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Is It SEP Yet?

by

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Abstract

This paper is a presentation of the results of recent studies indicating that solar electric propulsion can be implemented in a Discovery-class scenario to permit an affordable exploration of comets and asteroids in the very near future. Gallium arsenide solar array technology, the availability of space-qualified ion and plasma thrusters, and appropriate power conditioning equipment are cited as enabling factors for an exciting class of missions that can permit exploration of a number of asteroids and short-period comets, using the Delta launch vehicle, before the turn of the century. Launch requirements are about 993 kg to $C_3 = 10 \text{ km}^2/\text{s}^2$ for an assumed 50 to 75 kg complement of science instruments. An advantageous feature of electric propulsion is that the high installed power level, unnecessary for propulsion during rendezvous, enables high science data rates from most potential targets.

Introduction:

Advanced studies of the theory and application of solar electric propulsion (SEP) have been in progress for over 20 years. Advocates of this powerful technology have tried repeatedly to obtain approval for a deep-space SEP mission. The problem has been the classic one of a system without a mission or a mission without a system. A mission that requires SEP cannot be approved because it requires the development and space-qualification of SEP while a SEP mission, in itself, cannot be considered because there is no approved mission that demands the capabilities of such a system. If we had some ham, we could have some ham and eggs, if we had some eggs. This field has been plagued with the ham and egg syndrome almost from its infancy.

SEP has been further hampered by well-meaning advocates who have tried to promote the technology as an efficient means of transportation from LEO to GEO, an argument that appears valid until one works out the penalties for shadowing and radiation damage during passage through the Van Allen belts. When all the factors have been accounted for, SEP usually comes up a poor second to conventional propulsion for Earth orbit raising missions unless unrealistic assumptions are made with respect to potential traffic, specific power, and spacecraft radiation hardness. Only for very long duration transfers (6 months to 1 year) from LEO to GEO, using special radiation-tolerant materials, is SEP economically viable for this application. Of course, solar electric propulsion is ideal for GEO stationkeeping and is used routinely for that purpose. These limitations of SEP for LEO missions no longer apply in deep space out to about the distance of Jupiter's orbit.

In the following sections, we present a fundamentally sound spacecraft and system design concept capable of carrying out many of the deep-space science missions advocated by the principal scientists of the planetary science community -- the study of extraterrestrial bodies in sufficient detail to permit generalizations regarding the structure, energetics, and long-term stability of the ecological balance of our own planet. An important part of this study is the understanding of primitive bodies of the solar system such as asteroids and comets, their history, morphology, differentiation, and composition so as to permit valid comparison of the small-body structure and dynamics with those of the Earth.

It is our intention to show that the technology for Discovery class missions to the asteroids, short-period comets, and many inner solar system targets is now available without the usual uncertainty associated with new programs because much of the technology has already been developed. As professional engineers, we assume technology only insofar as we have been able to obtain verified or reasonably extrapolated data and test results. As long-standing members of the space exploration community, we are concerned that the existence and ready availability of solar electric propulsion technology is apparently not being included in advance planning for the exploration of the solar system.

I. Missions and Methods

The literature of space exploration is filled with reports of studies of optimal transfer from one body to the other, within the inner solar system, using various forms of "low-thrust" propulsion systems. The usual response to proposals to develop such technologies, such as nuclear electric, solar electric, nuclear thermal, and solar sailing, has been that there is no need for such technologies because there are no identified missions requiring these technologies. Solar Electric propulsion (in which the power is derived from solar panels and the propulsion is provided by ion thrusters) has received the most study for deep space missions and is, perhaps, the most mature of the technologies due to the on-going study of ion-thruster technology and the rapidly advancing technology of solar panels and power conditioning equipment¹⁻⁴. Russian stationary plasma thruster technology has recently come under consideration in the literature⁵ and a number of ion thruster types are under consideration for potential nuclear electric applications⁶.

One of us (Reinert) recently suggested that the combination of advances in solar panel technology and the recent interest in development of moderate to small spacecraft for deep-space exploration might make a Discovery-class asteroid rendezvous mission feasible, especially if such a mission could be launched by the Delta II 7925 launch vehicle. This suggestion was sufficiently exciting to us all that we began to review the literature for missions that could be achieved by such a system. We were surprised to find that a great many missions, formerly considered "big missions," could be achieved using existing technology. Our only reservations were ones of thruster lifetimes which have been extrapolated from less than 1000 hour tests to

10000 hour expected lifetimes. We opined that, even with pessimistic estimates for thruster life, we could still accomplish a number of important deep-space missions by carrying additional small thrusters to provide granularity of thrust and as substitutes for short-lifetime units. We estimated that inner-main-belt asteroid rendezvous could be accomplished by a 7 kw Solar Electric Explorer (SEEX) vehicle, running at an Isp of about 2800 seconds and delivering a total deep-space impulse between 8 and 10 km/s.

Our first try at a spacecraft design, called SEEX-1, is a machine made up of existing systems that can be obtained from experienced sources and integrated within about 2 years, to provide a very capable workhorse for inner solar system exploration. The spacecraft would deliver about 60 kg of science instruments to the inner main belt asteroid Vesta using a 2.5 year transfer with an Earth departure energy (C_3) of about $10 \text{ km}^2/\text{s}^2$, on the Delta II 7925 launch vehicle. More exciting, from a scientist's viewpoint, is the high data rate (about 40 kbps) that could be sustained during rendezvous due to the unprecedented power level available to the transmitter and the X-band capability built into the spacecraft and now available through the DSN 70 meter network.

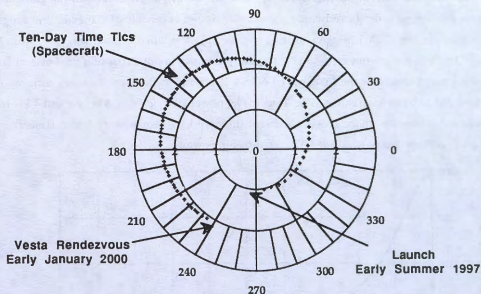


Fig. 1 Non-Optimal SEEX-1 Rendezvous with Vesta
 ($C_3 = 10 \text{ km}^2/\text{s}^2$; $T_0 = 0.298 \text{ n}$; $I_{sp} = 2678 \text{ s}$; Eng. Eff. = 64%; $\Delta V = 8800 \text{ m/s}$)

Fig. 1 shows a transfer trajectory from Earth to Vesta for the 7 kw SEEX 1 vehicle. It was assumed that 4.478° of the required inclination change (with respect to the ecliptic) was provided by the launch vehicle by launching at a time that would yield a 52° declination of the outgoing asymptote with respect to the ecliptic (28.5° Equatorial + 23.5° obliquity). This is a scenario that could be achieved by selecting the time

of day for the launch and would require no degradation of performance for a typical Delta II launch from the Eastern Test Range at Cape Canaveral. The initial spacecraft mass at separation from the launch vehicle was assumed to be 993.3 kg ($C_3 = 10 \text{ km}^2/\text{s}^2$) for the Delta II 7925 and the mass delivered to Vesta rendezvous was 710 kg, a substantial payload including the 60 kg of science payload and the residual 1500 watts of electrical power available for data downlink. In the SEEX-1 design, we have assumed that only about 340 watts of this residual power is used for data transmission because of the high mass of the transmitters required to take advantage of the excess power during cruise and rendezvous operations. On-board storage of data and modern compression techniques could enhance the throughput by a factor of two or four.

Several computer programs and formulations are available for optimizing these low-thrust trajectories. Some are "direct" or hunting procedure programs but the most capable are those based on the Calculus of Variations. These tools have been developed by our colleagues at Analytical Mechanics Associates, the Jet Propulsion Laboratory, Science Applications International, and AdaSoft Inc. Some of them can be obtained through the COSMIC library. As these tools were not immediately available to us, and because the kind of analysis required is so abundant in the literature, we chose to perform a few numerical integrations with simplified thrust profiles, in order to show that a particular design is capable of achieving the suggested missions. For example, the SEEX-1 design was used to develop a simplified transfer trajectory from Earth ($C_3 = 10 \text{ km}^2/\text{s}^2$) to Vesta rendezvous on a 2.5 year trajectory using an impulse (per unit final mass) of about 8800 m/s. The engine efficiency for the 7 engines (1 NASA 30 cm and 4 UK-10 cm thrusters with two spare UK-10s) was assumed to be a conservative 64%. Thus the jet power ($= 1/2 Tc$) was only about 4 kw from a designed 7 kw power subsystem. In the above, c is the exhaust speed, $c = g_{sp}$, where g is the acceleration due to gravity at Earth's surface (9.8066 m/s^2) and T is the thrust in newtons.

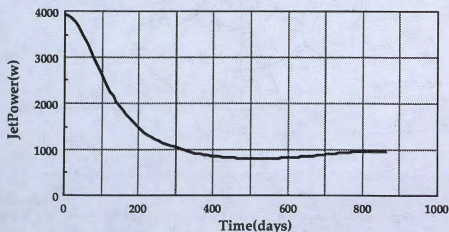


Fig. 2 Power Profile for SEEX-1 Vesta Rendezvous

Fig 2 shows the jet power profile during the Earth-Vesta transfer; the engine efficiency required to achieve such a profile is only 64%, a value easily achievable, for the thrusters considered, at an I_{sp} of 2678 sec. The time history of the thrust magnitude is obtainable from Fig. 2; simply multiply the jet power by 2 and divide by the exhaust speed in m/s (26262.1 for the SEEX-1 vehicle). The initial thrust is 0.298 n (1 NASA 30 cm thruster + 4 UK10 cm thrusters) requiring an input power to the thrusters of 6224 watts at 1 au. The initial acceleration of the vehicle is 0.3 mm/s^2 .

In the calculations used to generate the trajectory of Figs. 1 and 2, we have assumed that the power falls off as the inverse square of the distance from the sun. In previous studies of silicon solar arrays, the inverse square profile was found to be overly conservative because of temperature/efficiency considerations⁴. It is not clear that gallium arsenide cells will exhibit the same positive response to lower temperature as the earlier technology. We have assumed an inverse square power law in all the simulations and calculations considered here. We consider this a conservative assumption for outbound trajectories.

After verification of the Vesta rendezvous, we learned that our colleagues at the Jet Propulsion Laboratory⁷ had been working on similar studies and had discovered a Vesta rendezvous opportunity using the Delta II 7925 launch vehicle. Recent conversations^{8,9} have revealed that the JPL analysts have identified a Vesta rendezvous achievable using a solar electric power system of only 5 kw. In this option, the launch vehicle provides more of the energy ($C_3 \approx 20 \text{ km}^2/\text{s}^2$) for the transfer than our ($C_3 = 10 \text{ km}^2/\text{s}^2$) trajectory. The (optimized) transfers of the JPL studies achieved rendezvous using only 7500 m/s of impulse supplied by the solar electric propulsion system. It is expected that these more complete studies will appear in the literature soon, and we are grateful to receive these inputs that corroborate our own analyses.

Mission Impulse Requirements

It became obvious, upon study of our preliminary calculations, that missions even more ambitious than the Vesta rendezvous were available within the classification of the Discovery program, generally taken to represent missions with funding requirements under \$150 million 1990 U.S., including the launch vehicle. It is clear to us, after this short study, that many scientifically valuable missions are possible, within the funding limitations of the Discovery program, that make use of solar electric propulsion.

Studies of multiple asteroid rendezvous missions, comet rendezvous, and asteroid and comet sample return missions were conducted in the early 1980s. These studies revealed that missions, capable of significant advancement of our scientific knowledge, were possible using solar electric spacecraft in the 15 to 30 kw range. It has been known for many years that such spacecraft are viable. These capabilities are related to

the requirements for deep-space impulse necessary for rendezvous with several asteroids. In the past, it has been generally accepted that main-belt asteroids were of more scientific interest than the more accessible Apollo/Amor groups of objects. In the following paragraphs, we present our estimates of missions, in various classes of dynamical difficulty, that will be enabled by the development of the three levels of SEP spacecraft, called SEEX-1, 2, and 3, to be presented later in this paper.

Apollo/Amor Asteroid Multiple Rendezvous

These Earth/Mars crossing asteroids are among the most accessible solar system targets and can be easily reached (individually) by ballistic rendezvous techniques. Several of them (3 to 5) could be visited in a single mission, using the 11 km/s ΔV capability of the 7 kw SEEX-1 vehicle, provided that their orbital planes are not more than a few degrees from the ecliptic and that the rendezvous transfers do not require very long periods of low power thrusting to use up the thruster lifetime. There are so many of these objects that such a mission, if deemed scientifically worthwhile, could be easily identified.

Short-Period Comet Rendezvous

Comets like Temple 2, Kopf, and Encke are accessible to SEP rendezvous from Earth but with somewhat more difficulty than the Apollo/Amor objects. The authors of Ref. 3 have identified a 1994-launched Temple 2 rendezvous using an optimized ΔV of about 8500 m/s from an Earth-launch C₃ of 10 km²/s². In Ref. 4, the authors present a similar scenario requiring about 8200 m/s but with higher launch energy requirements. Thus, the SEEX-1 vehicle will have more than enough capability to provide a rendezvous with Temple-2 at a later apparition. It should be noted that SEP trajectories are much less sensitive to specific launch and arrival dates than ballistic trajectories and, unless the target is very far out of the ecliptic, the SEP performance variation from apparition to apparition, is relatively unimportant.

Main-Belt Asteroid Rendezvous

Vesta is considered a main-belt asteroid but is near the inner edge of the "main" belt. Even the largest of the main-belt asteroids, Ceres, is accessible to a rendezvous using the SEEX-1 vehicle with a ΔV requirement of about 9 km/s on an 850 day transfer from Earth launch to C₃ of about 14 km²/s². Thus, the 11 km/s capability of the SEEX-1 vehicle could easily provide a Ceres rendezvous and might even be able to make it to another main-belt asteroid such as 447 Valentine as identified in Ref. 3. Although such a mission would push the SEEX-1 vehicle to its limits, the SEEX-2 design, with its 14.3 km/s ΔV capability, launched perhaps on an Atlas II/Centaur, could easily accomplish multiple main-belt asteroid rendezvous. SEEX-3, with 17.8 km/s of ΔV capability, could achieve most of the multiple main-belt asteroid missions identified in Ref. 3.

Comet/Asteroid Sample Return and/or Tagging

Although the SEEX designs were not intended for sample return missions, they provide much more than enough ΔV to accomplish such missions. We are convinced that asteroid tagging and sample return missions, while probably not within the Discovery class, can be readily composed of various features of the three designs presented here. Our intent is to show that an affordable program can be established to expand our exploration capabilities and that such an expansion can be reasonably built on the preliminary designs of this paper. We have taken the attitude that we are proposing this design to a wider than usual range of potential entrepreneurs. We consider the design mature enough that it might be considered by private companies who may wish to claim a small asteroid by tagging it with a long-life transponder, and, after tracking it continuously for a substantial time, stake a claim to its resources at some future time when commercial operations are viable. Short period comets offer no less exciting possibilities for future interplanetary prospectors. The asteroid sample return missions of Ref. 3 require about 8 or 10 km/s for (direct return) sample return missions from near-earth asteroids and over 20 km/s for the multiple main-belt asteroid sample return missions. There is no doubt that many such missions, involving the return of a selected sample of an asteroid or comet, can be achieved with the 17.5 km/s ΔV capability of the SEEX-3 design.

Table I shows a set of potential missions in the inner solar system that could be carried out by one or more of the SEEX vehicles. Also shown is the estimated ΔV required and potential launch vehicles.

Mission	ΔV (km/s) / Vehicle	Launch Vehicle
Multiple Apollo/Amor Ast. Rendez.	5-10 / SEEX-1,2	Delta, Atlas
Short Period Comet Rendezvous	7-10 / SEEX-1,2	Delta, Atlas
Main-Belt Asteroid Rendezvous	7-15 (Multiple) / SEEX-1,2,3	Delta, Atlas, STS/TOS, Titan III
Ast/Comet Sample Return/Tagging	10-20 SEEX- 2,3	Atlas, STS/TOS, Titan III, Titan IV

Table I Some Candidate Missions for Solar Electric Propulsion

The considerations above have been incorporated into a preliminary set of requirements for what we consider a viable Discovery class solar electric spacecraft design. These preliminary requirements are listed in Fig. 3.

Req't Area	Parameter	Value	Comments
Launch Date	Calendar year	1998-2002	depends on target selection
Mission Duration	Prelaunch I&T, (Mo, max)	3	
	Propulsive Cruise, (Yr, Max)	3	
	Non-propulsive Science (Mo, Max)	6	12 is goal
Probability of success	Probability of System Successful Operation through EOM	TBD	Goal is $P_s = 0.80$ @ 4 yr
Launch Segment	Launch Vehicle Type	Delta 7925	9.5 or 10 ft Faring; Launch from ETR
Mission Orbit Compatibility	Propulsive Cruise: Periapsis/Apoapsis distance, (km/Au, min:max)	1.05E+8/0.7: 3.740E+8/2.5	Covers most main belt asteroids
	Science Operations, distance to Earth (Km/Au, max)	5.236E+8/3.5	Most main belt asteroids
	Science Operations, distance to Sun (Km/Au, max)	3.740E+8/2.5	Covers 4 comets, Most main belt asteroids
Payload Accommodation Characteristics	Maximum Mass (kg)	50-100	Mission specific
	Max Volume (ft ³ /m ³)	30/1	Delta Faring Limits
	P/L pointing Accuracy, 3-axis, deg, 3-sigma	0.05	Stellar reference
	P/L pointing Knowledge, 3-axis, deg, 3-sigma	0.01	
	P/L power, average (Science ops phase), W, max	250	
Propulsion/ Maneuver	design ΔV , m/s, max	10000-18000	with min (50 kg) P/L
	Vernier maneuver ΔV , m/s, min	25	Science ops mode
	RCS impulse, N-S, min	2200 (TBR)	Science ops mode only
Comm Segment	Type	DSN	70 and 34 M net req'd
	Frequency	X-Band	As MO, NEAR
	Cruise Coverage, # & Frequency of passes	1/Mo, 34 or 70M	Up to 4 Yr; 70M for range > 2 Au
	Science ops	70 M	Continuous
Ranging/ Navigation	Cruise, (Type)	DSN Ranging	Accuracy as MO, GLL
	Rendezvous, Recon and Approach	DSN+Optical	Relative Accuracy < 2 km, 2m/s
Mission Ops Segment	Type (Baseline)	Ball STACCS	Workstation Based & flight Proven
	Location	U of A, JPL	

Figure 3 SEEX S/C Mission Requirements Summary

II. SEEX Operational System Design

Requirements

The SEEX Vehicles must be compatible with the mission requirements outlined above. The spacecraft must also be compatible with all mission environments and meet the functional requirements described in Fig. 4.

Subsystem	Mission Phase	Functional Requirement
General	Launch	Mass at separation < 1279 kg
Mechanisms	All	Articulate HGA over 2 pi ster. centered on -Y axis (post-deploy) Articulate Array wings 360 deg around Z-axis Provide HGA mast release and deploy 90 deg around Z-axis
ADCS	Cruise	Point Z axis Perpendicular to sunline & orbit plane with 2 deg accuracy
	Maneuver/ Rendezvous :	Point inertial with < 0.10 deg accuracy Provide Earth pointing vector to HGA
	Science/	Point inertial with < 0.05 deg accuracy
	Recon:	Provide Earth pointing vector to HGA
Telecom	All:	Accept 8 bps minimum command uplink
	Cruise:	Provide ≥2800 bps Tlm downlink
	Science ops:	Provide ≥40 kbps Tlm downlink
C&DH	All	Provide decoding, storage and distribution for up to 512 cmds
		Provide 1 Gbit of data storage
		Acquire, format, store and downlink P/L and housekeeping data
Electrical Power	Cruise:	Provide survival power to P/L
		Provide 100 W avg to subsystems
		Provide ≥ 13 Kw max to SEP subsystem
	Maneuvers/ Science ops	Provide survival power to P/L Provide up to 500 W to subsystems
Propulsion/ RCS	Cruise	Provide startburn Acceleration ≥ $3.0 \times 10^{-4} \text{ m/s}^2$
		Provide up to 18000 m/s of ΔV
		Gimbal thrusters to provide torques for cruise mode attitude control
	Science Ops	Provide RCS Torques, TBD N-M-S angular impulse and minimum impulse bit A/R for science mode attitude control Provide 25 M/s of delta-V in < 3600 sec for vernier maneuvers
Thermal Control	All	Maintain all subsystem and P/L units within temp limits
		Minimize heater power

Fig. 4 SEEX S/C Operational System Functional Requirements Summary

System Design

The SEEX design approach combines state of the art spacecraft design (similar in many respects to the NEAR spacecraft) with Xenon ion thruster and solar array technology that is either launch ready or on the verge of readiness to demonstrate the performance advantages available from electric propulsion in missions of the mid to late nineties. To illustrate the range of performance available, we have defined three vehicles with essentially common subsystems but with increasing levels of array power.

The SEEX-1 configuration uses space-proven array technology to minimize required development and allow an IOC of 1995. The 7 kW available from the existing array limits performance to the low end of the available range, yielding a characteristic velocity of 11.6 km/s. SEEX-2 uses the Advanced Photovoltaic

Solar Array (APSA) under development by TRW for JPL¹. The increased level of installed power (10.8 kW) allows the characteristic velocity to be increased to 14.3 km/s. This vehicle could also be available by 1995 if the APSA development schedule described in reference one is realized. SEEX-3 is essentially identical to SEEX-2 except for the use of GaAs/Ge solar cells which increase the array output to 14.5 kW, allowing the characteristic velocity to increase to a little over 18 km/s. A SEEX-3 IOC of 1996 was assumed to allow for backfit of the more efficient cells to the baseline APSA array.

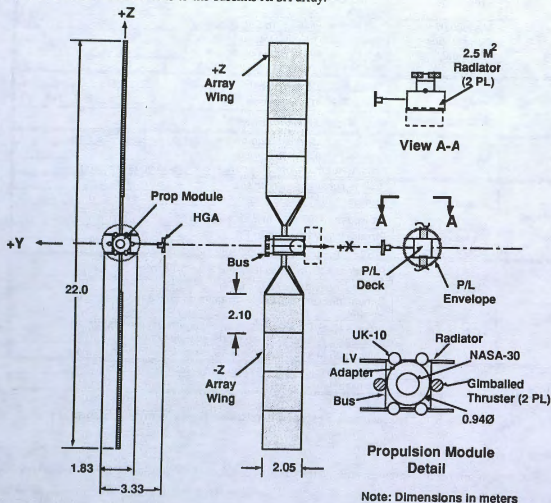


Fig. 5 SEEX 1 General Arrangement Showing TELECOM-2 Array Configuration

General Arrangement

The SEEX-1 S/C configuration in its operational configuration and its coordinate system is shown in Fig. 5. The SEEX 2 and 3 configurations are shown in Fig. 6. Key features shown for each include the rectangular primary structure; The two radiators used for vehicle thermal control; the plus and minus Z solar array

wings, and the 1 m diameter mast-mounted high gain antenna (HGA). A cylindrical adapter on the core's -X face provides the Launch Vehicle interface and mounts the SEPS's thrusters and propellant tank. The SEP electrical units and the S/C subsystems are mounted to the core's plus and minus Y faces. The Z faces mount the thermally isolated radiator panels and the solar array wings.

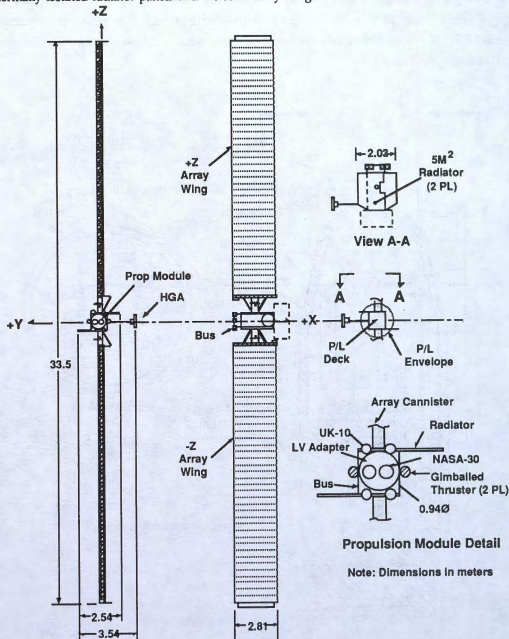


Fig. 6 SEEX 2-3 General Arrangement

Launch System Interface

Figure 7 shows the stowed SEEX 2-3 S/C system installed in the Delta 7925 ten foot fairing. The spacecraft sits on a stretched version of the existing MDAC 3712 Payload Attach Fitting (PAF), and is separated using an MDAC supplied marmon-clamp separation system. The volume available in the fairing allows use of fixed radiator panels. The HGA stows by folding its mast 90 degrees to tuck it up against the spacecraft -Z face. The smaller SEEX-1 configuration uses the same PAF and separation system, but is compatible with the 7925's 9.5 foot fairing.

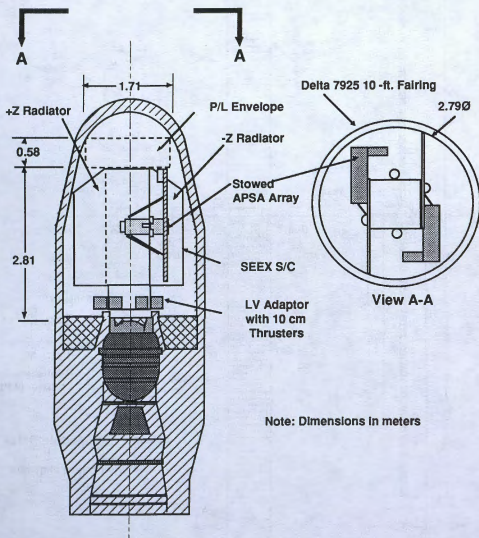


Fig. 7 SEEX 2-3 Vehicle Stowed for Launch in Delta 10 ft. Fairing

Functional Configuration

The Spacecraft functional design minimizes mass by centralizing spacecraft command, control and data handling functions in redundant microcomputers. Reliability is provided by a combination of block and functional redundancy that eliminates mission critical, single point failures.

Figure 8 is a functional diagram that shows the subsystems, their key elements, and their functional interfaces. The diagram indicates the use of the MIL-STD-1750A based microcomputers in the redundant Spacecraft Control Unit (SCU) to run the entire system. During the bulk of cruise operations, only the basic C²&DH elements, ADCS sensors, and the SEP system are active. The SEP system and the Telecom subsystem's 40w RF amplifier draw power directly from the solar arrays; other spacecraft units are supplied with 28 VDC power by DC-DC converters in the PCU.

Structure

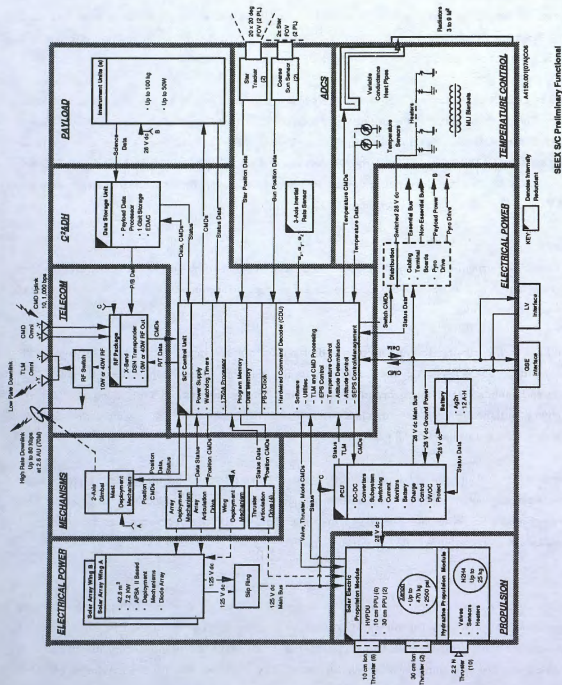
The SEEX vehicles' structure and mechanisms subsystem has to mount and support all subsystem and payload units during launch and ascent induced loadings, and then deploy the antennas and solar arrays into their operational orientations.

The SEEX 2 and 3 primary structure is composed of a 1x1x2.8 m rectangular core fabricated from GFRP honeycomb to minimize mass. A cylindrical adapter on the -X face provides the interface to the Delta 7925 3rd stage, and mounts the SEP thrusters and the Xenon tank. The Structure for SEEX-1 is identical in its basic arrangement, but the primary structure is shortened to 2 meters length to save weight and provide compatibility with the Delta 7925 9.5 foot fairing .

Mechanisms for array and antenna deployment are based on space-proven motor and spring driven actuators and pyrotechnically actuated release mechanisms. Torque margins are at least 400% for all actuators and release mechanisms are redundant. The HGA two-axis gimbal is a Schaeffer magnetics unit flight designed for the IRIDIUM spacecraft; it will be space-proven by 1995.

Telecom

The telecom subsystem receives commands and downlinks telemetry during all mission phases. All communications are at X-band. The integrated system uses four low gain command and telemetry omnis and a mast mounted 2-axis gimballed HGA that works with redundant DSN transponders and 10 and 40w power amplifiers. The 40w amplifier is powered from the 112 V main bus, and is not used during powered cruise. This subsystem is common to all three SEEX configurations.



Commands are received by the omnis, which provide 4 pi steradian coverage. The system supports command rates of 8 bps for the cruise phase using the 34M subnet out to 1 AU, and the 70M subnet for longer ranges. During the science phase, commands are uplinked through the HGA at 1,000 bps. Telemetry is supported by the omnis out to 10^8 km at 2860 bps, and the HGA at greater ranges. Cruise telemetry uses the HGA and 10w amp working into the DSN 34 M net at about 2800 bps. During the science phase, the 32 dBi gain HGA working with the 40w power amps provides a minimum downlink rate of 46 kbps at 3.5 AU working into the DSN 70 m subnet. (see Fig. 9)

Mission Phase	Range to Earth, Max (Au/km)	S/C RF Output Pwr (W)	Down/Uplink Antenna Gain, min, (dBi)	DSN Subnet Usage	Downlink Data rate min, (bps)	Uplink Data rate min, (bps)
Launch & Initiate	1E6 km	10	0/0	34M/cont.	2860	8
Cruise	3.5/5.24E8	10	32/0	34M/ 1 pass/mo	2860	8 (6)
Science Lo	3.5/5.24E8	40	32/32	34M/cont.	11440	1000
Science Hi	3.5/5.24E8	40	32/32	70M/cont.	45760	1000

Notes:

- 1) All links X band; 2) $10E-5$ BER, max; 3) Antenna Pointing Accuracy 0.5 deg, 3 sigma; 4) 3 dB margin for all links, min 5) All links convolutionally encoded (6) 70M @ $r > 1$ au

Fig. 9 Communications Link Performance

Electrical Power

The Electrical Power subsystem provides power to the SEPS and to spacecraft subsystems during cruise, and to subsystems and payload during science operations. The subsystem generates power for each of the SEEX Vehicles using two deployable arrays with up to 43 m² area each, which together generate up to 14.4 kw total power at 1.0 AU. Each array is independently articulated about the S/C Z-axis.

The SEEX 1 array design is based on the substrate and deployment mechanisms used on the Fokker built TELECOM 2 comsat array. It uses 18.5% efficient cells manufactured by ASEC to generate 7035 w at one AU. SEEX-2 uses the APSA array described in Ref. 1. This is a mast deployed fanfold array with a total area (2 wings) of 86 m². With the APSA baseline 13.8% efficient Silicon cells, it generates 10,800 w at 1 AU. For SEEX-3, the APSA substrate and deployment mechanism is backfitted with the 18% ASEC cells to increase power available at 1 AU to 14.5 kw.

Power from the array at 125 Vdc goes directly to the SEPS HVPU and the Telecom 40w RF amp for maximum efficiency. Redundant 28 VDC DC-DC converters power the rest of the spacecraft subsystems and the payload to allow use of existing components designed for this voltage. Electrical energy storage for array deployment and limited duration off-sun and shadow operation is provided a 12 A-H AgZn battery chosen for minimum mass. Battery charge control, low-voltage switching, and undervoltage and overcurrent protection is provided by the power control unit (PCU).

Command, Control, and Data Handling (C²&DH)

The C²&DH provides all command, telemetry and control functions. It consists of a redundant high-rel Mil-Std 1750A based microcomputer, the software executive and application packages that run in it, a 1 Gbit solid-state mass memory, real-time clock, and the I/O circuitry and MIL-STD-1553B data bus used for interfacing with the SEPS, payload and the other subsystems. A limited hardwired command decoder provides processor-independent control over processor status and safety critical functions.

Software

The C²&DH software load provides all the functions implemented by the microcomputer. It is coded in ADA, using algorithms directly derived from the Ball GEOSAT Follow-On (GFO) spacecraft, with functions modified and supplemented to accommodate the unique aspects of SEPS management and the science objectives of the selected SEEX mission.

Guidance

Guidance for all modes is provided from the ground by commanding an appropriate inertial attitude sequence and burn duration, using DSN tracking of the S/C, and target bearings determined from ephemerides and the SEEX 2/3's payload camera (if carried).

Attitude Determination and Control Subsystem (ADCS)

The SEEX ADCS keeps the vehicle pointed to satisfy the requirements of the SEPS and the science payload. During powered cruise, the vehicle's Z-axis is maintained perpendicular to the sunline, with the X-axis oriented to provide the correct thrust vector direction while the arrays are articulated around the Z-axis to track the sun. The vehicle's -Y axis is kept on the Earth side of the spacecraft to allow the HGA to see the Earth. during science operations, the vehicle's +X axis can point to any direction in inertial space, with a combination of roll and array articulation being used to keep the array wings pointed to the sun and the HGA to the Earth.

The ADCS uses 4 sun sensors, a star tracker, and a strapdown gyro Attitude Reference Unit (ARU) to provide attitude determination. Attitude control during powered cruise is provided by vectoring the thrust of two of

the 10 cm ion thrusters. During science operations with the ion thrusters off, attitude control is provided by the eight 2.2 N thrusters of the Propulsion/RCS subsystem. ADCS software application packages running in the C²&DH processor accomplish all attitude determination and control computations, sequencing and thruster control. All components are qualified or space-proven.

Thermal Control

The SEEX thermal control subsystem must keep the SEPS, payload, and all subsystem units within their temperature limits over a range of power dissipation from 150 to 1700W and over a solar range of 0.7 to 2.6 AU. The subsystem accomplishes this by using multi-layer insulation blankets, variable conductance heat pipes and 5 to 10 m² radiators oriented normal to the S/C Z-axis which is kept 90 degrees from the sun at all times. Heat pipe temperature control is augmented by thermostatically controlled heaters used for tighter temperature control of propellant tanks and batteries and for the thruster catalyst beds.

Propulsion

The SEEX propulsion subsystem must provide up to 18000 m/s of primary delta-V needed for reaching its destination along with 50 m/s for vernier maneuvers and RCS functions with the SEPS shut down.

The primary delta-V is provided by the SEPS, which is described in the following section. The vernier maneuver/RCS functions are provided by a simple blowdown monoprop system with a capacity of up to 25 kg of hydrazine and 10 thrusters. Two 2.2 n thrusters oriented along the X-axis provide X-axis translation for vernier maneuvers. Eight 2.2 n monoprop thrusters are arranged to provide three axes of rotation using pure couples, and Y and Z axis translation if required.

Mass and Power Budgets

The SEEX S/C system design reduces risk by providing margins in the key performance areas of mass and power in addition to the growth contingencies built into the design. Additional "margin" is provided by the high propulsive performance capability of the SEPS.

Mass margins are shown in Fig. 10 for the three SEEX spacecraft concepts. Masses for the S/C subsystems have been estimated from existing Ball spacecraft and from NEAR spacecraft weights. The SEPS mass estimates were derived from data accumulated by the SDIO TOPAZ program for thruster masses, and from parametric weight estimating relationships contained in Ref. 2. Startburn masses for each SEEX version were set by loading enough Xenon propellant to provide a 5% margin on the specified Delta throw weight. Many missions would allow propellant to be offloaded, providing larger margins or the possibility of larger payloads.

	SEEX-1	SEEX-2	SEEX-3
	Estimated	Estimated	Estimated
Subsystem/Item	Mass (Kg)	Mass (Kg)	Mass (Kg)
Structure	40.3	53.7	53.7
Mechanisms	8.9	8.9	8.9
Electrical Power & Distribution	31.4	31.4	31.4
Command, Control & Data Handling	21.1	21.1	21.1
Communications	36.7	36.7	36.7
Thermal Control	6.6	6.6	6.6
Attitude Determination & Control	9.8	9.8	9.8
RCS/Maneuver Propulsion (Dry)	12.5	12.5	12.5
SEPS (Dry) (incl solar array)	337.6	414.3	458.2
Subtotal, Bus, dry	504.7	594.8	638.7
Mass Growth Contingency @ 10%	50.5	59.5	63.9
Total, ICM Spacecraft Bus (Dry)	555.2	654.3	702.6
Payload	60.0	60.0	60.0
Total, Spacecraft (dry)	615.2	714.3	762.6
Propellants	384.0	505.0	464.0
Xenon	357.0	478.0	437.0
Hydrazine & Pressurant	27.0	27.0	27.0
Total, SEEX S/C @ Separation	999.2	1219.3	1226.6
Delta-II Throw Weight Capability (1)	1050.0	1290.0	1290.0
Margin on throw weight (5%)	50.8	70.7	63.4
(1) Notes:			
SEEX-1 launched to C3=10			
SEEX-2, 3 launched to C3=0			

Fig. 10 SEEX S/C Estimated Mass Margins

Figure 11 shows the energy budget for SEEX Cruise operations. The design of the SEEX subsystems keeps power consumption during cruise to under 100w to maximize the propulsion power available. The total startburn power level has been purposely "tuned" to the power available from the array at one AU by setting the number and specific impulse of the thrusters in operation at startburn.

	SEEX-1	SEEX-2	SEEX-3
	Average	Average	Average
Subsystem/Item	Power (W)	Power (W)	Power (W)
Mechanisms	2.0	2.0	2.0
Electrical Power & Distribution	7.0	7.0	7.0
Command, Control & Data Handling	16.0	16.0	16.0
Communications	16.0	16.0	16.0
Thermal Control	5.0	5.0	5.0
Attitude Determination & Control	31.3	31.3	31.3
RCS/Maneuver Propulsion (Dry)	0.0	0.0	0.0
Subtotal, SEEX S/C bus	77.3	77.3	77.3
SEPS @ Startburn	6226.0	9650.0	13017.0
Subtotal, Bus power, cruise	6303.3	9727.3	13094.3
Power Growth Contingency @ 10%	630.3	972.7	1309.4
Total, Bus Power (cruise)	6933.6	10700.0	14403.7
Payload	10.0	10.0	10.0
Total, Spacecraft Power @ startburn	6943.6	10710.0	14413.7
BOL Array Power @ 1 Au	7035.0	10800.0	14468.0

Fig. 11 SEEX S/C Cruise Power Budgets

Fig. 12 shows the power budgets for the science mode. In this mode, the bulk of available power goes directly to the Telecom subsystem's 40w RF power amps to maintain high data rate communications with the Earth. The payload power also increases from the 10w standby level allocated during cruise to the full 50w allocated for science operations, raising the overall spacecraft power to about 500w. A benefit from the use of SEP, illustrated in Fig. 12, is the substantial margin on electrical energy provided by the large array even at the 500w level of usage by the spacecraft. This excess energy is available to support even higher rate downlink communications, or for science payloads requiring more than the allocated 50w.

	SEEX-1	SEEX-2	SEEX-3
	Average	Average	Average
Subsystem/Item	Power (W)	Power (W)	Power (W)
Mechanisms	2.0	2.0	2.0
Electrical Power & Distribution	7.0	7.0	7.0
Command, Control & Data Handling	16.0	16.0	16.0
Communications	320.0	320.0	320.0
Thermal Control	5.0	5.0	5.0
Attitude Determination & Control	31.3	31.3	31.3
RCS/Maneuver Propulsion (Dry)	5.0	5.0	5.0
Subtotal, SEEX S/C bus	386.3	386.3	386.3
SEPS Off	0.0	0.0	0.0
Subtotal, Bus power, Science mode	386.3	386.3	386.3
Power Growth Contingency @ 10%	38.6	38.6	38.6
Total, Bus Power (Science Mode)	424.9	424.9	424.9
Payload	50.0	50.0	50.0
Total, Spacecraft Power: Science Mode	474.9	474.9	474.9
Array Power @ 2.5 Au	1126.0	1728.0	2318.0
Margin, W	651.1	1253.1	1843.1
Science Mode Margin, %	137%	264%	388%

Fig. 12 SEEX S/C Science Mode Power Budgets

III. Solar Electric Propulsion System

This section contains discussions of our thruster selection criteria, tradeoff considerations for the propulsion module, and ion propulsion technology maturity. In all cases, it is concluded that the sources, methods, and testbed heritage of all necessary technologies are sufficiently mature as to permit the rapid (18-30 months) integration of a workable version of the SEEX-1 design, and the not so rapid, but equally certain, development of the SEEX-2 and -3 spacecraft.

Thruster Selection

The propulsion concept for the spacecraft discussed above is based upon use of the NASA Lewis Research Center 30 cm ion thruster and the British UK-10 10 cm ion thruster. Hughes has similar 25 cm and 13 cm versions of their XIPS thruster. Other electric thrusters are available including the Russian SPT series of "Hall effect" or stationary plasma thrusters and the German RIT-35 ion thruster. Table II presents characteristics of several candidate thrusters. Fig. 13 presents a drawing and a photograph showing details of the NASA 30 cm ion thruster. Fig. 14 shows the basic components of an ion thruster, and Fig 15 is a block diagram of a suitable propulsion module showing the major functions of an electric propulsion module and the necessary interfaces to the spacecraft.

Designation	SPT-199	SPT-100	NASA Ion-30	RIT-35	UK-10	Hughes XIPS
Aperture	14-20 cm	10 cm	30 cm	14-30 cm	10 cm	13 cm
Engine Type	Xe SPT	Xe SPT	Xe Ion Thruster	Xe Ion Thruster	Xe Ion Thruster	Xe Ion Thruster
Power (kw)	4.15	1.35	4	4	0.625	0.497
Thrust (mn)	300	84	190	160	25	17.8
No. of Engines	1	3	1	1	1	1
Isp (seconds)	1800	1600	2800	3000	3200	2600
Tot. Pwr. (kw)	4.15	4.05	4	4	0.625	0.497
Tot. Thrust (mn)	300	252	190	160	25	17.8
Flow rate (kg/s)	1.7 e-5	1.61 e-5	6.92 e-6	5.44 e-6	7.97 e-7	6.98 e-7

Table II Characteristics of Candidate Thrusters (After Ref. 6)
(Preliminary Estimates for APL Conceptual Design Studies)

The LeRC thrusters were chosen because the high specific impulse is well adapted to the type of deep space missions discussed here and because the thrusters are reasonably well characterized both in the laboratory and in the literature. The UK-10 thrusters were chosen for their high Isp, and their thrust level which permits them to be grouped into clusters of four or six and provide the granularity of thrust vector magnitude

necessary for inner solar system transportation under varying conditions of available input power. Ion thrusters of this type ionize the propellant medium and then accelerate the ions through an electrostatic potential to the desired exhaust speed.

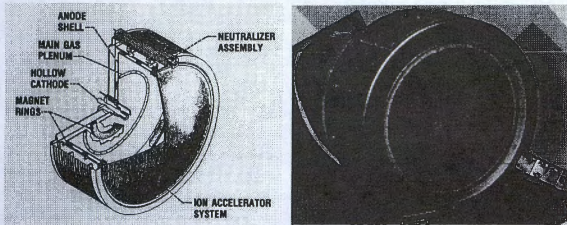


Fig. 13 Diagram and Photograph of NASA 30 cm Ion Thruster
(Courtesy NASA Lewis Research Center)

By the nature of the concept, the specific impulse of the thrusters can be adjusted over wide limits but practical considerations dictate that we must select a nearly constant value for a given mission. For a fixed available power, the thrust decreases as the square of the specific impulse. Thus, a very high specific impulse will only be gained at the cost of very low thrust to weight and nearly always increases the mission time. An optimum specific impulse and thrust-to-weight can usually be computed for a given mission/thruster combination. No specific optimization has been attempted for the present study although we were aware that previous studies have shown the best Isp for this kind of mission to be in the range from 2800 to 3400 seconds. It can be expected, for an actual mission, that the thrust levels, specific impulse, and propellant loadings can be optimized to give somewhat better performance than what is quoted in this paper.

The requirement for interplanetary operation at different distances from the Sun causes a wide variation in the power available to the spacecraft power supply. This means that the thrust must be effectively throttled to be compatible with the available power. Although it is possible to operate ion thrusters at beam currents considerably less than the design point, while still maintaining Isp by holding the grid voltage nearly constant, the power conditioning electronics for such variable thrust/constant Isp propulsion

is substantially complicated over the fixed designs with which we have the most experience. Experts in the field^{7,10} generally agree that throttling the thrusters over a range of 4 or 5 to one is possible but considerably complicates power control and adds a requirement for sophisticated propellant flow control. Throttling of ion thrusters does not appear to have been well explored and, for the sake of conservatism, we have assumed no deep throttling of individual thrusters. Power variations are handled by clustering a number of thrusters of two different sizes and operating them in selected combinations to match the available power. This selection process, of course, results in step functions in the power required. It is probably possible to throttle the individual thrusters enough to smooth out the steps and thus maintain the

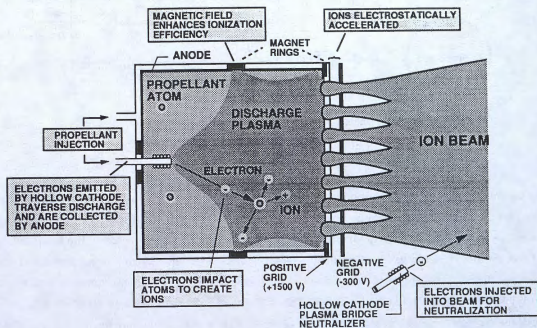


Fig. 14 Ion Thruster Elements and Functions

(Courtesy J.Sovey, Space Propulsion Technology Division, NASA LeRC)

maximum available thrust at any given time. In any case, the excess performance, for the missions cited as candidates for our preliminary designs, obviates the need for such fine tuning of the thrust/power profiles. And we do not wish to fall into the familiar trap of creating a system so complicated, untested, and expensive that we optimize ourselves out of consideration for these important applications.

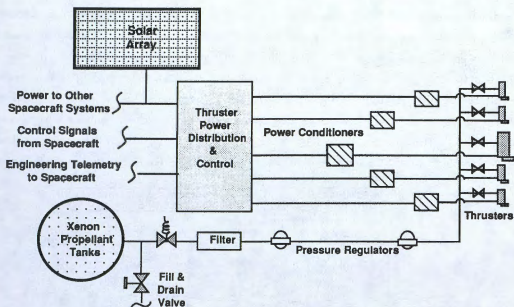


Fig. 15 Block Diagram of Electric Propulsion System

Because the energy required for ionization is lost to the propulsive process and radiated away as heat, it is desirable that the propellant medium be easily ionizable. Greatest propulsive efficiency is obtained with heavy ions. A number of suitable candidate propellant media exist including mercury, the heavy alkali metals, and the heavy noble gases. For many years, cesium, and later, mercury were considered the propellants of choice. More recently, due at least in part to environmental and toxicity concerns, xenon and other heavy noble gases have become preferred. This is a potential problem for large vehicle applications because of the rarity of the material and the expense of obtaining it in sufficiently pure form. Because of the limited amount of propellant proposed for these missions and the fact that the available thrustors have been life-tested with xenon, it is assumed to be the propellant.

Thruster lifetime was used as an initial design parameter for the propulsion system. It was estimated that missions to the asteroids and short-period comets would require thruster-on times of 4500 to 5000 hours (slightly over 1/2 year). This is well within the predicted life expectancy of the current technology thrustors. Operating lifetimes of 10,000 to 12,000 hours are expected from extrapolation of wear and degradation experienced in ground tests. These test durations have been substantially less than the predicted maximum life of the thrustors and the extrapolated lifetime of the thrustors must be taken with some caution. However, because of the use of multiple thrustors, the required operating life for the missions being considered here is so much shorter than the predicted life that the mission design may be approached with some confidence (see discussion below.)

Propulsion Module

Two propulsion module configurations have been considered. The first, used on the SEEX-1 configuration, uses one NASA 30 cm thruster and six UK-10 thrusters. The other configuration, used on both SEEX-2 and SEEX-3, uses two NASA thrusters and six UK-10s. In both cases, the large thrusters are center-mounted with the UK-10s in a ring surrounding the larger units. The UK-10s will be tuned so that each operates at about 1/7 the power of the NASA thrusters. This will facilitate the matching of thruster power requirements to power available from the solar arrays.

As an example, assume that a solar array BOL power output of 12 kw is available at 1 au and that X thrusters can be operated simultaneously at that distance from the Sun. When the distance to the Sun increases to 1.414 au, X/2 thrusters will be operable at that distance, and similarly, X/4 thrusters at 2 au.

The xenon propellant is carried at high pressure in aluminum lined, kevlar overwrapped spherical tanks. The tanks are 66 cm (SEEX-1) and 71 cm (SEEX-2-3) in diameter. The xenon is stored at a pressure of 2500 psi. A potential mass saving could be realized by storing the xenon as a solid, subliming the required flow rate at a modest pressure suitable for the thrusters. This approach has not been incorporated here but stands as a potential performance enhancement.

Power processing for the thrusters is a vital function. Once the number of thrusters (primary plus spares) is finally chosen, reliability analyses need to be made in order to determine how many (if any) power processing unit spares may be required over the maximum number of thrusters planned to be operated at any given time. In any case, thrusters and power processors will be cross-strapped so that in the event of failures, units that are still functional can be matched to maximize system reliability. For the purposes of this paper, it is assumed that one power processor per thruster will be used.

The power processors will be mounted on the -Y equipment panel immediately adjacent to the propulsion module to minimize cabling mass and losses. Even with modern, highly efficient power processors operating at up to 92% efficiency (we have assumed 90% for these studies) substantial heat will have to be dissipated. The amount of heat to be dissipated for the design study amounts to about 800 w for SEEX-1, 1100 w for SEEX-2, and 1500 w for SEEX-3. Since this heat must be rejected at a temperature compatible with the electronics in the processor, say 50°C, the radiator area is large, (5 m² for SEEX-1 and 10 m² for SEEX 2-3). This requirement and the desirability of keeping direct sun off the radiator surfaces is a major configuration and operations driver.

Two of the UK-10 thrusters will be independently gimbal-mounted to provide pitch, yaw and roll control during thrusting. For maximum flexibility and reliability, it would be desirable that each thruster be

independently actuated. This aspect of design should be studied very carefully, as the design matures, to ensure that any credible combination of failures does not unacceptably compromise control capability.

High pressure xenon could be used for attitude control during non-thrusting periods even though such an application yields poor performance due to the high atomic weight of the propellant. Because of the long duration of the missions, attitude control total impulse requirements will be fairly high. For this reason, attitude control during non-thrusting periods will be accomplished using a hydrazine monopropellant system. This is integrated into the same package as the electric propulsion unit, conveniently placing all the hazardous, high pressure systems in a unit separable from the remainder of the system.

Propulsion Technology Maturity

The thrusters assumed for this study are at a fairly high level of technical maturity, prototypical units having been built and subjected to extensive testing. While some additional design, development, and test would be required to bring the units to the level of deliverable hardware, the thrusters are adequately mature that a mission can be designed around them with confidence.

The power processors are less mature. Power processors used in the development of prototypic thrusters have typically been laboratory equipment at the breadboard level of maturity. Basically then, the circuit designs exist but have not been developed into prototypes of operational units. A considerable amount of engineering will be required to develop space qualifiable power processor designs. On the other hand, this is well understood technology and should present a minimum of unexpected pitfalls.

Similarly, control of the propellant flow into the thruster is critical. The flow rate just be closely matched to the power level of the thruster. To date, this has been done largely with laboratory equipment. It will be necessary to develop and qualify operational hardware. As with the power processors, this is fairly straightforward engineering and should cause no substantial problems.

Thruster lifetime is predicted to be on the order of 10,000 hours. It should be noted however, that no thruster has actually run this long. Life expectancy is extrapolated from the degradation observed during shorter life tests. For this reason, thruster life is approached cautiously. As a goal, no thruster will operate more than 5000 hours during the baseline mission. (An interesting possibility is an extended engineering mission wherein one or more thrusters is run to failure, if propellant allows, following completion of the primary mission.) There also appears to be some discrepancy between degradation experienced during ground tests and that experienced in actual space operations. All these uncertainties contribute to the desire to use multiple thrusters and to keep operating lifetime to half or less of the predicted values until more data are obtained.

Conclusions

A credible design for a Discovery class solar electric vehicle, launchable on the Delta 7925, and capable of carrying out a number of important missions to comets and asteroids, has been presented. Also included were upgraded versions of the SEP vehicle capable of capturing missions previously thought to be far too ambitious for consideration in the class of missions under \$500 million U.S. The design, while still preliminary, is sufficiently detailed to leave little doubt in the mind of the experienced spacecraft designer that such a vehicle could be brought to launch readiness in a few years of concerted effort and for relatively low funding requirements. Discussions of our concerns about thruster lifetime, power processor maturity, and propellant management technology were included and, wherever we had such concerns, we have made conservative assumptions to offset potential development and scheduling problems. After this short study, it is our unanimous conclusion that this powerful technology should be reconsidered as a mechanism for timely and affordable exploration of the inner solar system.

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