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SMALL BODY RENDEZVOUS MISSION USING SOLAR ELECTRIC ION PROPULSION: LOW COST MISS1ON APPROACH AND TECHNOLOGY REQUIREMENTS

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ABSTRACT

This paper shows that existing solar electric ion propulsion (SEP) technology can deliver substantial payloads to important small bodies for an effective cost. SEP, using hardware being validated by the NASA SEP Technology Application Readiness (NSTAR) program, can deliver significantly more mass in a dramatically shorter period of time than a chemical propulsion system launched from the same Delta II launch vehicle. Analysis of three rendezvous missions shows that NSTAR hardware can deliver a payload (spacecraft with science) of 364 kg to asteroid Vesta, 280 kg to the outer main belt asteroid Ceres, and 291 kg to comet Kopff. The paper begins with a discussion of why SEP is now ready for space science missions, the NSTAR program, benefits of ion propulsion, the range of SEP applications, a detailed SEP mass breakdown, and the cost and other considerations of using an ion propulsion system. A discussion of SEP navigation and a new start development schedule concludes the paper.

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Several researchers at LeRC have made important contributions to the development the material in this paper. The enthusiastic cooperation by Leon Gefert, John Hamley, Steven Oleson, Vincent Rawlin, and James Sovey has enabled the presentation of this paper.

This paper would not be possible without the low thrust and ballistic trajectory designs furnished by Carl Sauer. Carl provided detailed comet and asteroid rendezvous trajectory designs using his EPITOP[4] computer program. We especially thank Carl for his guidance in my mission design efforts and in developing and maintaining the VARITOP[5] and EPITOP low thrust programs through the ups and downs of low thrust propulsion development. Without his determination, much of the work in low thrust trajectory design would have been lost many years ago.

We would also like thank Dr. John Brophy[6] for his insightful review of this paper.

WHY IS SEP NOW READY FOR SPACE SCIENCE MISSIONS?

For thirty years, SEP has tantalized mission designers by its potential for a specific impulse 10 times greater than that attainable by traditional chemical propulsion systems. A number of programs attempted to harness this potential, but were halted by immature technologies that could not perform to the overreaching expectations. The cost of an ion propulsion system was also perceived to be prohibitive

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[5] Variable Trajectory Optimization Program
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because of the required development effort. The cost and schedule risk of a new technology development effort, primarily due to the long testing times required for hardware qualification, were also considered unacceptable. Fortunately, by conservatively derating today's mature technology, it is possible to perform significant and cost effective SEP missions. Only the negative perceptions left over from previous overreaching efforts need to be changed. This is best done by economically demonstrating and validating a SEP system that can be used for robust space science missions.

Cost Effective NSTAR Ion Propulsion Program

The NSTAR program is designed to resolve these issues by building and testing ion propulsion subsystems, culminating in the flight validation of an operational ion propulsion system[7]. The program includes a ground test and a space demonstration of the ion propulsion system that will examine all key aspects of ion propulsion. Some of these aspects include spacecraft integration, ion system and spacecraft. interaction, ion system performance, potential science interactions, autonomous spacecraft operation, and mission operations.

To minimize cost and risk, the NSTAR program is using derated ion thruster technology, derived from 30 years of research and development, to develop the ion propulsion system. The sacrifice of potential performance due to engine derating is acceptable because of the exceptional performance of modern ion propulsion systems. To make ion propulsion available and affordable for future missions, NSTAR is developing modular elements that can be mixed and matched as needed. Such a design will allow potential system designers to mix and match appropriate parts for their mission. Some of these modular elements include the appropriate size solar array, and the number of thrusters and power processors needed to meet the total impulse, attitude control, and mission reliability requirements.

NSTAR will validate a module with a maximum input power to the power processing unit (PPU) of 2.5 kW, resulting in an Isp of 3,300 seconds with a thrust of 91 mN. The thruster can be throttled over a continuous range of 0.75 kW to 2.5 kW. in this configuration, Isp and thrust level will vary non linearly over the throttle range. Subsequent to the NSTAR program, it is expected that the development of ion propulsion for a specific mission will only require the acquisition of subsystem modules, and the concurrent integration of the system with the spacecraft and solar array.

BENEFITS OF ION PROPULSION FOR ASTEROID AND COMET RENDEZVOUS MISSIONS

The chief benefit of ion propulsion is its ability to quickly rendezvous robust spacecraft to targets of interest using small launch vehicles. The high performance of ion propulsion systems permits the use of very low launch energies (C_3 of 3 to 15 km^2/s^2). Most SEP asteroid and cornet rendezvous missions can use the low cost Delta 11 (7925)[8] or possibly use the new and mm-e affordable Medium/Light launch vehicles[9] once they become available.

Available ion propulsion technology allows for the rendezvous of a 200 to 450 kg payload (spacecraft and science instruments) to almost all of the main belt asteroids, and also to most short period comets[10]. This substantial performance capability permits the delivery of a significant science payload by a low cost spacecraft using conventional, and possibly off-the-shelf hardware.

in addition to superior performance, ion propulsion reduces the mission operations cost of asteroid and con-ret rendezvous missions because of the short flight times relative to chemical ballistic trajectories. Most of the asteroid rendezvous missions arrive 2.5 years after launch, while the comet rendezvous missions take about 3.5 years.

The use of a low cost launch vehicle, combined with a shorter mission operations period and the use of conventional, off-the-shelf spacecraft systems, all contribute to a low life cycle cost.

[7] NSTAR Project Plan

[8] JPL Launch Vehicle Summary for Mission Planning (JPL D-6936, Rev. C)

[9] Private communication from LeRC to JPL.

[10] From a comparison of Kopff Vesta, and Ceres post processed EPITOP results to VARITOP analysis on 30 main belt asteroids and 7 comets.

RANGE OF MISSION APPLICATIONS USING ION PROPULSION

Target Ranges of Solar Electric Propulsion

The solar array size and minimum engine throttling, characteristics define the solar range limit of SEP operation. Solar array s are generally ineffective for propulsion at heliocentric distances greater than 3 AU because of the low power available beyond this point. This constraint does not mean that missions are limited to targets inside 3 AU. For example, a SEP Solar Pioneer spacecraft using a Jupiter gravity assist would thrust inside 2.5 AU and coast beyond 2.5 AU.

Applications of Ion Propulsion

An ion propulsion system with its very high specific impulse is ideal for high AV missions if the AV can be applied over a long period of time. Applications include both Earth orbital and space science missions. Earth orbital applications include station keeping, mbit repositioning, and mbit raising and lowering. Space science missions that benefit from SEP application include, but are not limited to asteroid rendezvous, comet rendezvous, fast planetary fly-bys and probes, solar probes, and possibly comet anti asteroid sample returns. Earth vicinity and lunar missions will also benefit from SEP. In addition, SEP may also benefit fly-by, orbiter, lander, and sample return missions to Mars, Venus, and Mercury.

Ideal missions for SEP are rendezvous missions with asteroids and comets. These missions require large AVS difficult to obtain using chemical propulsion, even with multiple planetary gravity assists. SEP asteroid trajectories use a simple heliocentric spiral from the Earth to the target. Because a spiral is used, the approach velocity at the asteroid amounts to only a few meters per second. Trajectories for comet missions are a bit more complex because of the eccentricity of most comet orbits. However, the closing velocity at closest approach is again only a few meters per second. This slow approach enables a safe crossing of the comet's debris field.

The SEP stage generally permits the launch of missions without the constraint of multiple planetary alignments that enable multiple gravity assists. As a result, SEP performance remains relatively constant from year to year because they cto not depend on gravity assists. 1 lowever, this is not to say that mission designers would not use SEP with gravity assists because many SEP space science missions will benefit from planetary swing-bys.

Finally, a significant associated benefit of SEP is its use of a large solar array. The array may allow for the elimination of the radioisotope thermoelectric generators (RTGs), and radioactive thermal units (RHUs) used on spacecraft that fly between 3 and 10 AU, or farther out. Also, when the SEP propulsion system is not in use, the large solar array creates the potential for high powered science experiments such as imaging radars and active spectrometers.

PERFORMANCE OF ASTEROID AND COMET RENDEZVOUS MISSIONS

As mentioned previously, asteroid and comet rendezvous missions are ideal for SEP because of the large AVS required. Figure 1 compares SEP and chemical ballistic performance for missions to asteroid Ceres, asteroid Vests, and comet Kopff, all launched from a Delta II[11]. The figure shows the net spacecraft mass (delivered mass minus propulsion) and flight time. Notice that the SEP system, assuming hardware being validated by NSTAR, always gives significantly better mass performance with about a 2 to 5 year shorter flight time.

Ceres and Vests, two of the largest main belt asteroids, are used in this example because they are of scientific interest and from a performance point of view, they represent a typical mission (Vests), and a difficult mission (Ceres). Figure 2 shows a typical trajectory. Notice that the SEP trajectory to the asteroid uses a simple heliocentric spiral from the Earth. On the other hand, the corresponding trajectory used by the chemical system employs a Mars-Mars gravity assist (MMGA). This path introduces a much longer flight time> because of the multiple Mars fly-bys. Furthermore, launch opportunities occur infrequent 1 y.

Members of the Comet Rendezvous/Asteroid Flyby (CRAF) Science Working Group (SWG) selected comet Kopff as a prime target. Figure 1 shows that a SEP system can deliver about 300 kg (net spacecraft mass) to Kopff. Two other prime comets selected by the CRAF SWG, Tempel 2 and Wild 2,

[11] SEP data obtained from post processed 1[1'1'1'01' data. Chemical ballistic data from MIDAS program by Carl Sauer.

Figure 1: SEP vs. Chemical Ballistic Performance

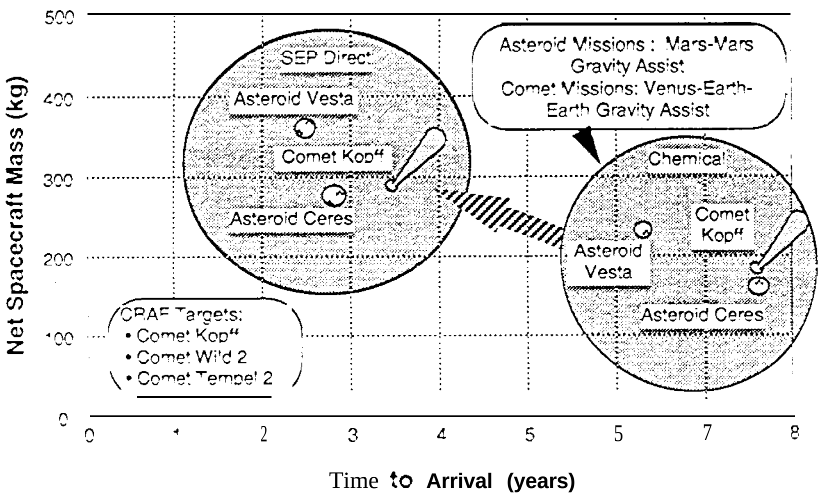


Figure 2: Typical SEP Asteroid Rendezvous Trajectory

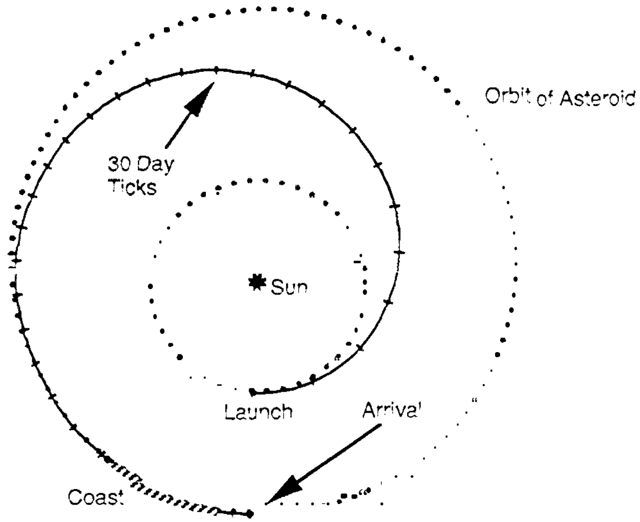


Figure 3: Typical SEP Comet Rendezvous Trajectory

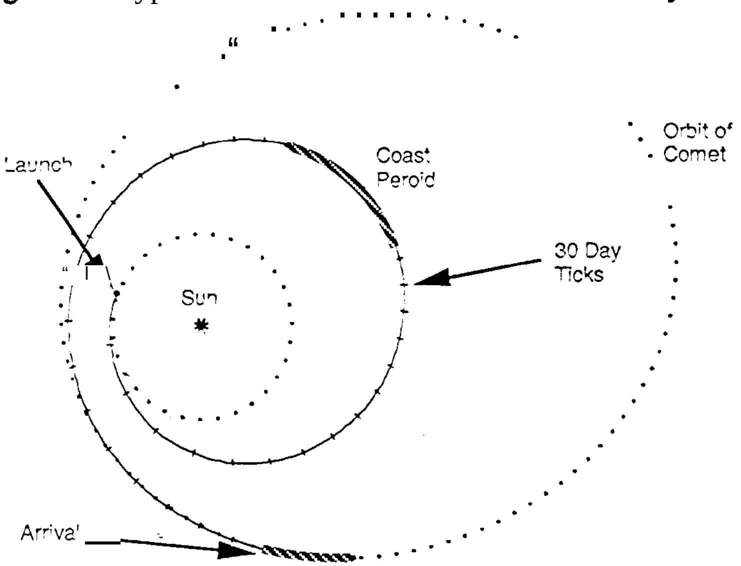
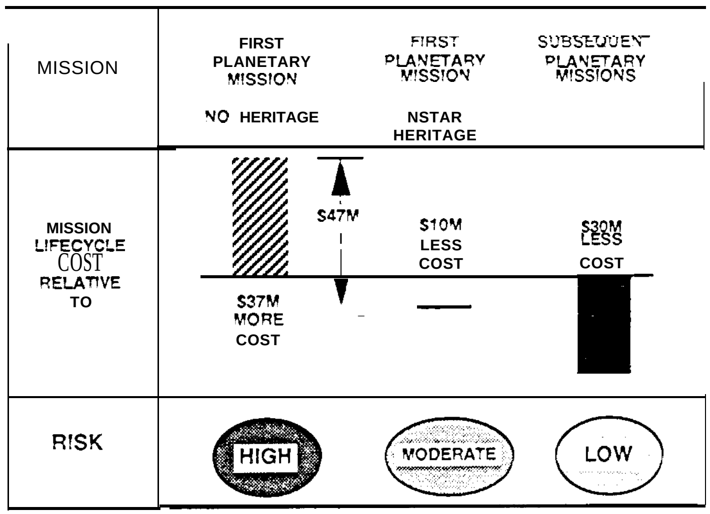


Figure 4: SEP Cost Savings



(Baseline Mission is Main Belt Asteroid Rendezvous) possess similar orbital characteristics to Kopff[12]. Consequently, the SEP performance for missions to these two comets will be similar to Kopff[13]. SEP trajectories for comet missions are generally more complex than for asteroid missions because of the large eccentricity of most comet orbits. The comets of interest typically orbit with a perihelion about 1.5 AU and an aphelion around 5 AU. Because of the limited solar array power at heliocentric distances greater than 2.5 AU, the SEP trajectories involve an elliptical shaped loop around the Sun that catches up with the comet on its outbound leg. Figure 3 shows the trajectory to Kopff. Again, not using gravity assists enables frequent launch opportunities to comets of scientific interest.

DETAILED SEP MASS BREAKDOWN AND PERFORMANCE

Every kilogram of mass in the propulsion subsystem of a SEP spacecraft decreases the amount of payload delivered to the final target. Consequently, NSTAR team members have made a concerted effort to identify the key elements that contribute to the total subsystem mass. The following three tables show the results obtained so far.

Table 1 shows the mass of the solar array and articulation system. The size of solar array is affected by the radiation dose received, caused primarily by solar flare protons for the case of interplanetary missions. Solar array radiation dose is usually defined in effective electron dose greater than 1 MeV. The first row in Table 1 shows the effective electron dose for an active solar period. Subsequent rows show the items needed to determine the mass of the solar array and the solar array power needed at launch. One factor contributing to the total solar array mass comes from the articulation device. On an interplanetary mission, the solar array must be able to rotate along a single axis because the thrust vector may point in any direction with respect to the Sun. The articulation device performs this rotation and passes power using slip rings. All three missions examined in the study used arrays in the 100 kg range.

Table 2 shows the mass breakdown of the entire SEP system with the solar array power shown for reference. Propellant mass was determined by post processing trajectory optimization data from EPTOP, a program developed by Carl Sauer. In this study, two thrusters are used simultaneously to minimize the mass of the SEP system. The next row shows the number of engines needed for the mission. This number is defined by the propellant throughput of the engine before it wears out. NSTAR is designing the engines to have a total throughput of at least 85 kg. One extra engine is added for redundancy. Subsequent rows show the mass of one engine, the combined mass of all of the engines with contingency, and the mass of a gimbal needed to point an engine. A two axis gimbal is assumed for each engine to provide three axis control authority, when paired with another thruster.

The number of PPUs is determined by the number of thrusters operating simultaneously plus one extra for redundancy. Other hits related to the PPUs include a thermal system needed to cool the 250 watts dissipated by the PPU at full power, and the switch unit that allows a PPU to power any engine.

The Digital Control/Interface Unit/Power Supply is the controlling and monitoring infrastructure for the SEP system. It controls the engines, PPUs, and feed system. The feed system itself is composed to three parts that includes a feed element associated with each thruster, a fixed mass feed element that meters the system, and a propellant storage system. The tanks are much lighter than that for a chemical system because SEP uses composite tanks filled with high density xenon stored as a critical fluid. One way to reduce the feed system mass involves flying "yet to be" qualified devices because presently qualified devices were developed for flow rates much larger than currently needed.

An analysis of previous propulsion stages yielded an estimate of the structure needed to hold the SEP subsystems together. This figure amounted to roughly 6.5% of the wet SEP mass. Finally, an allowance for cabling and thermal masses (less the PPU thermal system) was added. Notice that the total SEP system dry mass for these three cases varied from 338 to 372 kg. The SEP system can deliver more than 677 kg to the three targets in this study. Subtracting the SEP system dry mass from the delivered mass results in a net spacecraft mass (spacecraft plus science minus SEP) of over 280 kg.

[12] From the Mariner Mark II/CRAF mission plan

[13] From a comparison of VARITOP trajectory solutions of Kopff, Tempel 2, and Wild 2 to post processed EPTOP solutions for comet Kopff.

Table 1: Solar Array Mass Calculation

	Dose to get to Kopff	Dose to get to Vesta	Dose to get to Ceres
Electron Total Dose at End of Cruise Burn	1.461'.114	1.43E+14	1.43E+14
Solar Array Loss Due to Electrons	8.9%	8.8%	8.8%
Total Loss at End of Cruise Burn	15.9%	1 5.8%	15.8%
Additional Spacecraft Power Needed Above 250 W Allocated (W)	250.00	250.00	250.00
Max. Solar Range With Engine On (AU)	2.4?	2.57	2.64
on Power Needed at Max. Solar Range, Ref. to AU (kW)	8.00	8.00	10,.30
End of Life APSA Power (kW)	9.46	9.65	1204
Beginning of Life APSA Power (5"/. contingency kW)	10.97	11.18	13.95
Beginning of Life Specific Mass of Solar Array W/kg)	135.93	136.65	143.91
Solar Array Mass (kg)	80.68	81.80	96.91
Solar Array Articulation Device (kg)	12.00	12.00	12.00
Total Solar Array System Mass (kg)	92.68	93.80	108.91

Radiation dose data from JPL IOM 5215-93-37 "1 Mev-electron Equivalent Fluence for Solar Activity", Martin Ratliff to Gerry Murphy. Used 1 year interval (more conservative than the 3 year case).

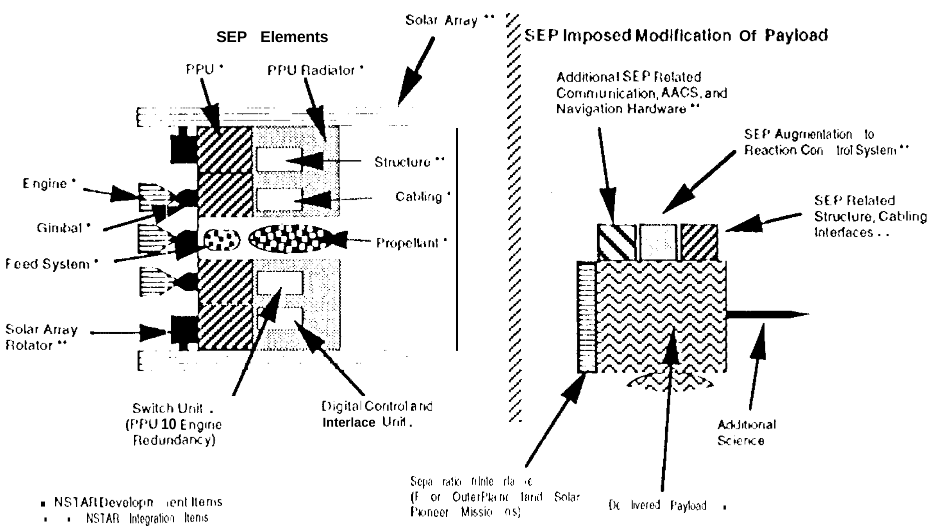
Table 2: SEP Performance Using NSTAR Hardware

Item	Contingency	Value	Vesta Rendezvous	Ceres Rendezvous	Kopff Rendezvous	COMMENTS
Solar Array Power			11.0 kW	11.2 kW	13.9 kW	Solar Array Power at Launch, 250 W Spacecraft Power, Unmodified APSA with Land Temperature, 7% Degradation for UV, Cover Glass Darking, Radiation Dose Over Mission
Propellant Mass			246 kg	342 kg	255 kg	Propellant Mass Used
# of Engines			4	5	4	Number of Engines Defined by Total Egi.. Propellant Equals Total Propellant Divided by 85 kg Rounded Up, Plus 1 for Redundancy
30 cm. Engine	30%	7.0	36 kg	46 kg	.36 kg	Present Specification for 11 RC
Gimbals	30%	40	21 kg	26 kg	21 kg	Estimate
# of PPUs			3	3	3	Number of Engines On at any Time, Plus 1 for Redundancy
Power Processing Unit (PPU	30%	119	46 kg	46 kg	46 kg	Present Specification for LFRC
PPU Thermal	30%	6.5	25 kg	25 kg	25 kg	Variable Conductance Heat Pipes
Switch Unit	30%	20	3 kg	3 kg	3 kg	From TRW Vesta Study
Digital Control/Interface Unit/Power Supply	30%	6.0	8 kg	8 kg	8 kg	From 1 KW Vesta Study
Feed System Mass/Engine	30%	3.0	16 kg	20 kg	r6 kg	From Dave Stevens First Cut Estimate Class A Single Variable Flow System May be reduced by a factor of 2 by using devices not yet qualified
Fixed Part of Feed System Mass	30%	114	15 kg	15 kg	15 kg	From Dave Stevens for Class A,. 114 kg May be reduced by a factor of 2 by using devices not yet qualified
Feed System Tankage	30%		10 kg	13 kg	11 kg	From Dave Stevens for Class A, 50 deg C Ma., 2000 psi, = O .3413 + O 0413* Prop Mass 7.33e-5 * Prop Mass^2 + 1 517.7. Prop Mass^3 .1 169e 10 " Prop Mass^4
200 Volt APSA Solar Array Rotator Mass	5% for Solar Array	Calculated	93 kg	94 kg	109 kg	Uses APSA Scaling Data from 1 RW; 5% TRW Contingency; Corrected for Radiation Dose to Time of Arrival, Uses Ratliff Radiation Data at 95% Confidence Rotator From 1 R W Study
SEP Subsystem Subtotal			273 kg	296 kg	289 kg	
Cabling	Included	5.0%	r4 kg	15 kg	14 kg	From Rule of Thumb, % of SEP Subsystem Total
Thermal Subsystem Less PPU Thermal	Included	5.0%	14 kg	15 kg	14 kg	From Rule of Thumb, % of SEP Subsystem Total
Structure	Included	6.5%	38 kg	46 kg	40 kg	From Propulsion Stage Analysis, % of Wet SEP Mass
Total SEP System Dry Mass			338 kg	372 kg	3$8 kg	
Trajectory (1 1,11019 Performance			708 kg	680 kg	677 kg	Constant Beam Voltage and Analog Throttling Will Improve Performance. Hold Up and Residual Included as Part of Improvement
Net Spacecraft Mass			370 kg	308 kg	319 kg	

Table 3: Net Spacecraft Mass, Contingencies, and Margins

Elements	Contingency	Discovery-Class Spacecraft with Science			Enhanced Spacecraft with Enhanced Science		
		Vesta Rendezvous	Ceres Rendezvous	Kopff Rendezvous	Vests Rendezvous	Ceres Rendezvous	Kopff Rendezvous
Pluto Spacecraft with Navigation/Imaging Camera Including 407'0 Contingency	40%	125 kg	125 kg	125 kg	125 kg	125 kg	125 kg
SEP Related Communication AACS, Power, and Thermal (Allocation)	30%	26 kg	26 kg	26 kg	26 kg	26 kg	26 kg
Additional Science Package (Allocation)	30%	26 kg	26 kg	?6 kg	65 kg	65 kg	65 kg
SEP Augmentation to Reaction Control System (Allocation)	30%	26 kg	26 kg	26 kg	39 kg	39 kg	39 kg
Additional Mass for SEP Related Structure @ 30%	30%	30 kg	30 kg	30 kg	51 kg	51 kg	51 kg
SEP/Spacecraft Separation system @3%		0 kg	0 kg	0 kg	0 kg	0 kg	0 kg
Resultant Net Spacecraft Mass		233 kg	233 kg	233 kg	306 kg	306 kg	306 kg
Available Net Spacecraft Mass for Delta II (7925) = Total Delivered Mass - SEP Mass		370 kg	308 kg	319 kg	370 kg	308 kg	319 kg
Margin Available for Additional Enhancements		137 kg	75 kg	86 kg	65 kg	3 kg	14 kg

Figure 5: SEP System Functionality



'1'able 3 shows two examples of possible spacecraft mass breakdowns. The first is a Discovery class spacecraft, and the second is an enhanced spacecraft. The examples use the Pluto Fast Fly-by spacecraft with SEP related communications, attitude control, power, and thermal modifications added[1] Additional science is added where appropriate to meet the mission goals[15]. A reaction control augmentation is also added to control the spacecraft when the SEP is not operating. Finally, structure is added to hold the additions to the Pluto Fly-by spacecraft.

The last three rows show the resultant spacecraft mass, the available spacecraft mass from the previous table, and the margin that may be used for other enhancements. Notice that all of the enhanced. missions have positive margins.

COST OF USING ION PROPULSION

Costs for using ion propulsion include the ion propulsion hardware, long operation of the ion propulsion system, and the integration of the SEP system into the spacecraft.

Cost Comparison of Ion Propulsion Missions with Chemical Propulsion Missions

in a study performed by Jerry M. Olivieri using the JPL cost model, the life cycle costs of ion propulsion missions subsequent to NSTAR are favorable to chemical ballistic missions because SEP permits the use of small launch vehicles, has faster trip times, and delivers enough mass to permit the use of conventional, and possibly off-the-shelf spacecraft.

Figure 4 shows the life cycle cost summary. It is important to note that if a planetary SEP mission is attempted without any ion propulsion heritage, that program may cost an additional $47 million and incur a considerably higher risk, increased cost, and longer schedule than a chemical ballistic mission. However, if the NSTAR program is successful, and if NSTAR heritage is used on the first planetary mission, then that program may have a reasonable amount of risk, and may cost $10 million less than a comparable chemical ballistic mission. Subsequent SEP missions will also be significantly less expensive because of the' shorter mission durations and / or the use of smaller launch vehicles.

The major assumptions used in the comparison are: FY 1994 dollars, Class "B" mission, protoflight approach with partial spares, project start in FY 1996, and launch in FY 2000 aboard a Delta II (7925). The SEP system uses a 10 kW solar array, the thrusters fire for 2.5 years, the mission life is 3.5 years. ('heroical systems used in this study were assumed to have a specific impulse of315 seconds.

Cost of Mission Operations

The operations cost of an ion propulsion spacecraft is conservatively estimated by two studies to be 30% higher than a conventional spacecraft because the spacecraft is almost always thrusting[16]. This constraint is perceived to give a larger load to the navigation and spacecraft team. However, the cost may be more perceptual than real because the timing of any particular maneuver period (typically one week in duration) is not critically important to the outcome of the mission because SEP systems have good trajectory recovery capability.

Low thrust spacecraft do not require a repeated return to a prescribed trajectory as ballistic missions using gravity assist encounters. Instead, when an inaccuracy in the trajectory exists, a new trajectory can be plotted, This flexibility enables the autonomous operation of the spacecraft, as long as an anomaly is fail safe, and provides for limited monitoring?, by the spacecraft and navigation team.

Low thrust spacecraft also follow a narrow "random walk" type of trajectory due to perturbation in thrust vector induced by the attitude and articulation control system (AACS) and the propulsion system. Fortunately, there is simple navigation solution for each point along the trajectory. The spacecraft is tracked by conventional radio navigation means, and a new navigation solution is uplinked to correct the trajectory as needed.

[14] IAF-93 <).5.410, "Pluto Mission Progress Report: Lower Mass and Flight Time Through Advanced Technology Insertion," Robert L. Staehle, et al. 44th Congress of the International Astronautical Federation. October 16, 1993, Graz, Austria.

[15] Rendezvous missions may require a greater science complement than a fly-by mission. For example, a gamma ray spectrometer is proposed for asteroid and comet missions to measure the elemental composition of the body.

[16] From a study by Jerry M. Olivieri and a study by Ronald Salazar (JPL Members of the Technical Staff).

Cost of Spacecraft integration

Assuming that SEP is to be integrated with a conventional spacecraft, a choice of integration method must be made. If the spacecraft is large enough to have the SEP integrated internally, then no significant additional structure is necessary. On the other hand, if the spacecraft is too small to include all of the SEP system, it will be necessary to add external structure to hold the ion propulsion system together. The SEP elements that must be integrated into a spacecraft are shown functionally in Figure 5. Primary elements include a solar array (SA), power management and distribution, power processing units (PPU), power processor thermal radiators, ion engines, propellant management and feed system, structure, and mechanisms (gimbals and solar array actuators). An electrical and mechanical network holds these subsystems together. In addition, a spacecraft utilizing SEP must be compatible with the system, resulting in a number of possible modifications.

For example, the operation of an active SEP system requires that the attitude and articulation system continuously control the thrust vector of the spacecraft. During the operation of the SEP, attitude control propellants are not needed because the SEP stage provides the control authority. However, during coast periods, the spacecraft must supply the control authority needed to articulate the solar arrays and point the spacecraft. Figure 5 shows the affected elements. Other integration considerations include the power bus voltage, the electromagnetic interference Of the SEP system (conducted and radiated), the thermal impact of the large solar array upon the spacecraft, field of view limitations caused by the large solar array, impact of charge exchange plasma on the solar array, and spacecraft. contamination from the ion engine.

Most cornet and asteroid missions require SEP operation beyond 0.9 AU. Some of the trajectories also require the thrust vector of the SEP system to point in any direction. These constraints require that the spacecraft permit solar illumination on three sides of the spacecraft for long periods of time. Chemical propelled spacecraft usually use some form of a solar shield against long term solar illuminations when inside 0.9 AU. Instead, a SEP spacecraft may use heat pipes, heaters, and other like devices for thermal compensation.

NAVIGATION

Navigation of Interplanetary Trajectories

The navigation of a SEP spacecraft is quite different from present chemical propulsion spacecraft because the SEP system uses a low, but continuous thrust. "The thrust duration for an interplanetary SEP mission is measured in years, not hours as in a chemical propulsion mission. Operations costs will be affected if the spacecraft does not have sufficient autonomy. One way to minimize the impact to mission operations cost is for the spacecraft to follow a weekly thrust vector profile. This scheme is possible because the low thrust of the SEP system changes the trajectory very slowly. Also, possible errors in maneuver execution are not serious because the spacecraft need not return to the original trajectory. An entire family of new trajectories that will meet the mission requirements exists for each point along a trajectory.

The orbit determination process for a SEP mission is also quite different from a chemical propulsion mission because the constant thrust of the spacecraft results in an acceleration four orders of magnitude larger than the stochastic accelerations from a chemical propulsion spacecraft during cruise. Unfortunately, use of conventional radio navigation techniques will result in larger spacecraft position uncertainty as compared to chemical systems. However, this uncertainty is not a problem because there is no need for high precision navigation, except during planetary fly-bys. Then, the ion propulsion system can be turned off, permitting accurate orbit determination by conventional means.

During a rendezvous approach to a comet or asteroid, large position uncertainties can be eliminated by optical navigation. Approach navigation is simple because the closing velocity is very small, and small maneuvers can be used to obtain "bearings only" navigation solutions.

Navigation of Orbits Around Large Asteroids

The navigation around large irregular shaped bodies, such as asteroids, presents a challenge for both chemical and SEP systems because the irregular mass harmonics will cause very large perturbations in the orbit. Spacecraft using either propulsion system will require some form of autonomous navigation to detect the Orbits] perturbations. The SEP system has an advantage because the propulsion system has enough energy to correct for many of the perturbations resulting from the irregular mass harmonics. Chemically propelled spacecraft may need to stay much further from the surface, and may need to carry larger batteries to cope with the long shadow periods resulting from being at a higher altitude[17].

Navigation Around Small Astercrids.anti Comets

Navigation around small asteroids and comets is much easier than around large asteroids because of the low mass. This differential results in much longer orbital periods, making the pace of the navigation solutions slower. Although the AV needed to change orbit altitude and inclination is extremely small (because of the low mass), only a SEP system will permit investigation of a comet coma that extends over hundreds of thousands of kilometers. Optical navigation will be the primary method of navigation around the'sc' bodies.

DEVELOPMENT SCHEDULE

The development schedule for a low cost SEP mission is determined by the NSTAR schedule because without the successful validation of the SEP system, the development cost, schedule, and risk are perceived to be excessive. The NSTAR validation will be completed in FY 1999. Ground testing should be completed in FY 1998 enabling the new start of a SEP program in the FY 1998 to FY 1999 time frame.

[17] Mass harmonics have less effect at higher altitudes. A spacecraft with limited AV may need to be at a higher altitude. Also, depending on the magnitude of the mass harmonics, the orbit may be forced into the shadow of the body. Since the shadow period is longer for higher altitudes, a larger battery may be required.