Electric Propulsion for Solar System Exploration

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The use of ion propulsion for deep-space missions will become a reality next year with the flight of the ion-propelled New Millennium Deep Space 1 spacecraft. The anticipation of this event is stimulating the call for improved ion propulsion technologies, a trend which is expected to continue. This paper describes a suggested roadmap for the development of advanced solar electric propulsion technologies based on the expectation that these technologies will provide significant benefits for projected near-, and mid-term solar system exploration missions, The advanced technologies, include high performance derivatives of the NSTAR technology, quarter-scale NSTAR systems, and direct-drive Hall-effect thruster with anode layer (TAL) systems. The results of this study indicate that significant near-term benefits can be obtained by the development of improved versions of the DS1/NSTAR ion propulsion system components. In addition, if the projected current trend to smaller planetary spacecraft continues, then the missions flying these smaller spacecraft will benefit substantially from the development of a scaled-down system which is approximately 1/4th the sue of the NSTAR hardware, and which incorporates advanced technologies in the ion engine and the propellant feed system. The performance of the direct-drive TAL systems is potentially superior to that of all other mid-term options, but has the highest development risk. It is recommended that the technology programs seek to remove these risks to enable reduced trip times to small bodies and the outer planets.

Introduction

In July of 1998 NASA will launch the New Millennium Deep Space 1 (NM DS 1) spacecraft to flyby the asteroid McAuliffe, Mars, and the comet West-Kohoutek-Ikemura [1]. This spacecraft will mark the first use of an ion propulsion system 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 for Deep Space 1 is being developed by the NSTAR (NASA Solar electric propulsion Technology Applications Readiness) program [2] and is based on NASA's 30-cm diameter xenon ion engine [3]. The NSTAR system technology has been shown to be capable of accomplishing many deep space missions of interest [4], However, this technology was intentionally conservative to maximize the probability of successful implementation, It is expected that future missions will benefit from improvements to or derivatives of the NSTAR technology, In addition, it is expected that the demonstration of Solar Electric Propulsion (SEP) on NM DS 1 will stimulate the consideration of more propulsively difficult (i.e., higher AV) missions requiring improved SEP systems. Indeed, this process has already begun with SEP being baselined on the Champollion/DS4 mission [5], where there are significant mission benefits enabled by an ion engine technology which has a larger total impulse capability than the NM DS 1 NSTAR engine.

This paper describes an investment strategy in advanced ion propulsion technology that is expected to provide the greatest benefit to future deep space missions and is based in part on a recent propulsion trades study [6,7]. This study evaluated propulsion options according to the needs of projected future missions including those in the recent solar system exploration planning activities [8]. In general, there are two objectives for furture electric propulsion technology developments for deep-space missions: reduce mission costs and reduce flight times,

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Near-Term Missions

Near-term, deep-space missions are defined in this paper as those which are launched before the year 2005. Missions in this category which could potentially benefit from the use of electric propulsion include: Europa Oribiter, Pluto Flyby, and Champollion/Deep Space 4 (DS4). For the Europa Orbiter and Pluto Flyby missions, the SEP system would be used as a high-energy upper stage which is jettisoned prior to reaching the final destination. The Champollion/DS4 mission will rendezvous with a short-period comet and demonstrate the technology necessary to return a sample to Earth. SEP is an enabling technology for this mission and will be used for both the outgoing and Earth-return phases of the mission. For the Europa Orbiter and Pluto Flyby missions, total mission cost is one of the principle drivers and SEP will be used for these missions only if it enables in lower overall costs.

For near-term missions only two types of SEP technologies will likely be available; those based on the NSTAR component technology as it will fly on NM DS 1 and those based on improved-performance derivatives of the NSTAR components.

NSTAR Technology

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

Table 1 NSTAR/DS1 SEP Component Masses

Component	Current Best Estimate Mass (kg)
Ion Engine	8.2
2-Axis Engine Gimbal	14.3
Power Processor Unit	11.9
Digital Interface and Control Unit	1.9
Fixed Feed System	7,2
Feed System per Engine	1.2
Propellant Tankage*	7.7
Structure/Cabling per Engine	5.9

*for a maximum storage of 82.5 kg of xenon

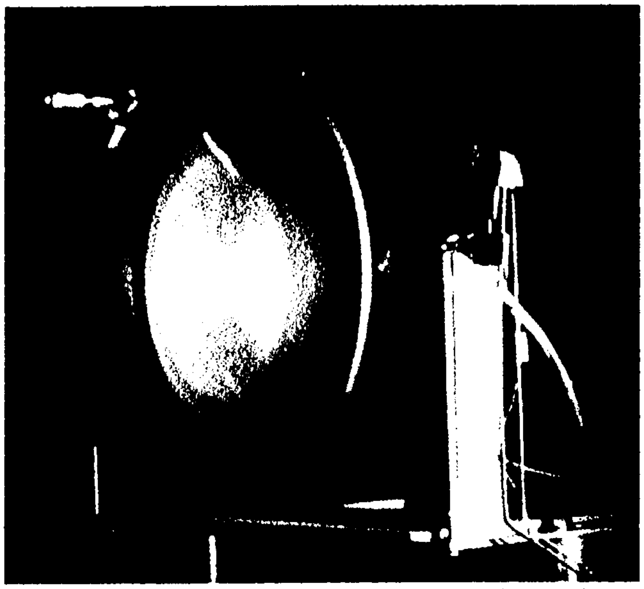

Fig. 1 The NSTAR 30-cm diameter, xenon ion engine operating at 2.3 kW.

The NSTAR ion engine is based on the NASA 30-cm diameter engine (shown in Fig. 1) and is the result of over 25 years of technology development on 30-cm sized electron-bombardment ion engines. The engine body, which is maintained at anode potential during engine operation, consists of a light-weight titanium structure and three rings of sumarium-cobalt permanent magnets..

The ion optics system uses two dished molybdenum electrodes mounted to a titanium support ring The inner electrode facing the discharge chamber (called the screen grid) is 360 μm thick and has approximately I S,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 diameter 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.

The main discharge chamber and neutralizer cathodes are comprised of Mo-Re tubes with an outside diameter 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 are being developed at the Lewis Research Center to reduce differential charging on the international space station and have been successfully operated in the laboratory for more than 25,000 hours [Patterson].

The thruster discharge chamber is surrounded by a fine screen (called a plasma screen) which shields the thruster from its externally induced plasma During operation the temperature of the downstream surface of the thruster

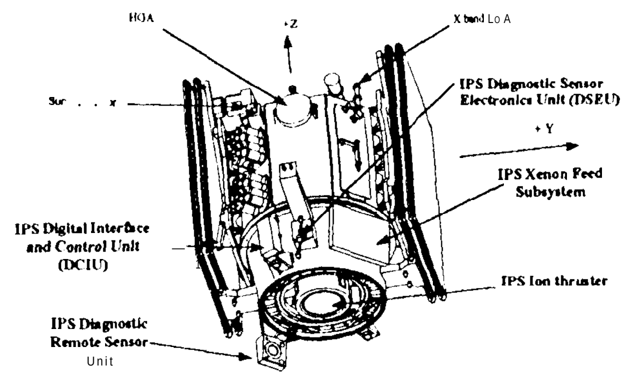

Fig. 2 New Millennium Deep-Space 1 in the stowed configuration.

plasma screen must be kept between -93 and 138°C. The non-operating temperature limits for the thruster are the same as the operating limits.

The NSTAR power processing unit can operate with input voltages over the range of 80 to 160 V with an efficiency of greater than 92°/0 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 non-operating 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.

The digital interface and control unit (DCIU) provides the interface between the ion propulsion system and the spacecraft and has the capability to operate up to 6 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 the 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 ability to operate over 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. [3]. A diagram of the NSTAR hardware integrated onto the Deep Space I spacecraft in the launch configuration is given in Fig 2.

improved-Performance NSTAR Systems

There are many potential ways in which the baseline NSTAR technology could be improved on and the objectives of the technology programs are to improve those aspects which provide the greatest mission benefits. Technology improvements can be roughly lumped into two categories: those which reduce the dry mass of the ion propulsion system; and those which improve one or more of its performance parameters (efficiency, thrust, or Isp). Engine service life and engine reliability are subsumed in the propulsion system dry mass category since deficiencies in either engine life or reliability must be overcome by adding additional engines (and associated hardware) at the expense of increased dry mass. These two system improvement categories are coupled since improved engine performance may be obtained at the expense of engine life or reliability and visa versa.

The projected end-of-fife performance of the NSTAR ion engine 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 its efficiency at the higher power levels. In addition, the current engine performance after 7,000 hours is significantly higher than that given in this table at the higher power levels. The maximum engine specific impulse,

Table 2 End-of-Life Engine Performance Projections

Engine Input Power (W)	Thrust (mN)	Isp (s)	Efficiency	Total Flow Rate (mg/s)
Projected NSTAR EOL Performance				
2305	91.6	3055	0595	287
2044	81.1	3068	0597	260
1.671	66.1	3097	0.601	2.06
1307	51,3	3024	0561	1.65
0951	357	2645	0,523	1.21
O 482	lQ.3	2015	0396	1,00
Projected Performance of the 14-cm Thruster				
730	27	3200	0575	0.85
670	24	3210	0.557	075
570	18	3500	0.552	054
520	17	3300	0.537	054
520	19	2900	0.52	067
450	16	3000	0.512	054
450	16	2750	0493	061
350	13	2500	0.457	054
270	9	2400	0396	039
250	9	2400	0413	037
200	6	2250	0365	029
120	4	2000	0327	020

however, is a mission driven parameter and NSTAR'S Isp of 3300 s is appropriate for the near-term missions of interest. Future, more demanding, higher AV missions will benefit from the development of higher Isp ion engines, and this must be done without compromising the engine service life capability.

Since there is little to be gained by attempting to improve the efficiency of the NSTAR 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 improving the engine total impulse capability.

Design and fabrication refinements could reduce the NSTAR engine mass by approximately 1 kg, Integrating the NSTAR PPU circuitry directly into a multifunctional bus may reduce the PPU mass by approximately a factor of two. 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 heavy plenum tanks. A complete flow controller-based feed system could be designed using the multi-function valves (MFV) and micro gas rheostats (MGR) currently under development by Marotta Scientific Controls, Inc. [13]. Such a system is expected to provide nearly a factor of three reduction in propellant feed system mass along with a substantial reduction in volume relative to the NSTAR feed system

The multi-function valves use a poppet actuated by a Terfenol-D magnetostrictive expansion rod and provide positive isolation, a 2:1 throttling capability, and are simple, rugged and have *a* high sealing force. The micro gas rheostat is comprised of micromachined capillary flow passages in a silicon chip contained within a metal housing. It provides a 3:1 throttling capability achieved by heating the chip to control the viscosity of the xenon, The MGR has no moving parts and is very small and light weight.

An ion engine gimbal design based on the Mars Pathfinder high gain antennae gimbal has been recently developed which should enable a factor of 2.5 reduction in gimbal mass relative to the NM DS 1 gimbal. This gimbal configuration has a +/-15 degree 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 8,000-hours at full power corresponding to a total impulse capability of 2.6×10^6 N-s and a maximum propellant consumption of 83 kg. Increasing the total impulse capability per engine by 50 % to 100 % would result in a corresponding reduction in the number of engines required to accomplish the mission, The near-term stimulus for this performance enhancement is the Champollion/DS4 mission, but it is expected that many other missions will also benefit from this performance enhancement.

The NSTAR program is currently executing a long-duration test of the NSTAR ion engine with the objective of demonstrating the full total impulse capability of the engine corresponding to 8,000 hours of operation at full power, This test, as of August 1, 1997, has successfully demonstrated 7,000 hours of operation and continues to run extremely well. There has been very little performance degradation observed over this time. Polk, using an in situ laser profilometer, has measured the erosion on the downstream surface of the accelerator grid as a function of time during the test. These data indicate that this grid erosion, once believe to be the major life-limiting mechanism for the thruster, is significantly lower than expected. In addition, the internal discharge voltage at full power is less than 24 V suggesting very low internal erosion rates. At the completion of this test a second long-duration test will be performed, The objective of this test is 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 due to 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 described above and probabilistic modeling of the principle failure modes [Polk, Brophy]. 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 due to operation at a significantly higher specific impulse.

Multimission SEP Module (MMSEP)

The DS4 mission requires the use of solar electric propulsion Therefore, this program must develop an SEP propulsion module capable of performing, at a minimum, a comet rendezvous mission. To potentially save money on other deep-space missions a multimission SEP module is being studied with the objective of using this for the Europa Orbiter and Pluto Flyby missions, as well as the DS4 mission for with little or no changes. Significant cost savings are anticipated through the reduction of non-recurring costs if such a MMSEP module is possible.

Analyses performed to date suggest that a MMSEP module for the Europa Orbiter, DS4, the Pluto Fly missions is possible and these missions can all be performed using a Delta II launch vehicle (the Pluto and DS4 missions would use the Delta 11 7925 and the Europa Orbiter would be launched on the higher performance Delta 11 7925 H). The current MMSEP module configuration consists of four NSTAR 30-cm diameter ion engines capable of processing a minimum of 120 kg of xenon each, two repackaged NSTAR power processing units, a repackaged DCIU, the NSTAR propellant feed system, multiple NSTAR-like xenon propellant tanks, and a two-axis gimbal mechanism for each thruster. The number of xenon tanks is variable depending on the required propellant loading for each mission.

The MMSEP module includes a light-weight solar array assumed to have a specific power of 100 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 is 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 tiring 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 have a submerged ocean of liquid water. One of the science objectives of this mission is to look for evidence of this liquid water ocean, The accomplishment of this goal requires orbiting Europa. The use 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. The chemical propulsion system, after the SEP system is jettisoned, is then used to perform the Jupiter orbit insertion maneuver and eventually deliver the spacecraft into orbit around Europa. The baseline, non-SEP mission is examining the use of an Atlas HAR launch vehicle to deliver a 260 kg spacecraft to Europa with a direct trajectory in about three years or a Delta H 7925 launch vehicle to deliver over 300 kg with a triple Venus gravity assist trajectory in just over six years.

The MMSEP module system with a 6-kW solar array can deliver bet ween 260 to 290 kg to Europa in 3.5 to 4 years using a Solar electric Venus-Venus Gravity Assist (SeVVGA) trajectory and a Delta 117925 launch vehicle. Over 300 kg can be delivered to Europa orbit using the

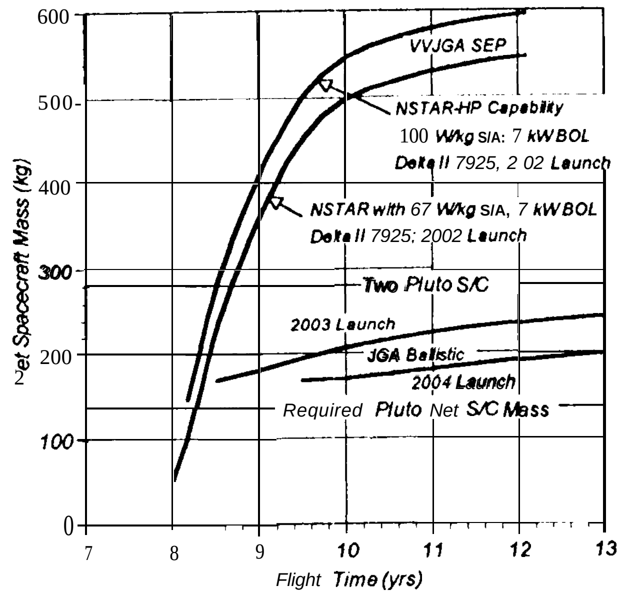

Fig. 3 **SEP performance for the Pluto Flyby mission.**

MMSEP module and the Delta 11 7925H launch vehicle. The low-thrust AV 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 one year trip time penality or a two-year reduction in trip time relative to the Delta H 7925 launched triple-Venus gravity assist chemical trajectory.

Pluto Flyby. Pluto is the only planet in the solar system which 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. Pluto may also be the first or best known Kuiper-belt object so a Pluto Flyby may also be a Kuiper-belt object flyby. The baseline non-SEP mission for Pluto uses a Delta 117925 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 to 12 years.

The common SEP module system with a 6.75-kW solar array at BOL can deliver the Pluto spacecraft in approximately 8.5 years using a Delta 11 7925 launch and a SeVVJGA (solar electric Venus-Venus-Jupiter gravity assist) trajectory for a launch in 2002 as indicated in FIG, 3. The low-thrust AV for this trajectory is about 9 km/s. Significantly, this SEP system could deliver two 145 kg spacecraft to Pluto in about 9.5 years using the same launch vehicle.

The common SEP module appears to be an attractive option for this mission especially if there is sufficient interest in the delivery of two spacecraft to Pluto. The SEP

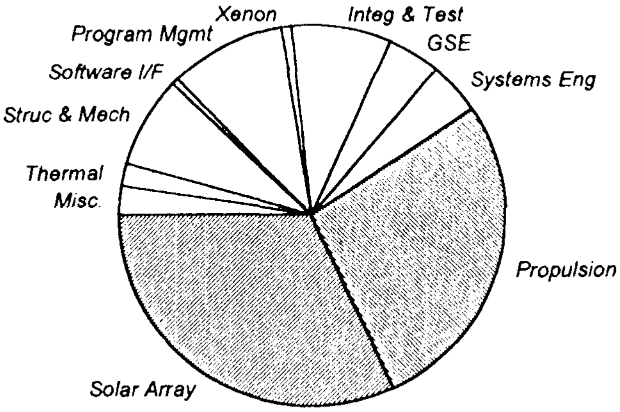

Fig. 4 SEP module cost distribution (first unit).

performance for a launch in 2004 is significantly poorer than in 2002. This is a result of Jupiter moving away from a position which provides significant gravity assist AV. After about 2005 Jupiter will not be available for gravity assists to Pluto for about 10 years. Missions to Pluto in this time frame will require more advanced (i.e. lighter) SEP systems.

Cost. Since SEP is not mission enabling for either the Europa Orbiter or Pluto Flyby missions it will be used only if provides a significant cost savings relative to non-SEP options. Potential cost savings from the use of SEP include enabling 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 have accrued during longer flight times. These savings must be sufficient to more than overcome the non-trivial cost of the SEP system.

Preliminary estimates of SEP module costs indicate that approximately 1/3 of the total cost is in the solar array, another 1/3 in the electric propulsion hardware (thrusters, PPUS, etc.) and 1/3 in everything else (engineering, integration, testing, program management, ground support equipment (GSE), etc.) as shown in Fig 4. Therefore, to reduce the cost of a SEP module it would appear to be most fruitful to concentrate on reducing solar array costs and the cost of the electric propulsion components. In addition, the MMSEP module is expected to significantly reduce non-recurring costs if the same or nearly the same design can be used for multiple missions.

Mid-Term Missions

Mid-term, deep-space missions are defined herein as those launching between 2005 and 2020. For mid-term missions two additional types of SEP systems appear to be desirable; one based on a scaled-down derivative of the NSTAR ion engine, and the other based on a high-Isp thruster with anode layer (TAL) [9, 10] operated direct-drive from a high voltage solar array

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 small planetary spacecraft continues. If it does, then a smaller, lighter SEP system based on a quarter-scale NSTAR ion engine, has been shown to provide significant benefits for these missions. Specifically it is expected that a quarter-scale SEP system will facilitate the use of launch vehicles smaller than the Delta 11 7326 for deep-space missions with small spacecraft. The quarter-scale engine in Ref. [Brophy] is assumed to be a 14-cm diameter ring-cusp ion engine with a modified magnetic circuit scaled down from the 30-cm diameter NSTAR engine. While Ref. [Brophy] considered only 14-cm diameter engines it is believed that the conclusions are not sensitive to the exact engine size within the range of approximately 12- to 18-cm diameter.

A technology program to develop a new ion engine should, in addition to simply scaling down the NSTAR engine, seek to improve certain performance aspects as well. Specifically, higher Isp operation, greater engine service life, and reduced engine mass appear to be attractive goals. The higher Isp is expected to benefit the higher AV missions anticipated in the mid-term time frame. Greater service life, as discussed above reduces the propulsion system dry mass by reducing the number of engines required. Finally, reducing the engine mass should be part of an overall effort to reduce the dry mass of ion propulsion systems.

Erosion-resistant carbon-carbon electrodes hold significant promise to enable increased engine life at the higher applied voltages necessary to obtain increased specific impulses without compromising engine life. 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. Carbon-carbon grids and their associated support structures are also significantly lighter than the same sized molybdenum grid based *accelerator* systems, thus, this technology will also help to significantly reduce the overall mass of ion engines since a significant fraction (almost 25°/0) of the engine mass is in the ion optics assembly as indicated in Fig. 5. 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

Figure 5 also indicates that almost 50% of the total engine mass is in the engine body (which includes the magnet rings) plus the plasma screen (labeled "Plasma Shield" in Fig. 5). A new approach to ion engine body fabrication is being developed under an SBIR contract with Energy Sciences

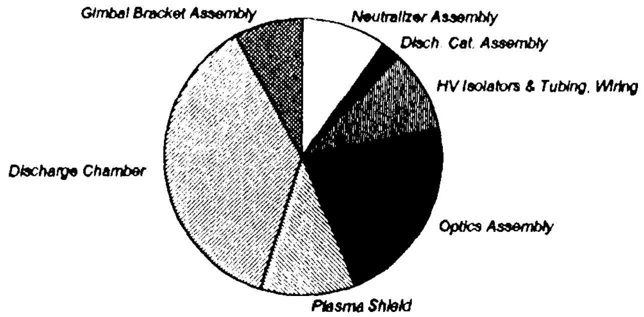

Fig. 5 NSTAR ion engine mass distribution.

Laboratory, Inc. [12]. This approach makes use of a fiber-core composite structure which is both strong and very light weight, as well as electrically insulating. The fiber-core composite consists of two very thin (50 μm thick) aluminum face sheets which 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 created by the flocked fibers and characterized by approximately one million nodes per cm^3. This produces a composite which has the strength of 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 1, OOO-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 atleast factor of two reduction in the combined engine body and plasma screen mass, as well as a reduction in the cost of engine fabrication due to a reduced parts count.

For mission performance assessment 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 diameter NSTAR engine (4.5 to 1 max. to min. 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 off a Taurus XL/Star 37. The masses in this table assume the use of the advanced propellant feed system described above and a very light-weight thruster gimbal mechanism. The 2-kg engine mass is very conservative and does not reflect the use of the fiber-core composite unibody fabrication approach, Mass mockups using the fiber-core unibody construction suggest that a 14-cm diameter ion engine could be fabricated with a mass of about 1 kg

Comet Rendezvous. The solar system has many different and interesting comets. To visit a variety of these different comets requires a propulsion system which can deliver a small science spacecraft to a comet from a small, inexpensive launch vehicle, Comparison of performance, in terms of the net spacecraft mass delivered, is given in Fig. 6 for the baseline NSTAR technology and the quarter-scale systems. 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 1.8 kW. The original trajectory calculations were performed by Sauer assuming a Delta 117925 and were scaled to the Taurus XL using the approach described in Ref. [15] The results in Fig. 6 show that the quarter-scale systems could deliver a 40 kg sciencecraft to many different comets. Note, the net spacecraft mass in this figure does not include the SEP system mass or the mass of the solar array. The flight times for these comet rendezvous range from 2,0 to 3.7 years with low-thrust AVS ranging from 7.5 to 14.5 km/s.

The significant advantage of the quarter-scale systems relative to the baseline NSTAR system is largely a result of the larger physical size of the NSTAR hardware and its ability to process more power than is required for these missions performed using the Taurus launch vehicle. The rest of the difference is attributable to the lighter-weight quarter-scale thrusters. Other analyses indicate that the full-sized NSTAR systems provide reasonable performance for Delta 117326 and larger launch vehicles.

Direct-Drive TAL Systems

To significantly shorten flight times to small bodies such as main-belt asteroids or comets requires the combination of high power, light-weight electric propulsion systems with small, light-weight spacecraft. Direct-drive Hall thruster systems offer the promise of providing significantly lower specific masses than conventional ion engine-based systems. This results from the fact that Hall thrusters require essentially only a single voltage input to operate steady-state, and that these thrusters have significantly higher thrust densities than gridded ion engines making them physically smaller and lighter, Direct-drive refers to a system in which the *single* 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, Otherwise, during steady-state operation, the solar array voltage is passed through unregulated. This significantly reduces the overall mass of the required power processor unit.

Table 3 Quarter-Scale NSTAR System Mass Breakdown

Two Engine Operation and One Redundant Engine

BOL Solar Array Power (kW) = 1.6 EOL Power (kW) = 1.2

Item	QTY	Unit Mass CBE (kg)	Contingency	Total CBE + Cont. (kg)
Ion Engines (14-cm dia.)	3	2.0	30	7.6
Gimbals (30% of Engine mass)	3	1.0	30	3.9
Digital Control I/F Unit (DCIU)	1	1.9	10	2.1
Power Processor Unit (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	included	8.0
Structure/Cabling per Engine	3	1.5	22	5.5
IPS Subtotal				48.0
Non-PPU Thermal Control (5% of Subtotal)	1	2.4		2.4
IPS Dry Mass				50.4
Solar Array Mass** (at 20 kg/kW)	1	31.2	included	31.2
Total Dry Mass				81.6
Propellant Mass	1	80	N/A	80
Total Wet Mass				*161.6*

IPS Specific Mass (kg/kW) 41.97

Total Specific Mass (kg/kW) 67.97

'*10%* of Propellant Mass Stored

● 'includes Articulation

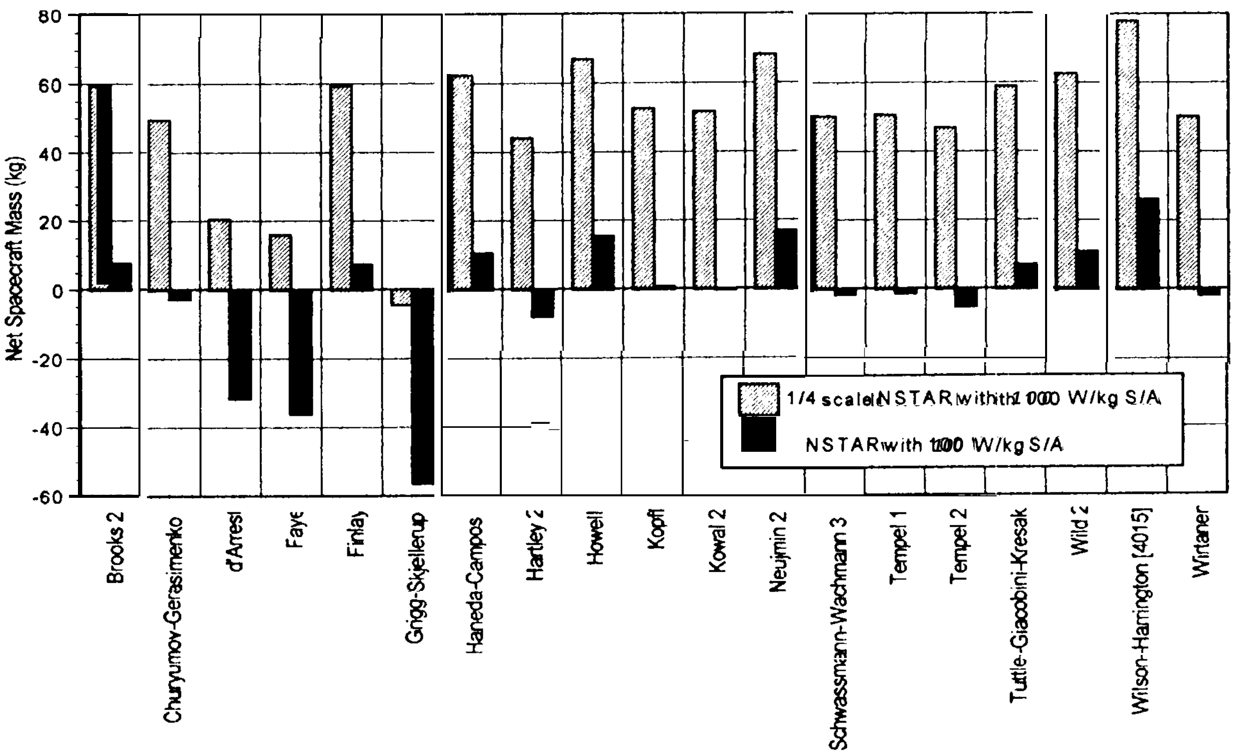

Fig. 6 Performance comparison between the quarter-scale NSTAR system and the baseline NSTAR for comet rendezvous missions assuming the use of a Taurus XL/Star 37 launch vehicle and a 1.8-kW BOL solar array.

The concept for direct-drive systems has been around for a long time, but has only recently became more realizable as a result of theavailability of the high performance Russian Hall thrusters. The first coupling of the direct-drive concept with Hall thrusters was proposed by Brophy in 1993 in recognition of the synergy between the ability of the linear concentrator solar array (under development by the Ballistic Missile Defense Organization) to operate at high voltages and the input voltage requirements of the Hall thruster. This concept was subsequently successfully tested at the Lewis Research Center.

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 around 2,500s is desirable for fast, small-body rendezvous missions. Space-charge limitations of the ion optics of gridded ion engines makes it very difficult for them to be capable of processing high powers at the relatively low Isp of 2,500 s. A single Hall thruster, however, could relatively easily be designed to operate at 10 kW and 2,500 s. This thruster would require an input voltage from the solar array of between 500 and 600 V. In contrast, a 10-kW ion engine would have to operate a much higher specific impulse with an applied voltage from the solar array of greater than 1500 V.

An unregulated, 10-kW, direct-drive PPU has been estimated to have a specific mass of 0.5 kg/kW [12] which is about a factor of ten less than the NSTAR PPU. This direct-drive PPU provides the ability to soft-start and shutdown the thruster, as well as providing fault protection. A breadboard direct-drive PPU has been fabricated and tested with the D-100 Hall thruster at up to 4.5 kW and 600 V. The measured efficiency of this PPU is over 99%.

While the thrust density and power handling capability of Hall thrusters is impressive, thruster life at a specific impulse of 2,500 s is a major concern. The authors believe that the thruster with anode layer [TAL] offers the best path to achieving the desired thruster lifetimes at this specific impulse. This opinion is based on the experience obtained in endurance testing the SPT- 100, D-55, and T-100 Hall thrusters, as well as running the D-100 at Isp's up to 2,700s.

A complete direct-drive TAL system requires a light-weight propellant feed system and light-weight thruster gimbal. The feed system based on the MFV and MGR discussed above can also be used with Hall thrusters. The combination of the relatively lightweight TALs with the direct-drive PPU and lightweight feed system is expected to result in these systems having the smallest specific masses of any mid-term SEP technology. An example point design for a DD-TAL system is given in Table 3 (for a Vesta sample mission return off a Taurus XL).

Table 3 Direct-Drive TAL Spacecraft Mass Breakdown

Item	Mass (kg)
10-kW TAL Engine	8.0
Engine Gimbal	1.9
Direct-Drive PPU	5.0
PPU Thermal Control	3.0
Digital Interface & Control Unit	0.5
Propellant Tank	32
Fixed Feed System	2
Feed System per Engine	1
Power Management & 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**

High Voltage Solar Arrays. The SCARLET linear concentrator solar array will fly on the New Millennium DS 1 spacecraft and produce an output voltage of between 90 and 120 V. In its DS1 configuration this array has a specific mass of about 50 W/kg, Technology improvements are expected to improve this to 70-80 W/kg The PAPS+ experiment [REF] tested several cell technologies in space for one year at +/- 500 V and showed that concentrator arrays were stable and resistant to radiation damage and plasma interaction. The present 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.

In addition, on outbound interplanetary trajectories the solar array output voltage will increase as the array temperature and output power decrease. 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.

Mission performance. To perform fast rendezvous missions to small bodies with inexpensive launch vehicles requires the use of a high power SEP system and a small spacecraft. 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 a fourth of this to Earth escape. The mass delivery capability to Earth escape is too small to accommodate a high power SEP system and

Table 4 Direct-Drive TAL Mission Performance

Item	Vesta	Ceres	Kopff
Flight Time (yrs)	1.3	13	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

its payload, 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 limited to less than 80 days.

The resulting mission performance for rendezvous missions to the main belt asteroids Vesta and Ceres and the comet Kopff are given in Table 4. The procedure used to calculate these performance values is described in more detail in Ref. [20]. 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 AV required for Earth escape. 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.

Finally, packaging the 10-kW solar arrays along with the SEP system and the spacecraft into the Taurus XL shroud is a major issue which was not addressed in this study, although there is a possibility that the significantly larger Delta 11 shroud could be used on the Taurus XL [24].

Conclusions

Ion propulsion is on the verge of entering the mainstream of propulsion options available for planetary missions. Continued investment is needed to develop high performance NSTAR derivatives to meet the needs of emerging, more demanding missions such as the Champollion/DS4 mission, This will provide benefits to other deep-space missions of interest, especially if spacecraft masses don't decrease significantly in the first decade of the next century. Simultaneously, investment in the development of the a scaled-down, advanced-technology NSTAR system with engines capable of operating at 1000 s higher Isp than the NSTAR engines will be needed to meet the anticipated needs of future small spacecraft and enable higher AV 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 a 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 which have a greater total impulse and higher specific impulse,

Direct-drive TAL systems offer the potential for the best performance at the expense of the highest development risk. A unique role for high-performance direct-drive TAL 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. The electric propulsion technology programs should investigate reducing the unknowns associated with high-Isp, direct-drive Hall thruster systems,

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