AIAA 97-2782

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33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit
July 6-9, 1997 / Seattle, WA

AIAA-97-2782

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The use of ion propulsion for deepspace missions is becoming a reality with the flight next year of the ion-propelled New Millennium Deep Space 1 spacecraft. This event is already stimulating the call for improved ion propulsion technologies, a trend which is expected to continue. This paper describes the examination of advanced solar electric propulsion technologies to determine their potential 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 and are compared to a baseline represented by the ion propulsion system which till fly on DS1. The results of this study indicate that significant benefits can be obtained by the development first of Improved versions of the DS1 ion propulsion system components. In addition, if the projected current trend to smaller planetary spacecraft continues, then the missions flying these small spacecraft will benefit substantially from the development of a scaled-down system approximately l/4* the size of the NSTAR which incorporates advanced technologies in the ion engine and the propellant feed system. The performance of the direct-drive TAL systems, while potentiality the superior to that of all other options did not appear within the constraints of this study to offer sufficient performance gains to offset the development risks.

Introduction

In the summer of 1998 NASA will launch the New Millennium Deep Space 1 (NM DS1) 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 improvement 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, where there are significant mission benefits enabled by an ion engine technology which has a larger total impulse capability than the NM DS1 NSTAR engine.

To determine the investment strategy in advanced ion propulsion technology that will be of the greatest benefit to future deep space missions a trades study was initiated. This study was part of a larger trades study looking at the relative benefits of advanced chemical propulsion, advanced SEP and solar sails. These propulsion options were evaluated according to the needs of projected future missions including those in the recent solar system exploration planning activities [5] This paper describes only the evaluation of SEP technologies for the larger propulsion tl-tides study and includes consideration of the following missions:

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1. Europa Orbiter
2. Pluto/ Kuiper Flyby
3. Solar Probe
4. Comet Rendezvous (19 different comets are included)
5. Main belt asteroid rendezvous (Vests and Ceres)
6. Comet Nucleus Sample Return (Tempel 1 and Tempel 2)
7. Main belt asteroid sample return (Vests)
8. Jupiter multiprobe
9. Mars sample return
10. Neptune orbiter
11, Mercury orbiter

Advanced SEP Technologies

Four different electric propulsion technology options are considered:

1. NSTAR: Baseline ion propulsion systems based on the SEP components as they will fly on NM DS1,
2. NSTAR-HP: High Performance NSTAR-derivative systems based on the 30-cm diameter engine and characterized by an engine with a higher totat xenon throughput capability.
3. NSTAR-QS: Quarter-Scale NST.AR-denvative systems based on a 14-cm diameter ion engine,
4. I) D-TAL: Direct-drive systems based on the use of a high-Isp version of the thruster with anode layer [6,7],

Baseline NSTAR Systems

The baseline NSTAR systems assume the use of the NSTAR hardware exactly as it wilt fly on NM DS 1 with the exception that multiple thrusters are allowed. The NSTAR hardware is capable of multiple thruster operation, but onty 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.

An example point design for a baseline NSTAR system is given in Table 1. In this table the current best estimates (CBE) for the NSTAR hardware are used along with the NSTAR project's estimates of the appropriate associated contingencies. In all of the point designs the SEP systems are assumed to be integrated into the spacecraft The structure required to support the SEP system components is assumed to be 7.5% of the SEP hardware dry mass and is book-kept elsewhere. In addition, the mass of the power management and distribution hardware is assumed to be part of the spacecraft and not part of the SE P system, The solar arrays are assumed to be gimbaled and the mass for this hardware is assumed to be included in the solar array mass estimates. The engine gimbal hardware mass is based on the four-bar gimbal mechanism designed by Haag for the 30-cm thruster [8] No mass allocation is made for unusable xenon propellants since this effect is expected to be small (the NSTAR NM DS1 xenon storage and distribution system is expected to have only 2 kg of unusable xenon). In addition, none of the point designs include xenon propellant contingencies beyond that represented by the conservative assumptions used in the trajectory analyses as described below.

NSTAR-HP Systems

The NSTAR-HP systems represent an extension to the baseline NSTAR technology that is characterized primarily by a significant increase in the engine total xenon throughput capability, The NSTAR-HP engines are assumed to be capable of processing 50 to 100°/0 more xenon relative to the baseline NSTAR engines. Therefore, each N'STAR-HP engine can process between 120 and 160 kg of xenon, 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,

In addition, in some cases the NSTAR-HP systems were assumed to use NSTAR engines with twice the maximum input power and thrust level of the baseline NSTAR engines. In other cases the effect on mission performance of having a high Isp NSTAR engine was examined

The NSTAR-HP systems are assumed to use lighter xenon feed systems based on the use of active propellant flow controllers. These feed systems are assumed to be the same as the ones assumed for the NSTAR-QS systems and are described in more detail below.

NSTAR-QS Systems

The NSTAR-QS systems are based on a quarter-scale NSTAR-derivative ion engine and were selected for inclusion in this study on the expectation that spacecraft for deep-space missions would continue to decrease in size and, therefore, a smaller, lighter SEP system would show a significant benefit for these missions relative to the basetine NSTAR systems, Specifically it was expected that a quarter-scale SEP system could enable the use of launch vehicles smaller than the Delta II 7326 for deep-space missions with small spacecraft

The quarter-scale engine 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 this study considered a 14-cm diameter engine it is believed that the end results are not sensitive to

Table 1 Example of a Baseline NSTAR System Point Design

Two Engine Operation and 3 to meet the total impulse requirements				
BOL Solar Array Power (kW) =	5.33	EOL Power (kW) = 4 1		
Item	**QTY**	**Unit Mass CBE (kg)**	**Contingency**	**Total CBE + Cont. (kg)**
Ion Engines (30-cm dia.)	3	8.2	10	27.1
Gimbals (30% of Engine mass)	3	2.5	30	9. 6“
Digital Control I/F Unit (DCIU)	1	1.9	10	2.1
Power Processor Unit (PPU)	2	11.9	—7	25.4
PPU Thermal Control	2	3.5	20	8.4
Fixed Xenon Feed System Mass	1	6.5	10	7.1
Feed System Mass per Engine	3	1.2	10	4.1
Propellant Tankage*	1	15.5	included	15.5
Structure/Cabling per Engine	3	5.9	22	21.4
IPS Subtotal				**120.7**
Non-PPU Thermal Control (5% of *Subtotal*)	1	6.0		6.0
IPS Dry Mass				126.7
Solar Array Mass** (at 15 kg/kW)	1	79,95	included	80.0
Total Dry Mass				206.6
Propellant Mass	1	155	N/A	155
Total Wet Mass				***361.6***
IPS Specific Mass (kg/kW)	30.90			
Total Specific Mass (kg/k W)	***50.40***			
*10% of Propellant Mass Stored				
**Includes Articulation				

Table 2 Example of a NSTAR-QS (Quarter Scale) System Point Design

Two E n - 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	7,8
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				

Table 3 Example of a Direct-Drive TAL System Point Design

Two Engine Operation and One Redundant Engine				
BOL Solar Array Power (kW) =	11.7	EOL Power (kW) =	9.0	
Item	QTY	**Unit Mass CBE (kg)**	**Contingency**	**Total CBE + Cont. (kg)**
TAL (2.3-kW)	4	2.5	10	11.0
Gimbals (30% of Engiine mass)	4	0.8	30	3.9
Digital Control I/F Unit (DCIU)	1	1,0	10	1.1
Direct Drive Power Processor Unit (PPU)	4	1.3	30	6.5
PPU Thermal Control	4	0.5	20	2.4
Fixed Xenon Feed System Mass	1	1 6	10	1 8
Feed System Mass per Engine	4	1.2	10	5.4
Propellant Tankage*	1	30.0	included	30.0
Structure/Cabling per Engine	4	4.0	22	19.5
IPS Subtotal				***81.6***
Non-PPU Thermal Control (5% of *Subtotal*)	*1*	*4.1*		*4.1*
IPS Dry Mass				***85.7***
Solar Array Mass** (15 kg/kW)	**1**	175,5	included	175.5
Total *Drv Mass*				**261.2**
Propellant Mass	**1**	300	N/A	300
Total Wet Mass				***561.2***
IPS Specific Mass (kg/k W)	***9.52***			
Total Specific Mass (kg/k W)	***29.02***			
*10% of Propellant Mass Stored				
● *Includes Articulation				
Assumes 2.3 kW max and 100 kg Xc/engine				

the exact engine size within the range of approximately 12- to 18-cm diameter.

Two sets of characteristics for the quarter-scale engine are assumed, one which covers the same Isp range as the baseline NSTAR engine and a second one which is assumed to operate at specific impulses up to 1000 s higher than the maximum NSTAR engine Isp of 3300 s. Carbon-carbon electrodes are assumed to be used to enable sufficient engine life at the higher applied voltages necessary to obtain a maximum specific impulse of 4300s for this second case.

The use of carbon composite electrodes and supporting structure for the ion accelerator system also helps reduce the overall mass of the ion engine, An engine mass of 2 kg was assumed with a 30°0 contingency. This represents a conservative estimate characteristic of what could be built using conventional NSTAR-like fabrication techniques and is consistent with the masses of 14-cm dia. laboratory model engines built at JPL.

A new approach to ion engine body fabrication is being developed under an SBIR contract with Energy Sciences laboratory. Inc [9]. 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 sandwich structure is approximately 5 mm thick and the resulting microtruss structure created by the flocked fibers has approximately one million nodes per cm^3. This produces a very strong, lightweight composite, With the use of glass or quartz fibers the aluminum face sheets are electrically isolated from each other. If successful, this will allow the inner surface to be at the roughly 1,000-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 a factor of two reduction in engine mass, as well as a reduction in the cost of engine fabrication due to a reduced parts count,

The maximum input power to each quarter-scale PPU is assumed to be 770 W; 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. This scaled-down PPU is assumed to be internally redundant where appropriate.

The quarter-scale xenon propellant feed system is based on the use of active propellant flow controllers, The complete feed system assumes the use of the multi-function valves (MFV) and the micro gas rheostats (MGR) currently under development by Marotta Scientific Controls, Inc. [10], and 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 example point design for an NSTAR-QS system is given in Table 2.

Direct-Drive TAL Systems

The direct-drive TAL systems are assumed to use high-Isp versions of the thruster with anode layer. In each of these systems the thrusters are assumed to be driven directly off a high voltage solar array. During steady-state operation the TAL requires only a single high voltage input to run. (Two electromagnet power supplies are also required during steady-state operation, but the mass and power of these supplies are small.) Operating the thrusters directly off a high voltage solar array eliminates most of the mass of the power processing unit mass. An unregulated direct-drive PPU has been estimated to have a specific mass of 0.5 kg/kW or 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 TAL at up to 4,5 kW. The measured efficiency of this PPU was over 99°/0,

Two different kinds of direct-drive TAL systems are considered in this study, The first assumes the use of T. AI, engines with a maximum input power of 1.1 kW and a total xenon throughput capability of 50 kg. The second assumes TAL engines with a maximum input power of 2,3 kW and a total xenon throughput capability of 100 kg. Both engine types are assumed to operate with specific impulses in the range of 1700 to 3000s.

The feed systems for the direct-drive TAL systems are assumed to be based on the same flow control components described for the NSTAR-QS and NSTAR-HP systems. 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 dry masses of any of the SEP technologies included in this study.

Missions

To investigate the effect of spacecraft size on the SEP system requirements and the relative benefits of the different SEP technologies, five different launch vehicles were included in the study: the Atlas IIAS, the Delta II 7925, the Delta 117326, the Taurus/Star 37, and the Pegasus XL/Star 21'. All of the launch vehicle/upper stage combinations are assumed to take the spacecraft and SEP system to Earth escape with a small hyperbolic excess velocity. The combination of 5 different launch vehicles, 4 different SEP technologies, and 32 different missions resulted in the need to generate several hundred SEP system point designs, three of which are shown in Tables 1-3.

Mission Analyses

For almost all of the missions considered in this study low-thrust trajectories were computed using SEPTOP [12] by Carl Sauer. SEPTOP currently represents the best low-thrust trajectory calculation tool available and includes realistic engine throttling characteristics, launch vehicle performance models, models of solar army characteristics versus solar range, and the ability to do multiple gravity assist trajectories[13]. All the trajectories considered in this study (except where noted) were performed using either the Delta II 7925 or Delta II 7326 launch vehicles, These trajectory results were then scaled to other launch vehicles based on their relative mass delivery capability to the same C3 for each trajectory. The scaling factors were determined by Kakuda [4] and are based on the five missions shown in Table 4. As indicated in this table the scaling factors are relatively insensitive to C3. This is to be expected provided the specific impulses of the chemical injection stages are similar for the different launch vehicles,

The SEPTOP trajectory calculations include the following deratings:

1. The launch vehicle is derated 8 to 10%
2. The beginning-of-life (BOL) solar array power is 1.3 times the required end-of-life (EOL) power referenced to 1 AU to account for radiation and micrometeroid aging effects. This 30°0 solar army degradation is assumed to take place at the start of the mission (i.e., the end-of-life solar array power is used throughout the mission).
3. End-of-life N STAR thruster performance is assumed for the entire mission The NSTAR thruster is currently being endurance tested and has completed over 6,500 hours of a planned 8,000-hr test [12].

Table 4 Launch vehicle scaling by injected mass

Launch Vehicle	Solar Probe	Pluto Express	Vests Rendezvous	Ceres Rendezvous	Kopff Rendezvous	COST
	(SEVVJGA)	(SEVVJGA)	(direct)	(direct)	(direct)	FY'99 Dollars
Flight Time (years)	55	10	25	3	32	
Total ΔV (km/s)	5.6	6.9	8.5	9.1	9 . 8	
C3	3.5	347	9.13	1.4	7.68	
	Injected Mass (ka)					
Atlas HAS	2500	2500	2250	2600	2300	105 to 145
Delta II 7925	1187	1188	1059	1237	1091	54
Delta II 7326 (Medlite)	608	608	534	636	552	44
Taurus/Star 37	266	266	235	281	243	35
Pegasus XL/Star 27	100	100	88	104	91	20
	Scaling Factor -- Injected Mass Relative to the Delta II 7326					Average
Atlas IIAS	4.11	4.11	4.21	4.08	4.17	4.14
Delta H 7925	1.95	1.95	1.96	1.94	1.98	1.96
Delta H 7326 (Medlite)	1	1	1	1	1	100
Taurus/Star 37	0.441	0.441	0.440	0440	0.440	0.44
Pegasus XL/Star 27	0.164	0164	0.165	0.163	0.165	0.16

4. Many of the trajectories derate the SEP system duty cycle by 1 0% to account for navigational coast periods or other spacecraft activities which may reduce the duty cycle of the SEP system.

Engine Performance Scaling

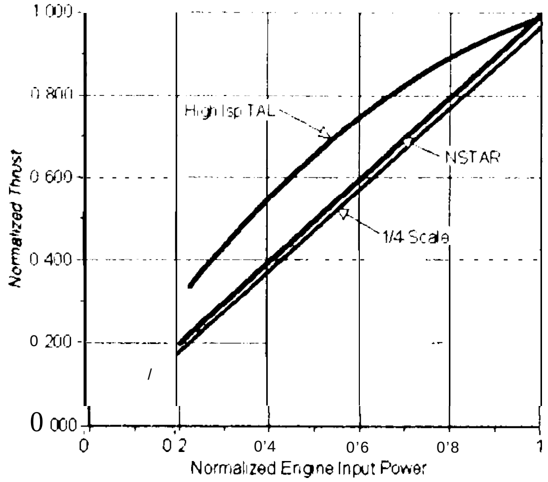
All Of the trajectory analyses were performed using the characteristics of the NSTAR engine (more specifically the projected end-of-life NSTAR performance characteristics). This is a result of the fact that JPL has now established a large database of low thrust SEPTOP trajectories to interesting destinations in the solar system based on the use of NSTAR engine-based SEP systems. It was not feasible to run new trajectories for each launch vehicle and SEP technology combination included in the study. Therefore, the behavior of each engine (thrust, Isp, and efficiency) vs. input power was artificially assumed have the same functional form as the NSTAR engine. In addition, the power per initial wet mass was assumed to be the same for each SEP technology for a given trajectory. These assumptions result in the same vehicle accelerations so that each spacecraft follows the same trajectory regardless of the SEP technology on board, In doing this the benefits of advanced SEP systems manifest themselves in terms of reductions of the SEP system dry mass only, which translates into larger delivered net spacecraft masses at the target. It also implies that if an improved SEP technology could provide significant reductions in trip time, this methodology would not reveal that capability.

The relative fictional forms for the assumed engine characteristics are given in Fig. 1 for the NSTAR, ¼-scale NSTAR, and high-Isp TAL thrusters, The data m these plots are normalized with the m aximum values for each parameter for each SEP technology These data indicate that the TAL exhibits a slower decrease m thrust and a faster decrease m Isp than the two ion engines. The efficiency variations of all three thrusters are not widely different. The fact that the curves in Fig. 1 do not all lie on top of each indicates that the analyses herein which assumes that they do will be in error. This error is appears to be the greatest for the TAL systems.

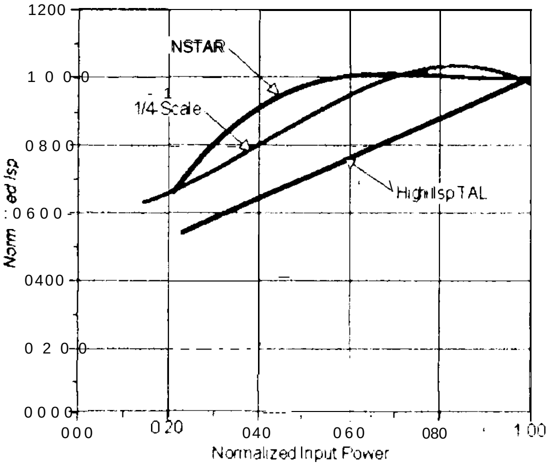
An added discrepancy for the TAL systems is that if the direct-drive PPU is unregulated, the increasing solar array output voltage with increasing distance from the sun will be passed onto the engine resulting in a higher applied voltage to the thruster. This will cause the engine to produce higher specific impulses at the same time the input power is decreasing. This is different from the ion engine throttling curves which show the Isp constant at the higher throttle levels before it decreases at the lowest throttle levels. The increasing solar array output voltage with solar range will make it harder to operate the TAL thrusters over the required dynamic power range (assumed to be 4.5 to 1). It is not clear to what extent TAL technology can be developed to meet the requirements assumed herein, but these requirements on the thruster could be relaxed substantially through the use of a "direct-drive" PPU which regulates the output voltage to the engine. This would come at the expense of an increase in the PPU mass, Nevertheless, direct-drive TAL systems are expected to offer the best overall mission performance, but at the expense of the highest development risk.

Solar Array Characteristics

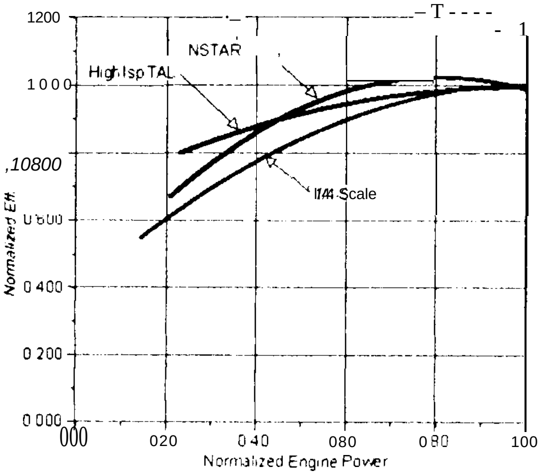
Two sets of solar array characteristics, in terms of specific mass vs power, were used in the analyses: a conservative set, and an advanced performance set. These specific mass curves are given in Fig. 2 The trajectory analyses assume that the solar array characteristics with solar range correspond to the APSA array with silicon cells. In addition, the high voltage solar arrays required for the direct-drive TAL systems are assumed to have the same specific masses as the lower voltage arrays. This is an added technology risk for the direct-drive systems



a. Normalized thrust.



b. Normalized Isp.



c. Normalized efficiency.

Fig. 1 Comparison of thruster characteristics.

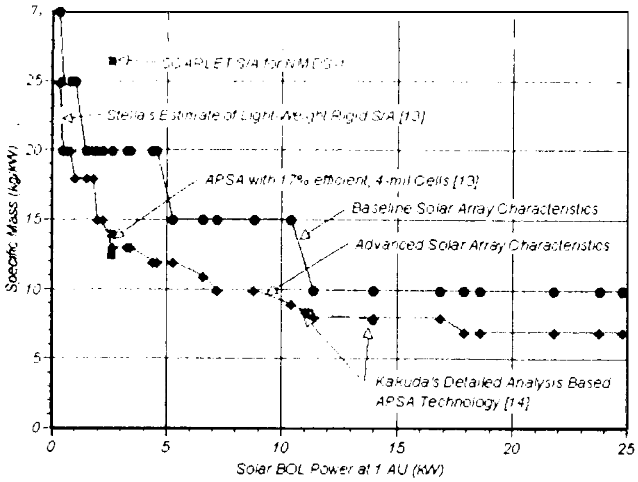


Fig. 2 Assumed solar array characteristics; baseline characteristic (circular symbols), and advanced characteristic (diamond symbols).

Results

The results comparing the performance capabilities of the different SEP systems are given in the following sections organized by destination,

Europa Orbiter

Europa is one of the moons of Jupiter and is suspected of have a submerged ocean of liquid water. One of the objectives of this mission is to look for evidence of this liquid water ocean, The accomplishment of this goat can best be done by orbiting Europa. The use of SEP 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 is 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 IIAR launch vehicle to deliver a 260 kg spacecraft to Europa with a direct trajectory in about three years or a Delta II 7925 launch vehicle to deliver over 300 kg with a triple Venus gravity assist trajectory in just over six years.

The baseline NSTAR system with a conservative solar army at 7-kW beginning of life (BOL) can deliver between 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 II 7925 launch vehicle. Note, the SE P system actually delivers the Europa spacecraft and a large chemical propulsion module to Jupiter. The SEP module is then separated from the spacecraft and jettisoned prior to the Jupiter orbit insertion maneuver.

An NSTAR-HP system with an advanced solar array at 7-kW BOL can deliver between 290 and 315 kg in the same time and the same trajectories. For this mission the propellant loading is such that the added throughput capability of the NSTAR-HP system does not help significantly and most of the performance improvement is obtained from the use of the advanced (i.e , lighter) solar array, Given that this mission is a near-term mission, (the trajectories in Fig. 3 assume a **2002** launch) the baseline NSTAR system technology appears to be an attractive option that may enable downloading the spacecraft from an Atlas IIAR to a Delta II 7925 (with a launch vehicle cost savings of roughly $50M) with only a slight trip time penalty.

The NSTAR-QS and D-TAL systems were not considered for this mission because it was believed that the technology could not be made available to support a possible 2002 launch and also because the near-term NSTAR or NSTAR-HP technologies are well suited to doing this mission

Pluto/Kuiper 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 II 7925 launch in 2002 or 2004 and a Jupiter gravity assist trajectory to deliver a 135 kg spacecraft to Pluto in approximately 10 years.

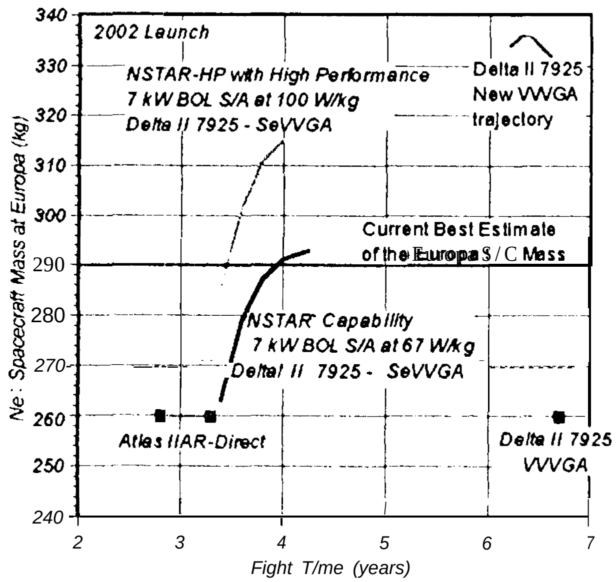


Fig. 3 SEP performance for the Europa Orbiter mission.

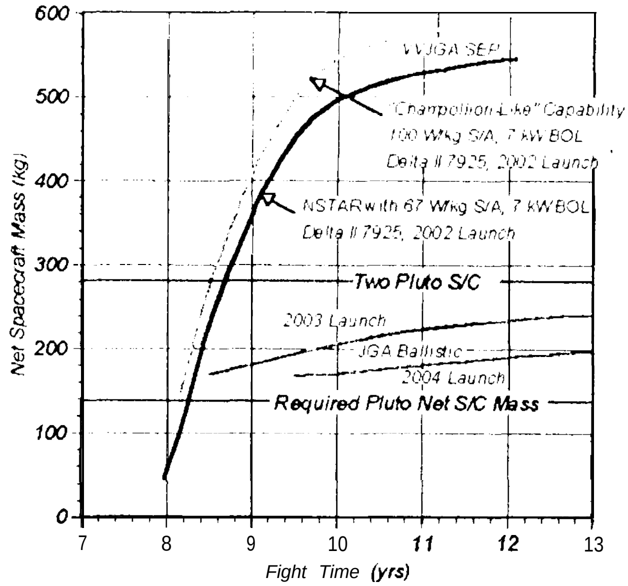


Fig. 4 SEP performance for the Pluto Flyby mission.

The baseline NSTAR system with a conservative solar array at 7-kW BOL 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. Significantly, this SEP system could deliver two 135 kg spacecraft to Pluto in about 9 years using the same launch vehicle. The NSTAR-HP system with an advanced solar array at 7-kW BOL shows better performance than the baseline NSTAR system, but the additional performance is not very compelling.

The baseline NSTAR system with a conservative solar army appears to be an attractive option for this mission especially if there is sufficient interest in the delivery of two spacecraft to Jupiter.

Like the Europa Orbiter mission the NSTAR-QS and DD-TAL systems were not considered for the Pluto Flyby mission because it was believed that the technology could not be made available to support a possible 2002 launch. In addition the near-term NSTAR or NSTAR-HP technologies are well suited to doing this mission,

Solar Probe

The solar probe mission seeks to deliver a spacecraft to within 4 solar radii of the sun at a 90 deg. Inclination. Two classes of trajectories to accomplish this mission with SEP have been examined The first is a solar electric Venus-Venus-Jupiter gravity assist trajectory where Jupiter is used to do the inclination change This trajectory requires the SEP system to provide a total ΔV of a little over 5 km/s. The second trajectory uses no gravity assists and never goes beyond about 2.5 AU from the sun This trajectory, however, requires a total ΔV from the SEP system of about 15 km/s or three times the SeVVJGA trajectory. The advantage, however. is that the flight time is only about 3.6 years instead of the 5.5 years for the SeVVJGA trajectory,

The baseline solar probe mission is investigating a Delta 11 7925 launched chemical/ballistic Jupiter gravity assist trajectory with a flight time of about 3.5 years. Ibis system approach can deliver the 150 kg solar probe spacecraft. The SeVVJGA SEP approach can deliver much more mass (about a factor of three) from the Delta II 7925 assuming a baseline NSTAR system, but at the expense of a 2 year longer flight time. Alternatively, a baseline NSTAR system could detiver the solar probe spacecraft in 5.5 years from the less expensive Delta II 7326 launch vehicle.

The non-gravity assist trajectory for this mission was identified in the 1970's, At that time, this trajectory made very optimistic assumptions regarding the mass of the SEP system components including the solar array. In addition, this study considered the delivery of very a large spacecraft (of order 1000 kg net spacecraft mass) with an initial wet mass of 5560 kg. in the 1990's version of this mission, the required mass to be delivered is 150 kg instead of 1000 kg and the initial wet mass is determined by the capability of the Delta 117925 which can inject approximately 900 kg to the same C3 used in the 1970 study to start the SEP trajectory (10.5 km^2/s^2). Seating the required power for the SEP system by the ratio of initial wet masses (900/5560) reduces the SEP system power requirement from 50 kW to a much more manageable 8.1 kW.

The other interesting thing that has happened in the last 25 years is that solar array technology has improved to the point where the very optimistic solar array assumptions used back then are now very reasonable. The end result is that the use of SEP for solar probe flying the high AV trajectory looks promising, However, the high ΔV requirement for this trajectory means that a higher Isp from the ion propulsion system would be desirable, Indeed, the 1970's study assumed a maximum Isp for the ion engines of 4300 s, nearly 1,000 s greater than the current NSTAR design Fortunately, for ion propulsion it is relatively easy to increase the specific impulse even by an amount as large as 1,000 s. To do so, however, requires redesigning the PPU (to output a higher voltges) and a re-evaluation of the engine grid life due to the higher applied voltages

An estimate of the net spacecraft mass delivery capability to 4 solar radii for the high ΔV SEP trajectory is given in Fig. 5 for three cases: a high Isp (4300 s) version of the NSTAR engine with the conservative solar, a high Isp NSTAR engine with an advanced solar array; ancl a high Isp NSTAR-QS system with an advanced solar array. This chart indicates that the NSTAR-QS system can deliver approximately 80 kg more than the current chemical/ballistic mission using the same launch vehicle. Current planning

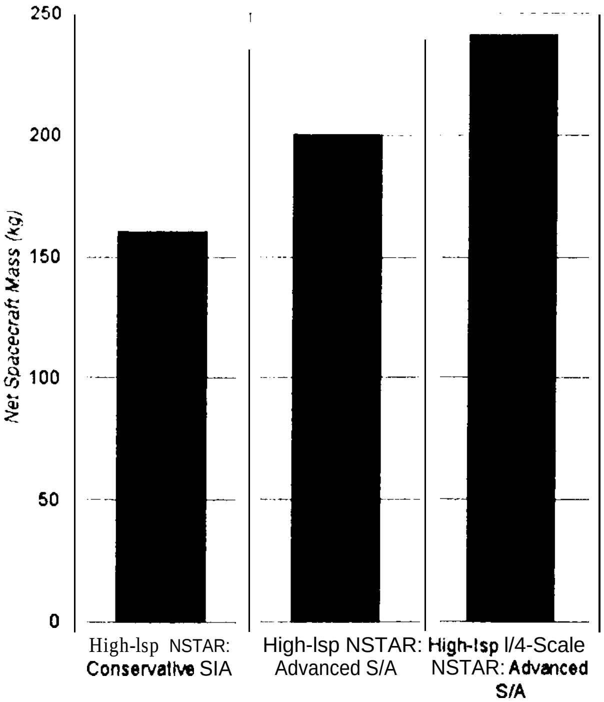


Fig. 5 Solar Probe net spacecraft masses for different SEP technology options assuming a Delta II 792S launch, 8.1-kW BOL solar array and a 3.6 yr flight time.

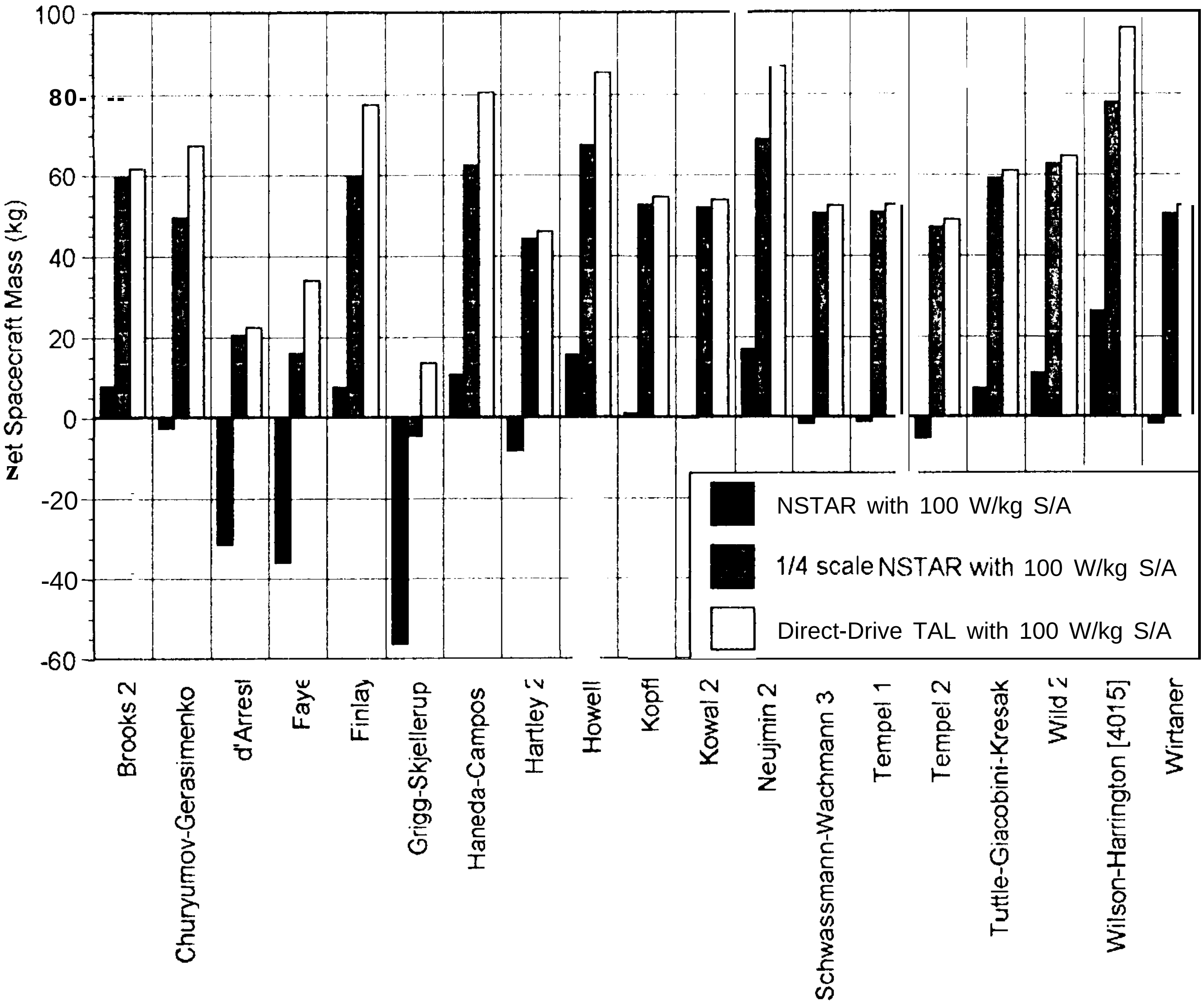
has the solar probe mission flying sometime in the year 2006 giving plenty of time to develop the NSTAR-QS system.

It should be noted that the earlier SEP trajectory calculations are not as accurate those that could be obtained using SE PTOP. Therefore, the high ΔV trajectory needs to be re-run using SEPTOP before the evacuation of using SEP for solar probe can move ahead.

Comet Rendezvous

The solar system is awash in comets, To visit many different comets requires a propulsion system which could 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, NSTAR-QS, and DD-TAL systems. These data were obtained assuming a Taurus XL/Star 37 launch vehicle and an advanced solar array with a BOL power level of 1.8 kW and show that the NSTAR-QS systems could deliver a 40 kg sciencecraft to many different comets.

The significant advantage of the NSTAR-QS system relative to the baseline NSTAR system is largely just a result of the larger physical size of the N STAR hardware and its ability to process more power (ban is required for these missions performed using the Taurus launch vehicle. The



Fig, 6 Comet rendezvous mission performance comparison for a Taurus XL/Star 37 launch and a 1.8-kW BOL advanced solar array.

NSTAR systems appear to provide reasonable performance relative to the other SEP technologies for the Delta II 7326 and larger launch vehicles. The flight times for these comet rendezvous range from 2.0 to 3.7 years.

The DD-TAL systems are shown to offer better performance than the NSTAR-QS systems, but not sufficiently better to warrant the investment in the DD-TAL technology for these missions. Finally, packaging the 1.8-kW solar arrays along with the SEP system and the spacecraft into the Taurus XL/Star 37 shroud is a major issue which was not addressed in this study.

Comet Nucleus Sample Return

The first comet sample return mission will be Champollion/New Millennium DS 4 which will demonstrate the technologies required to perform a comet sample return. Solar electric propulsion is enabling for this mission which wilt use the SEP system for both getting to and departing from the comet. There is a significant benefit to this mission obtained by using higher throughput ion engines. The near-term reality of this mission is the primary driver behind the need to develop the NSTAR-HP technology.

Main belt asteroid rendezvous

The solar system is also loaded with main belt asteroids. Like cornets, to visit a lot of them requires inexpensive delivery of many small spacecraft to different asteroids. SEP mission per formances are compared in Fig. 7 for Vests and Ceres rendezvous missions using a Taurus XL/Star 37 launch and a 2.()-WV advanced solar array. The flight time to Vesta is 2.5 years and 3 years to Ceres. The Ceres trajectory also includes a Mars gravity assist.

These data show a significant benefit for the NSTAR-QS systems relative to the baseline NSTAR systems. This, again is simply a result of the larger physical size and mass of the NSTAR hardware, The NSTAR systems show good performance for launch vehicles the size of the Delta 117326 or larger. The DD-TAL systems exhibit the best performance, but again the performance gain does not appear sufficient compelling to warrant the development risk.

Main belt asteroid sample return

SEP system performance is compared for a sample return mission to the main belt asteroid Vesta in Fig. 8 for a Delta 117326 and 4.5 kW of power from an advanced solar army. The figure of merit here is the net spacecraft mass at Earth return. The net spacecraft mass refers to everything that is not SEP (the solar array is assumed to be part of the SEP system). This trajectory assumes that the SEP system

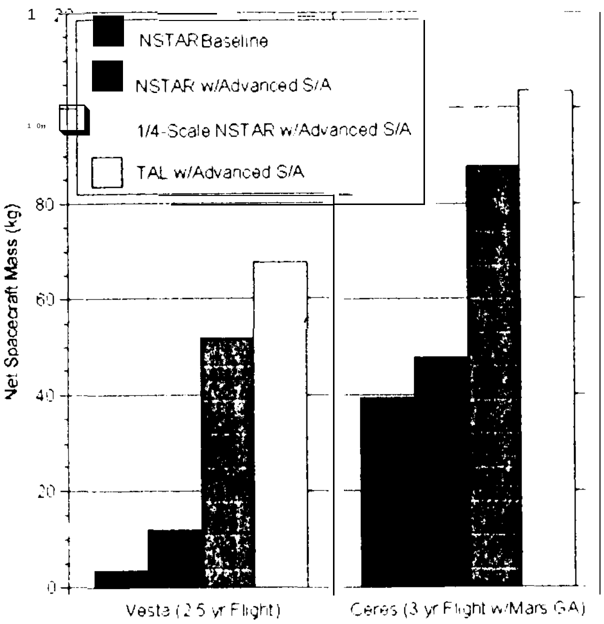


Fig. 7 **Comparison of SEP technologies for Vesta and Ceres rendezvous missions assuming a Taurus XL/Star 37 launch and a 2.0-kW BOL advanced solar array.**

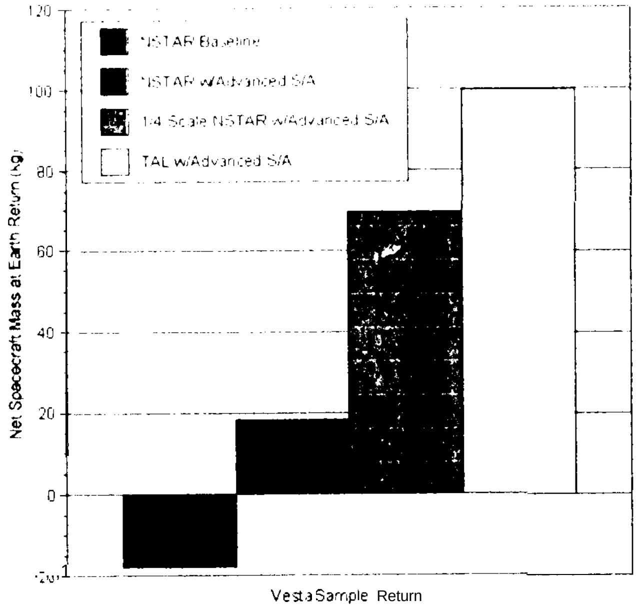


Fig. 8 Relative SEP performance for a Vesta sample return assuming a Delta 117925 launch and a 4.5-kW advanced technology solar array.

is used for both the outbound and return legs of the mission. The total flight time is 5 years. Significantly higher net spacecraft masses at Earth arrival could be achieve by going up to the next size in launch vehicle to the Delta II 7925 and increasing the BOL solar array power to 8.8 kW.

Jupiter multiprobe

The Jupiter mulitprobe mission seeks to distribute multiple probes into Jupiter's atmosphere. If an SEP system is used for this mission the performance and SEP technology benefits will be similar to those for the Europa orbiter mission. That is, this mission could be accomplished well using the baseline NSTAR hardware. The advanced SE P technologies considered here are not significantly beneficial for this mission as shown in Fig. 9 for a Delta II 7925 launch vehicle and a BOL advanced solar array of 6.5 kW The SE P flight time for this mission is 3.5 years for SeVVGA trajectories.

Uranus Orbiter

A Delta 117326 launch and a BOL solar array of 4.5 kW results in the SEP mission performance given in Fig. 10 for solar electric Earth-Earth gravity assist trajectories. The significantly longer flight times to Uranus compared to Pluto are the result of the Jupiter gravity assist for the Pluto trajectories. The net spacecraft mass, in this case refers to the mass delivered by t he SE P system (not including the

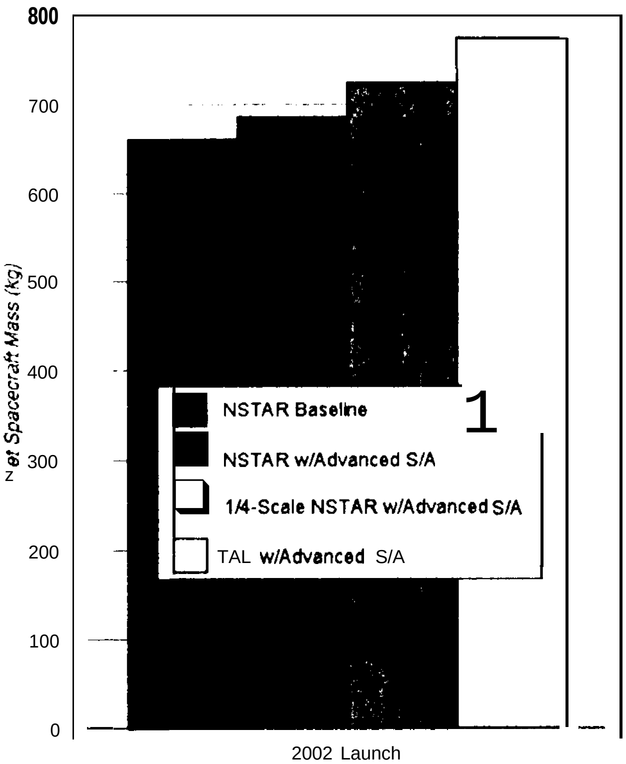


Fig. 9 Comparison of SEP technologies for a Jupiter multiprobe mission and a Delta II 7925 launch vehicle and a BOL power of 6.5 kW.

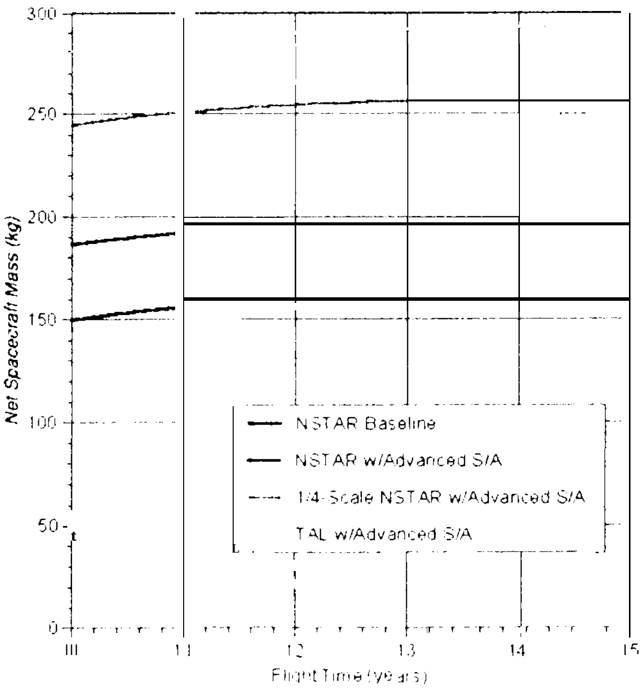


Fig. 10 SEP performance comparison for a Uranus Orbiter mission for a Delta 117326 launch vehicle and a BOL solar array power of 4.5 kW. The net spacecraft mass is prior to the Uranus orbit insertion maneuver.

SEP system) prior to the Uranus orbit insertion maneuver The orbit insertion could be accomplished chemically m with an aeroassist approach There is little effect of flight time on the SEP delivered mass The NSTAR-QS shows a significant benefit relative to the baseline NSTAR for this mission and launch vehicle. For the larger Delta II 7925 launch vehicle there is essentially no difference in the NSTAR baseline and the NSTAR-QS performances.

Neptune Orbiter

SEP missions to Neptune would fly the same SeVVJGA trajectories that are used for the Pluto missions, and since Neptune is a approximately the same distance from the sun as Pluto the trip times should be comparable. To get into Neptune orbit will require a combination of aeroassist technologies and chemical propulsion. Shorter flight times to Neptune may be possible using the high power concept described by Noca [15] in which a small spacecraft is propelled by a high power SEP system.

Mercury Orbiter

Mercury orbiter mission performance is given in Fig. 11 for a solar electric Venus gravity assist (SeVGA) trajectory using a Delta 117326 launch vehicle and a BOL power of 1.7 kW. The NSTAR-QS system shows about a factor of two better performance than the baseline NSTAR system. However, an NSTAR-derivative system which uses the 30-

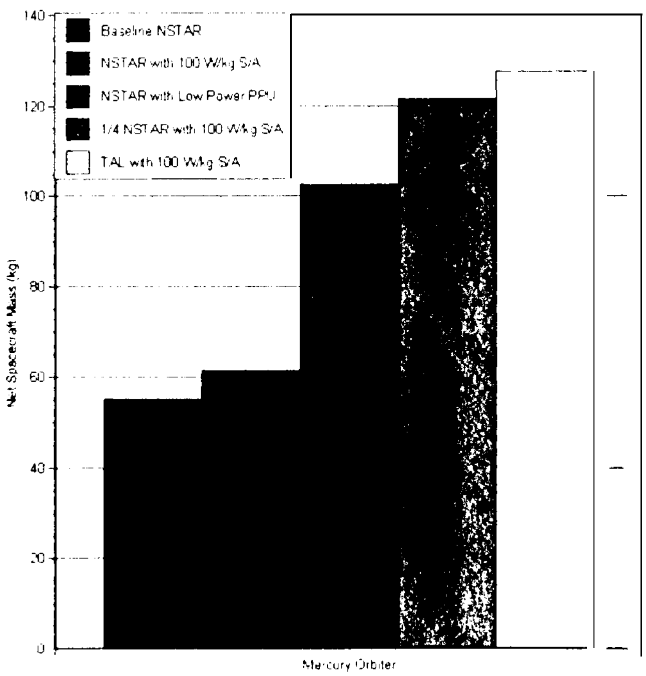


Fig. 11 Mercury orbiter SEP performance comparison for a Delta II 7326 launch, a '2.1 yr night time and a BOL solar array power of 1.7 kW.

cm diameter sized thruster, but redesigns the PPU to process a fixed input power of 1.7 kW instead of 2 5 kW results in a significant improvement m mission performance. A fixed input power is feasible for inbound trajectories where the solar array power increases during the mission. In this case the NSTAR thruster would always be running at an power of 1.7 kW input to the PPU. This throttled operation should facilitate keeping the ion engines cool on the inbound trajectory and is probably the preferred approach

Future Advancements

The future for electric propulsion is likely to be characterized by larger, lighter, less expensive solar arrays. Improving solar array technology will enable new mission/trajectory approaches. For example the use of high performance, high power **SEP** systems on small spacecraft will enable the use of SEP to start deeper in the Earth's gravity well. This will significantly extend the payload capabilities of small launch vehicles. A good candidate for the high performance SEP systems required for this are the DD-TAL systems described in this paper. Noca [18] has shown that such DD-TAL systems can provide very rapid transportation of small spacecraft (or order 50 kg) to the main asteroid belt and to short period comets.

Conclusions

The following conclusions and recommendations are made based on this propulsion trades study. NASA should invest in the development of the high performance NSTAR-HP technology first as required by 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 fist decade of the next century

Simultaneously, NASA should invest in the development of the quarter-scale NSTAR-QS technology with engines capable of operating at 1000 s higher Isp than the NSTAR engines. This will meet the anticipated needs of future small spacecraft and enable higher ΔV missions (> 15 km/s). Also, the use of multiple smaller engines facilitates operating engines pairs to provide spacecraft role control and reducing gimbal requirements.

If the trend toward smaller spacecraft continues in the future and ifit is desirable to launch these smaller spacecraft from launch vehicles smaller than the Delta II 7925, then there is a significant payoff from the development of a scaled-down N STAR-derivative technology. This study considered a ¼-scale NSTAR system, but the results are expected to be relatively insensitive to the exact scale-down Factor. More difficult missions contemplated for the future such as comet and asteroid sample returns, solar probe, and the multiple main belt asteroid rendezvous, require the development of NSTAR-derivative engines which have a greater total impulse and high specific impulse

Direct-drive TAL systems offer the potential for the best performance (and also the highest development risk), but the performance gains over the nearer-term NSTAR-derivative systems are probably not compelling enough to warrant the required investment to develop this technology. However, a unique role for high-performance direct-drive TAL systems may be in enabling very short trip time missions to be performed from Pegasus 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 solar system exploration,

Acknowledgments

The author **thanks the** numerous people who contributed to the information contained in this paper including Roy Kakuda, Carl Sauer, Robert Gershman, Jon Sims, Muriel Noca, John Beckman, Tom Haag, Hoppy Price, Dara Sabahi, and Stacy Weinstein.

The work described in the paper was conducted at the Jet Propulsion Laboratory under contract with the National Aeronautics and Space Administration.

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