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Direct Flight to Pluto Using Solar Electric Propulsion

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High-performance spacecraft propulsion and lightweight integrated science payloads promise to enable extensive exploration of the outer solar system. Previous analysis has shown that a small, yet highly capable, spacecraft launched on a Delta II could be propelled by solar-powered ion thrusters on an Earth-gravity-assist trajectory that reaches Pluto in just 7–8 years. Whether additional development of solar electric propulsion technology could improve the power-to-mass ratio sufficiently to enable using a direct trajectory is determined. The advantages would be a slower flyby velocity at target encounter and a much reduced operating time required of the propulsion system. This approach becomes feasible with the implementation of two anticipated advances: a lightweight, foldable-blanket solar array operating at 1000 V, and thruster ion optics made with high-durability carbon–carbon composite material.

Introduction

THE outer reaches of our solar system have been barely explored, yet they hold keys for understanding its formation. The planetary bodies far from the sun, Pluto, the Kuiper Belt objects, and perhaps the rings and moons of Neptune, likely remain primeval, and probing them closely with spacecraft will reveal much about early conditions in the solar system. The vastness of the regions these bodies occupy argues for diverse exploration, but today the costs for launching such far-reaching missions preclude planning for more than a single mission to fly by Pluto and a Kuiper Belt object.

To put a relatively small spacecraft on a 7-year trajectory to Pluto would entail the expense of a Titan-IVB/Centaur launch vehicle carrying two solid-propellant kick stages.[1] Potential technology advances aimed at reducing spacecraft mass would only lessen the needed launch capability to that of a Proton or the space shuttle.[2] Allowing a greater trip time considerably lessens the launch requirement, but so too is the perceived value of a mission lessened when the scientific returns take a decade or more.

A promising approach for reducing launch costs is to use solar electric propulsion (SEP) to provide the high velocity required. Rapid advances in microelectronics and materials technologies are enabling the design of highly compact and efficient science and telemetry payloads with superior capabilities than their larger predecessors. For lightweight payloads, ion propulsion only requires modest power levels, that is, 5–15 kW, to provide substantially better mission performance than chemical propulsion, and thus solar-powered ion propulsion becomes a viable option for enabling the use of small or medium launch vehicles.

A previous analysis found that SEP employed on an Earth-gravity-assist (EGA) trajectory can achieve a mission time under 8 years with launch on a Delta II (J. Sercel and C. Sauer, private communication, NASA Jet Propulsion Laboratory, November 1993). The power requirement is 5–10 kW, and the required thruster technology[3] is the current state of the art.

Several other studies have evaluated the use of ion propulsion for missions to the outer planets, but with nuclear power rather than solar power.[4–7] An evaluation of a transfer stage using solar thermal propulsion indicated that it also could perform a Pluto flyby mission using the Delta II launch vehicle.[8]

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This paper presents a study done to identify the advances in ion thruster and solar array technology needed to enable sending a spacecraft on a fast direct trip (no gravity assist) to the outer solar system, using the Pluto mission as an example. The objective was to achieve a trip time of 8 years or less using a Delta-II class launch vehicle, matching the performance obtained with an EGA trajectory. The motivation for exploring the direct trajectory is that it has a slower flyby velocity at target encounter and can entail much less operating time for the SEP system. The EGA trajectory involves a 2-year swing through the inner solar system, much of it under thrust, to set up the EGA.

We also evaluated EGA mission performance using the technologies applied to the direct-trajectory missions. This showed to what degree the reduced thrusting time obtained in the direct-trajectory analysis was due to the trajectory and how much to the SEP technology improvements themselves.

Direct-Trajectory SEP Mission Concept

The study assumed a mission comprising a lightweight science spacecraft mated to a separable SEP module. This module incorporates solar arrays, ion thrusters, power processing units (PPUs), an attitude control system, avionics, and communications antennas. An expendable launch vehicle boosts this combination to a speed substantially exceeding Earth escape velocity, whereupon the SEP module commences a period of continuous thrusting lasting until the solar power available drops to an unusable level at between 3 and 4 astronomical units (AU) from the sun.

Mission Spacecraft

The spacecraft configuration used in this assessment is shown in Fig. 1. The science payload is a digital camera, infrared and ultraviolet spectrometers, and a radio science experiment and has a mass of just 7 kg. We assumed the total mass of the spacecraft to be 145 kg, midway in the range from 120 to 165 kg spanned by various proposed concepts.[9] We included an allowance for a hydrazine propulsion system sized to provide a ΔV of 350 m/s, of which about 300 m/s is for correcting trajectory insertion errors.

Propelling the spacecraft for a sustained period of time with a low-thrust SEP system eliminates the need for the spacecraft to carry hydrazine for trajectory correction. With negligible impact on the amount of xenon propellant required, trajectory insertion errors can be removed during the long thrusting period. Eliminating 300 m/s of the spacecraft's ΔV capability reduces the onboard propulsion system mass by 20 kg, and so for our analyses we assumed the mass of the spacecraft to be 125 kg.

SEP Module

We configured the SEP system as a propulsion module rather than an independent stage. Because guidance, navigation, and attitude

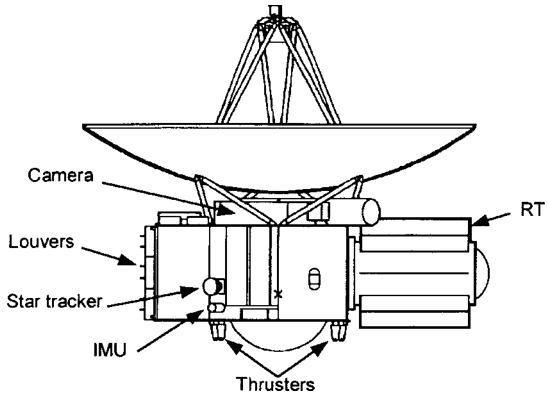

Fig. 1 Mission S/C.

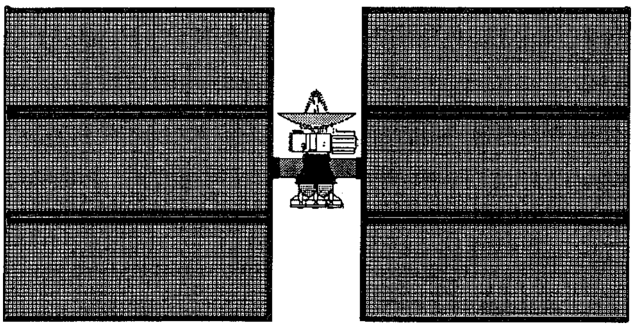

Fig. 2 Conceptual configuration of a 15-kW SEP module mated to the mission S/C.

control calculations could be done by the spacecraft computer. Because the module would block the spacecraft's view toward Earth and the sun, it includes additional sun sensors for attitude determination. Cold-gas thrusters provide attitude control. The spacecraft antenna cannot be pointed at Earth while the SEP module is operating, so the module also includes a pair of omnidirectional antennas and an X-band amplifier for communications. This configuration minimizes the impact on the spacecraft design and makes the SEP module readily adaptable to other spacecraft.

The evaluation encompassed several combinations of array and thruster technologies. Planar and concentrator solar arrays with performance ranging from 60 to 160 W/kg were assessed. Power levels ranged from 5 to 25 kW. At the higher power levels, both the planar and the concentrator arrays have to be a blanket type of array to stow compactly enough to fit in the Delta II 2.9-m fairing. Figure 2 shows a 15-kW SEP module mated to the spacecraft. In this example, the booms holding the arrays are short to aid in stowage and to minimize the rotational moment of inertia, but this will subject the array to impingement by the fringes of the ion beams during the initial part of the trajectory, when the thrust vector will be nearly coplanar with the array. Vertical fences along portions of the inside edges of the array, or telescoping booms, would be needed to avoid damage to the array.

Methods have been proposed for operating solar arrays at voltages as high as 1000 V. If this can be done, a significant benefit ensues from a dramatic lessening of the mass of the PPUs. Concentrator arrays, in particular, lend themselves to this because the cell area is small and most of the electrical insulation and grounding needed to avoid high-voltage arcing are provided as a consequence of other elements of the design. Protecting a planar array is considerably more difficult, but techniques for doing so with ground screens and encapsulation have been described.[10,11]

For the propulsion, the performance characteristics of a lightweight 30-cm ion thruster were assumed.[12,13] Two types of ion optics, or grids, for the thrusters were considered: conventional molybdenum grids and carbon–carbon (CC) grids, which are under development.[14–17] With molybdenum, grid erosion imposes significant operating constraints. To achieve the requisite lifetime, the thruster must be operated at a lower thrust level, that is, at a lower beam density, than it would otherwise be physically capable of. CC grids are expected to provide two advantages: The grid spacing can be tighter because thermal distortion is nearly eliminated, and the grid erosion rate may be only $\frac{1}{10}$th as much as with molybdenum.[18] The operational benefit is that the thruster can produce a higher maximum beam density due to the tighter grid spacing, without incurring rapid grid erosion and early failure. This reduces the number of thrusters needed for a given total power input, and thruster efficiency is boosted by the higher throughput.

For operation with molybdenum grids, we set the maximum thruster input power to 4.5 kW at a specific impulse I_{sp} of 3500 s, the value that our mission simulations showed to yield the shortest trip time. Although this power level exceeds the expected 10,000-h qualification ratings for ion thrusters, it is commensurate with the demonstrated capability of the thruster[12] and is believed acceptable for this application because the required operating time for each thruster is only 4000 h or so, much of it below full power. With CC grids, we assumed that a 30-cm thruster could be operated continuously at 5 kW at an I_{sp} of 3000 s, near the highest demonstrated power with this thruster at that I_{sp}. The increased beam density that this represents is expected to boost the efficiency to 70% (Ref. 19). We constrained the I_{sp} to 3000 s because we elected not to impose a voltage requirement greater than 1000 V on the high-voltage arrays assumed for the direct drive system configurations. An I_{sp} of about 3500 s appears to be optimum for this system also, but the penalty incurred by operating at 3000 s is slight.

PPUs rated for 2.5–5.0 kW are expected to have an efficiency of 92% and a specific mass of 4 kg/kW when the input voltage is 150 V (Ref. 20). With a solar array operating at 1000 V, however, the PPU need not have a dc-to-dc converter to produce the high voltage needed by the thrusters. This eliminates the principal section of the PPU, and as a result the PPU efficiency and specific mass are projected to be 98% and 1 kg/kW, respectively (L. R. Pinero and V. K. Rawlin, private communication, NASA John H. Glenn Research Center at Lewis Field, and R. J. Kay, private communication, Olin Aerospace Corporation, March–June 1993).

Analysis Method

We postulated three SEP systems comprising various combinations of the technologies just described: 1) 160-W/kg planar solar array (at 150 V) with molybdenum-grid thrusters, 2) 140-W/kg concentrator solar array (at 1000 V) with molybdenum-grid thrusters, and 3) the system 2 solar array with CC-grid thrusters. The assumed performance parameters for these systems are listed in Table 1. (For reference, the solar arrays on the International Space Station have a specific power of 40–45 W/kg.) In Table 1, the lower power level and efficiency indicated for the thruster in system 2, relative to system 1, is due to the lower I_{sp} imposed by the 1000-V constraint on the system operating voltage. In system 3, even though the I_{sp} is the same as in system 2, the thruster power level and efficiency are considerably higher because the carbon grids enable greater propellant throughput.

The first part of the analysis investigated the SEP performance level needed to achieve a trip time less than 8 years using the Delta

Table 1 SEP system characteristics used in the direct-trajectory mission analyses

System	1	2	3
Solar array			
Type	Planar	Concentrator	Concentrator
Specific power, W/kg	160	140	140
Voltage, V	150	1000	1000
Thruster			
Grids	Molybdenum	Molybdenum	CC
Size, cm	30	30	30
I_{sp}, s	3500	3000	3000
Power, kW	4.5	3	5
Efficiency, %	68	65	70
PPU			
Specific mass, kg/kW	4	1	1
Efficiency, %	92	98	98

Table 2 Parameters used in estimating the mass of the SEP module

Component	Mass	Comment
Delta II adapter	3% of SEP + S/C mass	Composites
Structure	10% of SEP dry mass	Composites
Propellant tank	12% of propellant mass	Supercritical gas
Feedsystem	3.1 kg + 1 kg per engine	
Thermal control	2% of tank, feed, avionics	Insulation, heaters
Engine gimbals	1 kg per engine	Nitinol actuator
Array positioners	2×3.75 kg	1 axis
Cable	5.0 kg	
Attitude control	2.1 kg	Xenon cold gas
Communication	3.5 kg	
Avionics	1.5 kg	
S/C separation	1.0 kg	Nitinol release
Mass growth	30% of SEP dry mass	

II launch vehicle. To determine the trip time, we used a trajectory simulation code that modeled the spacecraft motion resulting from accelerations imposed by the sun, the planets, and the thrust of the spacecraft, starting from spacecraft separation from the launch vehicle. The code used the Bulirsch–Stoer method (see Ref. 21) to integrate the equations of motion. To verify the accuracy of the integrator in the code, we duplicated a test case with the OTIS code,[22] a more sophisticated and computationally intensive code that was well validated. Table 2 gives the parametric relationships used to estimate the mass of the SEP module as a function of power level and propellant mass. Earth-escape energy C_3 as a function of launch mass was determined from performance curves for the Delta II Model 7925 upgraded with the addition of three air-lit solid rocket motors (the configuration for the Earth Observing System satellite launches). Use of the Delta-II 2.9-m payload fairing was assumed.

For each condition, the trajectory simulation was run iteratively while varying the launch mass until the payload delivered by the SEP module converged to 125 kg. The launch window used was December 1999 through January 2000. Targeting was done by varying the launch date and the angle of the thrust vector. The steering law was a simple one with the thrust aligned with the velocity vector except for about a 20-deg yaw angle out of the orbital plane of the spacecraft, which propelled the spacecraft out of the ecliptic to intersect the plane of Pluto's orbit.

The power output of the solar array was decreased in inverse proportion to the square of the distance to the sun. Radiation degradation was assumed negligible, and no adjustment was made for gradually improved array efficiency as the array cooled. The efficiency is expected to increase by as much as 20%, but this benefit is counteracted by a concurrent 15–20% decrease in thruster efficiency due to throttling. [Strictly, this 20% increase is expected only for arrays having a high concentration ratio ($50X$–$100X$). At 1 AU such concentrator arrays operate at a relatively high temperature, and at 3 AU their solar cells still produce a current equal, or nearly equal, the cell current in planar arrays at 1 AU. As a concentrator array cools with increasing distance from the sun, the resistive and reverse-current losses are predicted to decrease more than the forward current, resulting in a net increase in efficiency. The relationship with temperature is nonlinear, and planar arrays, which at 1 AU start at a lower temperature, do not benefit from proportionally as much reduction in losses when the array cools. Also, planar arrays may get cold enough to reduce the mobility of the semiconductor charge carriers and impede generation of forward current. Thus, far from the sun, the efficiency of planar arrays drops. Our simulation did not include such detail, and consequently the results for SEP system 1 are optimistic.] The propulsion subsystem included enough thrusters to utilize all of the power produced by the array at 1 AU. For maximum performance, the number of thrusters would be less, but the approach we used provides a measured amount of redundancy. A thruster failure early in the mission is not crippling and would only impose a moderate increase in trip time. In our simulation, as the power output declined, the thrusters were throttled and shut down one by one. The throttling was by propellant flow control that regulated current draw to maintain constant array voltage. The I_{sp} of the thrusters was thus considered fixed throughout the mission.

The second part of the analysis was an evaluation of SEP technologies applied to EGA trajectories. No trajectory simulations were run in this phase. From unpublished trajectory data provided to us (C. Sauer, private communication, NASA Jet Propulsion Laboratory, November 1993), we derived the ΔV, thrust time, and average solar flux for each trajectory and used them as inputs in an SEP parametric model developed separately for orbit-raising analyses.[19] Once again we assumed the launch performance of the upgraded Delta II and applied the SEP mass model in Table 2.

Afterward, we briefly examined operating the SEP module well beyond the distance at which the array power drops below 1 kW. For this, the SEP module would carry a second set of thrusters sized to operate efficiently at a power level as low as 100 W. It must also have a concentrator solar array that directs sufficient light onto the individual solar cells to keep them functioning far from the sun. When the solar array output drops to 1 kW, the main thrusters and their PPUs are staged from the SEP module, and the smaller thrusters and their PPUs take over, operating at higher efficiency than the main thrusters could when throttled deeply. When the thrusting time is increased to 3 years, this strategy potentially makes better use of the SEP power system. Even though the total thrusting time is long, the operating time of each of the thrusters is less than 10,000 h if four small thrusters are used. Those that have been shut down as the power declines can be sequentially restarted as the operating thruster in turn reaches its rated lifetime.

Additional analyses of direct and EGA trajectories using launch vehicles other than the Delta II are discussed in Ref. 23.

Mission Performance

Direct Trajectory

The difficulty with the direct trajectory is that the spacecraft moves quickly outward from the sun, limiting the time during which much power is available for thrusting. We found that this necessitates an initial power level greater than 10 kW coupled with the most advanced SEP performance that we postulated.

For each of the direct-trajectory SEP system options, we did a parametric survey of trip time vs initial power to determine an appropriate initial power level at which to size the SEP module. Too much power actually lengthens the trip by making the SEP module too heavy for the launch vehicle to provide much escape velocity from Earth, and the initial part of the trip goes slowly. With too little power and a light SEP module, the departure velocity is high and the spacecraft reaches the point of thrust termination quickly, leaving too little time for the SEP system to impart much of the mission's total impulse. A power of 15 kW at 1 AU turned out to be the best one for comparing the three SEP systems. It is nearly optimum for system 1 and yields a trip time less than 8 years with system 3.

Table 3 lists the trip times obtained for the three systems. It is apparent from the 10.6-year trip time of system 1 that a highly advanced solar array producing 160 W/kg does not by itself provide the desired SEP performance. With system 2, adding a high-voltage array to eliminate the need for a dc-to-dc converter in the PPU drops the trip time only to 8.6 years. The thruster technology included in system 3, CC grids to improve the power density and efficiency of the thruster subsystem, is necessary to bring the trip time under 8 years.

Increasing the power to 20 kW enables system 3 to achieve even faster trips, as indicated in Table 4. For reasons of practicality and cost effectiveness, however, 15 kW is probably a better size, so we selected the 15-kW system 3 to describe here in further detail. Figure 3 shows graphically the SEP portion of the flight trajectory to Pluto and lists the mission parameters specific to this case. The

Table 3 Direct-trajectory trip times for the three SEP systems[a]

System no.	Trip time, yr
1	10.6
2	8.6
3	7.6

[a]Array power at 1 AU is 15 kW.

Table 4 System 3 trip time vs array power at 1 AU

Array power, kW	Trip time, yr
10	8.1
15	7.6
20	7.1

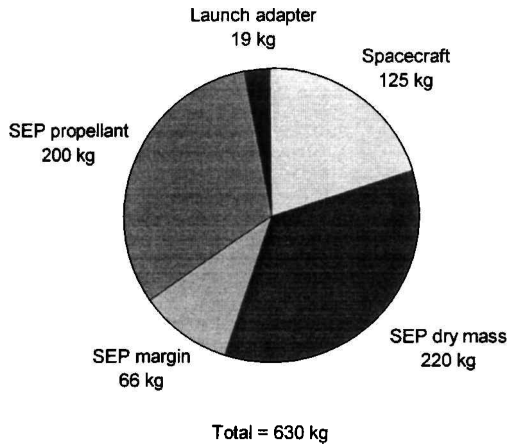

Fig. 3 Mass distribution for the 15-kW system 3 SEP module.

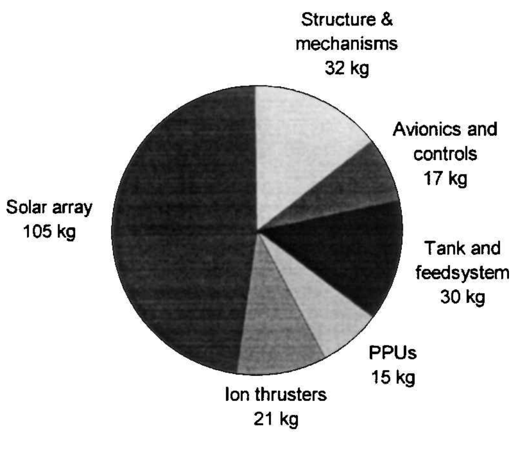

Fig. 4 Mass distribution among SEP module subsystems.

ΔV imparted by the SEP module is 11.4 km/s, the thrusting time is 6200 h, and the flyby velocity is 17 km/s.

15-Kilowatt SEP Module

The estimated mass of the 15-kW system 3 SEP module is 485 kg, including propellant. Of this, about 70 kg, or 30% of the dry mass, is margin. The xenon propellant load is 200 kg, including a small margin equivalent to 1% of the ΔV required. (A margin as small as this may well be adequate, when, as in this instance, the consequence of a shortfall is merely a slightly longer trip.) With the spacecraft (S/C) and the launch vehicle adapter added, the total launch mass was 630 kg. Figure 4 summarizes how this is apportioned, and Fig. 5 shows a breakout of the primary SEP subsystems. The solar array and ion thrusters make up about 60% of the dry mass of the module, whereas the PPUs make up a relatively small 7% proportion because of the use of the high-voltage array. In this instance, this saves a total of about 60 kg, including concomitant reductions in the structure and mass margin. Equally important, it increases the overall system efficiency by 4 percentage points.

Two-Stage SEP

It appears that a two-stage SEP module could attain a short trip with a much smaller array, although by forsaking a short thrusting time. We did not simulate this approach using a system 3 SEP module, but we did compare one- and two-stage modules while searching for a less technology intensive way to achieve a trip time under 8 years. We extrapolated those results to estimate the performance of a two-stage mission using system 3 technology.

Specifically, we evaluated one- and two-stage SEP modules using an 120-W/kg, 150-V array and thrusters with CC grids. Although the solar array in this two-stage example was a concentrator array to enable the SEP module to function out to 10 AU, it was not a high-voltage array. This power system is relatively heavy, and launch on a Delta II is inadequate. To achieve the trip-time goal, the simulation required an Atlas IIA with a Star-48 kick stage. With the single-stage SEP module, we found that 10.6 years is the shortest trip time achievable and 10 kW is the optimum array size. With two stages, the optimum power is 12 kW and the trip time is 8.0 years. (In this case, the Atlas/Star-48 combination boosts 750 kg to a C_3 of 60 km^2/s^2.)

In light of the data trend shown in Table 4, this result suggests that a two-stage system 3 SEP module with 5–10 kW of power could achieve a trip time of $7\frac{1}{2}$ years or better when launched on a Delta II. As the following section shows, this power requirement is the same as that for an EGA trajectory with a similar trip time.

Table 5 EGA trajectory trip times

	Trip time, yr	
Power, kW	60-W/kg array	120-W/kg array
5	8.2	7.7
8	8.0	7.4
10	7.8	7.1

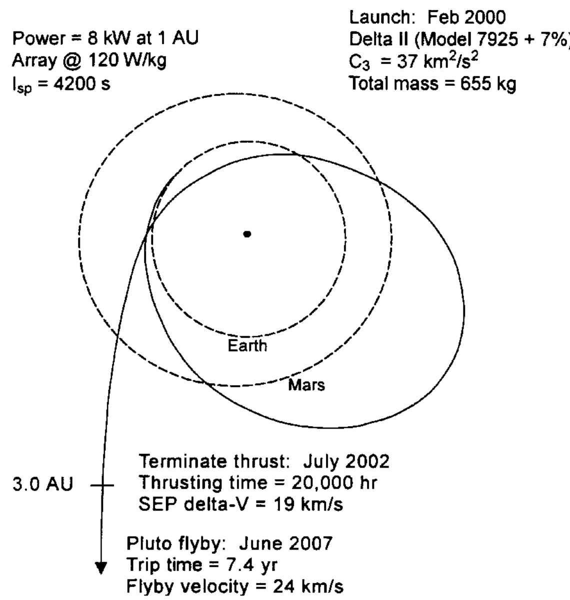

Fig. 5 Mission data for 7.4-year EGA trajectory.

EGA Trajectory

Using an EGA enables a trip time under 8 years with less advanced SEP technology. Table 5 lists the trip times achievable with system 1 thrusters and PPUs combined with solar arrays with specific powers of 60 and 120 W/kg. Compared to the direct-trajectory results, the needed power levels are lower, and adequately short trip times are possible with a 60-W/kg array. For the missions listed, the SEP ΔV ranges from 17 to 22 km/s, and the optimum I_{sp} is higher than for the direct trajectory, ranging from 4000 to 5000 s. Figure 6 shows an EGA mission having a 7.4-year trip time.

The thrusting times for the cases listed in Table 5 range from 19,000 to 21,000 h. This imposes a need for two sets of thrusters, assuming that the qualified full-power lifetime for the thrusters is

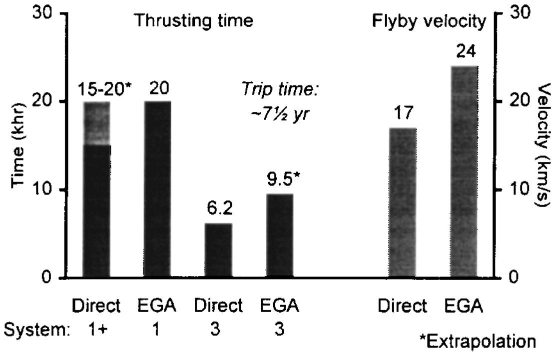

Fig. 6 Thrusting times and flyby velocities.

10,000 h. Because the power available during much of the trajectory will be significantly less than the system rating and the thrusters will be often throttled or shut down, two full sets actually provides substantial redundancy as well.

We also investigated the thrusting time required for the EGA trajectory if the advanced technology SEP system 3 were used. In the case of the 7.4-year EGA mission, this enables the size of the SEP system to be increased from 8 to 15 kW while maintaining the same launch mass. This mission then requires an estimated 9500 h of thrusting time. (Note that a secondary consequence of increasing the power level is a reduction in the ΔV requirement. To make this estimate of thrusting time, the ΔV reduction was assumed to be roughly 5%, from 19 to 18 km/s. A trajectory simulation would be needed to determine the thrusting time accurately.) With a similar power level and trip time, this EGA mission is closely comparable to the 7.6-year system 3 direct trajectory mission. Thus, the thrusting time for the EGA trajectory is only 50% more than that for the direct trajectory when identical technology is applied to both trajectories. (The ratio between the thrusting times is about the same as the ratio between the ΔV for the two cases: 11 vs 18 km/s.) Therefore, it is not the direct trajectory per se that yields most of the reduction in the thrusting time; it is the high-power, lightweight SEP system.

As noted in Fig. 6, the flyby velocity at Pluto with the 7.4-year EGA trajectory is 24 km/s, approximately 40% higher than that for the 7.6-year direct trajectory. We did not assess the impact of this difference on the science observations or the postencounter data telemetry. Figure 6 summarizes the flyby velocities and needed thrusting times for the missions that yielded a trip time of $7\frac{1}{2}$ years or so.

Conclusion

SEP promises to enable missions to the outer solar system that only require a Delta-II class launch vehicle. Transit to Pluto in under 8 years is achievable with either a direct or an EGA trajectory. For a direct trajectory, a 15-kW direct-drive SEP system nearly 70% efficient and having an overall specific mass less than 10 kg/kW is needed. Although these performance characteristics are beyond present capabilities, they are projected to be attainable with the introduction of high-voltage concentrator solar arrays and CC ion thruster grids.

The EGA trajectory, on the other hand, provides $2\frac{1}{2}$ years of travel close enough to the sun for SEP operation. The advantages of this are that as little as 5 kW of power is adequate, a highly advanced solar array is unnecessary, and thrusters with molybdenum grids may be used. Relative to the direct trajectory, however, this approach necessarily has a 40% higher flyby velocity at target encounter and entails SEP operation for three times as long.

The benefits of lower flyby velocity and thrusting time achieved with the direct trajectory must be weighed against the apparent cost advantages conferred by the reduced power and technology requirements of the EGA trajectory. Although the balance today between them may well favor the EGA approach, the mission trends found in this study remain for consideration in planning outer solar system exploration in the future.

Acknowledgments

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