NEXT GENERATION ION ENGINES: MISSION PERFORMANCES

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In 2001, NASA released a Research Announcement for the Next Generation of Ion Engine Technology. As specified in this Research Announcement, significant technology advances over the NSTAR DS1 ion engine were sought, especially an increase in specific impulse, total impulse, power and efficiency, and a decrease in propulsion dry mass. Two ion engine designs, one based on a derivative of the NSTAR 30-cm and the other one based on a 40-cm ion engine design, were identified as potential next generation technologies. This paper summarizes the characteristics of the three technologies in questions, and their mission performances for Solar System Exploration and Primitive Bodies Exploration missions: Neptune Orbiter, Titan Explorer, Jupiter/Europa mission, Comet Kopff Rendezvous, Asteroids Vesta-Ceres Rendezvous, and Comet Nucleus Sample Return. These new mission analyses using the next generation of ion engines show enabling benefits to Discovery type and New Frontiers type missions.

INTRODUCTION

After a development history spanning nearly forty years, the first use of solar electric propulsion (SEP) for primary propulsion on a deep-space mission began with the launch of the Deep Space 1 (DS1) spacecraft on October 28, 1998 [1]. This event marks a major milestone in the development of advanced propulsion for deep-space missions. The DS1 spacecraft uses a single-engine ion propulsion system (IPS), provided by the NASA Solar electric propulsion Technology Applications Readiness (NSTAR) project, as the primary on-board propulsion system. This propulsion system is designed to deliver a total ΔV of 4.5 km/s to the 486-kg (initial wet mass) DS1 spacecraft while consuming only 81 kg of xenon.

Several scientifically interesting and energetically demanding deep-space missions are now looking to the use of ion propulsion to significantly reduce total mission costs. Because these missions are more difficult, from a propulsion standpoint, than those used to justify the development of the NSTAR IPS technology, they benefit significantly from improvements to the ion propulsion technology that flew on DS1. Typically, the greatest overall benefit comes from increasing the total impulse capability per engine. As the engine total impulse capability is increased, fewer engines are required for a given mission resulting in substantial savings in mass and cost. Additional savings may be obtained for some missions by increasing the maximum engine specific impulse, resulting in significant propellant mass savings.

In 2001, a NASA Research Announcement for the next generation of ion engine technology was released. This paper describes the results of a trade study that was performed to identify the propulsion technology that would best fit various missions representative of potential future flight projects. Six missions were picked to cover a wide range of mission profiles: two "Discovery" type missions, two "New frontiers" type missions, and two "Flagship" type missions. The approach to evaluate the mission benefit of the ion propulsion option will be to calculate the net delivered mass (mass without the propellant or propulsion system) at destination for various EP power levels at a fixed optimized flight time. In order to quantify the net delivered mass, a model of the ion propulsion dry mass is constructed. This model will be used with the trajectories. This paper will discuss first the propulsion technologies, followed by a description of the ion propulsion module design assumed. Finally, the mission trade results will be presented.

PROPULSION SYSTEMS DESCRIPTION

Assumptions on the Ion Engine Technologies

Three propulsion technology performances, as identified in Table 1, have been investigated against the demonstrated NSTAR technology to determine which engine and system technology improvements would provide the greatest mission benefits without introducing unacceptable technical risk. These technologies include a high specific impulse (Isp) advanced version of the NSTAR thruster, a high-power derivative of NSTAR and high power (10-kW) 40-cm diameter ion engine. The characteristics of the NSTAR technology can be found in many references [1, 2, 3, 4]. The two derivatives of NSTAR have been presented in [5], and were named NGN-2 and 3 for "Next Generation NSTAR". Finally, the details of the high power (10-kW) 40-cm diameter ion engine can be found in [6] and have been the basis for this analysis. More recent results on this last technology can also be found in [7, 8], but they were published after this analysis was performed. These thruster options are characterized by differences in four major parameters: engine diameter, engine input power, maximum specific impulse, and engine total impulse (or throughput) capability. Figure 1 shows the Isp as a function of PPU input power and thruster efficiency of NSTAR and the ***projected*** Isp and efficiencies of the three thrusters in question.

Table 1: Ion engine main projected characteristics

	NSTAR	NGN-2	NGN-3	10-kW 40-cm
Max. thruster processed power (kW)	2.3	3.1	5	10
Engine diameter (cm)	30	30	30	40
Maximum Isp (sec)	3100	3800	5000	3900
Xe throughput (kg)	130	200	250	550

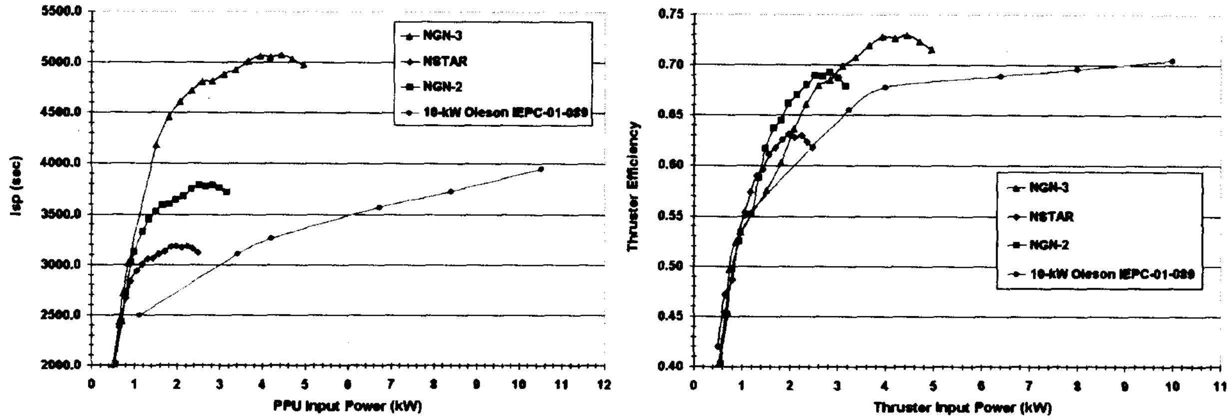

Figure 1: Ion thrusters specific impulse and efficiency projections

As can be seen in Figure 1, the thrusters chosen for this analysis cover a wide range of Isp. Their diversity will help determine which mission they benefit the most. They also represent a significant jump in thruster efficiency compared to NSTAR.

Systems Description

One of the objectives of this study was to include in the trade as much detail as possible of the ion propulsion system (IPS). Thus the ion propulsion system was designed more as a propulsion module than just thrusters and power processing units. Figure 2 shows a simplified block diagram of the NSTAR IPS (single string). To that basic configuration was added redundancy, structural and thermal considerations. Figure 2 also shows an example of what the IPS module designed here could look like.

As figure 2 shows, the NSTAR ion thruster uses xenon propellant delivered by the Xenon Feed System (XFS) and is powered by the Power Processing Unit (PPU), which converts power from the solar array to the currents and voltages required by the engine. The XFS and PPU are controlled by the Digital Control and Interface Unit (DCIU), which accepts and executes high-level commands from the spacecraft computer and provides propulsion subsystem telemetry to the spacecraft data system. To accommodate variations in the solar array output power with distance from the sun, the NSTAR IPS was designed to operate over a PPU input power range of 580 W to 2500 W, with input voltages in the range of 80 to 160 V.

We will now describe the solar array technology assumed in this study. A layout of the assumptions that went into conceptually designing the ion propulsion module is provided next. This model of the ion propulsion system will then be used with the trajectories to determine the mass left to the spacecraft.

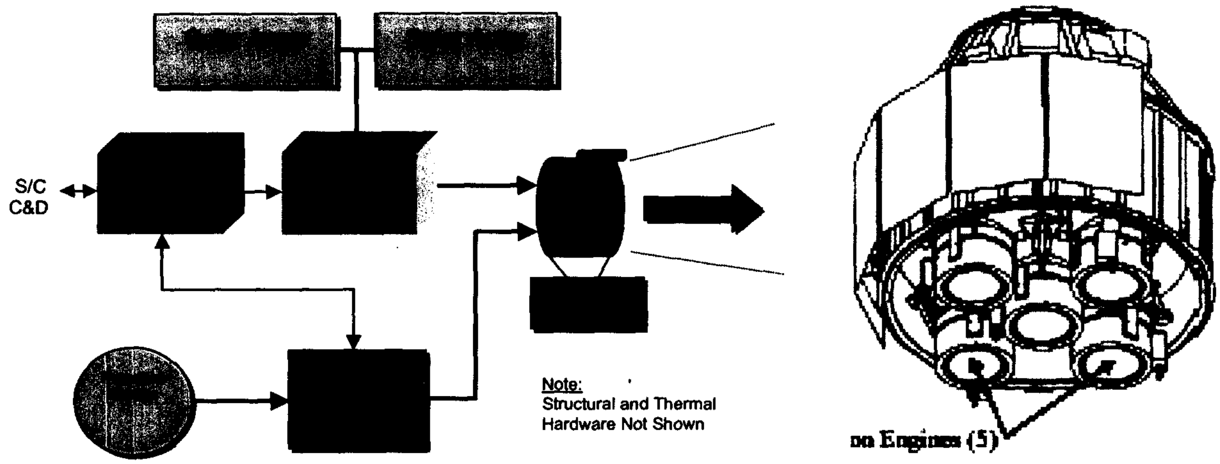

Figure 2: Ion propulsion module block diagram and conceptual configuration for system sizing

Solar Arrays

The solar arrays were sized based on a projection of the AEC-Able Ultraflex array capability. Since this array technology scales with power from ~ 1 kW up to ~ 30 kW, it was used as a representative potential technology for SEP applications. The decrease in specific mass (kg/kW) of this array was also taken into account since there is a scaling benefit for increased power for this array type: the specific mass was assumed 100 W/kg at 1 kW and 174 W/kg at 20 kW. A 14% degradation factor was applied to the array Beginning-of-Life (BOL) power to account for various degradation phenomenon. Also, in order to support power demand during launch, a primary battery was used prior to solar array deployment.

System masses

Since the system masses are function of mainly power level, launch mass and propellant mass, each trajectory was uniquely considered and had a system mass associated with it. The component and subsystem sizing assumptions are given in Table 2. To be consistent with the JPL conceptual design guidelines, 30% mass contingency was applied to all spacecraft subsystems, and a 10% launch vehicle margin was assumed.

The number of thrusters and PPUs was calculated on the basis of power requirements by the trajectory and thruster propellant throughput. The system architecture followed a conventional approach with parallel strings of PPUs and thrusters. Each PPU is cross-strapped to two engines. One spare ion engine, one spare PPU and DCIU were also included for single-fault tolerance. Each thruster was gimbaled separately. The PPUs were assumed to be 95% efficient.

The tankage fraction was calculated assuming cylindrical composite tanks. Those tanks have a propellant storage efficiency (Tank Fraction TF) of about 2.5% for Xenon when stored as a supercritical gas (~2000 psia). Furthermore, a 10% propellant contingency was added to the deterministic propellant mass to account for flow rate characterization, residuals, attitude control and margin.

The structures/cabling mass are not based on a specific design but rather on a percentage of the subsystems to which the structures apply. These percentages are based on historical data. In addition, the SEP module structure includes mass to hold the spacecraft above during launch loads, mass to hold the tanks, and mass for system assembly hardware (bolts, epoxies, tie downs...). The spacecraft side of the launch vehicle adapter is also included in the SEP module structure.

Table 2: Ion propulsion module masses (without contingency)

	NSTAR 2.3 kW	NGN-2 3.1 kW	NGN-3 5-kW	10-kW 40-cm
Thruster mass (kg)	3	7	5	12
PPU mass (kg)	13.6	15	15	27
Radiator mass (kg)	5	3.5	5.1	9.8
Other module subsystems:				
Attitude control	Fixed mass for driver motors			
Power	Solar array + high voltage power management			
Other propulsion elements				
Digital Control Interface Unit	New design			
Tank	2.5% of propellant mass			
Feed system	Fixed part, variable part = f(# of thrusters) + tubing, fittings...			
Thermal	Radiators per PPU + thermal blankets + heaters...			
Structure				
Propulsion	26% of propulsion system + gimbals and actuators			
Propellant tank	4% of propellant mass			
Power and other subsystems	16% of power and other subsystems			
Payload support and interface	6% of payload mass			
System assembly hardware	2% of all structure			
S/C adapter	1.5% of launch mass			
Cabling	Various harnesses + PPU to engine cables + fraction of other subsystems			

MISSION RESULTS

Once the ion propulsion module was modeled, trajectories could be optimized to determine the net delivered payload at destination and the corresponding flight time. Carl Sauer (JPL) ran a trajectory optimization code named SEPTOP for Solar Electric Propulsion Trajectory Optimization Program, which is based on the calculus of variations. This code optimizes two body interplanetary trajectories and can model discrete numbers of operating Xenon thrusters throughout the trajectory. The number of operating thrusters is switched by SEPTOP in an optimal fashion. Additionally, SEPTOP throttles the thrusters in power as required by that available from the Sun. Carl allowed for a coast time duty cycle of 10% to simulate times when the spacecraft is not thrusting due to housekeeping activities, and assumed a constant 250 W from the solar arrays for the spacecraft.

Thus, the four ion thruster technologies were evaluated against six different missions. Two of these missions correspond to the "Discovery" class missions: a Comet Kopff rendezvous and a two-asteroid rendezvous mission namely Vesta and Ceres. Both these missions need relatively small spacecraft (~300 kg). The two other missions, a Comet Tempel 1 Nucleus Sample Return and a Jupiter mission could be classified within the "New Frontier" type missions. Larger spacecraft masses are expected for these missions (~ 1000 kg). Finally, two "Flagship" missions were also evaluated: a Saturn/Titan Explorer and a Neptune Orbiter mission. It is expected that both last missions would be combined with aerocapture for orbit insertion at destination. Discovery missions are selected every 18 months. The New Frontiers missions have a longer recurrence, while Flagship missions would be flown once per decade.

With the appropriate thruster models, trajectories were run parametrically as a function of power level for an optimized flight time. Results are in terms of net delivered mass. The net delivered mass is defined as the spacecraft dry mass minus the dry mass of the ion propulsion system, or here as explained above, the dry mass of the ion propulsion module. Therefore the net delivered mass is everything on the spacecraft that isn't propellant or part of the ion propulsion module.

All figures are located at the end of the paper.

Comet Kopff rendezvous and Asteroids Vesta-Ceres rendezvous

Given the constraint of launch vehicle, namely the Delta II 7925 to reduce launch cost, only one operating thrusters was considered for each thruster option and each mission. The optimum flight time for the Comet Kopff mission is 3.1-3.6 years, whereas it is 7.8 years for the Vesta-Ceres mission (much longer since it has to rendezvous with Vesta and then depart Vesta to rendezvous with Ceres). A similar characteristic of both missions is that critical thrusting is done far away from the Sun (2-3 AU). That fact tends to advantage thrusters that can operate at low power levels. Figure 3 and 4 show the net delivered masses for both missions. As can be seen in these figures, all three advanced ion thruster offer much greater capability than NSTAR. The NGN-2 offers the highest delivered mass at low power while the NGN-3 offers the highest delivered mass at higher power. The NGN-2 provides around 100 kg more than NSTAR at the same power level.

Comet Tempel 1 Nucleus Sample Return

The same comments apply for this mission as for the two others. The optimum flight time is around 7 years and here again critical maneuvering is required far from the Sun. The trajectories assume a net drop mass at the comet of 25 kg and a minimum stay time at the comet of 60 days. Figure 5 shows that both NGN-2 and NGN-3 have clear benefits over NSTAR, with more obvious advantage of NGN-3 at higher system power levels. One to two operating thrusters were used on a Delta II 7925 launch. The 100 kg difference between NGN-2 and NSTAR remains at comparable power levels. The 10-kW thruster suffers a low Isp at low power level and a relatively high minimum power level. One caveat though is that the chosen Isp curve for the 10-kW corresponds to the high-thrust profile, leading to lower Isp at low power. A high Isp path would most certainly show better performances.

Jupiter/Europa Fly'by with Venus Gravity Assist

A larger launch vehicle was used for this mission, the Delta 4240, and the optimum flight time was around 3 years. Maximum 2 operating engines were used for the 10-kW thruster, while up to 4 (or 6) engines were used for the other thruster options. As can be seen in Figure 6, this mission benefits greatly from the 10-kW engine, with 400-500 kg more delivered mass than NSTAR at comparable power levels. With the Venus Gravity Assist, this mission requires most of the thrusting close to the Sun. The 10-kW, with its high power and high thrust per engine, is very well suited for this type of maneuvers. Both NGN-2 and NGN-3 have higher Isp at high power than the 10-kW thruster and therefore need a high launch C3 to provide a 3 year flight time (flight time constraint). This high launch C3 penalizes their launch mass and thus the net delivered mass. The results are based here as well on high-Isp profiles for both NGN-2 and NGN-3. They would most certainly show better results if their high-thrust profiles were considered.

Saturn/Titan and Neptune missions

Both Saturn and Neptune trajectories were run with the Delta 4240 as launch vehicle. The optimal flight time was 7.75 years for Saturn and 10.75 years for Neptune. Target spacecraft net masses were 1400 kg for Saturn and 850 kg for Neptune. Both destinations were reached after a Venus Gravity Assist. Maximum 2 operating engines were used for the 10-kW thruster, while up to 4 engines were used for the other thruster options. Figures 7 and 8 show that the 10-kW engine provides the greatest benefit, with around 400 kg more mass delivered for the same power than NSTAR. Clearly, this technology is very appropriate for Outer Planet missions. The NGN-3 technology suffers from its too high an Isp at low power levels, and a high-thrust profile for the engine would be more appropriate.

Engine throughput requirements based on this mission set

Setting up the requirements for engine throughput (total propellant mass that is processed by one thruster) and lifetime is one of the most important parameter for engine development since it will determine the amount of testing required and also which engine wear-out failure modes are critical. In order for this requirement set to be complete, one would also need the profile of the power processed as a function of time. In other terms, one needs to know at which power level the engine would run the longest during the trajectory, since the failure modes are also a function of power level of the thruster. However, due to time and funding constraints, this last requirement was not looked at during this study. Only the total throughput requirements will be shown here. Table 3 shows the amount of propellant required by each mission for each ion engine option. A range of propellant mass is shown since the propellant mass will vary for different power levels. It also shows how many engines are required to process the available power.

Figure 9 shows the desired throughput per engine for the most demanding mission case compared with the projected throughput found in Table 1. The desired throughput is the throughput the mission needs to have no more engines than what is required to process the power. It would provide the lightest ion propulsion module, and thus the greatest net delivered mass. The Vesta-Ceres mission is the most demanding in terms of propellant requirements for NGN-2 and NGN-3 while the Neptune Fly'by mission is the most demanding for the 10-kW 40-cm technology option. The advanced thruster technologies seem appropriately sized in terms of propellant throughput between the projections and the needs. The NGN-2 predicted throughput is close to the actual need, and thus can be expected to match the need if it gets to be developed.

Table 3: Propellant throughput requirement for the set of missions

	NSTAR	NGN-2	NGN-3	10 kW 40 cm
Comet Kopff Rendezvous				
Number of engines required for power	1	1	1	1
Total Xe load (kg)	160-190	140-180	120-160	200-300
Desired throughput per engine (kg)	160-190	140-180	120-160	200-300
Asteroids Vesta-Ceres Rendezvous				
Number of engines required for power	1	1	1	1
Total Xe load (kg)	240-280	210-240	150-220	340
Desired throughput per engine (kg)	240-280	210-240	150-220	340
CNSR Tempel 1				
Number of engines required for power	2	2	2	2
Total Xe load (kg)	260-370	140-340	150-300	160-410
Desired throughput per engine (kg)	130-185	70-170	75-150	800-205
Jupiter Fly'by				
Number of engines required for power	4	4	4	2
Total Xe load (kg)	500-800	350-620	280-500	510-720
Desired throughput per engine (kg)	125-200	90-155	70-125	255-360
Saturn/Titan Fly'by				
Number of engines required for power	4	4	4	2
Total Xe load (kg)	560-670	450-610	350-530	570-700
Desired throughput per engine (kg)	140-170	110-150	90-130	285-350
Neptune Fly'by				
Number of engines required for power	4	4	4	2
Total Xe load (kg)	400-800	300-730	320-590	500-880
Desired throughput per engine (kg)	100-200	75-180	80-150	250-440

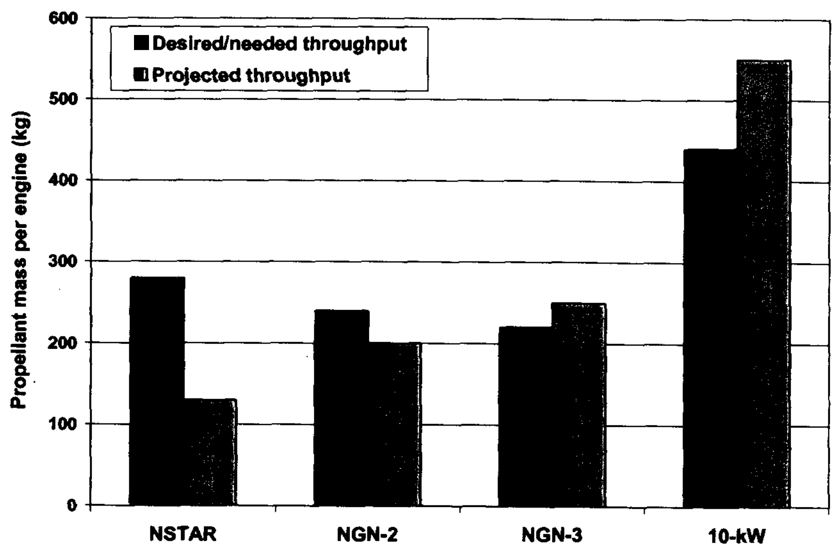

Figure 9: The predicted propellant throughput is appropriate for the most constraining mission for the 3 advanced technologies

CONCLUSIONS

This paper investigates the mission benefits of 3 potential ion thruster technologies. Two of these technologies use the NSTAR 30-cm engine body but with improved performances (Isp, efficiency) and power level, namely NGN-2 and NGN-3 and the other uses a 40-cm diameter body with high Isp and power (10 kW). The projected characteristics of these advanced engines are shown, and they were used to evaluate ion thruster technologies for 6 different missions. The six missions chosen cover the range of potential future flight projects, with two belonging to the "Discovery" class mission, Comet Kopff Rendezvous and Asteroids Vesta-Ceres Rendezvous, two belonging to the "New Frontiers" mission category, a Comet Temple 1 Nucleus Sample Return and a Jupiter/Europa Fly'by, and finally two belonging to the "Flagship" mission, the Saturn /Titan and Neptune missions.

Results show that all three advanced thruster technologies have greater mission benefits than NSTAR. Unfortunately not one technology fits all missions. The NGN-2 technology is the most beneficial for primitive bodies type missions (Comets, Asteroid rendezvous or sample return), as they require critical maneuvering at low power levels. On the other hand, the 10-kW 40-cm technology provides the most benefits to bigger missions to the Outer Planets, where high thrust is required for relatively short trip times. The NGN-3 technology could most probably be very applicable for Outer Planet mission as well if it was used with a high-thrust thruster profile instead of the high-Isp profile as considered in this study. The overall lifetime predictions of the future engines are well suited for the mission set studied here.

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REFERENCES

[1] Rayman, M. D., "Results from the Deep Space 1 Technology Validation Mission," IAA-99-IAA.11.2.01, presented at the 50th International Astronautical Congress, 4-8 Oct 1999, Amsterdam, The Netherlands.

[2] Polk, J. E., et al., "In-Flight Performance of the NSTAR Ion Thruster Technology on the Deep Space One Mission," AIAA-99-2274, presented at the 35th AIAA/ASME/SAE/ASII Joint Propulsion Conference, June 21-23, 1999.

[3] Christensen, J. A., et al., "The NSTAR Ion Propulsion System for DS1," AIAA-99-2972, presented at the 35th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Los Angeles, CA, June 1999.

[4] Ganapathi, G. B. and Engelbrecht, C. S., "Performance of the Xenon Feed System on Deep Space One," *J. Spacecraft and Rockets*, Vol. 37, No. 3, pgs. 392-398, May-June 2000.

[5] Brophy, J.R, "Ion Propulsion System Design for the Comet Nucleus Sample Return Mission," AIAA-2000-3414, presented at the 36th AIAA/ASME/SAE/ASII Joint Propulsion Conference, July 2000.

[6] Patterson, M.J, et al., "Next-Generation 5/10 kW Ion Propulsion Development Status," IEPC-01-089, 27th International Electric Propulsion Conference, Pasadena, California, October 15-19, 2001.

[7] Patterson, M.J., et al., "NEXT: NASA's Evolutionary Xenon Thruster," AIAA-2002-3832, presented at the 38th AIAA/ASME/SAE/ASII Joint Propulsion Conference, 7-10 July 2002, Indianapolis, Indiana.

[8] Oleson, S., et al., "Mission Advantages of NEXT: NASA's Evolutionary Xenon Thruster," AIAA-2002-3969, presented at the 38th AIAA/ASME/SAE/ASII Joint Propulsion Conference, 7-10 July 2002, Indianapolis, Indiana.

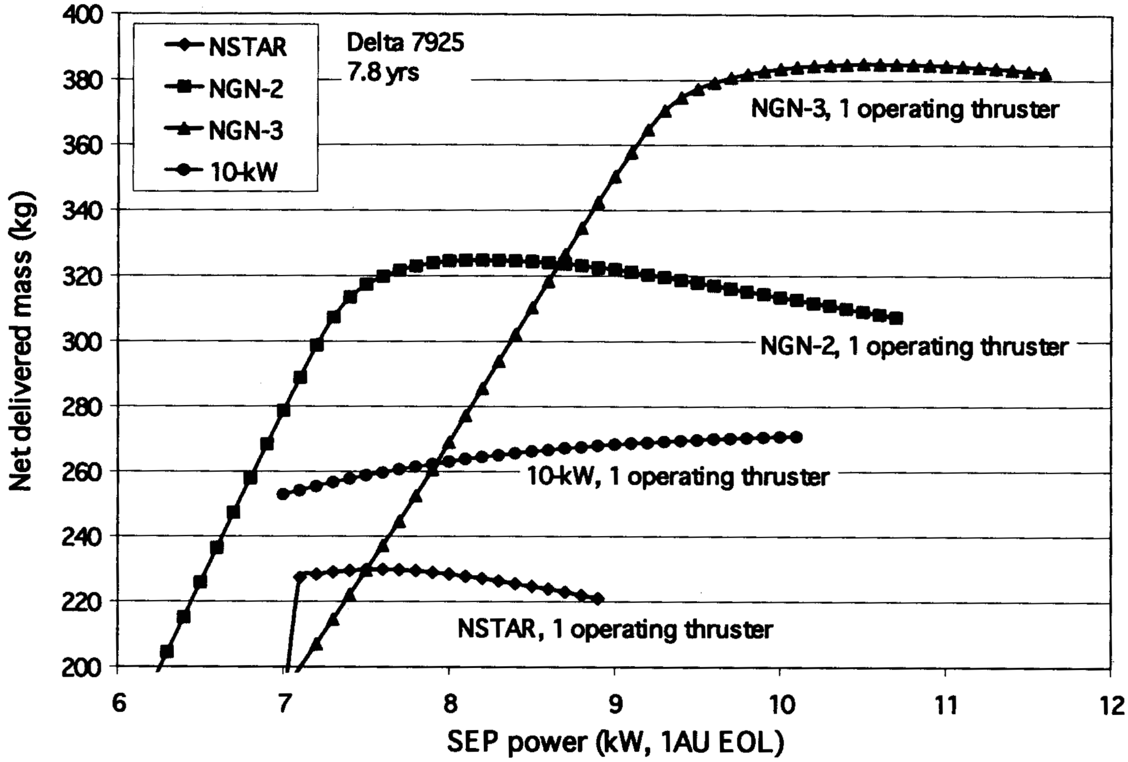

Figure 3: Comet Kopff Rendezvous

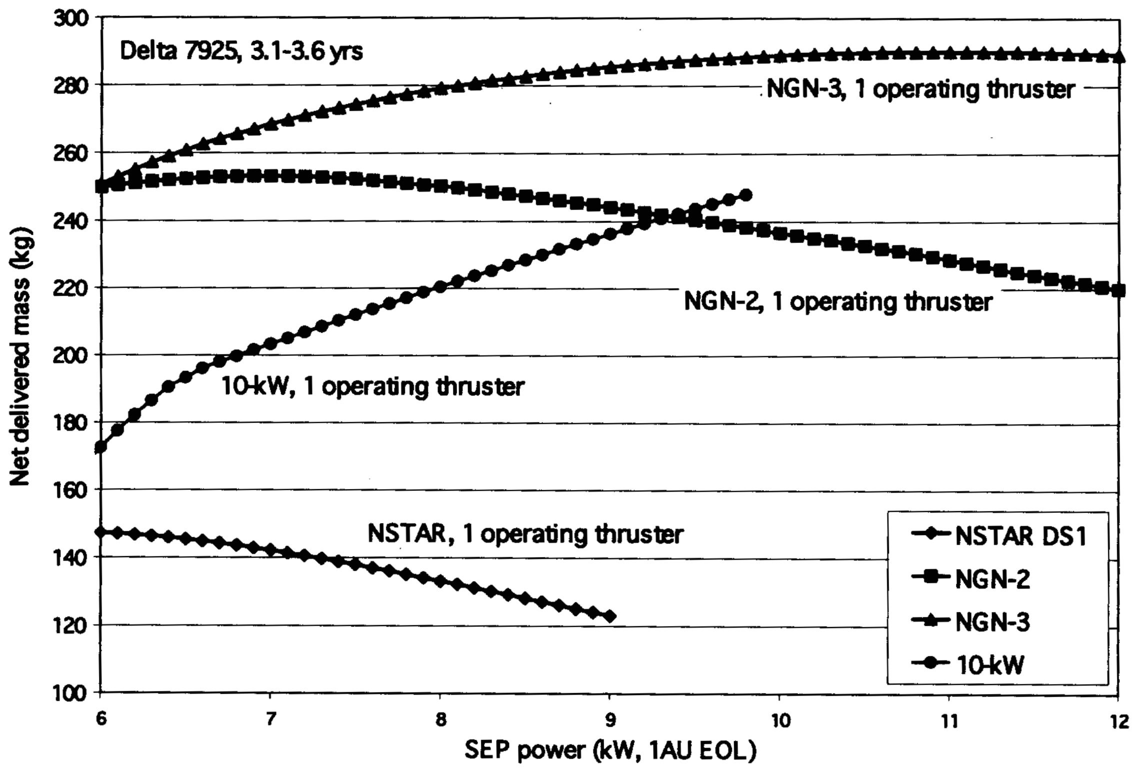

Figure 4: Asteroids Vesta and Ceres Rendezvous

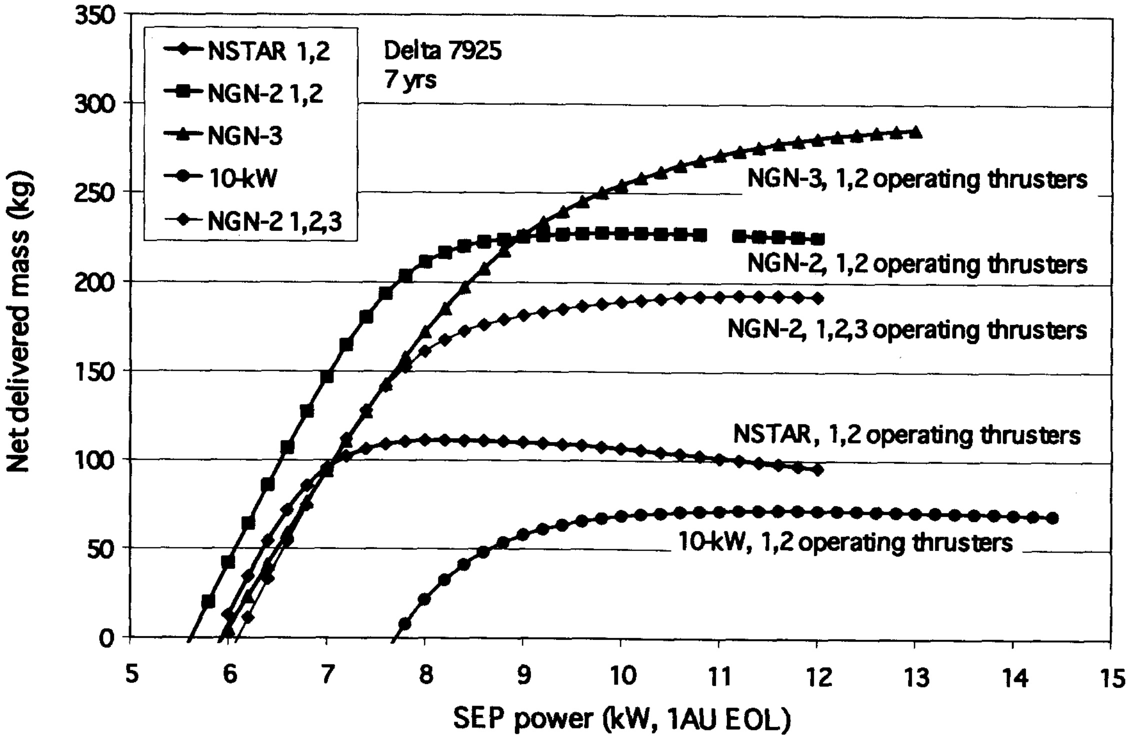

Figure 5: Comet Tempel 1 Nucleus Sample Return

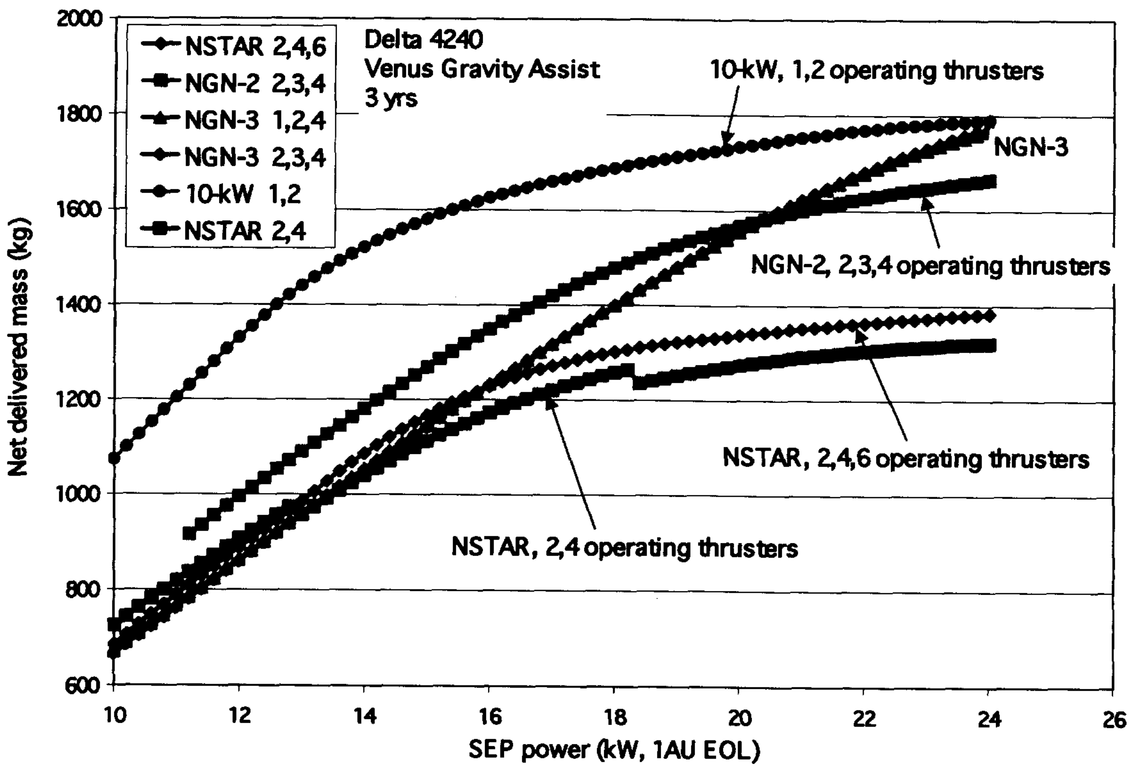

Figure 6: Jupiter/Europa Fly-by

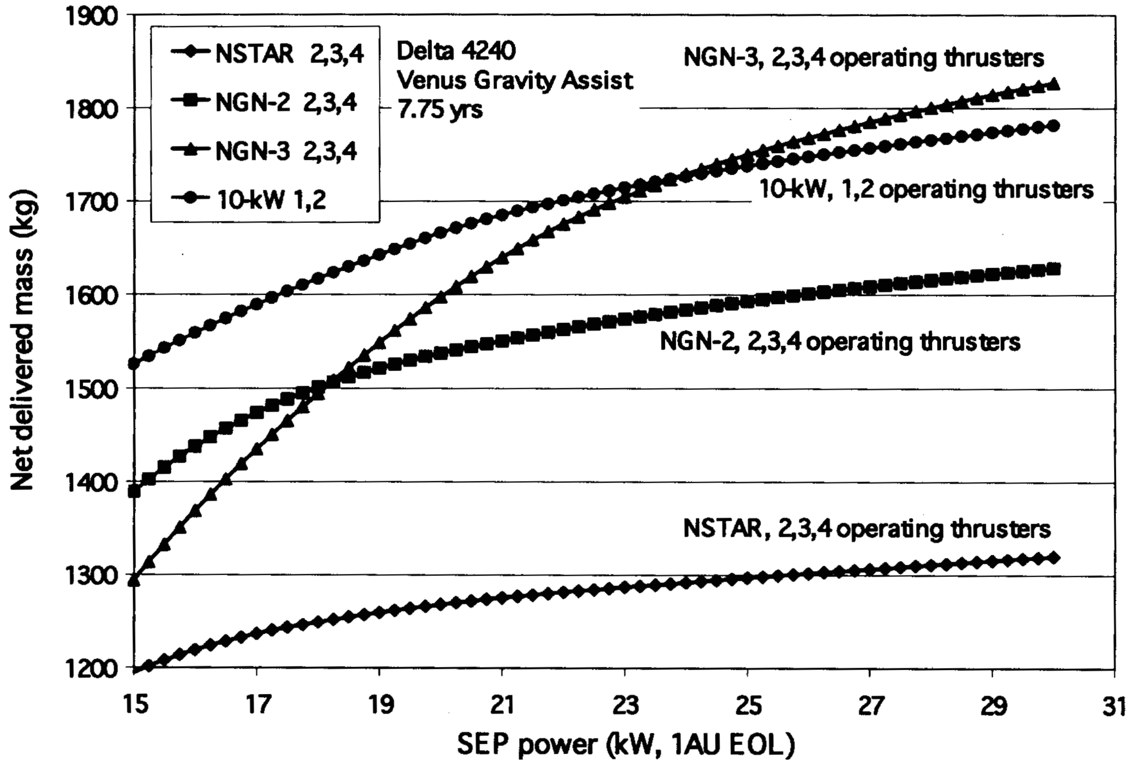

Figure 7: Saturn/Titan Fly-by net delivered mass for an optimized flight time of 7.75 years

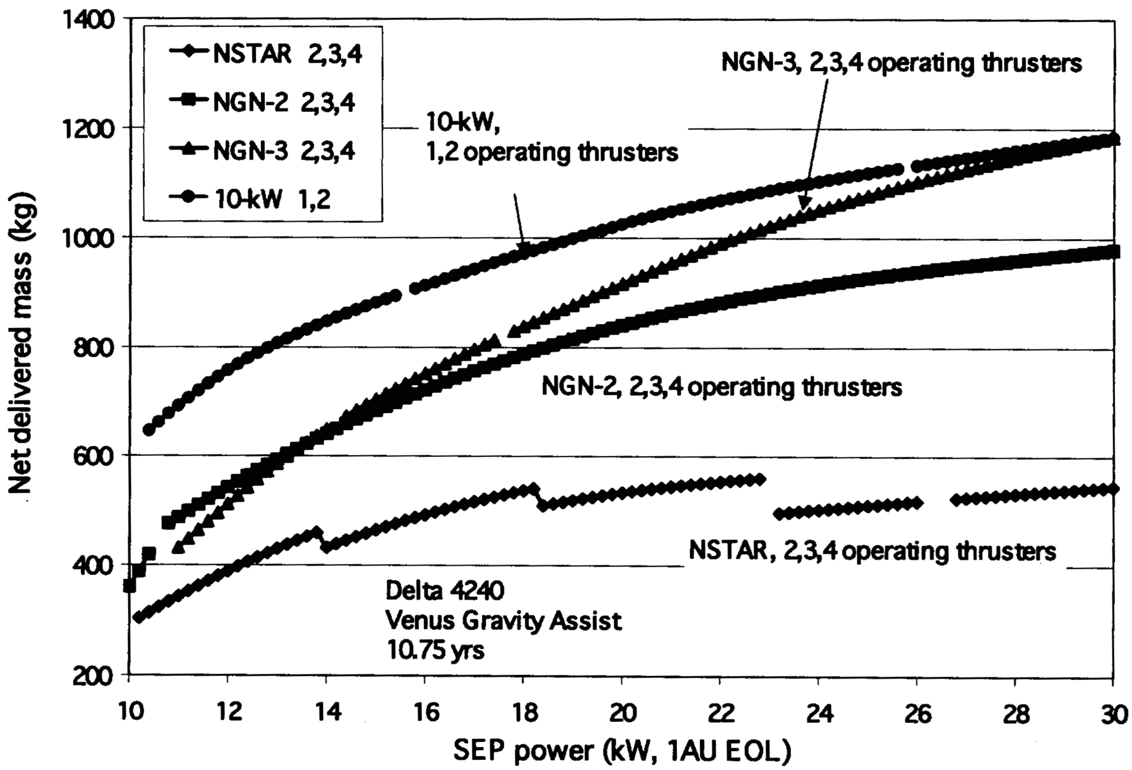

Figure 8: Neptune Fly-by net delivered mass for an optimized flight time of 10.75 years