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A SEGMENTED ION ENGINE DESIGN FOR SOLAR ELECTRIC PROPULSION SYSTEMS*

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Abstract-Solar electric xenon ion propulsion ean be used to deliver a substantial quantity of science instruments (of order 100 kg) in rendezvous missions to small bodies such as cornets and main belt asteroids with an Atlas IIAS launch vehicle. The performance of the ion propulsion system enables it to deliver typically more than twice the total mass (spacecraft plus science) to the destination in less than half the trip time relative to a chemical/ballistic approach using the same launch vehicle. A new ion engine design referred to as a segmented ion engine is shown to reduce the required ion source life time for these small body rendezvous missions from 18,090 h to approximately 8000 h. This breakthrough, together with the use of SAND ion optics for the engine accelerator system substantially reduces the cost of demonstrating the required engine endurance, a demonstration which has never been accomplished for primary ion propulsion. A flight test of a 5-kW xenon ion propulsion system on the ELITE spacecraft would enormously reduce the cost and risk of using ion propulsion on a planetary vehicle by addressing systems level issues associated with flying a spacecraft radically different from "traditional" planetary vehicles.

INTRODUCTION

Ion engines have been under research and development for over 30 years. Very early in this development, the electron bombardment engine demonstrated performance capabilities (thrust, specific impulse and efficiency) which were of interest for space propulsion applications. In 1964 the SERT 1 (Space Electric Rocket Test 1) sub-orbital test flight was launched only four years after the first broad-beam, electron bombardment ion engine was operated in the laboratory [1, 2]. This flight test successfully demonstrated that the positive ion exhaust beam could be neutralized in space, ending the major controversy of the day regarding the usefulness of ion propulsion.

Six years later in 1970, the SERT II experiment to demonstrate long term engine operation in space was launched [3]. This experiment, while not entirely successful in its original objectives (both thrusters failed before their goal of 6 months of operation), was ultimately operated over a period of 21 years and returned a wealth of information which substantially exceeded the original plans [3]. Among the accomplishments of SERT 11 was the successful operation of botb neutralizer cathodes for over 17,000 h in space [3]. Operation of the neutralizer cathodes was terminated due to the exhaustion of the mercury working fluid, so that 17,000 h dots not represent the maximum lifetime of these cathodes.

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With these successes and the development of improved ion accelerator system designs (using closely spaced, thin, dished molybdenum electrodes) to improve ion engine thrust densities and thrust per unit input power capabilities, onc may wonder why ion propulsion has never been used for primary propulsion on a planetary spacecraft. There have been numerous (some may say too numerous) studies demonstrating the substantial benefits of ion propulsion for solar systcm exploration, yet still the technology has not been used. Many explanations have been offered for this including the following: conservatism on tbe part of program managers who may be reluctant to risk their very expensive spacecraft on an "exotic" new propulsion technology; unsubstantiated concerns over mercury contamination of the spacecraft from the engine exhaust (SERT II showed that this was not a problem); tbc availability of many potential missions of interest which don't "need" the capabilities of ion propulsion; the cleverness of trajectory specialists who through the usc of "planetary billiards" can compensate (at the expense of increased trip time) for the limited capabilities of chemical propulsion; and the prohibitive cost and unacceptable programmatic schedule risk to the first user of ion propulsion.

The prohibitive cost and schedule risk arise primarily from the facts that engine lifetime required for a planetary mission has not been demonstrated and that the engine reliability is unknown, that is, that the technology simply was not ready. It is remarkable that no ion engine to be used for primary propulsion has ever been successfully operated for its full design life. Typical lifetimes of the order of 15,000 h are required for planetary missions of interest. The cost and schedule risk of performing engine endurance tests of this duration under a flight project arc unacceptable. In addition, the cost to perform the endurance testing under existing technology programs is too high as well. This has resulted in the present situation in which neither the existing technology programs nor any potential flight project can afford to endurance test the engines, and no flight project will usc ion propulsion unless the engines have been successfully endurance tested for at least their full design life.

This paper addresses solutions to the technical problems impeding the application of solar electric ion propulsion to solar system exploration, namely, required engine life, engine endurance testing, throttling and engine reliability evaluation.

SEGMENTED ION ENGINE

The segmented ion engine divides a large engine into several identical smaller ion sources configured to have the same active grid area as the original larger ion source. For example, a 5-k W, 30-cm diameter ion engine may be replaced by a segmented ion engine consisting of four 15-cm diameter ion sources which are operated from a single power conditioning unit in the manner suggested in Fig. 1. The chief advantage of doing this results from the necessity to throttle the engines for planetary spacecraft powered by photovoltaic solar arrays. Rendezvous missions to small bodies such as comets and main belt asteroids result in large decreases in solar array power as the spacecraft trajectory carries it away from the sun. To perform these missions the propulsion system must be capable of operating over approximately and order of magnitude variation in input power. This translates into a requirement to throttle the engines over a comparable input power variation. Throttling of conventional ion engines may be accomplished in two ways, with fixed propellant flow rate, and with variable propellant flow rates [4, 5]. With fixed flow rate the input power is varied by changing the net accelerating voltage at a constant beam current. This technique has been shown to enable engine throttling over approximately a 3,8 to 1 input power variation [6]. However, deeply throttling the engine in this manner probably severely compromises the accelerator grid life time due to the necessity to maintain a constant total voltage as the net voltage is decreased [6].

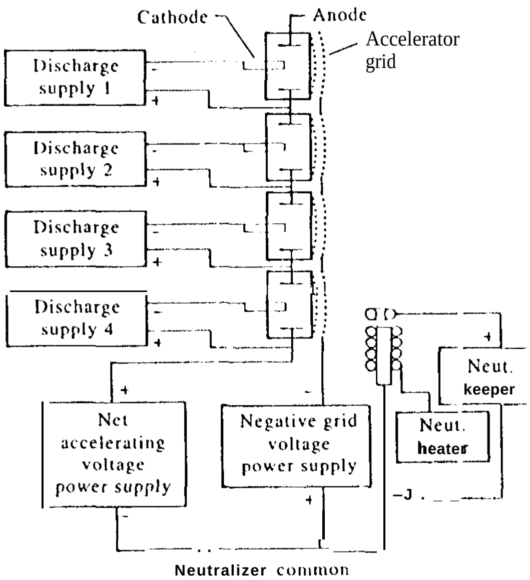

Fig. 1. Power conditioning unit block diagram for the segmented ion engine.

The variable flow rate throttling reduces both the beam current and the net accelerating voltage. Reductions in beam current are accomplished by reducing the main, cathode and neutralizer flow rates, as well as the discharge current. A simplified version of this reduces only the main flow while maintaining constant values for the cathode and neutralizer flow rates [5].

The segmented ion engine is throttled through a combination of techniques. Gross throttling is accomplished by shutting off individual segments. Power is removed from the segment and the segment is electrically isolated from the high voltage power supplies. The propellant flow to the segment is also halted by closing the latch value to that segment (see Fig. 2), For a segmented ion engine consisting of four segments, throttling the engine in this way enables a 4 to 1 input power variation (which is accomplished by shutting the segments off one by one). Finer throttling control is achieved by throttling the individual segments using the fixed flow throttling approach over a 2 to 1 input power range. The result is an overall engine input power throttling capability of 8 to 1.

The flow control system shown in Fig. 2 regulates the pressure fed to the engines by cycling the solenoid valve upstream of the accumulator. The regulated pressure is determined by the valve duty cycle and the accumulator volume. A typical regulated pressure would be of order 10^5 Pa. The remaining pressure drop to the level required by the engines is accomplished though the use of flow restrictors at the engine (not shown in Fig. 2). The potential advantages of this system are that it eliminates the need for a pressure regulator which are expensive, have long lead times, and are prone to failure. This system may also be used to vary the pressure fed to the engines. This would enable engine throttling to be accomplished by the variable flow rate approach. This advantage was not explored in the present work, but warrants examination.

The individual segment fixed flow throttling characteristics arc given in Fig. 3. The beam current reduction from 0.09 to 0.79 A and then to 0.74 A is accomplished by reducing the discharge current at constant flow rate as indicated in Fig. 4. This was

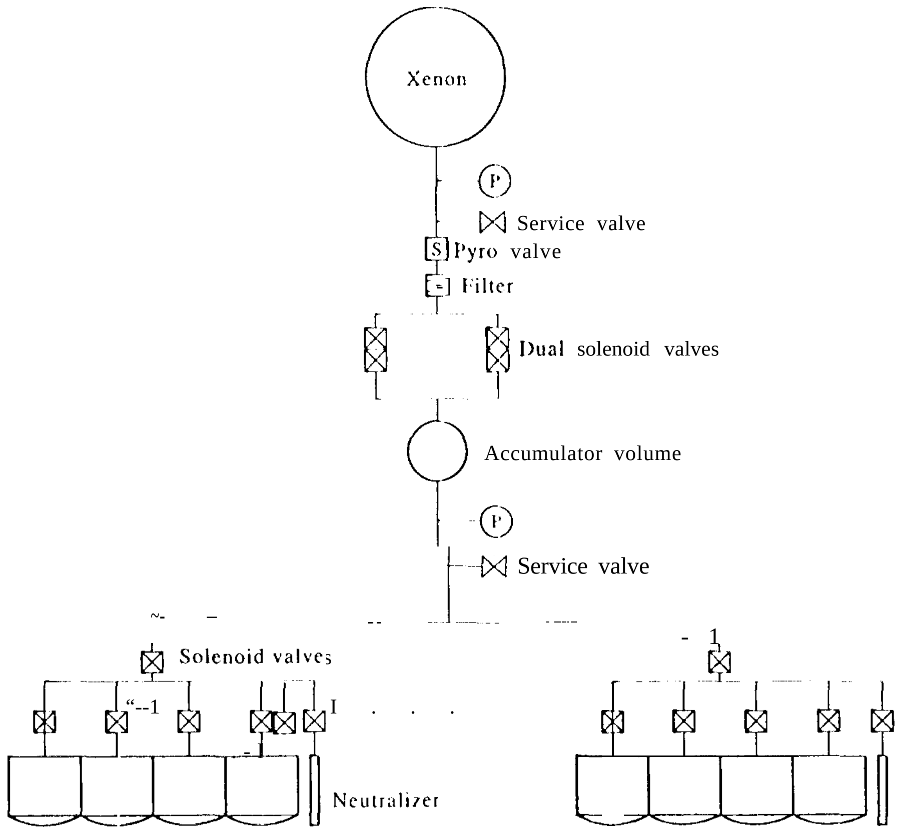

Fig. 2. Flow system for the 10-kW xenon ion propulsion system.

done despite the decrease in the propellant efficiency and corresponding decrease in overall engine efficiency in order to maintain the magnitude of the accelerator voltage less than 300 V (as indicated in Fig. 4). This criteria was selected in order to not compromise the accelerator grid life by operating with excessive accelerator grid voltages. The full 4-segment segmented ion engine throttling characteristics are given in Fig. 5.

Reduced component lifetime requirement

The major advantage of throttling the segmented ion engine in the manner described above comes not from its ability to be throttled over an 8 to 1 input power range, the 30-cm ring cusp engine can easily meet or exceed that range using the simplified variable flow rate throttling [5], but rather from the fact that the segments which have been shut-off in the throttling process are no longer subject to wear. This results in a substantial reduction in the required segment operating life for missions of interest such as comet and main belt asteroid rendezvous missions.

As an example we look at the comet Kopff rendezvous mission assuming the use of an Atlas IIAS launch vehicle, a 10-kW solar array, a spacecraft bus based on a derivative of the ELITE [7] bus, and conventional 5-kW, 30-cm diameter ion engines. The required engine lifetime as a function of the number of engines in the propulsion system is given in Fig. 6. Two engines are required to process the maximum power available from the array. Furthermore, it is assumed that it is necessary to have two engines on during powered flight in order to provide attitude control functions for the spacecraft.

A system of four 30-cm engines requires an engine life of 13,600 h if it is assumed there are no engine failures. To tolerate one engine failure at the beginning of the mission the engine lifetime requirement increases to 18,200 h. It should be noted that much of this operating time is spent in a throttled state which reduces the erosion rates on the engine components, and consequently, the engine technology may be capable of lifetimes of this magnitude [8]. However, demonstrating engine operation for this duration is extremely difficult and has historically been an intractable problem.

The use of a 5-kW segmented engine design consisting of four 15-cm dia. ion sources in place of the conventional 30-cm dia. ion engine significantly reduces the ion source component lifetime requirement for comet and asteroid rendezvous missions of interest. The required component lifetimes for the segmented engine are compared to those for the conventional engine for the comet Kopff rendezvous

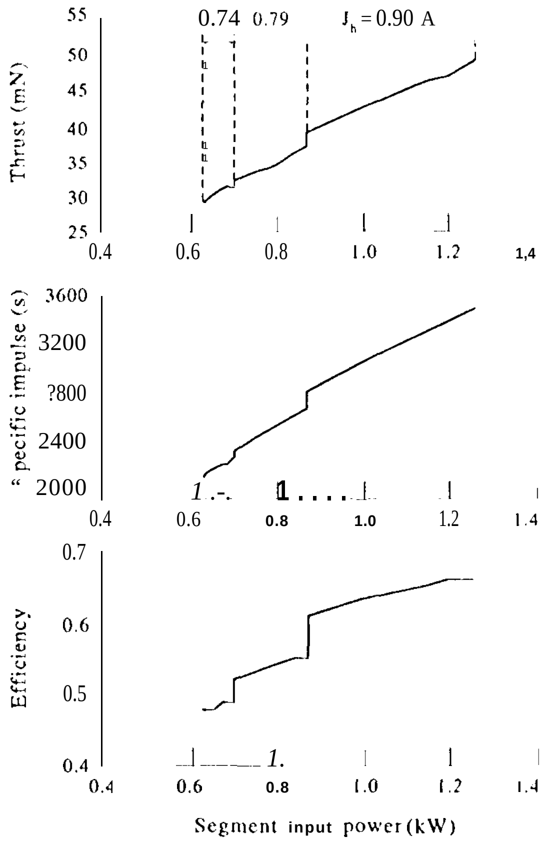

Fig. 3. Throttling characteristics for one segment of the segmented ion engine.

mission in Fig. 7. Throttling the segmented ion engine by sequentially turning off individual segments reduces the required ion source component lifetime for a single-fault-tolerant system from 18,200 to 8200 h. Similar results are obtained for a variety of comets, as WCH as the main belt asteroid Vesta as indicated in Fig. 8.

Each segmented ion engine is equipped with a single neutralizer cathode. This cathode, therefore, must operate for the full lifetime of the engine which is greater than the required life of the individual segments. In fact, for the example given in Fig. 7 the neutralizer cathode for the segmented ion engine must have an operating life of 18,200 h. This, however, is not a major problem for a couple of reasons. First of all, as mentioned earlier, the SUR1'11 mission successfully demonstrated over 17,000 h of operation in space for two neutralizer cat IN AS. Second, cathode testing is substantially easier and less costly 10 perform in ground tests than full engine tests due to the substantial decrease in required vacuum system pumping speed.

Ion engine endurance testing

The 10,000 h reduction in lifetime requirement enabled by the segmented engine configuration enormously facilitates the test program to demonstrate engine life. The segmented ion engine endurance test could be accomplished in a single year, while the conventional engine endurance test would cover nearly two and one-half years. Furthermore, the use

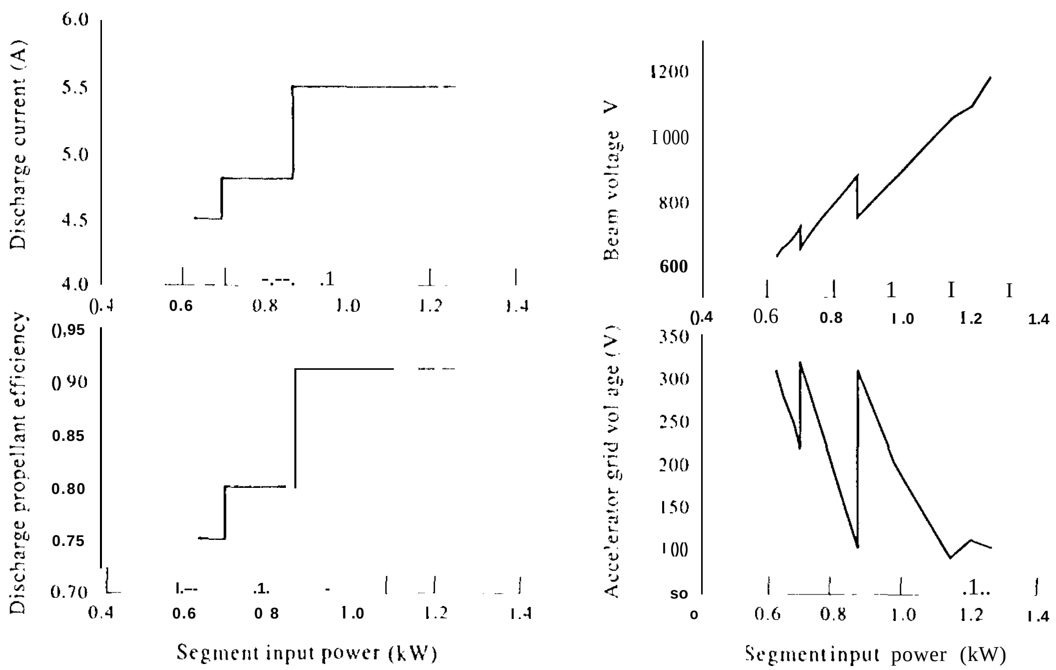

Fig. 4. Current, voltage and efficiency characteristics for throttling one segment of the segmented ion engine,

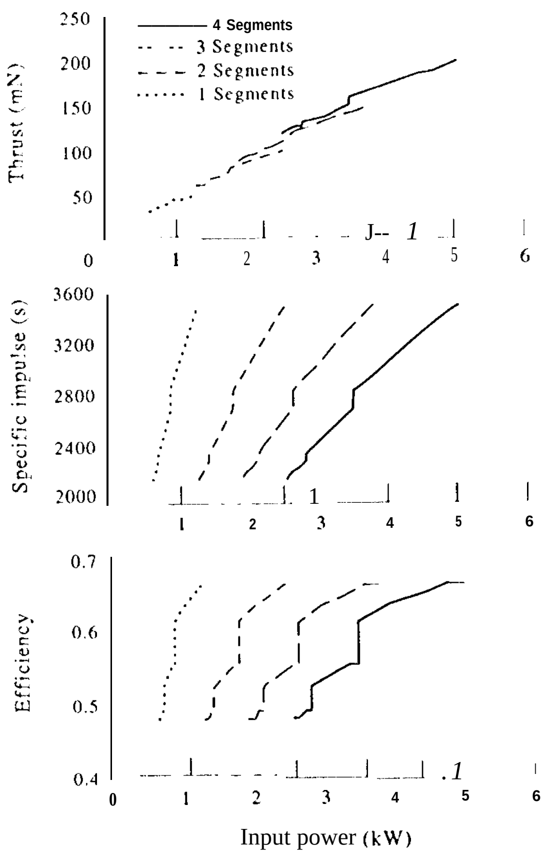

Fig. 5. Throttling characteristics for the full 4-segment segmented ion engine.

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of SAND optics [9] enables ion engine endurance testing at vacuum chamber pressures an order of magnitude higher than previously required, This enables endurance testing to be performed in relatively modest vacuum facilities.

A new facility has been developed at JPL to take advantage of this development. The vacuum tank for this facility is 3-m dia. x 5-m and is pumped by three 1.22-m dia. cyropumps. This facility will be dedicated to ion engine endurance testing.

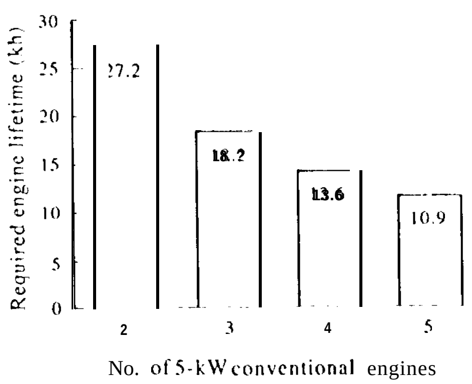


Fig. 6. Required service life for conventional 30-cm ion engines used on the Kopff rendezvous mission assuming a 10-kW solar array and an Atlas IIAS launch vehicle.

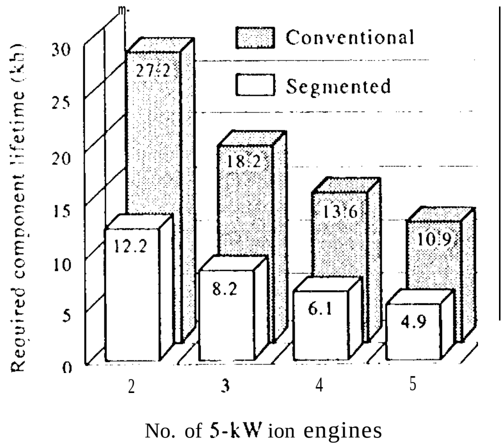
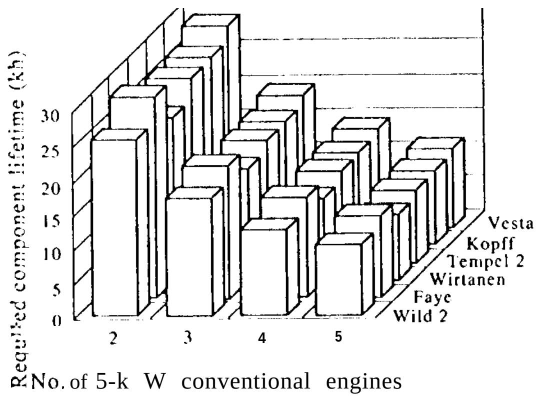


Fig. 7. Comparison of required service life for conventional and segmented ion engine components for the Kopff rendezvous mission indicating the substantial reduction in service life enabled by the segmented engine configuration.

The shortened engine life requirement greatly increases the probability of successfully demonstrating the required lifetime and the capability to perform endurance tests at high tank pressures reduces the

(a)



(b)

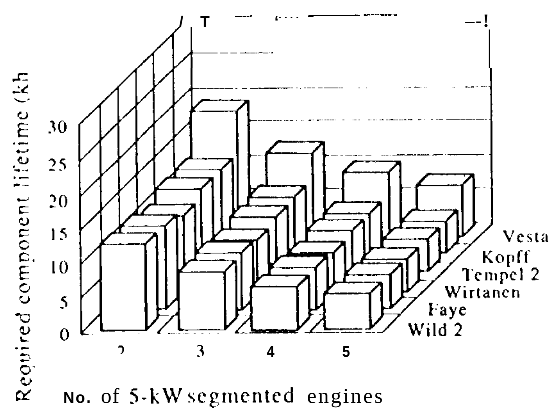


Fig. 8. (a) Required service life for conventional 30-cM ion engine base systems for the main belt asteroid Vesta and several comet rendezvous missions. (b) Required service life for segmented ion engine base systems for the main belt asteroid Vesta and several comet rendezvous missions.

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cost of development testing to the point where it is easily affordable within existing technology programs. With these developments, in approximately two years the technology for ion propulsion should will have an engine with a demonstrated a lifetime sufficient for planetary missions.

It should be noted that the simple demonstration that an ion engine can be operated for its mission lifetime requirement is insufficient to guarantee that the engine reliability for this mission will be high. Such a reliability determination can be performed rigorously through the application of probabilistic failure methodologies (PFM) [1 O]. In the framework of PFM, engine endurance testing is performed primarily to identify failure mechanisms and to determine performance variations versus run time.

MISSION PERFORMANCE

Solar electric propulsion with xenon ion enginesis capable of delivering a full suite of science instruments (of order 100 kg) to many comet and main belt asteroid targets of interest with an Atlas II AS launch vehicle. Mission performance calculations were performed assuming the use of a spat.cc;aft bus approximately the size of the ELITE spacecraft [7,11], a 10-kW APSA [12] solar array, and anion propulsion system based on the segmented ion engine.

The resulting performance for rendezvous missions to the main belt asteroid Vesta and the comets Kopff, Tempel 2, Wirtanen, Faye and Wild 2 arc given in Fig. 9. All of these missions can be performed in less than 3.5 years. Shorter trip times can be achieved by delivering less than the maximum possible mass. The injected energies (C_3) for each mission provided by the launch vehicle are shown in Fig. 9(a), and the characteristic velocity (delta-V) provided by the electric propulsion system in Fig. 9(b). The launch vehicle C_3 and the low thrust trajectories were optimized to maximize the mass delivered to the comet.

The maximum mass delivery capabilities are given in Fig. 9(c) and the propellant consumed in Fig. 9(d). The solid line in Fig. 9(c) indicates the projected total dry mass so the SEP spacecraft (1020 kg) including: the spacecraft bus, 120-kg of science instruments, the 10-kW solar array, and the electric propulsion system (all of which include amass contingency factor of 20% as indicated in Table 1). The electric propulsion system in Table 1 consists of four 5-kW segmented ion engines, four power conditioning units, a digital control and interface unit, and a propellant storage and feed system. these masses are itemized in Table 2.

The difference between the solid line and the delivery capability in Fig. 9(c) represents the mass growth margin which can be accommodated in the spacecraft dry mass while still being capable of accomplishing the mission. These margins range from

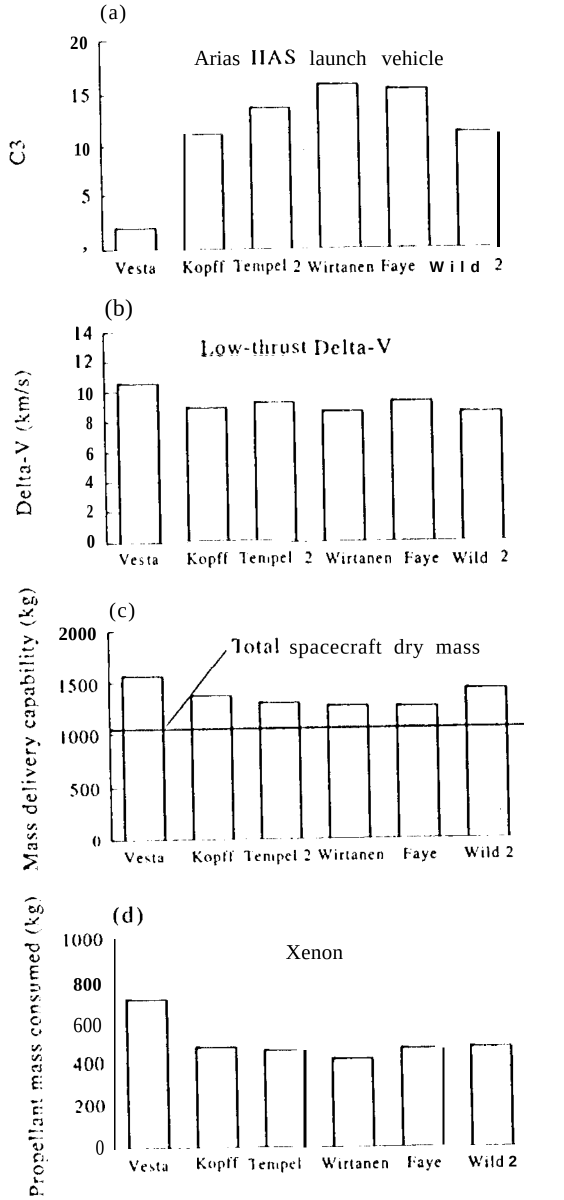

Fig. 9. (a) Optimized C3 provided by launch vehicle for rendezvous missions for maximum delivered mass. (b) Low-thrust delta-V provided by the electric propulsion system. (c) Maximum mass delivery capability for the SEP vehicle compared to the projected vehicle dry mass (including the payload and propulsion system). (d) Xenon propellant required for the indicated rendezvous missions.

Table 1. Mass summary of the SEP planetary spacecraft

Item	Mass (kg)
Spacecraft bus	430
10-kW APSA array (130 W/kg)	77
Science instruments	100
EP system	243
Contingency (20%)	169
Total dry mass	1019

Table 2. Mass summary of ion propulsion system

Item	QTY	Unit mass (kg)	Total mass (kg)
Segmented ion engine	4	8	32
Power conditioning unit	4	28	112
Propellant storage and feed system (for max. of 703 kg of xenon)	1	84	84
Digital control and interface unit	1	3	3
		subtotal	231
Cabling (5% of subtotal)		12	12
		Total	243

21 to *53%* of the projected dry mass (beyond the 20% contingency already included).

The mass delivery capabilities for the SEP system are compared to those of a chemical/ballistic system in Fig. 10 for the Vesta, Kopff and Tempel 2 missions in which both systems are assumed to use an Atlas IIAS launch vehicle. The ballistic options assume a dual Mars fly-by for Vesta, an Earth–Venus–Earth–Earth–Kopff trajectory for Kopff and an Earth–Venus–Earth–Earth–Tempel 2 trajectory for Tempel 2. These trajectories increase the delivered mass capability for the ballistic systems at the expense of significantly increased trip times. Even so the SEP systems deliver significantly more useful mass to the destination than the ballistic systems.

The trip times for these three missions for the SEP and ballistic systems are compared in Fig. 11, where it is seen that in addition to increased mass delivery capability the SEP systems cut the trip times by approximately a factor of two. Furthermore, the opportunities for the complex multiple fly-by ballistic trajectories are rare, significantly reducing the available launch opportunities.

SUGGESTED DEVELOPMENT PLAN

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The advantages of electric propulsion for planetary missions are well known. Identification of these advantages has so far been sufficient to warrant the large expenditure required to bring the ion propulsion system to flight readiness. In other words, the major programmatic impediment to the application of ion propulsion to planetary spacecraft is that the cost to the first user is prohibitive. This is in part due to the fact that an SEP spacecraft with 10 kW or more of installed power, very large flexible solar arrays, and a low-thrust propulsion system is vastly different from "conventional" planetary spacecraft. Key systems issues such as, attitude control, guidance and navigation, and autonomous control and health monitoring must be addressed for such a new and different SEP spacecraft.

A low-cost opportunity exists for resolving these issues on a flight experiment. The joint Air Force/TRW ELITE program is designed to address these issues. The propulsion system for ELITE is based on the 30-kW ammonia arcjet derated to 10 kW [13, 14]. The ELITE mission can accommodate a secondary experiment [11, 15]. For the development of ion propulsion, an ideal secondary experiment would be to fly a 5-kW xenon ion propulsion system. Such an experiment would provide many results which may ordinarily be unaffordable to obtain at this power level.

These benefits include:

1. Demonstration of a high power (5-kW), long duration (> 1000 h) xenon ion engine in space. No high power ion engine (> 1 kW) has ever been flown.
2. Subsystem technologies (solar array, spacecraft bus, ion propulsion system) are tested together in a relevant environment.
3. Experience gained in the integration of the ion propulsion system with the spacecraft.
4. Autonomous guidance, navigation and control demonstrated with the ion propulsion system.

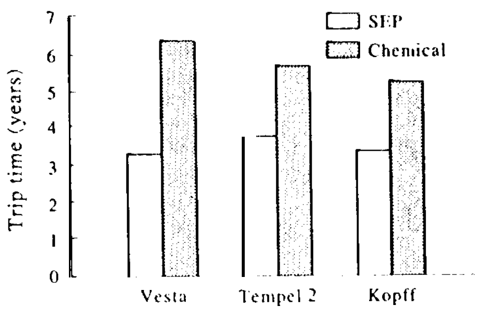

Fig. 10. Trip time comparison for SEP and chemical systems for rendezvous mission with an Atlas IIAS launch vehicle. Shorter trip times could easily be achievable by reducing, the mass delivered.

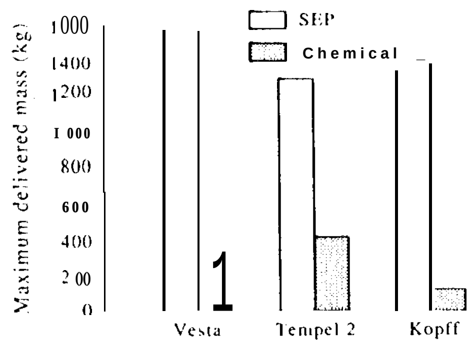

Fig. 11. Comparison of maximum mass delivery capability for the SEP and Chemical systems consistent with the trip times given in Fig. 10.

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5. The cost and risk to the first planetary user of ion propulsion arc significantly reduced.
6. Results in a technology readiness level in which only "delta-qual" programs arc required for planetary mission application.

A program plan to develop ion propulsion to "flight readiness" may be briefly outlined as follows:

1. Develop the 5-kW segmented ion engine, endurance one segment for 8000 b followed by an *8700* h (1 year) endurance test of the full 4-segment engine at 5-kW.
2. Perform an 18,000 h bell-jar test of a neutralizer cathode.
3. Apply the techniques of probabilistic failure analyses [14] to access the reliability of the segmented ion engine including the neutralizer cathode.
4. Begin the immediate development ofa 5-kW power conditioning unit for the segmented ion engine.
5. Fly a 5-kW ion propulsion system as a secondary experiment on ELITE. The propulsion system should consist of one 5-kW segmented ion engine, a power conditioning unit, a digital interface and control unit, a propellant storage and feed system, and possibly a two-axis gimbal mechanism for the thruster.

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Ion propulsion offers enormous mission benefits for rendezvous missions to small bodies such as comets and asteroids including a reduction in trip time by a factor of two with a simultaneous factor of two increase in the mass delivery capability relative to chemical/ballistic systems using the same launch vehicle. These benefits, while certainly not a ncw discovery, can be obtained with the usc of a moderately sized 10-kW SEP system including the APSA solar array technology, xenon ion propulsion, and an Atlas IIAS launch vehicle.

The usc of the segmented ion engine design reduces the required ion source operating life for these small body missions from 18,000 to 8000 h. This reduction in required life enormously facilitates flight qualification of the engines. In addition, the usc of SAND optics enables the engine endurance testing to he performed in modest vacuum facilities substantially reducing the cost of the endurance test program still further.

Finally, a fright test of 5-k W xenon ion propulsion system as a secondary experiment on the ELITE spacecraft will substantially reduce the cost and risk to the first planetary user of ion propulsion by addressing key system issues associated with flying a high power, low-thrust, spacecraft with large flexible solar arrays.

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