology development program or procuring flight qualification units as part of the development program. It can also be addressed by allowing/creating cost-sharing arrangements that fund the flight qualification effort outside the mission cost-cap limit.

Finally, two observations have been made regarding the general performance of EP systems on SEP missions:

1) In general, the best mass performance comes from the EP systems that best use available solar array power.

When excess array power is available, thrusters with the widest PPU input power range generally perform the best, because they can convert more of the available electrical power into kinetic energy over the course of the mission. The choice of specific impulse also influences delivered mass, but this influence is secondary compared with the thruster's minimum–maximum power range.

2) EP systems that operate with high efficiency at low power can greatly benefit Discovery-class missions.

Low-power performance is useful for Discovery missions, because these missions are cost-limited, and so the solar array is generally made as small as possible to lower program costs. This results in missions with extended periods of low-power operation. The proposed low-power Hall system operates at much higher efficiency and at lower power than the ion system and offers significant performance and cost benefits to some Discovery missions. In particular, the device's relatively high thrust at low power enables it to accomplish a Dawn-like Vesta–Ceres rendezvous, using a solar array that is 20% smaller than the ion systems.

It should be noted that this study uses reference missions derived from current mission concepts that use the state-of-the-art NSTAR ion thruster. Discovery missions cover a wide range of scientific goals, and so these reference missions do not encompass the full range of possible destinations and challenges. Principal investigators are continually asking for more challenging missions within the Discovery program, and advanced electric propulsion technologies may enable a class of missions more challenging than those considered in this study.

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