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ELECTRIC PROPULSION OPTIONS FOR 10 KW CLASS EARTH-SPACE MISSIONS

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Abstract

Five and 10 kW ion and arcjet propulsion system options for a near-term space demonstration experiment have been evaluated. Analyses were conducted to determine first-order propulsion system performance and system component mass estimates. Overall mission performance of the electric propulsion systems was quantified in terms of the maximum thrusting time, total impulse, and velocity increment capability available when integrated onto a generic spacecraft under fixed mission model assumptions. Maximum available thrusting times for the ion-propelled spacecraft options, launched on a DELTA II 6920 vehicle, range from approximately 8,600 hours for a 4-engine 10 kW system to more than 29,600 hours for a single-engine 5 kW system. Maximum total impulse values and maximum delta-v's range from 1.2×10^7 to 2.1×10^7 N-s, and 3550 to 6200 m/s, respectively. Maximum available thrusting times for the arcjet propelled spacecraft launched on the DELTA II 6920 vehicle range from approximately 528 hours for the 6-engine 10 kW hydrazine system to 2328 hours for the single-engine 5 kW system. Maximum total impulse values and maximum delta-v's range from 2.2×10^6 to 3.6×10^6 N-s, and approximately 662 to 1072 m/s, respectively.

INTRODUCTION

A high level of interest has been developing for a near-term demonstration of primary solar electric propulsion[1-2]. The proposed "strawman" electric propulsion (EP) space experiment would act to assist transition of primary electric propulsion to operational status through development and flight qualification of realistic electric propulsion system(s). A successful demonstration of the electric propulsion system(s) would verify thruster system performance and lifetime, and establish the capabilities of this technology to accomplish high-energy missions. Other advantages of such a space experiment may include the demonstration of autonomous flight control of a low thrust spacecraft, investigations of spacecraft/plasma/EP system interactions, and concurrent demonstration of advanced solar array technologies.

Ion and arcjet propulsion are options for primary electric propulsion missions because of their specific impulse, efficiency, and development maturity. This paper examines the ion and arcjet thruster and propulsion system options available for a near-term space demonstration mission. First-order calculations of propulsion system performance and component masses are provided. Overall mission performance of the proposed EP systems is quantified in terms of total impulse, delta-v, and thrusting time capability based on a fixed mission model.

MODEL DESCRIPTION

Assessing a propulsion system requires selecting the thruster technology (performance and operating condition) and deriving the system parameters (component masses and architecture) based on that technology. This section describes the thruster and system mass models used for the ion and arcjet propulsion systems, and the mission model used in characterizing their performance.

ION PROPULSION SYSTEMS

Thruster - The ion thruster, shown in Figure 1, generates thrust by electrostatic acceleration of ions extracted from a plasma discharge. Predicting the performance (thrust, specific impulse, power level, etc.) of an ion thruster with various propellants is fairly straightforward because the device operation is determined by well known physical operating principles. In general, the performance of a given ion thruster can be specified to within 5 percent accuracy for most parameters prior to it's operation. The assumptions used in determining the thruster performance for this study are listed in Appendix A. Only 30 cm diameter (28.7 cm effective beam diameter) ion thrusters were considered in these analyses due to the relative maturity of this technology. A ring-cusp discharge chamber was assumed as it has demonstrated improved efficiencies compared to the SEPS divergent-field ion thruster[3-4]. Further, high power operation of the divergent-field thruster with xenon has shown unacceptable levels of discharge component erosion over a wide range of input power levels of interest[5]. This indicates that the divergent-field thruster has insufficient life for high total impulse missions using inert gas propellants. This high erosion has not been observed in high power ring-cusp ion thruster testing to date. Xenon was selected as the propellant because of its high atomic mass which gives a high thrust-to-power ratio, and because it is the propellant for which the majority of high power testing has been conducted. Figures 2 and 3 show thruster efficiencies, and thrust levels, respectively, versus specific impulse achievable with present 30 cm ion thruster technology on xenon propellant. The assumptions used in obtaining these values for both figures are listed in Appendix A. These predicted performance levels for 30 cm ion thruster technology correspond favorably to results obtained at NASA-LeRC and elsewhere[3-9]. As indicated in Figure 2, 30 cm xenon ion thrusters can run from approximately 2450 seconds to 4520 seconds specific impulse, which corresponds to 62 to 79% overall thruster efficiency. The efficiency of electrostatic thrusters increases with specific impulse, as an increasing fraction of the input power goes directly into the beam. For completeness, data were included over a range of discharge chamber propellant efficiencies (from 90 to 95%), which accounts for the vertical band in the efficiency curve. Thrust levels with xenon propellant range from approximately 0.04 to 0.42 Newtons, which corresponds to thruster input power levels from approximately 1 to 12 kW (Figure 3).

As indicated in Figure 3, the plot of thrust versus specific impulse defines an operating envelope for 30 cm ion thrusters. The boundaries for the envelope include:

1. a lower-limit range of specific impulse, defined by a net-to-total accelerating voltage ratio (R ratio) of 0.55. Reducing the R ratio below this value with two grid optics results in defocussing and direct-impingement of the ion beamlets onto the accelerator electrode. The lower limit to the specific impulse could be reduced further by the use of three grid optics which permit operation at lower values of R, but this would result in a reduction in the absolute value of the thrust.

2. an upper-limit range of specific impulse, defined by an R ratio of 0.85. Increasing the R ratio beyond this value results in electrons in the neutralized ion beam backstreaming to the screen (positive) electrode.

3. a maximum specific impulse, thrust, and thruster input power, defined by the electric field strength limit (or equivalently, ion extraction limit) of the ion optics.

4. an upper-limit to the thruster input power, and thrust, defined by the heat rejection capability of the thruster discharge chamber. As noted in Appendix A, ring-cusp ion thrusters incorporate rare-Earth cobalt magnets in the discharge, which are susceptible to irreversible losses when their surface temperature exceeds 300 °C. This corresponds to approximately 800 Watts discharge power for present 30 cm ring-cusp thruster designs. Figure 3 shows for xenon propellant, the discharge thermal limit and the ion optics extraction limit are reached at approximately the same condition. The thruster thermal limit may be increased by an improved thermal design (e.g.- larger diameter discharge chamber) or by an improved plasma containment geometry which reduces discharge losses.

Thirty centimeter diameter ion thrusters have operated for short duration tests at performance levels substantially higher than those shown in Figure 3[4]. However, the authors believe that the performance parameters indicated in this figure encompass the range of moderate risk operating conditions presently available, which lead to adequate thruster lifetimes. The xenon operating envelope spans the range of performance obtained from the 1.4 kW, 25 cm diameter thruster of Beattie, et al.[9], to the 10 kW, 30 cm ion thrusters presently under testing at NASA-LeRC.

It is noted that all the operating conditions within the performance envelopes of Figure 3 are not of equal risk, in terms of thruster endurance. Qualitatively, the minimum risk operating conditions are at the lowest power density and near the center of the specific impulse band. Reduced thruster life occurs by increasing input power (thrust), or by changing the specific impulse to a value near the prescribed limits. These occur when the total accelerating voltage across the optics is increased, when the beam current is increased (either by increasing discharge power or total accelerating voltage), or when the R ratio is decreased/increased to beyond 0.55/0.85. Ion thrusters are typically operated near 0.80 R ratio, below the point of electron backstreaming but high enough to maximize the engine thrust. Substantial reductions in the R ratio lead to defocussing and direct impingement (as previously discussed) and also lead to increased charge-exchange erosion of the accelerator grid electrode due to the higher operating voltages.

Although ion thruster lifetime at high power levels has not been verified, it is believed that projections based on ion erosion of the screen electrode to 50% of its initial thickness provides a conservative measure of thruster life. Based on the 10 kW wear test of a xenon divergent-field thruster[5], and short-term erosion rate measurements performed on a 1.4 kW xenon ring-cusp thruster[10], a 35,000 A-h screen grid (beam current times operating time) lifetime limit at 28 V discharge voltage is proposed as the best estimate for xenon thruster life presently available. Based on this criterion, the xenon ion thruster lifetimes associated with the operating conditions of Figure 3 would range from approximately 5,800 to 35,000 hours, corresponding to the range of input power levels from approximately 12 kW down to 1 kW.

System Mass Model - The propulsion system mass model used in this study was derived from the architecture/methodology proposed in reference 11. It's implementation is consistent with that of reference 12.

This mass model defines the propulsion system into a thrust module and an interface module (see Table I). The thrust module includes the thrusters (self radiating), gimbals, power processor units (ppu) and ppu thermal control, thruster structure, and propellant distribution. The interface module consists of converters, controllers, power distribution cabling, structural mass (comprising the 'dry' interface module mass), and the propellant and propellant storage and assembly (comprising the propellant module mass). The sum of the thrust module mass and the 'dry' interface module mass are defined here as the propulsion system dry mass. Reference 12 lists the equations used to calculate the component masses of the thrust and interface modules.

No redundancy was included in the propulsion system, except for the power distribution cabling, converters, and controllers. Also, the mass of the guidance/navigation system was assumed to be part of the spacecraft (payload) and hence was not included in the evaluation of the propulsion systems. The question of what is the correct number of active thrusters to incorporate into a system design is a function of the propulsion system power, mass, and volume allotment, as well as the tradeoffs in risk (thruster endurance) versus propulsion system cost (including qualification testing) and complexity. Based on mission model assumptions, systems ranging from 4 active thrusters processing a total of 5 kW to 1 thruster processing 10 kW were examined over the range of operating conditions identified in Figure 3.

ARCJET PROPULSION SYSTEMS

Thruster - Arcjet thrusters are electrothermal devices in which an arc is used to heat the propellant. A cutaway schematic of a typical laboratory model arcjet thruster is shown in Figure 4. A high-voltage pulse is used to start the arcjet. In operation the arc cathode attachment point is a small, molten spot at the cathode tip. The arc attaches in a diffuse manner in the divergent section of the nozzle which also acts as the anode. Propellant is injected into the chamber tangentially to provide swirl stabilization of the arc.

The current NASA-sponsored research program on low-powered arcjet thrusters was started in 1983. Under this program, operation in the 1 to 2 kW range has been demonstrated using hydrazine, or mixtures simulating it's decomposition products, as the propellant. Hydrazine was chosen because the near-term goal of this work is the application of the arcjet to North-South Station Keeping (NSSK) on geosynchronous communication satellites which currently use this propellant. Ammonia has also been tested in the range of interest.

A 1000 hour/500 cycle lifetest was successfully completed on a lab model thruster to demonstrate long-term reliability[13]. Typical performance measurements taken using this thruster are shown in Figure 5. These data show that there was no significant change in the thruster operating characteristics over the course of the test. While this test was run on mixtures simulating hydrazine decompostion products, other tests have shown the operating characteristics of the arcjet are virtually identical when operated on hydrazine[14]. In addition to the above, both the impacts of the plume[15-16] and electromagnetic interference (EMI) are under investigation.

An engineering model flight-type arcjet system is currently nearing completion and lifetests of this system are planned for the near future. A preliminary investigation of arcjet operating characteristics at increased power levels has also been started.

Additionally, the Jet Propulsion Laboratory (JPL) has been establishing a large performance and lifetime database for 30 kW-class ammonia arcjets[17-18]. This program supports the development of arcjet thruster technology for the SP-100 Flight Experiment for which arcjet EP has been baselined as the active load[1].

System Mass Model - The arcjet propulsion system mass model used in these analyses was based on the methodology developed in reference 11, and was consistent with that shown in Table I and published by Hardy et al.[19], except for the modifications identified in Appendix B. Three different thrusters and thruster power levels (approximately 1.5, 4.6, and 9.4 kW) were selected based on the database established with the 1 kW NASA-LeRC arcjet[13]. This criterion resulted in system designs which ranged from 1 to a maximum of 6 active thrusters available to process 5 to 10 kWe from the power source. The issue of redundancy was treated as described previously in the ion system mass model description.

MISSION MODEL

A mission model was used to calculate the capabilities of each of the low power ion and arcjet systems, once the dry masses of these propulsion systems were determined. The mission model is described in the following assumptions:

1. A low power ion or arcjet system would fly on a dedicated electric propulsion space experiment. The total payload mass (not including the low power ion or arcjet propulsion unit or propellant) would be 2950 kg. This would include the spacecraft bus, an energy storage module, solar arrays, an EP diagnostic package, and an autonomous EP control unit.

2. Either 5 or 10 kWe would be available from a solar array for the low power ion or arcjet propulsion module.

3. The EP spacecraft would be launched from the Eastern Space and Missile Center (ESMC) either on an ATLAS I, ATLAS II, DELTA II 6920, or DELTA II 7920 expendable launch vehicle. The ATLAS vehicles would do a direct planar ascent insertion of the EP spacecraft into a circular orbit of 550 km altitude. The DELTA II vehicles would insert the EP spacecraft into a circular orbit at 550 km altitude using a similar two-stage mission profile, but at a 28.7 degree inclination. The initial circular orbit altitude of 550 km was selected to ensure at least an order of magnitude in the thrust-to-drag ratio for all the proposed spacecraft. This altitude was determined assuming a maximum acceptable drag force of 5.0×10^{-5} N/m^2, using nominal atmospheric density projections for peak conditions during solar cycle 22 from reference 20. The payload capability of each vehicle to this orbit is 3600 kg, 4500 kg, 4600 kg, and 5100 kg, for the DELTA II 6920, ATLAS I, DELTA II 7920, and ATLAS II, respectively. These launch vehicle performance numbers were obtained from references 21 and 22.

4. The mission would require a total thrusting time demonstration sufficient to verify the performance and lifetime of the thruster technology. The thrusting time would be specified based on either the launch vehicle payload margin (termed "Mission Mode I") or by an estimate of future EP system mission requirements (termed "Mission Mode II"). In these analyses, both approaches were addressed. The performance of the low power ion and arcjet systems in Mission Mode I were characterized by the maximum total impulse, velocity increment, and thrusting time capability of each system, using the launch vehicle payload margin for the addition of propellant. Based on the evaluation of arcjet and ion system mass and thrust, and the lift mass of the selected launch vehicles, a thrusting time of 5000 hours for the ion systems, and 1000 hours for the arcjet systems was used for Mission Mode II. The two EP system types provide comparable values of total impulse for these thrusting times.

As low thrust trajectory analyses (including occultation, eclipse and drag penalties, array degradation, etc.) was beyond the scope of this investigation, the velocity increments were determined solely from the 'rocket equation' without consideration for thruster duty cycle/restart requirements, or throttling strategies.

RESULTS AND DISCUSSION

This section describes the ion thruster performance parameters, ion propulsion system options, and arcjet propulsion system options available for an for an electric propulsion space experiment. The mission performance of each system, in terms of total impulse, delta-v and thrusting time capability is also quantified and presented.

ION THRUSTER PERFORMANCE AND PROPULSION SYSTEM OPTIONS

As suggested in Figures 2 and 3, the operating conditions available to a an ion thruster with a single propellant are quite varied. Specifying the power delivered to the thruster or thruster ppu is not sufficient to define the thruster operating point, as several operating conditions (specific impulse, efficiency, thrust, etc.) are available at any one power level. The mission model assumes both 5 and 10 kW are available from a solar array to the ion propulsion module. Since the ppu efficiency and line losses were assumed to be 92% and 0.5%, respectively, the total power delivered to the ion thrusters was approximately 4.6 and 9.2 kW. As discussed earlier, the number of thrusters per power level would range from 1 to a maximum of 4. Consequently, a selection of thruster operating condition must be made so that an integral number of thrusters will process the power available, and yet be consistent with the operating envelope and criteria previously discussed.

One way to select the ion thruster operating conditions would be to select the specific impulse (I_{sp}) from within the available range determined by the engine parameters. Selecting the Isp specifies the beam voltage. To ensure that an integral number of thrusters would process all the available power requires that the thrusters be throttled by reducing the beam current. However, this could reduce the total propulsion system thrust significantly.

By judiciously selecting the total accelerating voltage and beam voltage, and thereby specifying the I_{sp} and power level, an integral number of thrusters can be obtained without throttling the beam current This method ensures that the minimum total voltage required is selected, the R ratio is kept near 0.80 (minimizing the accelerator voltage), and the thrust is maximized. Table II lists the 30 cm ion thruster performance parameters for 6 operating conditions with xenon propellant which arose from the 5 and 10 kW single and multiple engine system designs. The thruster input power levels range from approximately 1 to 9 kW, for an Isp range of approximately 3200 to 4400 seconds.

Table III outlines the top-level design parameters of the 5 and 10 kW xenon ion propulsion system options, including total propulsion system thrust, efficiency, and dry mass. Total thrust for the xenon propulsion systems range from 0.180 to 0.208 Newtons for the 5 kW options, to 0.329 to 0.392 Newtons for the 10 kW options. These thrust levels correspond to overall propulsion system efficiencies of 65 to 72%, respectively.

Table IV provides more system detail as it breaks down the 'dry' component masses of these ion propulsion systems. As indicated, the propulsion system dry masses are strongly dependent on total input power and number of thrusters. The propulsion system dry masses range from a maximum of 234 kg for a 4-engine 10 kW system, to a minimum of 79 kg for a 1-engine 5 kW system. Figure 6 presents the propulsion system dry mass as a function of specific impulse for these various systems. The number of thrusters per system is also identified in this figure.

Figures 7(a) and 7(b) show the dry mass distribution of the 5 and 10 kW xenon ion propulsion systems, respectively. The 10 kW ion thruster systems are approximately 12 to 28 percent heavier, for the same number of the thrusters, than the 5 kW systems. The mass distribution as a function of number of thrusters is generally the same for both power levels. The majority of the mass is in the power processing, constituting over 40% of the total dry mass for all the system options. The thruster/gimbal mass is the second most massive component, except for the single-engine 10 kW case where the thermal control mass exceeds the thruster and gimbal masses.

ARCJET THRUSTER PERFORMANCE AND PROPULSION SYSTEM OPTIONS

Performance estimates for the 1.5, 4.6, and 9.4 kW ammonia and hydrazine arcjets are listed in Table V. Listed values include thrust, I_{sp}, efficiency, and propellant mass flowrate as a function of thruster input power. The performance estimates used are based on conservative projections and are consistent with results obtained at NASA-LeRC[13] and at JPL[17].

Table VI outlines the top-level system parameters for the 5 and 10 kW ammonia and hydrazine arcjet propulsion system options. The parameters include total system thrust, efficiency, and total dry mass. Total arcjet system thrust with both propellants is typically a factor 2 to 3 higher than that projected for the xenon ion propulsion systems at the same input power. Hydrazine arcjet system thrust levels range from 0.423 to 0.593 Newtons for the 5 kW options, and 0.810 to 1.187 Newtons for the 10 kW options. These thrust levels correspond to system efficiencies of 30-32%.

As indicated in Table VI, the propulsion system dry mass is independent of propellant and depends only on total power and the number of thrusters. The propulsion system dry masses range from a minimum of 38 kg for a single-engine 5 kW system, to a maximum of 87 kg for a 6-engine 10 kW system. These dry mass values are a factor of 2 to 3 lower than those projected for the ion propulsion systems for the same power and numbers of thrusters (see Table III). Figure 8 shows the arcjet propulsion system dry masses as a function of specific impulse.

Table VII breaks the mass of the arcjet propulsion system options down into the subsystem level. For all of the arcjet systems, the power processor is the most massive system component, constituting anywhere from 28 to 34% of the total dry mass. The second most massive component is the dry interface module, except for the 6- and 2-engine 10 kW systems where the thermal control is the second heaviest component. Unlike the ion propulsion systems, the thruster/gimbal masses in the arcjet systems are fairly small, constituting only 6-8% of the total mass. Figure 8 shows the arcjet propulsion system dry masses as a function of specific impulse. Figure 9 shows the dry mass distribution of each of the arcjet propulsion systems.

MISSION PERFORMANCE

Values for total initial dry spacecraft mass for each combination of electric propulsion (EP) system option/payload were determined by individually adding each EP system dry mass to the 2950 kg payload. Subtracting these dry mass values from the mass lift capability of the launch vehicles defines a positive payload margin. The payload margin, in part or in whole, is then consumed by the addition of propellant and tankage (the propellant module mass) for the EP system.

Mission Mode I - In this mode the mission parameters for the EP options are determined by the launch vehicle payload margin. This is accomplished by filling the payload margin with propellant for the EP system to increase the thrusting time and delta-v capability of the ion and arcjet systems. Tables VIII (a) and (b) list the spacecraft system and mission parameters calculated for this extended-duration mission using the various ion and arcjet propulsion system options. The ammonia arcjet numbers are not shown, as the system and mission parameters with this propellant are comparable to those obtained with hydrazine propellant which are listed.

The xenon 4-engine 10 kW ion propelled spacecraft has the lowest positive payload margin of any system option when launched on the DELTA II 6920 vehicle. Yet, as indicated in Table VIII, there is still sufficient payload margin to on-load enough propellant to run the propulsion system for more than 8,600 hours, giving a delta-v of 3600 m/s. Enough propellant could be on-loaded on most of the ion propelled spacecraft to permit thruster operation greatly in excess of the projected thruster lifetimes. That is, the capacity to fully demonstrate and qualify the ion propulsion systems is not limited by the lift capacity of the proposed launch vehicles, but is restricted by the engine life. Maximum available thrusting times for the ion-propelled spacecraft on the DELTA II 6920 vehicle range from approximately 8,600 hours for the 4-engine 10 kW system to more than 29,600 hours for the single-engine 5 kW system. Maximum total impulse values and maximum delta-v's on the DELTA II 6920 range from 1.2×10^7 to 2.1×10^7 N-s, and 3550 to 6200 m/s, respectively. Ion propulsion system and spacecraft mission parameters obtained with launches on the ATLAS I, DELTA II 7920, and ATLAS II vehicles are also listed in Table VIII.

Since the specific impulse of the arcjet propulsion options is substantially lower than that of the ion systems, the launch vehicle payload margins are also much lower. Maximum available thrusting times for the arcjet propelled spacecraft on the DELTA II 6920 vehicle range from approximately 528 hours for the 6-engine 10 kW hydrazine system to 2328 hours for the single-engine 5 kW system. Maximum total impulse values and maximum delta-v's on the DELTA II 6920 range from 2.2×10^6 to 3.6×10^6 N-s, and approximately 662 to 1072 m/s, respectively. These values of total impulse and delta-v are approximately a factor of 5 to 6 lower than that obtained with the ion systems for the same payload and launch vehicle. It is noted that only three arcjet options provide total thrusting times in excess of 5000 hours. These are the single-engine 5 kW system on the ATLAS I (5736 hours), DELTA II 7920 (6096 hours), and the ATLAS II (7992 hours) launch vehicles.

The highest performance in terms of maximum delta-v for both the ion and arcjet systems are obtained with the 10 kW single-engine options. Figures 10 and 11 show the maximum thrusting time versus launch vehicle option, and maximum delta-v versus launch vehicle option for these systems. As indicated in Figure 10, the maximum thrusting time for the single-engine 10 kW ion-propelled spacecraft ranges from approximate 17,400 hours on the DELTA II 6920 to 64,200 hours on the ATLAS II. Corresponding values for the single-engine 10 kW arcjet spacecraft range from approximately 1248 hours to 4368 hours. Maximum delta-v's for this ion-propelled spacecraft range from 6.2 km/s on the DELTA II to 18.6 km/s on the ATLAS II. Corresponding values for the single-engine 10 kW arcjet spacecraft range from 1.1 km/s to 3.0 km/s.

Mission Mode II - In defining a total thrusting time for the EP systems (5000 hours for ion/1000 hours for arcjet), a delta-v, total impulse, and initial spacecraft mass can be determined. Tables IX (a) and (b) list the propulsion system and mission parameters calculated for this mission mode using the various ion and arcjet propulsion system options.

Under the constraint of a fixed thrusting time, the maximum spacecraft delta-v and total impulse are obtained with the 4-engine 10 kW xenon ion propulsion system, and (for arcjet) the 6-engine 10 kW hydrazine arcjet propulsion system. The xenon ion system provides a 2121 m/s delta-v capability, at a total impulse value of 7.1×10^6 N-s and total mass of 466 kg for a 5000 hour total burn-time. The hydrazine arcjet system provides a 1161 m/s delta-v capability, at a total impulse value of 4.3×10^6 N-s and total mass of 1161 kg for a 1000 hour total burn-time.

The propulsion system dry masses for the ion options are 2 to 3 times more massive than the arcjet options as previously indicated. However, the total EP system mass (including the propellant and tankage) of the arcjet systems is approximately a factor of 2 higher than that of the ion systems at the same power level and number of thrusters because of the lower values of specific impulse. The 10 kW total ion propulsion system masses range from 466 kg for the 4-engine option, to 257 kg for the single-engine option. The 10 kW hydrazine arcjet system masses range from 1161 kg for the 6-engine option to 536 kg for the single-engine option.

Table IX also lists the total spacecraft mass for all the EP system options, and the positive payload margin available on the four proposed launch vehicles with these spacecraft. The total spacecraft masses are most massive with the arcjet propulsion system options; consequently the payload margins are lower than with the ion system options. The payload margin with the 6-engine 10 kW arcjet propelled spacecraft on the DELTA II 6290 is negative, which indicates that this launch vehicle option is not available. To accomplish the baseline mission scenario of a 1000 hour thrusting-time would require that this system be launched on a heavier lift vehicle such as the ATLAS I. Positive payload margins for the arcjet propelled spacecraft range from a low of 57 kg on the DELTA II 6920 to a maximum of 1848 kg on the ATLAS II. All the payload margins for the ion propelled spacecraft are positive, ranging from a low of 184 kg on the DELTA II 6920 to a high of 1975 kg on the ATLAS II. That is, the baseline mission scenario (5000 hour total thrusting time) could be accomplished with the DELTA II 6920 for any of the ion system options.

Figures 12 and 13 show the spacecraft mass distribution for the 2-engine 10 kW ion and 6-engine hydrazine arcjet system options, respectively. As indicated in figure 12, the total ion propulsion system mass (including propellant and tankage) represents only 10% of the 3288 kg spacecraft mass. The propellant and tankage for the ion system is more than 55% of the total EP mass. The arcjet propulsion system mass (see figure 13) represents more than 28% of the 4111 kg total spacecraft mass, with the propellant and tankage constituting nearly 93% of the total EP mass.

CONCLUDING REMARKS

Ion and arcjet propulsion system options for a near-term space demonstration experiment were identified, and analyses were conducted to determine first-order propulsion system performance, and system component mass estimates. Overall mission performance of the EP systems was quantified in terms of total impulse, delta-v, and thrusting time capability based on a mission model which assumed a 2950 kg payload mass (including power) and either 5 or 10 kW available from a solar array to the EP system.

The thirty centimeter diameter ring-cusp engine operating on xenon propellant was baselined as the ion thruster option. This thruster can run from approximately 2450 seconds to 4520 seconds specific impulse, which corresponds to 62 to 79% overall thruster efficiency. Thrust levels range from approximately 0.04 to 0.42 Newtons, which corresponds to thruster input power levels from approximately 1 to 12 kW. Although ion thruster lifetime at high power levels has not been verified, it is believed a 35,000 A-h screen grid (beam current times operating time) lifetime limit at 28 V discharge voltage with xenon propellant is a conservative measure of thruster life. Ion thruster lifetimes ranging from approximately 5,800 to 35,000 hours, corresponding to the range of input power levels from 12 kW down to 1 kW, would be anticipated.

Both ammonia and hydrazine propellants were assumed for the arcjet. Thruster performance estimates on hydrazine range from 523 seconds to 800 seconds specific impulse, corresponding to 34% thruster efficiency. Thrust levels with hydrazine were estimated to be in the range of 0.20 to 0.81 Newtons for input power levels from approximately 1.5 to 9.4 kW.

The ion and arcjet propulsion systems were derived using comparable propulsion system mass and mission models. The system mass model used defined the propulsion system into a thrust module and an interface module. The mission model assumed that a low power ion or arcjet propulsion system would fly on a dedicated electric propulsion space experiment. Either 5 or 10 kW was assumed to be available from a solar array on the sapcecraft to the ion or arcjet propulsion module. Two mission modes were assumed; an extended-duration thrusting time dependent upon the launch vehicle payload margin, and a thrusting time demonstration of 5000 hours for the ion systems and 1000 hours for the arcjet systems. A DELTA II 6920, ATLAS I, DELTA II 7920, or ALTAS II launch vehicle deployment of the spacecraft to 550 km circular orbit was assumed.

Several candidate 5 and 10 kW ion and arcjet propulsion system options were identified. The majority of dry mass of the ion systems was in the power processor, which constituted greater than 40% of the total dry mass for all the system options. The arcjet propulsion system dry masses ranged from a maximum of 87 kg for a 6-engine 10 kW system, down to a minimum of 38 kg for a single-engine 5 kW system. For all of the arcjet systems, the power processor was the most massive system component, constituting anywhere from 28 to 34% of the total dry mass. The second most massive component was the dry interface module, except for the 6- and 2-engine 10 kW arcjet systems where the thermal control was the second heaviest component system component.

Values for total initial dry spacecraft mass for each combination of electric propulsion system option and payload were determined. This was done by individually adding each EP system dry mass to the 2950 kg payload and subtracting this mass value from the mass lift capability of the launch vehicle to define a payload margin. The payload margin, in part or in whole, was then consumed by the addition of propellant and tankage for the EP system.

The thrusting times and delta-v's could be determined by on-loading propellant to the spacecraft to fill the launch vehicle payload margin. Maximum available thrusting times for the ion-propelled spacecraft on the DELTA II 6920 vehicle range from approximately 8,600 hours for the 4-engine 10 kW system to more than 29,600 hours for the single-engine 5 kW system. Maximum total impulse values and maximum delta-v's on the DELTA II 6920 range from 1.2×10^7 to 2.1×10^7 N-s, and 3550 to 6200 m/s, respectively. Enough propellant could be on-loaded to most of the ion-propelled spacecraft to permit thruster operation greatly in excess of the projected thruster lifetimes.

Since the arcjet total propulsion system masses were substantially higher than that of the ion systems, the launch vehicle payload margins were much lower. Maximum available thrusting times for the arcjet propelled spacecraft on the DELTA II 6920 vehicle range from approximately 528 hours for the 6-engine 10 kW hydrazine system to 2328 hours for the single-engine 5 kW system. Maximum total impulse values and maximum delta-v's on the DELTA II 6920 range from 2.2×10^6 to 3.6×10^6 N-s, and approximately 662 to 1072 m/s, respectively. These values of total impulse and delta-v are approximately a factor of 5 to 6 lower than that obtained with the ion systems for the same payload and launch vehicle. Only three arcjet options provided total thrusting times in excess of 5000 hours. These were the single-engine 5 kW system on the ATLAS I (5736 hours), DELTA II 7920 (6096 hours), and the ATLAS II (7992 hours) launch vehicles.

The highest performance in terms of maximum delta-v for both the ion and arcjet systems were obtained with the 10 kW single-engine options. Maximum delta-v's for the single-engine ion-propelled spacecraft range from 6.2 km/s on the DELTA II to 18.6 km/s on the ATLAS II. Corresponding values for the single-engine 10 kW arcjet spacecraft range from 1.1 km/s to 3.0 km/s.

Under the constraint of a fixed thrusting time, the maximum spacecraft delta-v and total impulse were obtained with the 4-engine 10 kW xenon ion propulsion system, and (for arcjet) the 6-engine 10 kW hydrazine propulsion system. The xenon ion system provides a 2121 m/s delta-v capability, at a total impulse value of 7.1×10^6 N-s and total mass of 466 kg for a 5000 hour total burn-time. The hydrazine arcjet system provides a 1161 m/s delta-v capability, at a total impulse value of 4.3×10^6 N-s and total mass of 1161 kg for a 1000 hour total burn-time. The baseline mission scenario (5000 hour total thrusting time) could be accomplished with the DELTA II 6920 for all of the ion system options. However, to accomplish a 1000 hour thrusting time would require that the 6-engine 10 kW arcjet systems be launched on a heavier lift vehicle such as the ATLAS I.

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APPENDIX A

The assumptions used in generating predicted 30 cm ion thruster performance on xenon propellant were as follows:

1. The beam current (Jb) was calculated from an empirical equation which predicts the performance of 30 cm small-hole-accelerator-grid (SHAG) 2-grid ion optics on a laboratory model ring-cusp ion thruster with xenon propellant. This expression is given by

$$J_b = \frac{A \cdot 2.77 \times 10^{-5} \cdot (Vt)^{2.27}}{\sqrt{M}} \quad \text{for } 1000 \text{ V} \leq V_t \leq 2000 \text{ V} \tag{1}$$

   where A is the effective beam area, M is the propellant atomic mass unit, and Vt is the total accelerating voltage. This equation predicts 85%-of-maximum beam current attained experimentally for the optics described, at a cold-gap spacing of approximately 0.66 mm.

2. The total accelerating voltage across the optics ranged from 1000 to 2000 volts, inclusive. Two-grid SHAG optics were assumed, with an effective range of net-to-total voltage (R) of 0.55 to 0.85, inclusive. The maximum total accelerating voltage attainable was assumed to be 2500 V at 0.66 mm gap, or equivalently, 3800 V/mm maximum electric field strength.

3. A total thrust loss due to beam divergence (0.98, neglecting dependence on R-ratio) plus multiply-charged ions was estimated. The value of multiply-charged ions (J^{++}/J^{+}) was estimated based on a simple curve-fit of data obtained from 30 cm ion thruster beams, given by

$$\frac{J^{++}}{J^{+}} = -7.1078 \cdot N + 15.3317 \cdot N^2 - 14.7856 \cdot N^3 + 5.4228 \cdot N^4 + 1.2439 \tag{2}$$

   where N is the discharge chamber propellant efficiency.

4. The maximum discharge chamber propellant efficiency for xenon was assumed to be 0.95. The beam ion production cost was assumed to be 125 W/A. The discharge voltage was assumed to be 28 volts.

5. A fixed power loss of 0.050 kW was assumed.

6. A neutralizer mass flow rate of 3.2% of the beam current was assumed.

7. A thruster thermal limit of 800 watts maximum discharge power was specified to preclude the rare-earth cobalt magnets in the discharge chamber from exceeding 300 °C where irreversible losses in field strength may occur.

APPENDIX B

The modifications, from reference 19, employed in the low power arcjet propulsion system modeling include the following:

1. The arcjet mass, including gas generator, propellant valve, and mounting structure, was assumed to be a function of input power to account for changes in required radiating surface area. The system modeling identified 3 candidate power levels – ~1.5 kW, ~5 kW, and ~10 kW per thruster/ppu. The present 1 kW NASA-LeRC arcjet[13] is capable of operation up to approximately 2 kW. Consequently, this point design of 0.84 kg was used to estimate the 1.5 kW thruster mass. The thruster masses for the 5 and 10 kW power levels were estimated to be 1.70 and 2.95 kg, respectively.

2. The arcjet discharge supply mass (M_{ds}) in the power processor (in kilograms) was scaled based on the following equation

$$M_{d_s} = 0.455 + 4.095 \cdot \left(\frac{P}{1.5}\right)^{0.75} \tag{3}$$

where P is the input power into the power processor in kW. This equation is based on a single point design of a 1.4 kW flight-type arcjet power processor[14].

3. A power processor efficiency of 90% for ppu input power levels of ~1.7 kW was assumed. The power processor efficiency was assumed to be 93% at ~5 kW and 94% at ~10 kW. Line losses of 0.5% from the solar array to the ppu were assumed.

Table I. Propulsion System Model Description

THRUST MODULE

-- Thruster
-- Gimbal (fixed fraction of thruster mass)
-- Thermal Control (Power Processor only)
-- Power Processor (PPU)
-- Thruster Structure
-- Propellant distribution

INTERFACE MODULE

"Dry" Interface Module Mass:
-- Converter
-- Thrust System Controller
-- Reconfiguration Unit
-- Thermal Control
-- Housing structure

Propellant Module Mass:
-- Propellant
-- Tankage
-- Tankage Structure

Table II. Projected 30 cm Xenon Ion Thruster Performance

Beam Current, A	Thrust, N	Input Power, kW	Isp, s	Thruster Efficiency	V_t, V	R ratio	Mass flow kg/s
1.13	.052	1.14	3179	.71	1050	.80	1.66e-6
1.39	.067	1.52	3347	.73	1150	.81	2.05e-6
1.91	.098	2.27	3526	.75	1325	.78	2.82e-6
2.35	.126	3.03	3712	.76	1450	.79	3.46e-6
3.15	.180	4.55	3960	.77	1650	.79	4.64e-6
5.15	.329	9.04	4414	.79	2050	.79	7.60e-6

Table III. Xenon Ion Propulsion System Options

Total Power, kW	5				10			
Number of Thrusters	4	3	2	1	4	3	2	1
Specific Impulse, s	3179	3347	3526	3960	3526	3712	3960	4414
Total Thrust, N	.208	.201	.196	.180	.392	.378	.360	.329
System Efficiency	.65	.67	.69	.70	.69	.70	.70	.72
Dry Mass,kg	208	166	123	79	234	191	147	101
Specific Mass,kg/kW	41.6	33.1	24.5	15.8	23.4	19.1	14.7	10.1

Table IV. Xenon Ion Propulsion System Component Masses

Total Power, kW	5				10			
Isp,s	3179	3347	3526	3960	3526	3712	3960	4414
Number of Thrusters	4	3	2	1	4	3	2	1
Thruster/Gimbal,kg	59	44	29	15	59	44	29	15
Thermal Control,kg	11	11	11	11	21	21	21	2
Power Processor,kg	98	77	55	33	111	89	66	41
Thruster Structure,k	18	14	9	5	18	14	9	5
Dry Interface Module,kg	22	20	19	15	25	23	22	19
Total Dry Mass,kg	208	166	123	79	234	191	147	101

Table V. Projected Arcjet Thruster Performance

Thrust, N	Input Power, kW	Isp, s	Thruster Efficiency	Mass flow kg/s
NH_3				
.195	1.49	500	.32	3.97e-5
.428	4.63	750	.34	5.81e-5
.763	9.35	900	.36	8.64e-5
N_2H_4				
.198	1.49	523	.34	3.86e-5
.423	4.63	758	.34	5.69e-5
.810	9.35	800	.34	1.03e-4

Table VI. Arcjet Propulsion System Options

Total Power, kW	5				10					
Propellant	NH_3		N_2H_4		NH_3			N_2H_4		
Number of Thrusters	3	1	3	1	6	2	1	6	2	1
Specific Impulse, s	500	750	523	758	500	750	900	523	758	800
Total Thrust, N	.584	.428	.593	.423	1.168	.855	.763	1.187	.846	.810
System Efficiency	.29	.31	.30	.31	.29	.31	.34	.30	.31	.32
Dry Mass,kg	49	38	49	38	87	65	57	87	65	57
Specific Mass,kg/kW	9.8	7.6	9.8	7.6	8.7	6.5	5.7	8.7	6.5	5.7

Table VII. Arcjet Propulsion System Component Masses

Total Power, kW	5		10		
Number of Thrusters	3	1	6	2	1
Thruster/Gimbal,kg	3.4	2.3	6.8	4.6	4.0
Thermal Control,kg	13.4	9.4	26.9	18.8	16.1
Power Processor,kg	14.6	10.5	29.2	21.0	17.4
Thruster Structure,kg	1.0	0.7	2.1	1.4	1.2
Dry Interface Module,kg	16.0	15.0	22.0	19.0	18.0
Total Dry Mass,kg	~49	~38	~87	~65	~57

Table VIII. Ion and Arcjet Propulsion System and Mission Parameters for Extended-Duration Mission (Mission Mode I - ~Zero Launch Vehicle Payload Margin)

(a). Xenon Ion Propulsion

System and Mission Parameters		Launch Vehicle Option							
		Delta II 6920							
EP System Design	Power	5				10			
	No. of Thrusters	4	3	2	1	4	3	2	1
Initial Thrust/Weight		5.78e-5	5.61e-5	5.47e-5	5.00e-5	1.09e-4	1.05e-4	1.00e-4	9.15e-5
Max. Thrust Time, days		674	768	909	1237	361	433	538	727
Max. Total Impulse, N-s		1.21e7	1.33e7	1.54e7	1.92e7	1.22e7	1.41e7	1.67e7	2.07e7
Max. Δ V, m/s		3550	4000	4600	5800	3600	4200	5000	6200
EP System Dry Mass, kg		208	166	123	79	234	191	147	101
Xenon/Tankage Mass, kg		442	467	507	567	403	444	493	546
Total EP System Mass, kg		650	633	630	646	637	635	640	647
Payload Mass, kg		2950	2950	2950	2950	2950	2950	2950	2950
Total Initial S/C Mass, kg		3600	3583	3580	3596	3587	3585	3590	3597

System and Mission Parameters		Launch Vehicle Option							
		Atlas I							
EP System Design	Power	5				10			
	No. of Thrusters	4	3	2	1	4	3	2	1
Initial Thrust/Weight		4.62e-5	4.50e-5	4.37e-5	4.02e-5	8.74e-4	8.41e-4	8.01e-5	7.31e-5
Max. Thrust Time, days		2047	2231	2526	3156	1167	1318	1522	1928
Max. Total Impulse, N-s		3.68e7	3.87e7	4.28e7	4.91e7	3.95e7	4.30e7	4.73e7	5.48e7
Max. Δ V, m/s		9450	10200	11200	13000	10200	11200	12400	14400
EP System Dry Mass, kg		208	166	123	79	234	191	147	101
Xenon/Tankage Mass, kg		1344	1356	1408	1447	1301	1352	1396	1448
Total EP System Mass, kg		1550	1522	1531	1526	1535	1543	1543	1549
Payload Mass, kg		2950	2950	2950	2950	2950	2950	2950	2950
Total Initial S/C Mass, kg		4500	4472	4481	4476	4485	4493	4493	4499

System and Mission Parameters		Launch Vehicle Option							
		Delta II 7920							
EP System Design	Power	5				10			
	No. of Thrusters	4	3	2	1	4	3	2	1
Initial Thrust/Weight		4.52e-5	4.40e-5	4.28e-5	3.92e-5	8.56e-5	8.25e-5	7.87e-5	7.15e-5
Max. Thrust Time, days		2195	2394	2695	3401	1251	1405	1614	2062
Max. Total Impulse, N-s		3.94e7	4.16e7	4.56e7	5.29e7	4.24e7	4.59e7	5.02e7	5.86e7
Max. Δ V, m/s		10000	10800	11800	13800	10800	11800	13000	15200
EP System Dry Mass, kg		208	166	123	79	234	191	147	101
Xenon/Tankage Mass, kg		1440	1455	1502	1560	1395	1441	1480	1549
Total EP System Mass, kg		1648	1621	1625	1639	1629	1632	1627	1650
Payload Mass, kg		2950	2950	2950	2950	2950	2950	2950	2950
Total Initial S/C Mass, kg		4598	4571	4575	4589	4579	4582	4577	4600

System and Mission Parameters		Launch Vehicle Option							
		Atlas II							
EP System Design	Power	5				10			
	No. of Thrusters	4	3	2	1	4	3	2	1
Initial Thrust/Weight		4.08e-5	3.97e-5	3.86e-5	3.55e-5	7.71e-5	7.42e-5	7.07e-5	6.50e-5
Max. Thrust Time, days		2958	3210	3605	4441	1704	1906	2172	2677
Max. Total Impulse, N-s		5.32e7	5.57e7	6.10e7	6.91e7	5.77e7	6.22e7	6.76e7	7.61e7
Max. Δ V, m/s		12650	13600	14800	17000	13800	15000	16400	18600
EP System Dry Mass, kg		208	166	123	79	234	191	147	101
Xenon/Tankage Mass, kg		1941	1951	2010	2037	1900	1955	1992	2011
Total EP System Mass, kg		2149	2117	2133	2116	2134	2146	2139	2112
Payload Mass, kg		2950	2950	2950	2950	2950	2950	2950	2950
Total Initial S/C Mass, kg		5099	5067	5083	5066	5084	5096	5089	5062

(b). Hydrazine Arcjet Propulsion

System and Mission Parameters		Launch Vehicle Option				
		Delta II 6920				
EP System Design	Power	5		10		
	No. of Thrusters	3	1	6	2	1
Initial Thrust/Weight		1.65e-4	1.18e-4	3.30e-4	2.35e-4	2.25e-4
Max. Thrust Time, days		47	97	22	46	52
Max. Total Impulse, N		2.38e6	3.53e6	2.24e6	3.37e6	3.61e6
Max. Δ V, m/s		710	1051	662	1002	1072
EP System Dry Mass, kg		49	38	87	65	57
Hydrazine/Tankage Mass, kg		599	612	561	585	592
Total EP System Mass, kg		648	650	648	650	649
Payload Mass, kg		2950	2950	2950	2950	2950
Total Initial S/C Mass, kg		3598	3600	3598	3600	3599

System and Mission Parameters		Launch Vehicle Option				
		Atlas I				
EP System Design	Power	5		10		
	No. of Thrusters	3	1	6	2	1
Initial Thrust/Weight		1.32e-4	9.40e-5	2.64e-4	1.88e-4	1.80e-4
Max. Thrust Time, days		116	239	57	117	130
Max. Total Impul		5.96e6	8.72e6	5.81e6	8.55e6	9.08e6
Max. Δ V, m/s		1536	2245	1490	2198	2334
EP System Dry Mass, kg		49	38	87	65	57
Hydrazine/Tankage Mass, kg		1500	1512	1461	1484	1492
Total EP System Mass, kg		1549	1550	1548	1549	1549
Payload Mass, kg		2950	2950	2950	2950	2950
Total Initial S/C Mass, kg		4499	4500	4498	4499	4499

System and Mission Parameters		Launch Vehicle Option				
		Delta II 7920				
EP System Design	Power	5		10		
	No. of Thrusters	3	1	6	2	1
Initial Thrust/Weight		1.29e-4	9.20e-4	2.58e-4	1.84e-4	1.76e-4
Max. Thrust Time, days		124	254	61	125	138
Max. Total Impulse, N-s		6.36e6	9.29e6	6.22e6	9.14e6	9.69e6
Max. Δ V, m/s		1614	2358	1570	2312	2454
EP System Dry Mass, kg		49	38	87	65	57
Hydrazine/Tankage Mass, kg		1600	1612	1563	1584	1592
Total EP System Mass, kg		1649	1650	1650	1649	1649
Payload Mass, kg		2950	2950	2950	2950	2950
Total Initial S/C Mass, kg		4599	4600	4600	4599	4599

System and Mission Parameters		Launch Vehicle Option				
		Atlas II				
EP System Design	Power	5		10		
	No. of Thrusters	3	1	6	2	1
Initial Thrust/Weight		1.16e-4	8.30e-5	2.33e-4	1.66e-4	1.59e-4
Max. Thrust Time, days		163	333	80	164	182
Max. Total Impulse, N-s		8.35e6	1.22e7	8.20e6	1.20e7	1.27e6
Max. Δ V, m/s		1976	2880	1932	2836	3008
EP System Dry Mass, kg		49	38	87	65	57
Hydrazine/Tankage Mass, kg		2101	2110	2062	2084	2093
Total EP System Mass, kg		2050	2148	2149	2149	2150
Payload Mass, kg		2950	2950	2950	2950	2950
Total Initial S/C Mass, kg		5100	5098	5099	5099	5100

Table IX. Ion and Arcjet Propulsion System and Mission Parameters for Mission with Specified Thrust Time (Mission Mode II)

(a). Xenon Ion Propulsion

Total Power, kW	5				10			
Number of Thrusters	4	3	2	1	4	3	2	1
Isp,s	3179	3347	3526	3960	3526	3712	3960	4414
Initial Thrust/Weight	6.3e-5	6.2e-5	6.1e-5	5.8e-5	1.1e-4	1.1e-4	1.1e-4	1.0e-4
Thrust Time, days	208.3	208.3	208.3	208.3	208.3	208.3	208.3	208.3
Total Impulse, N-s	3.7e6	3.6e6	3.5e6	3.2e6	7.1e6	6.8e6	6.5e6	5.9e6
Δ V, m/s	1155	1138	1120	1053	2121	2088	2026	1887
EP System Dry Mass, kg	208	166	123	79	234	191	147	101
Xenon/Tankage Mass, kg	137	126	116	96	232	214	191	156
Total EP Mass, kg	345	292	239	175	466	405	338	257
Payload Mass, kg	2950	2950	2950	2950	2950	2950	2950	2950
Total S/C Mass, kg	3295	3242	3189	3125	3416	3355	3288	3207
Launch Vehicle Payload Margin, kg								
Delta II 6920	305	358	411	475	184	245	312	393
Atlas I	1205	1258	1311	1375	1084	1145	1212	1293
Delta II 7920	1305	1358	1411	1475	1184	1245	1312	1393
Atlas II	1805	1858	1911	1975	1684	1745	1812	1893

(b). Hydrazine Arcjet Propulsion

Total Power, kW	5		10		
Number of Thrusters	3	1	6	2	1
Isp, s	523	758	523	758	800
Initial Thrust/Weight	1.7e-4	1.3e-4	2.9e-4	2.4e-4	2.3e-4
Thrusting Time, days	41.7	41.7	41.7	41.7	41.7
Total Impulse, N-s	2.1e6	1.5e6	4.3e6	3.0e6	2.9e6
Δ V, m/s	643	484	1161	914	885
EP System Dry Mass, kg	49	38	87	65	57
Hydrazine/Tankage Mass, kg	537	264	1074	528	479
Total EP System Mass, kg	586	302	1161	593	536
Payload Mass, kg	2950	2950	2950	2950	2950
Total Initial S/C Mass, kg	3536	3252	4111	3543	3486
Launch Vehicle Payload Margin, kg					
Delta II 6920	64	348	-511	57	114
Atlas I	964	1248	389	957	1014
Delta II 7920	1064	1348	489	1057	1114
Atlas II	1564	1848	989	1557	1614

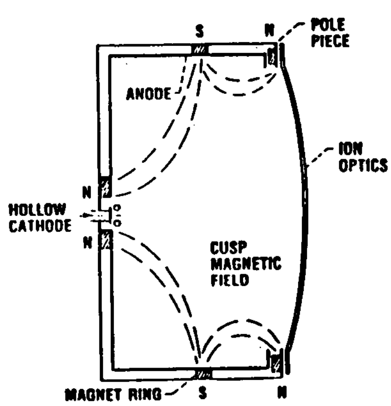

Figure 1. - Cross section of ring-cusp ion thruster.

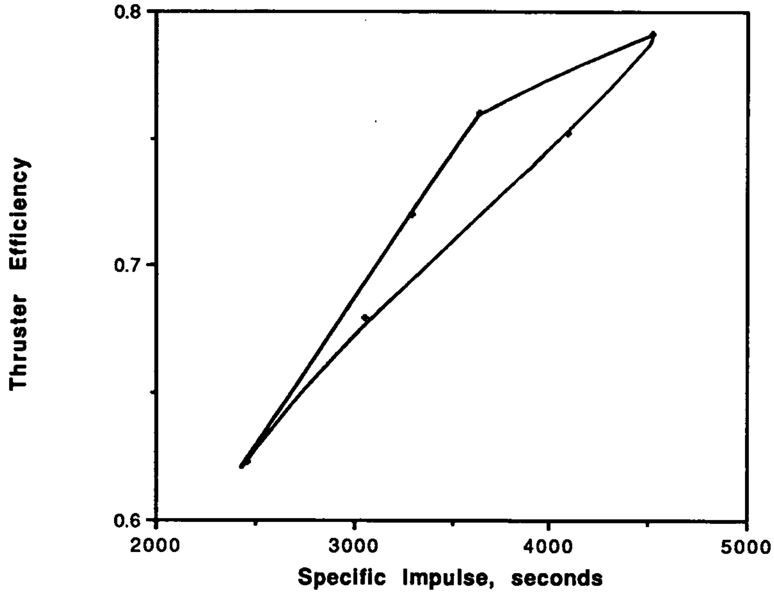

Figure 2. - Ion thruster efficiency versus specific impulse with xenon propellant.

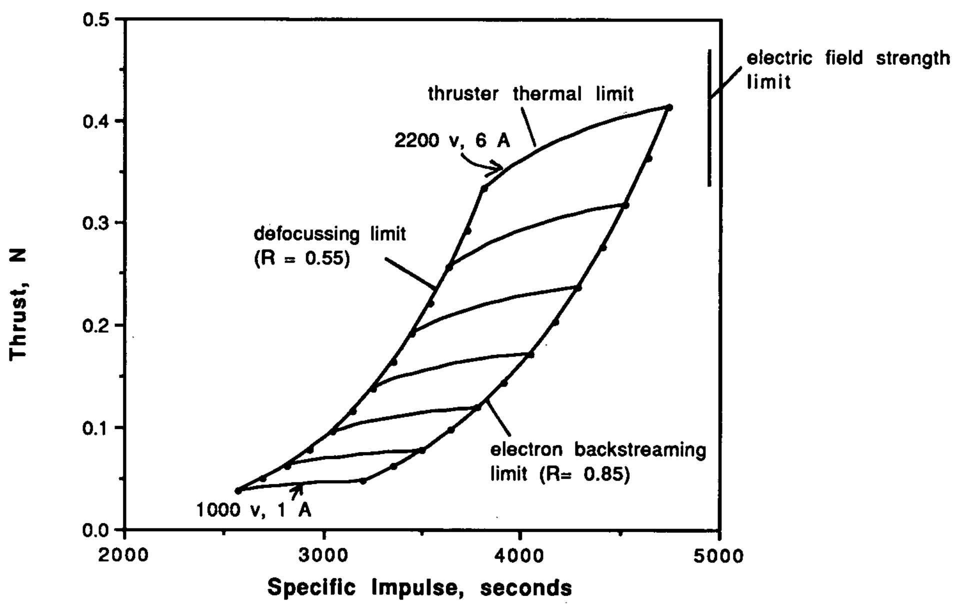

Figure 3. - Ion thruster thrust versus specific impulse.
Approximate operating envelope for 30 cm ring-cusp xenon ion thruster technology.

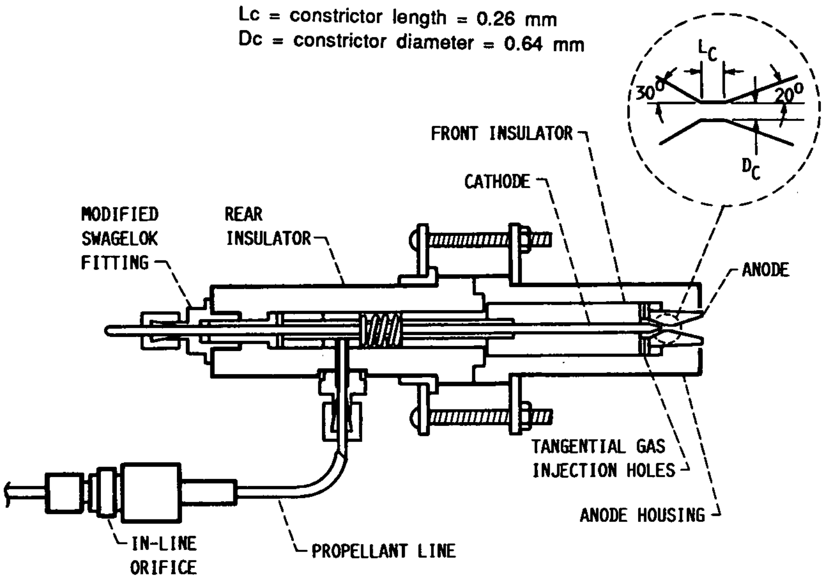

Figure 4. - Cutaway view of arcjet thruster with typical dimensions.

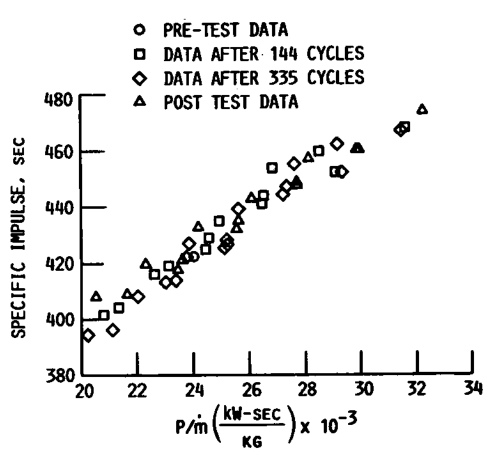

Figure 5. - Performance measurements before, during, and after a 1000 hr/500 cycle automated arcjet lifetest.

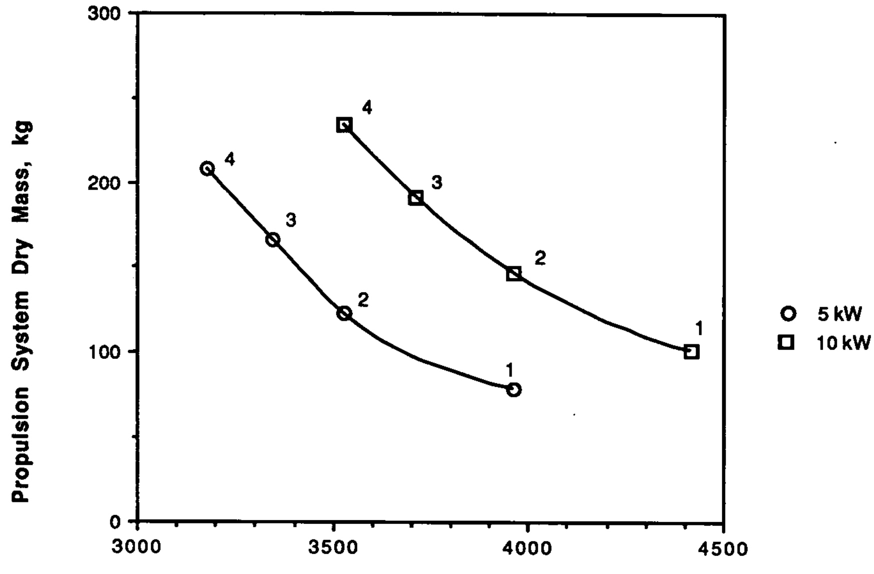

Figure 6. - Ion propulsion system dry mass versus specific impulse. Number of thrusters is indicated.

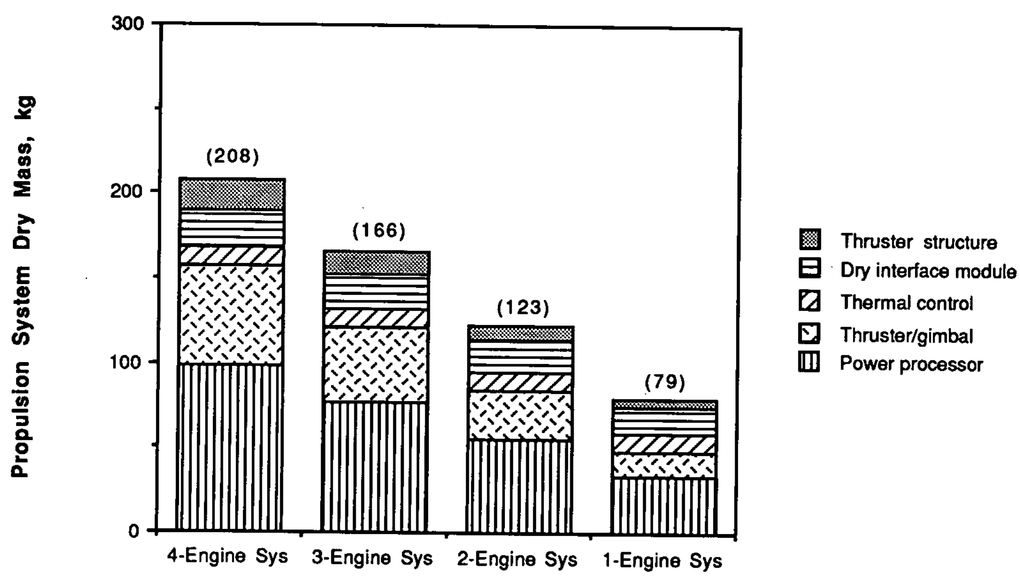

Figure 7. (a) - Dry mass distribution of 5 kW ion propulsion systems.

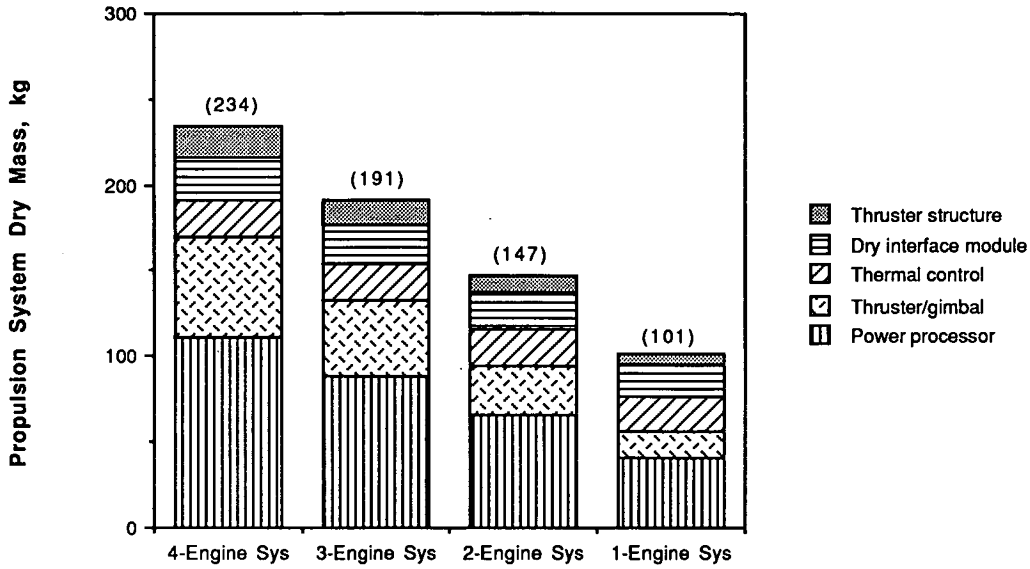

Figure 7. (b) - Dry mass distribution of 10 kW ion propulsion systems.

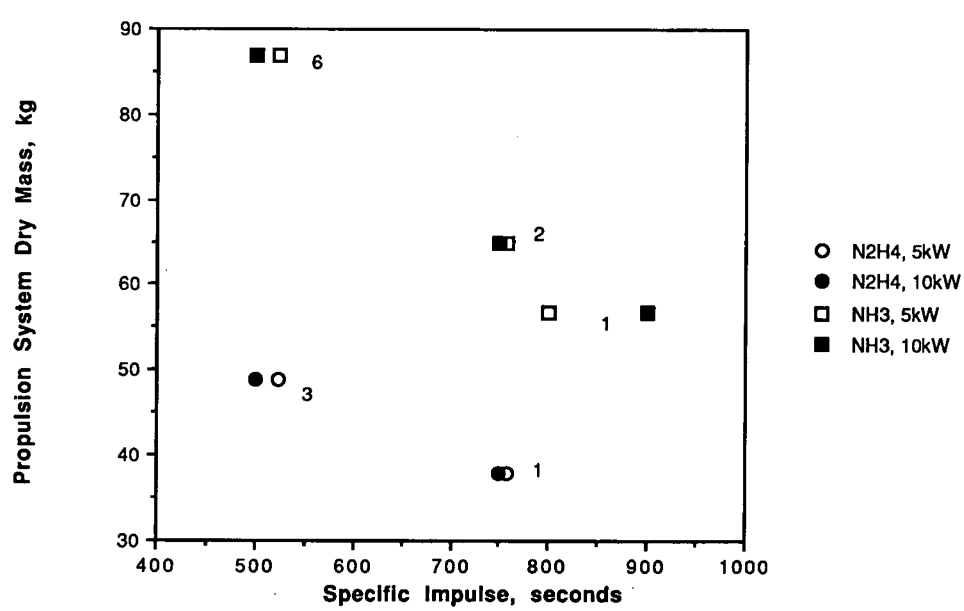

Figure 8. - Arcjet propulsion system dry masses versus specific impulse.
Number of thrusters is indicated.

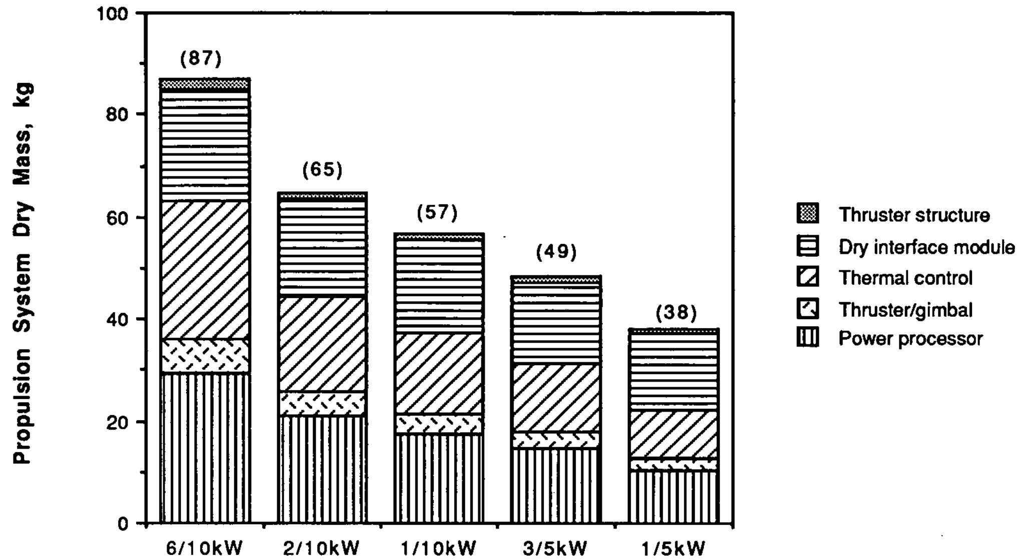

Figure 9. - Dry mass distribution of arcjet propulsion systems.

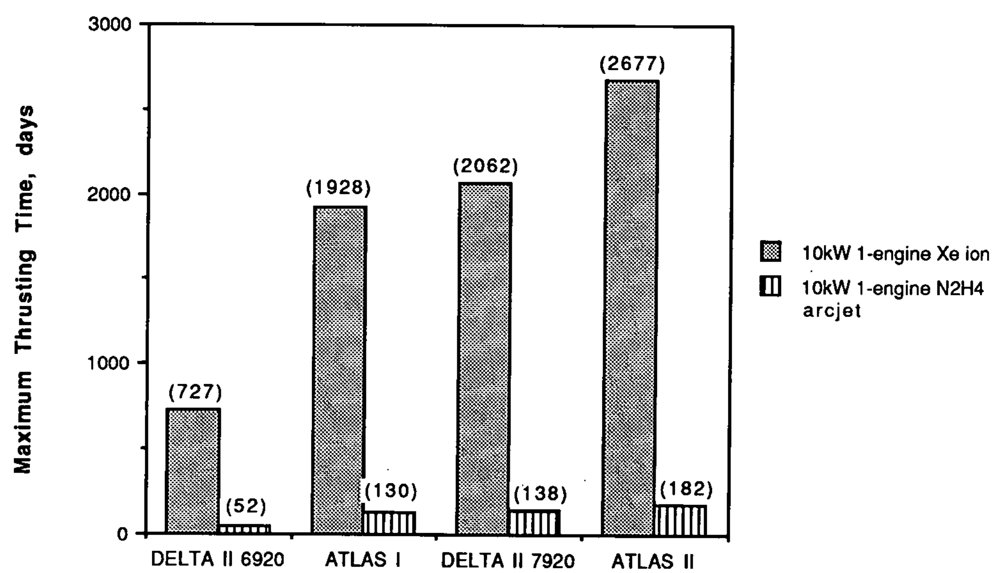

Figure 10. - Max. thrusting time vs. launch vehicle option for extended-duration mission. Mission Mode I

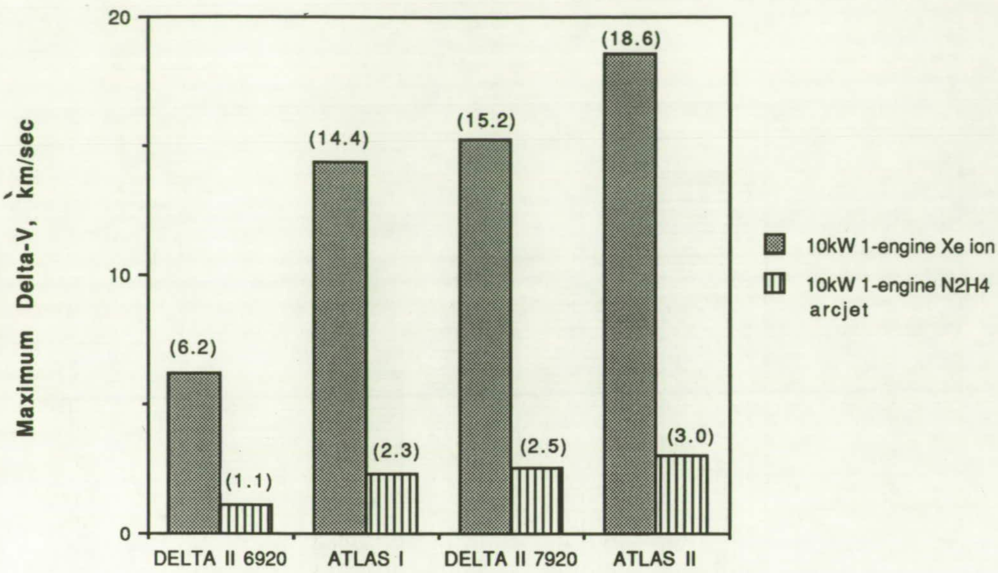

Figure 11. - Max. delta-v vs. launch vehicle option for extended-duration mission. Mission Mode I

Total propulsion system mass 338 kg.

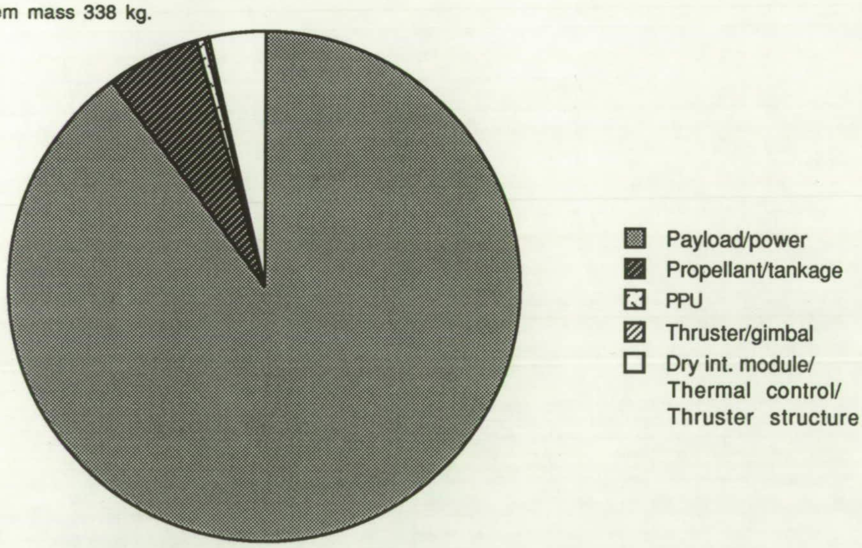

Figure 12. - Spacecraft mass distribution with 2-engine 10kW Xe ion option. 5000 hour thrusting time. Total s/c mass 3288 kg. Mission Mode II

Total propulsion system mass 1161 kg

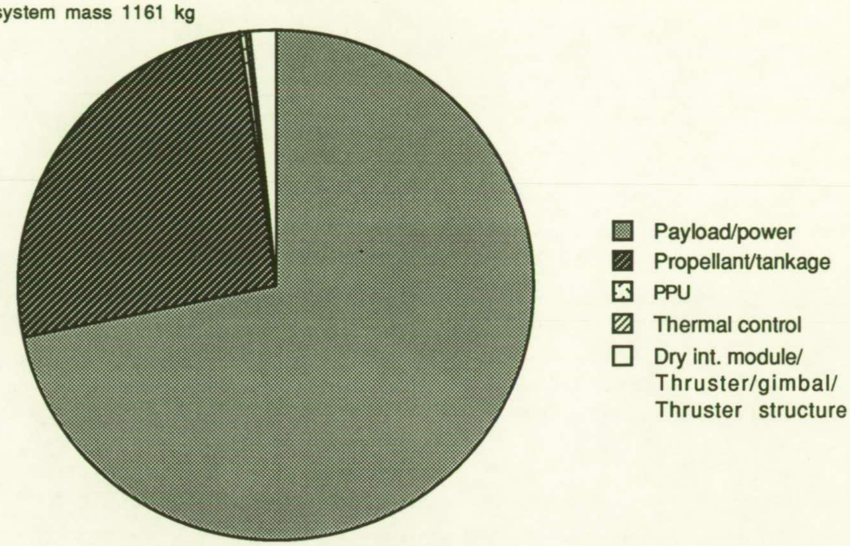

Figure 13. - Spacecraft mass distribution with 6-engine 10kW N2H4 arcjet option.
1000 hour thrusting time. Total s/c mass 4111 kg.
Mission Mode II