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# Electric Propulsion for Solar System Exploration

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**The use of ion propulsion for deep-space missions will become a reality in 1998 with the flight of the ion-propelled, New Millennium Deep Space 1 (DS1) spacecraft. The anticipation of this event is stimulating the call for improved ion propulsion technologies, a trend that is expected to continue. This paper describes the evaluation of possible advanced solar electric propulsion technologies and their potential benefits to projected near-term and midterm solar system exploration missions. The advanced technologies include high-performance derivatives of the DS1 ion propulsion technology, scaled-down DS1 systems, and direct-drive Hall-effect thruster systems. The results of this study indicate that significant near-term benefits can be obtained by the development of improved versions of the DS1 ion propulsion system (IPSs) components. In addition, if the current trend to smaller planetary spacecraft continues, then missions flying these smaller future spacecraft will benefit substantially from the development of scaled-down IPSs that incorporate advanced technologies in the ion engines and the propellant feed systems. The performance of the direct-drive Hall thruster systems is potentially superior to that of all other midterm options, but this technology has the highest development risk. Significantly reduced trip times to small bodies and the outer planets may be possible if technology programs work to retire these risks.**

## Introduction

IN October 1998, NASA will launch the New Millennium Deep Space 1 (DS1) spacecraft originally scheduled to flyby the asteroid McAuliffe, Mars, and the comet West-Kohoutek-Ikemura.[1] This spacecraft will mark the first use of ion propulsion to meet the primary propulsion requirements of a solar system exploration mission and will usher in a new era in the application of advanced propulsion for deep-space missions.

The ion propulsion system (IPS) for DS1 is being developed by the NSTAR [NASA solar electric propulsion (SEP) technology applications readiness] program and is based on NASA's 30-cm diam xenon ion engine.[2] The NSTAR technology has been shown to be capable of accomplishing many deep-space missions of interest.[3] However, this technology was intentionally conservative to maximize the probability of successful implementation. Therefore, significant improvements to the NSTAR IPS are possible, and it is expected that future missions will benefit from the development of these improvements. In addition, it is expected that a successful demonstration of SEP on DS1 will stimulate the consideration of more propulsively difficult, i.e., higher $\Delta V$, missions requiring improved SEP systems. This process has already begun with the selection of ion propulsion as the baseline system for the New Millennium Deep Space 4 (DS4) mission. Significant mission benefits for DS4 are enabled by an improved ion engine technology that has a larger total impulse capability than the DS1 NSTAR engine.

In general, there are two objectives for near- and midterm electric propulsion technology improvements for deep-space missions: reduce total mission life cycle costs and reduce flight times. This paper describes some near- and midterm electric propulsion technology advancements that could be developed and discusses their potential benefits to future deep-space missions. The results presented are based in part on a recent propulsion trades study.[4] This study evaluated advanced propulsion options against the needs of projected future missions included in solar system exploration planning activities.

## Near-Term Missions

Near-term, deep-space missions are defined in this paper as those that are launched before the year 2005. Missions in this category that could potentially benefit from the use of electric propulsion include the Europa Orbiter, Pluto Flyby, and the New Millennium DS4. Electric propulsion systems for the Europa Orbiter and Pluto Flyby missions, if they are used, would serve as high-energy upper stages that are jettisoned prior to reaching the final destinations. Total mission cost is one of the principle drivers and SEP will be used for the Europa Orbiter and Pluto Flyby missions only if it enables lower overall costs. For the DS4 mission, which will rendezvous with a short-period comet and demonstrate the technology necessary to return a sample to Earth, SEP is an enabling technology and will be used for both the outgoing and Earth-return phases of the mission. The only electric propulsion systems considered viable for near-term missions are those using the NSTAR hardware and those based on advanced technology derivatives of the NSTAR components.

### NSTAR Technology

The baseline NSTAR technology is defined as that which will fly on DS1. The NSTAR hardware is capable of multiple thruster operation, even though DS1 will fly only one thruster. The input power to each NSTAR power processor unit (PPU) can vary from a maximum of 2.5 kW to a minimum of 0.6 kW during thruster operation. In addition, each thruster can process a maximum of 83 kg of xenon independent of the throttle level. The NSTAR/DS1 SEP system component masses are given in Table 1. This table does not include mass for the engine gimbal electronics which, for DS1, is accounted for on the spacecraft side of the interface. Optimization of the ion engine gimbal mass for DS1 was not a concern resulting in a relatively heavy mechanism as indicated in this table. The end-of-life performance of the NSTAR ion engine over its full throttle range is given in Table 2.

The NSTAR ion engine is based on the NASA 30-cm-diam engine and is the result of over 25 years of technology devel-

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**Table 1 NSTAR/DS1 IPS component masses**

| Component | As-built mass, kg |
|---|---|
| Ion engine | 8.3 |
| Two-axis engine gimbal | 14.3 |
| PPU | 13.1 |
| PPU micrometeoroid shielding | 1.7 |
| PPU thermal control | 1.0 |
| DCIU | 2.5 |
| Fixed feed system | 12.4 |
| Feed system per engine | 1.2 |
| Propellant tankage[a] | 7.7 |
| Structure/cabling per engine | 6.3 |
| Component thermal control | 1.0 |
| Subtotal | 66.3 |

[a]For a maximum storage of 82.5 kg of xenon.

**Table 2 NSTAR end-of-life engine performance[a]**

| Engine input power, W | Thrust, mN | $I_{sp}$, s | Efficiency | Total flow rate, mg/s |
|---|---|---|---|---|
| 2.32 | 92.3 | 3313 | 0.646 | 2.84 |
| 2.08 | 83.3 | 3293 | 0.645 | 2.60 |
| 1.67 | 66.1 | 3291 | 0.640 | 2.05 |
| 1.37 | 52.8 | 3300 | 0.622 | 1.63 |
| 0.93 | 34.8 | 2974 | 0.544 | 1.19 |
| 0.57 | 21.8 | 2188 | 0.409 | 1.01 |

[a]After 8000 h of operation at 2.3 kW.

opment on electron-bombardment ion engines of this size. The engine body consists of a lightweight titanium structure and three rings of Sm–Co permanent magnets. The ion optics' system uses two dished molybdenum electrodes mounted to a titanium support ring. The inner electrode (called the screen grid) faces the discharge chamber and is 360 μm thick. This electrode has approximately 15,000 holes, each 1.91 mm in diameter, arranged in a close-packed, hexagonal pattern. The outer electrode (called the accelerator grid) is 510 μm thick and has 1.15-mm-diam apertures aligned with the corresponding holes in the screen grid. The grids are electrically isolated from each other and the discharge chamber body through the use of ceramic isolators. The cold grid separation is 0.61 mm. At full power a maximum electric field of 2100 V/mm is maintained between the grids (based on the cold grid separation).

The main discharge chamber and neutralizer cathodes are composed of Mo–Re tubes with an o.d. of 6.35 mm. For each cathode the electron emitter is a porous tungsten insert impregnated with a low-work function barium-oxide mixture. Similar cathodes have been developed at the NASA Lewis Research Center to reduce differential charging on the International Space Station and have been successfully operated in the laboratory for 28,000 h.[5]

A fine screen (called a plasma screen) surrounds the thruster discharge chamber and shields the thruster from its externally induced plasma. The operating and nonoperating temperature limits for the thruster are defined by the requirement to keep the downstream surface of the thruster plasma screen between −93 and 138°C.

The NSTAR PPU can operate with input voltages over the range of 80–160 V with an efficiency of greater than 92% over the full throttle range from 0.6 to 2.5 kW. During thruster operation the PPU baseplate temperature must be kept between −5 and 50°C. The nonoperating temperature limits for the PPU are −25 and 55°C. Thermal control for the NSTAR PPU on DS1 is provided by the spacecraft and is accomplished through the use of a radiator to reject the PPU waste heat during thrusting and heaters to keep the PPU from getting too cold during low-power operation and when the PPU is off.

A digital control and interface unit (DCIU) provides the command and control interface between the IPS and the space-

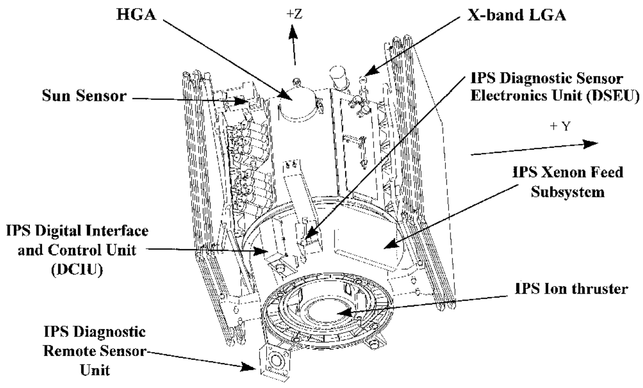


**Fig. 1 New Millennium DS1 in the stowed configuration showing the location of the IPS components.**

craft and has the capability to operate up to six thrusters simultaneously. The DCIU also controls the propellant feed system to establish the correct xenon flow rates into the thruster(s).

The propellant feed system consists of two parallel bang–bang pressure regulation systems and fixed flow restrictors to control the xenon flow rates. The DCIU reads the pressure in plenum tanks positioned between the xenon storage tank and the flow restrictors. If the pressure in the plenum tanks is too low the DCIU cycles solenoid valves upstream of the plenum tanks, resulting in the controlled addition of xenon. Three separate propellant feeds are required to operate the NSTAR engine: the main flow, the discharge cathode flow, and the neutralizer cathode flow. Throttling the engine requires the feed system to provide a range of propellant flow rates. The main flow is adjustable between 0.6 and 2.3 mg/s. The two cathode flows are fed from the same plenum tank and are adjustable between 0.24 and 0.36 mg/s. All flow rates are maintained to within ±3% over the entire throttle range.

Additional details on the NSTAR hardware are provided by Sovey et al.[2] A diagram of the NSTAR hardware integrated onto the DS1 spacecraft in the launch configuration is given in Fig. 1.

## Improved-Performance NSTAR Systems

There are many potential ways in which the baseline NSTAR technology could be improved and near-term technology development objectives should strive to improve those aspects that provide the greatest mission benefits. Technology improvements can be roughly lumped into two categories: those that reduce the dry mass of the IPS, and those that improve one or more of its performance parameters [efficiency, thrust, or specific impulse ($I_{sp}$).] Engine service life and reliability are subsumed in the propulsion system dry mass category because deficiencies in either engine life or reliability may be overcome by adding additional engines (and associated hardware) at the expense of increased dry mass. These two system improvement categories are coupled because improved engine performance may be obtained at the expense of engine life or reliability and visa versa.

The end-of-life performance of the NSTAR ion engine (after 8000 h of operation at full power) is already very good as shown in Table 2 and it is unlikely that it would be cost effective to invest in trying to significantly improve the efficiency of the engine's discharge chamber. On the other hand, meaningful performance improvements may be obtainable through the development of a cathode neutralizer technology that requires lower xenon flow rates. Missions that require extended thruster operation at the low power end of the thruster's throttle range, such as DS4, show attractive benefits, assuming the successful development of such a low-flow neutralizer.

The maximum engine $I_{sp}$ is a mission-driven parameter and NSTAR's $I_{sp}$ of 3300 s is appropriate for the near-term missions of interest. Future, more demanding, higher $\Delta V$ missions will benefit from the development of higher $I_{sp}$ ion engines. Ion engine operation at higher $I_{sp}$ is relatively easy to do, but this must be done without significantly compromising the engine service life capability.

Because, with the exception of the neutralizer, there is little to be gained by attempting to improve the efficiency of the NSTAR thruster components for near-term missions, improvements to the overall system must come in the form of dry mass reductions. Such reductions could come from reducing the mass of the engine, the PPU, the propellant tank, the propellant feed system, the gimbal mechanism, and/or improving the engine total impulse capability. Design and fabrication refinements may enable reductions in the NSTAR engine mass by approximately 1 kg, and integrating the NSTAR PPU circuitry directly into a multifunctional bus may reduce the PPU mass by perhaps as much as a factor of 2. There is, however, little prospect for significantly reducing the propellant tank mass, which for NSTAR is already outstanding at only 9.3% of the propellant mass stored.

Significant reductions in the propellant feed system mass and volume are possible through the use of active propellant flow controllers to eliminate the bang–bang pressure regulation scheme employed by NSTAR and its associated large and relatively heavy plenum tanks. Several different types of xenon flow control devices are currently under development. The use of active flow controllers may enable the design of xenon feed systems that could reduce the mass of the NSTAR system by a factor of 3 while also enabling substantial reductions in the feed system volume.

An ion engine gimbal design based on the Mars Pathfinder high-gain antennae gimbal has been proposed that may enable a factor of 2.5 reduction in the gimbal mass relative to the New Millennium DS1 gimbal. This gimbal configuration would have a ±15-deg capability in two axes and is projected to have a mass of 5.6 kg.

Finally, significant dry mass reductions can be achieved by increasing the ion engine service life capability or more specifically, its total impulse capability. The NSTAR ion engine has a design life of 8000 h at full power, corresponding to a total impulse capability of $2.6 \times 10^6$ N-s and a maximum propellant consumption of 83 kg. Increasing the total impulse capability per engine by 50–100% could result in a beneficial reduction in the number of engines required to accomplish high-$\Delta V$ missions. The near-term stimulus for this performance enhancement is the DS4 mission, but it is anticipated that many other missions will also benefit from this improvement.

The NSTAR program has successfully completed a planned 8000-h, long-duration test of the DS1 ion engine demonstrating the full total impulse capability of the engine.[6] This test was completed in October 1997 and was voluntarily terminated after demonstrating a total of 8194 h of operation at full power. Posttest inspection of the thruster indicated that erosion of the accelerator grid, once believed to be the major life-limiting mechanism for the thruster, was significantly lower than expected prior to the start of the test. As expected, very little interior discharge chamber erosion was observed because of low internal discharge voltage (less than 24 V at full power) observed throughout the test. Preparations for a second long-duration test are currently under way with the objective to demonstrate 150% of the engine's design total impulse and to do so through extended operation at throttled power levels.

A related and key part of the NSTAR program is the engine service-life validation task. This activity seeks to quantify the failure probability for the engine because of damage-accumulation failure modes as a function of run time. For high-reliability components such as the NSTAR ion engine it is not practical to establish the failure probability experimentally. Consequently, the engine service life is being established through a combination of the long-duration testing discussed earlier and probabilistic modeling of the principle failure modes.[7,8] In this framework the long-duration testing is performed to identify unknown failure modes, quantify engine performance changes vs run time, and validate the models of the principle wear-out modes. Once reliable models of the failure modes are established, the effects of changes in engine operation dictated by mission considerations can be readily quantified. For example, such models could be used to assess the failure risk of demanding a 100% increase in the engine total impulse capability, or the impact on the engine service life because of operation at a significantly higher specific impulse.

**Multimission SEP Module**

The DS4 mission requires the use of SEP. To determine the feasibility of saving money on other deep-space missions, a multimission SEP (MMSEP) module was studied. The objective of this study was to evaluate whether such a system, if it were developed for the DS4 mission, would be attractive for the Europa Orbiter and Pluto Flyby missions with little or no changes. Significant cost savings are anticipated through the reduction of nonrecurring costs if such a MMSEP module is possible.

Analyses suggest that it is possible to design a MMSEP module for the Europa Orbiter, DS4, the Pluto Fly missions, and fit within the capabilities of the Delta II launch vehicle. The study determined that the Pluto and DS4 missions could use the Delta II 7925 and that the Europa Orbiter mission would need the higher-performance Delta II 7925H. The current MMSEP module configuration consists of four 30-cm-diam NSTAR-derivative ion engines capable of processing a minimum of 120 kg of xenon each, two NSTAR PPUs, a digital interface unit, the NSTAR propellant feed system, advanced xenon propellant tanks, and advanced two-axis thruster gimbals. Further improvements could be obtained through the use of advanced xenon feed system components as discussed earlier.

The MMSEP module includes a lightweight solar array assumed to have a specific power of 70 W/kg beginning of life (BOL) referenced to 1 AU. Solar-array BOL powers range from 6 to 12 kW, depending on the mission, and are provided by a two-wing configuration. A gimbal mechanism for each wing is used to enable the solar array to be pointed at the sun while the ion thrusters are firing in the desired direction. Structure, thermal control, cabling, and separation mechanisms complete the MMSEP module. Additional hardware is added for the DS4 mission including telecom, a docking mechanism, and a hydrazine attitude control system to meet its specific mission requirements.

*Europa Orbiter*

Europa is one of the moons of Jupiter and is suspected of having a submerged ocean of liquid water. One of the science objectives of this mission is to look for additional evidence of this liquid water ocean. The accomplishment of this goal requires orbiting Europa. The application of the MMSEP module for this mission would be to deliver the Europa spacecraft and a large chemical propulsion system to the vicinity of Jupiter. After the SEP system is jettisoned, the chemical propulsion system performs the Jupiter orbit insertion maneuver and eventually delivers the spacecraft into orbit around Europa. At the time of this study, the baseline, non-SEP mission considered the use of an Atlas IIAR launch vehicle to deliver a 260-kg spacecraft to Europa with a direct trajectory in about three years, and a Delta II 7925 launch vehicle to deliver approximately 300 kg with a triple Venus gravity-assist trajectory in just over 6 years.

The MMSEP module system with a 6-kW (at end-of-life referenced to 1 AU) solar array can deliver between 260 and 290 kg to Europa in 3.5–4 years using a solar electric Venus–Venus gravity assist (SeVVGA) trajectory and a Delta II 7925 launch vehicle. More than 300 kg could be delivered to

Europa orbit using the MMSEP module and the Delta II 7925H launch vehicle. The low-thrust $\Delta V$ for this mission is about 6 km/s. Current mission planning calls for this mission to be launched between 2002 and 2004. The performance for the 2004 launch dates is slightly worse than the 2002 cases. The use of the MMSEP module for this mission allows a reduction in launch vehicle from the Atlas IIAR to the Delta II 7925H, with a 1-year trip-time penalty or a 2-year reduction in trip time relative to the Delta II 7925 launched triple-Venus gravity assist ballistic trajectory.

*Pluto Flyby*

Pluto is the only planet in the solar system that hasn't been visited by a U.S. spacecraft, and the Pluto Flyby mission is intended to be a low-cost mission to fill this void. At the time of this study, the baseline, non-SEP mission for Pluto assumed the use of a Delta II 7925 with a Star 30C upper stage launched in 2002 or 2004 and a Jupiter gravity-assist trajectory to deliver a 145-kg spacecraft to Pluto in 9–12 years.

The MMSEP module system with a 6.75-kW solar array at the beginning of life (referenced to 1 AU) can deliver the Pluto spacecraft in approximately 8.5 years using a Delta II 7925 launch and a SeVVJGA (solar electric Venus–Venus–Jupiter gravity assist) trajectory for a launch in 2002. The low-thrust $\Delta V$ for this trajectory is about 9 km/s. Significantly, this SEP system could deliver two 145-kg spacecraft to Pluto in less than 9.5 years using the same launch vehicle. The superior performance of the MMSEP system using SeVVJGA trajectories relative to chemical propulsion with ballistic Jupiter gravity-assist trajectories is shown in Fig. 2.

The MMSEP module appears to be an attractive option for this mission, particularly if there is sufficient interest in the delivery of two spacecraft to Pluto. The SEP performance for a launch in 2004 (not shown in Fig. 2) is significantly poorer than in 2002. This is a result of Jupiter moving away from a position that provides significant gravity assist $\Delta V$. After about 2005 Jupiter will not be available for gravity assists to Pluto for either SEP or chemical/ballistic trajectories for about 10 years. In this time frame solar electric propulsion may become enabling for missions to Pluto, and these missions will benefit from the development of more advanced, i.e., lighter SEP systems that could be made available by then.

*Cost*

Because SEP is not mission enabling for either the Europa Orbiter or Pluto Flyby missions it will be used only if it provides a significant cost savings relative to non-SEP options. Potential cost savings from the use of SEP include enabling

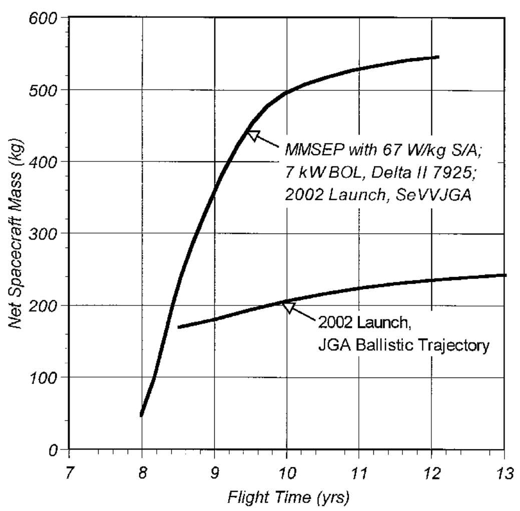


**Fig. 2 SEP performance for the Pluto Flyby mission relative to ballistic Jupiter Gravity Assist trajectories using conventional chemical propulsion.**

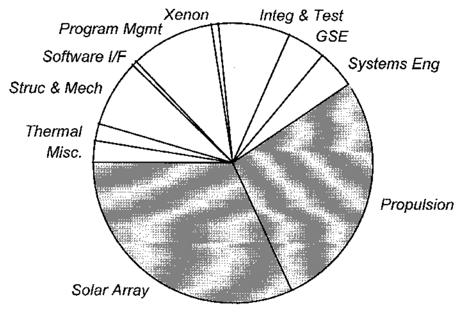


**Fig. 3 SEP module cost distribution (first unit).**

the use of smaller, less-expensive launch vehicles, and/or reductions in trip times. Shorter trip times save money through the elimination of the mission operation costs that would accrue during longer flight times. These savings must be sufficient to more than overcome the nontrivial costs of SEP systems.

Preliminary estimates of MMSEP module costs indicate that approximately one-third of the total cost is in the solar array, another one-third in the electric propulsion hardware (thrusters, PPUs, etc.), and one-third in everything else [engineering, integration, testing, program management, ground support equipment (GSE), etc.], as indicated in Fig. 3. Therefore, to reduce the cost of a MMSEP module, it would appear to be most fruitful to concentrate on reducing solar array costs and the cost of the electric propulsion components. The MMSEP module is expected to significantly reduce nonrecurring costs if the same or nearly the same components and system design can be used for multiple missions.

## Midterm Missions

Midterm, deep-space missions are defined as those launching between 2005 and 2020. For midterm missions two additional types of SEP systems were evaluated; one based on a scaled-down derivative of the NSTAR ion engine, and the other based on a high-$I_{sp}$ Hall thruster-operated direct-drive from a high-voltage solar array.[9,10]

### Scaled-Down NSTAR Derivative Systems

The development of a scaled-down NSTAR derivative system may be attractive for deep-space missions provided the current trend to smaller planetary spacecraft continues. If it does, then a smaller, lighter SEP system based on an NSTAR-derivative ion engine approximately one-quarter the size of the present engine can be shown to provide significant benefits for these missions.[4] Specifically, it is expected that a quarter-scale SEP system will facilitate the use of launch vehicles smaller than the Delta II 7326 for deep-space missions with small spacecraft. The quarter-scale engine in Ref. 4 is assumed to be a 14-cm-diam ring–cusp ion engine with a modified magnetic circuit scaled down from the 30-cm-diam NSTAR engine. Although Ref. 4 considered only 14-cm-diam engines, it is believed that the conclusions are not sensitive to the exact engine size, provided the engine is large enough to produce the required thrust levels but not so large that the system dry mass suffers from the added engine mass.

The development of a scaled-down NSTAR ion engine should, in addition to simply making the thruster smaller, also seek to increase the maximum $I_{sp}$, increase the engine service life, and reduce the engine specific mass. Higher specific impulses will be required for the higher $\Delta V$ missions anticipated in the midterm time frame. Greater service life, as discussed earlier, reduces the propulsion system dry mass by reducing the number of engines required. Finally, reducing the engine specific mass is necessary because electric propulsion systems are inherently more sensitive to dry mass than chemical propulsion systems. This is because each kilogram of propellant that is displaced by dry mass has a larger total impulse penalty than a kilogram of propellant in a chemical propulsion system.

Erosion-resistant, carbon–carbon electrodes for the ion engine accelerator system hold significant promise to enable increased engine life at the higher applied voltages necessary to obtain increased specific impulses. At the same incident ion energies, carbon–carbon composite electrodes may be seven times more erosion-resistant than the state-of-the-art NSTAR molybdenum grids.[11,12] Carbon–carbon grids and their associated support structures are also significantly lighter than the same-sized molybdenum-grid-based accelerator systems. Because a significant fraction (almost 25%) of the engine mass is in the ion optics assembly as indicated in Fig. 4, this technology will also help to reduce the overall mass of ion engines. Therefore, the development of carbon–carbon grid-based ion accelerator systems can potentially benefit three performance aspects; longer life, high specific impulses, and engine mass reduction.

Almost 50% of the total engine mass according to Fig. 4 is in the engine body, which includes the magnet rings, plus the plasma shield. Consequently, a new approach to ion engine body fabrication should be developed. One potential approach makes use of a fiber-core composite structure that is both strong and very lightweight, as well as electrically insulating. The fiber-core composite consists of two very thin (50-μm-thick) aluminum face sheets that are flocked with quartz or glass fibers and glued together with the flocked sides facing each other. The resulting composite is a microtruss sandwich structure approximately 5 mm thick. This composite structure has approximately the same bending strength of 0.76-mm-thick aluminum but only one-sixth the mass. With the use of glass or quartz fibers the aluminum face sheets are electrically isolated from each other. If this composite can be successfully developed, it will allow the inner surface to be at the roughly 1000 V potential of the ion engine discharge plasma while the outer surface is at spacecraft ground potential. Thus, the composite structure becomes both the engine body and the surrounding plasma screen. This unibody construction is expected to result in at least factor of 2 reduction in the combined engine body and plasma screen mass, as well as a reduction in the cost of engine fabrication as a result of a reduced parts count.

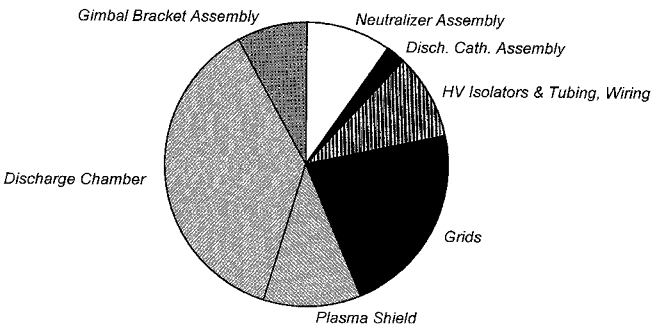


**Fig. 4 NSTAR ion engine mass distribution.**

For the trajectory analyses used to determine potential mission benefits, the maximum input power to each quarter-scale engine PPU is assumed to be 770 W, and the dynamic throttle range is assumed to be the same as for the 30-cm-diam NSTAR engine (4.5 to 1 maximum to minimum input power ratio). The PPU mass is scaled as the square root of the power ratio relative to the NSTAR PPU.

An example point design for a scaled-down SEP system is given in Table 3 for a comet rendezvous assuming the use of a Taurus XL/Star 37 launch vehicle. The masses in this table assume the use of an advanced propellant feed system described earlier and a very lightweight thruster gimbal mechanism. The engine mass in this table is based on mass mockups fabricated using the fiber-core unibody construction, which suggest that an advanced 14-cm-diam ion engine could have a mass as low as 1 kg. However, because this is an unproven technology, a large mass growth contingency is assigned as indicated in Table 3.

### *Comet Rendezvous*

The solar system has many different and interesting comets. One approach to studying a variety of these comets would be to develop a propulsion system that can deliver a small science spacecraft from a small, inexpensive launch vehicle. Many different comets could then be visited by launching many copies of the same spacecraft to different destinations. Comparison of performance, in terms of the net spacecraft mass delivered, is given in Fig. 5 for the baseline NSTAR technology and a quarter-scale, NSTAR-derivative system. These mission performance calculations were made assuming a Taurus XL/Star 37 launch vehicle and an advanced solar array (100 W/kg) with a BOL power level of 1.6 kW. The original trajectory calculations were performed by Sauer[13] assuming a Delta II 7925, and were scaled to the Taurus XL using the approach described in Ref. 4. The results in Fig. 5 show that the quarter-scale systems could deliver a 40-kg science craft to many different comets. The net spacecraft mass in this figure does not include the SEP system mass or the mass of the solar array so that the total mass delivered to each comet is significantly greater than that indicated in Fig. 5. The flight times for these comet rendezvous missions range from 2.0 to 3.7 years with low-thrust $\Delta V$ ranging from 7.5 to 14.5 km/s.

**Table 3 Quarter-Scale NSTAR system mass breakdown**

| Item | Quantity | Unit mass, kg | Growth contingency, % | Total mass, kg |
|---|---|---|---|---|
| Ion engine | 3 | 1.0 | 160 | 7.8 |
| Gimbals | 3 | 1.0 | 30 | 3.9 |
| DCIU | 1 | 1.9 | 10 | 2.1 |
| PPU | 2 | 6.9 | 7 | 14.8 |
| PPU thermal control | 1 | 0.7 | 20 | 0.8 |
| Fixed xenon feed system mass | 1 | 1.6 | 10 | 1.8 |
| Feed system mass per engine | 3 | 1.0 | 10 | 3.3 |
| Propellant tankage | 1 | 8.0 | 0 | 8.0 |
| Structure/cabling per engine | 3 | 1.5 | 22 | 5.5 |
| IPS subtotal | —— | | —— | 48.0 |
| Non-PPU thermal control (5% of IPS subtotal) | 1 | 2.4 | 0 | 2.4 |
| IPS dry mass | —— | —— | —— | 50.4 |
| Solar array mass (at 20 kg/kW) | 1 | 31.2 | 0 | 31.2 |
| Total dry mass | —— | —— | —— | 81.6 |
| Propellant mass | 1 | 80 | 0 | 80 |
| Total wet mass | —— | —— | —— | 162 |

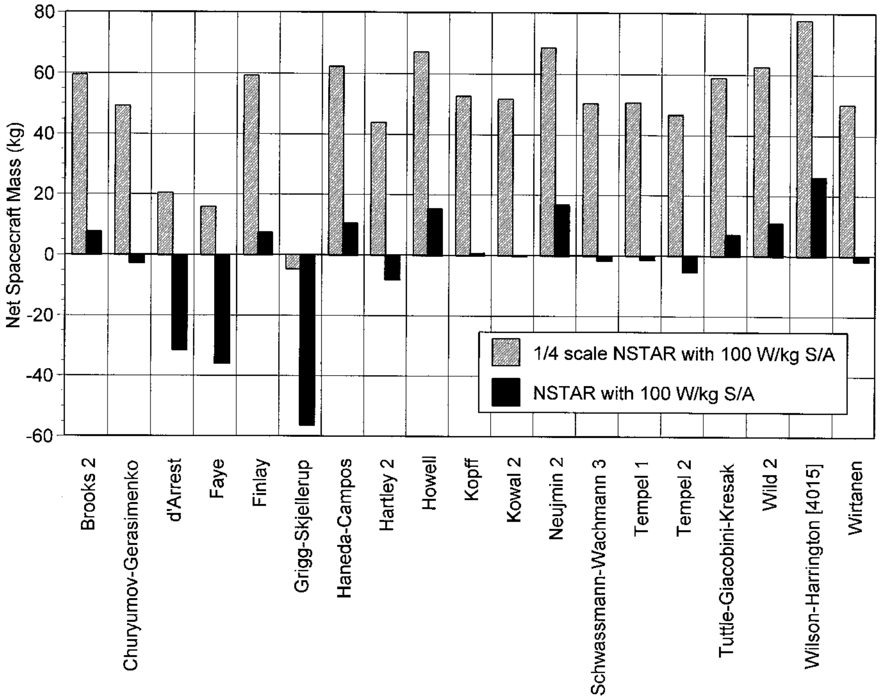


**Fig. 5 Performance comparison between the quarter-scale, NSTAR-derivative system and the baseline NSTAR for comet rendezvous missions assuming the use of a Taurus XL/Star 37 launch vehicle and a 1.8-kW solar array (at beginning of life referenced to 1 AU).**

The significant advantage of the quarter-scale system relative to the baseline NSTAR system is largely a result of the fact that the NSTAR hardware is physically larger and has the ability to process more power than is required for these missions, resulting in a significant dry mass penalty. Other analyses indicate that the full-sized NSTAR system provides an attractive performance for spacecraft-sized systems for launch on Delta II 7326 or larger launch vehicles.

**Direct-Drive TAL Systems**

A second objective of this study was to identify electric propulsion technology requirements that, if met, would enable significantly shorter flight times to small bodies such as main-belt asteroids or comets. Specifically, a factor of 2 reduction in flight time relative to that achievable using NSTAR-level technology was selected as the goal. Qualitatively, to achieve this trip-time reduction requires the combination of high-power, lightweight electric propulsion systems with small, lightweight spacecraft, i.e., a spacecraft with a high power-to-weight ratio. Direct-drive Hall thruster systems hold the potential of providing significantly lower specific masses than conventional ion engine-based systems for a given thrust level. This results from the fact that Hall thrusters require simpler power-processing hardware and have significantly higher thrust densities than gridded ion engines. Direct-drive refers to a system in which the high voltage input required by the thruster is supplied directly from a high-voltage solar array with only enough power-processing hardware in between to be able to start and stop the thruster and provide fault protection. During steady-state operation the direct-drive PPU passes the solar array voltage through unregulated to the Hall thrusters. This significantly reduces the overall mass of the required PPU. For Hall thruster systems, the power drawn from the solar array is controlled by adjusting the propellant flow rate to the thruster.

The concept for direct-drive systems has been around for a long time,[14] but has only recently become more realizable as a result of the availability of the high-performance Hall thrusters. The concept of using Hall thrusters in direct-drive systems was proposed in 1993 because of the expected synergy between the linear concentrator solar array (under development by the Ballistic Missile Defense Organization) with its projected ability to operate at high voltages and the input voltage requirements of Hall thrusters. This concept was subsequently successfully tested at the NASA Lewis Research Center.[15] Hall thrusters operate with good performance at lower applied voltages than gridded ion engines, enabling the development of direct-drive systems with significantly lower solar array voltages. In addition, mission analyses suggest that for small spacecraft a 10-kW propulsion system with a specific impulse of approximately 2500 s is desirable for fast, small-body rendezvous missions. Space-charge limitations of the ion optics of gridded ion engines make it very difficult to process 10 kW at the relatively low voltage corresponding to an $I_{sp}$ of 2500 s. On the other hand, the development of a single Hall

**Table 4 Direct-drive TAL spacecraft mass breakdown**

| Item | Mass, kg |
|---|---|
| 10-kW, high-$I_{sp}$, Hall engine | 8.0 |
| Engine gimbal | 1.9 |
| Direct-drive PPU | 5.0 |
| PPU thermal control | 3.0 |
| Digital interface and control unit | 0.5 |
| Propellant tank | 32 |
| Fixed feed system | 2 |
| Feed system per engine | 1 |
| Power management and distribution | 5 |
| **Subtotal** | **58.4** |
| Cabling (5% of subtotal) | 2.9 |
| Structure (15% of subtotal) | 8.8 |
| Thermal (5% of subtotal) | 2.9 |
| Solar array drive | 6.5 |
| **Total** | **73.0** |
| Contingency (30% of total) | 21.9 |
| Solar array | 103 |
| Residual xenon | 7 |
| **Total SEP dry mass** | **204** |
| Xenon propellant | 320 |
| Spacecraft | 50.0 |
| Launch vehicle adapter (2.5 % wet) | 14.7 |
| **Total S/C mass** | **589** |

Table 5 Direct-drive TAL mission performance

| Item | Vesta | Ceres | Kopff |
|---|---|---|---|
| Flight time, yr | 1.3 | 1.3 | 1.2 |
| Launch vehicle | Taurus XL | Taurus XL | Taurus XL |
| Solar array BOL power, kW | 9.0 | 10.0 | 9.9 |
| Initial thrust, N | 0.44 | 0.61 | 0.485 |
| Propellant mass flow rate, mg/s | 18.0 | 25.0 | 19.8 |
| Total propellant mass, kg | 321 | 520 | 376 |
| Total spacecraft wet mass, kg | 589 | 836 | 644 |
| Launch vehicle capability, kg, to 200 km, derated 10% | 1278 | 1278 | 1278 |
| Launch margin, kg | 689 | 442 | 634 |

thruster designed to operate at 10 kW and 2500 s appears to be a realistic goal; achieving the required engine life is the biggest technological challenge. Such a thruster would require an input voltage from the solar array of between 500 and 600 V.

An unregulated, 10-kW, direct-drive PPU has been estimated to have a specific mass of 0.5 kg/kW, which is about a factor of 10 less than the NSTAR PPU. Such a direct-drive PPU would provide fault protection and the ability to soft-start and shutdown the thruster. A breadboard, direct-drive PPU has been fabricated and tested with a Hall thruster at up to 4.5 kW and 600 V. The measured efficiency of this PPU is over 99% (when operating unregulated).

A complete direct-drive Hall system requires a lightweight propellant feed system and lightweight thruster gimbal. Such a system potentially has the smallest specific mass of any midterm SEP technology. An example point design for a direct-drive Hall system sized for a Vesta sample mission return assuming the use of a Taurus XL launch vehicle is given in Table 4.

### High-Voltage Solar Arrays

The SCARLET linear concentrator solar array will fly on the New Millennium DS1 spacecraft and produce an output voltage of between 90 and 120 V.[16] In its DS1 configuration this array has a specific mass of about 50 W/kg. Technology improvements may be expected to increase this to 70–80 W/kg in the near-term. The PAPS+ experiment[17] tested several cell technologies in space for 1 year at ±500 V and showed that concentrator arrays were stable and resistant to radiation damage and plasma interaction. The present direct-drive study assumed a solar array specific power of 100 W/kg and an output voltage of between 500 and 600 V. Achieving the 100 W/kg with the linear concentrator array will be a major technology challenge.

On outbound interplanetary trajectories the solar array output voltage will increase as the array temperature and output power decrease. This change in solar array voltage has not been accounted for in the analyses performed to date. In general, this effect will result in higher specific impulses and lower thrust for the available power. It may be possible to compensate for the increase in array voltage by adjusting the engine operation to pull the array voltage down to the desired level. In the trajectory calculations shown next in this paper it was assumed that the $I_{sp}$ and, hence, the solar array voltage, was constant over the mission.

#### *Mission Performance*

The feasibility of performing fast rendezvous missions to small bodies using small inexpensive launch vehicles together with high-power SEP systems and small spacecraft was examined assuming the availability of direct-drive SEP technology. A small, relatively inexpensive launch vehicle, such as the Taurus XL, can deliver over 1200 kg to a 200-km low-Earth orbit (LEO), but only about one-fourth of this to Earth escape. The mass delivery capability to Earth escape is too small to accommodate a high-power SEP system, and its payload with technologies is expected to be available in the midterm time frame. Therefore, the SEP system must be used for Earth escape as well as for the heliocentric transfer. With a sufficiently high-powered SEP system and small payload, the time spent spiraling through the Earth's radiation belts can be minimized.

The performance for rendezvous missions to the main-belt asteroids Vesta and Ceres and the comet Kopff are given in Table 4, assuming the 10-kW, direct-drive Hall thruster system and spacecraft masses listed in Table 5. The procedure used to calculate these performance values is described in more detail in Ref. 18. The flight times in this table include the time required for Earth escape. The large launch vehicle margins shown indicate that the launch vehicle should be used to place the spacecraft in a higher initial orbit. This will serve to reduce the time and $\Delta V$ required for Earth escape and, more significantly, will reduce difficulties with aerodynamic drag at very low orbit altitudes. Even so, the flight times shown are less than half those calculated for the conventional SEP approach, i.e., launch to Earth escape, and NSTAR technology.[13] Packaging the 10-kW solar arrays along with the SEP system and the spacecraft into the Taurus XL shroud is a major issue that was not addressed in this study.

## Conclusions

Ion propulsion is on the verge of entering the mainstream of propulsion options available for deep-space missions. Continued investment is needed to develop high-performance NSTAR derivatives to meet the needs of emerging, more demanding missions such as the New Millennium Deep Space 4 mission. Simultaneously, investment in the development of a scaled-down, advanced-technology NSTAR system, with engines capable of operating at higher specific impulses than the NSTAR engines, will be needed to meet the anticipated needs of future small spacecraft and enable higher $\Delta V$ missions (>15 km/s). If the trend toward smaller spacecraft continues in the future, and if it is desirable to launch these smaller spacecraft from launch vehicles smaller than the Delta II 7326, then there is a significant payoff from the development of scaled-down NSTAR-derivative technology. More difficult missions contemplated for the future such as comet and asteroid sample returns, solar probe, and the multiple main-belt asteroid rendezvous, will benefit from the development of NSTAR-derivative engines that have significantly greater total impulse and specific impulses.

Direct-drive Hall systems offer the potential for the best performance of any midterm SEP technology at the expense of the highest development risk. A unique role for high-performance direct-drive Hall systems may be in enabling very short trip time missions to be performed from Taurus XL-class launch vehicles where the SEP system use begins at LEO rather than after Earth escape. The use of SEP for planetary missions in this manner may be the next major advance in solar system exploration, but will require the development of lightweight, high-voltage solar arrays. The electric propulsion technology programs should investigate reducing the unknowns associated with high-$I_{sp}$, direct-drive Hall thruster systems.

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