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# Low-Power Ion Propulsion for Small Spacecraft

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# Low-Power Ion Propulsion for Small Spacecraft

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**Analyses were conducted which indicate that sub kW-class ion thrusters may provide performance benefits for near-Earth space commercial and science missions. Small spacecraft applications with masses ranging from 50 to 500 kg and power levels less than 0.5 kW were considered. To demonstrate the efficacy of propulsion systems of this class, two potential missions were chosen as examples; a geosynchronous north-south station keeping application, and an Earth orbit m-agnetospheric mapping satellite constellation. Xenon ion propulsion system solutions using small thrusters were evaluated for these missions. A payload mass increase of more than 15% is provided by a 300-W ion system for the north-south station keeping mission. A launch vehicle reduction from four to one results from using the ion thruster for the magnetospheric mapping mission. Typical projected thruster performance over the input power envelope of 100-300 W range from approximately 40% to 54% efficiency and approximately 2000 to 3000 seconds specific impulse. Thruster technologies required to achieve the mission-required performance and lifetime are identified.**

## Introduction

Analyses are ongoing to examine ion thruster scaling relationships in detail to determine system requirements, performance limits, and lifetime expectations. Specifically, electron-bombardment xenon ion thruster solutions are being evaluated for input power levels of several hundred watts. Solutions examined include thruster sizes ranging from about six to 10 cm in beam diameter, at input power levels in the range of 0.1-0.3 kW. The performance, design, and lifetime goals for this engine class are identified in Table I.

The impacts of low-power ion propulsion systems on commercial and science missions were considered. This was done to investigate the relative benefits of developing flight systems based on low power ion thrusters for application on small spacecraft.

North-South station keeping (NSSK) on a small (430 kg) geostationary satellite was considered as one target mission for low-power ion technology. A constellation of four 65-kilogram magnetospheric mapping spacecraft with a mission consisting of an orbit transfer from LEO to GEO was also chosen to investigate the viability of the small ion propulsion for small spacecraft.

This paper discusses these two mission applications, the benefits of, and technology requirements for low power ion propulsion as applied to small spacecraft for these commercial and science missions.

## Propulsion System

This section describes elements of the low power ion propulsion system used in this study including thruster, power processor, and propellant feed system.

### Thruster

Estimates for low-power electron bombardment xenon ion thruster operation were calculated for use in the mission analyses. The methodology and results are discussed here.

**Performance -** For purposes of this analysis, input power levels from about 100 W to a maximum of about 300 W input power into the thruster were assumed. Thruster configurations ranging from about 6 cm to 10 cm beam diameters were initially examined for this power range, with a final selection of 8 cm (beam diameter) for this study. The considerations driving the thruster size selection included: the maximum acceptable beam current density, which impacts grid life time; the minimum discharge electrical efficiency which impacts overall thruster efficiency; and the maximum acceptable operating discharge voltage which impacts both discharge chamber and screen grid lifetime.

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An 8-cm thruster operating at 200 W input power, and at comparable grid voltages to the 30-cm NSTAR thruster, will operate at about the same average current density as the NSTAR thruster (approximately 2.9 mA/cm$^2$). Hence, one would anticipate comparable accelerator grid lifetimes at this condition.

At about 3000 seconds specific impulse and at the maximum input power of 300 W, the beam current density of an 8 cm is about 4.1 mA/cm$^2$, or about 1.4 times that of the NSTAR thruster. As will be discussed in a following section, this increase in current density, while maintaining useful thruster lifetimes, is considered feasible.

In general, as total propellant throughput decreases, the discharge electrical efficiency also decreases. That is, the power to produce an ampere of beam ion current increases as the thruster is power-throttled down.[1] This is because the neutral density in the discharge decreases and, hence, the probability that energetic electrons will undergo inelastic collisions prior to being collected at anode surfaces decreases.

Because the primary electron containment length decreases as the beam diameter is reduced, the discharge electrical efficiency also decreases. To maintain a constant propellant efficiency the discharge must be operated at successively higher voltages as the thruster diameter is decreased.[2] Restated, if fewer electrons are available to ionize the gas (due to the higher loss rate of primary electrons), their mean energy has to increase to maintain a constant ionization rate.

For a beam diameter of 8 cm, very high discharge voltages (>32 V) are required to attain discharge propellant efficiencies of 90% or greater.[2] Here, a clear trade exists in thruster efficiency versus life time, in the trade of maximum propellant efficiency versus maximum discharge voltage.

For this analysis, it was assumed that the thruster would be operated at a maximum discharge voltage of 28 V at full power, to ensure adequate life time. This is consistent with past design criteria, including that used in the development of the NSTAR 30 cm thruster. A linear increase in discharge propellant efficiency with input power is expected, and propellant efficiencies ranging from about 78% at 100 W to about 82% at 300 W were assumed.

The discharge losses were estimated using a correlation established between discharge losses and thruster input power.[1] At an input power of 100 W, discharge losses of 333 W/A were estimated; at 300 W thruster input power, 266 W/A was assumed.

The ion optics performance for the 8-cm thruster was scaled from that demonstrated with 2-grid 30 cm optics. For this analysis, the perveance documented for the small-hole-accelerator-grid (SHAG) ion optics from reference 3 was used. For these optics, the beam current (in amperes) was found to be approximately $4.8\times10^{-5}(Vt)^{1.5}$, where Vt is the total accelerating voltage in volts.

Adjusting for the difference in beam area, the perveance-limited beam current for the 8 cm using SHAG optics ranged from about 57 mA to about 204 mA, over a range of 600 V to 1400 V total voltage. This total voltage range is approximately equal to that which is used for the NSTAR thruster. For performance estimations, the accelerator grid voltage was calculated assuming an R-ratio of 0.80, and the accelerator grid current was assumed to be equal to 0.50% of the beam current over the entire power-throttling range.

A hollow cathode with keeper was assumed to provide the beam neutralization. A fixed keeper current of 100 mA (yielding a conservative 3:1 total emission current ratio with/without beam extraction) at 20 V keeper voltage and 15 V coupling voltage was assumed. Applying an empirically-derived correlation of flow rates to emission current, a maximum flow rate of 36 eq. mA xenon (about 0.49 sccm) was estimated for the neutralizer.

Using the aforementioned assumptions regarding discharge chamber, ion optics, and neutralizer operation, performance estimates for an 8-cm thruster were obtained over an input power range of about 100 W to 300 W. These estimates are shown in Table II, and in Figures 1 and 2. Thrust losses associated with beam divergence and doubly-charged ions were accounted for in these estimates, using the methodology described in reference 3.

As indicated in Figure 1 and Table II, estimates of thruster efficiency range from about 37% at 1810 seconds specific impulse and 85 W input power, to about 54% at 2960 seconds and 300 W input power. These performance values are believed to be reasonable goals based on testing conducted to date.

A critical area necessary to achieve the goals and performance levels identified in Tables I and II is development of low-flow rate xenon hollow cathodes. The hollow cathode neutralizer performance has a significant impact on overall thruster efficiency at 100-300 W thruster power levels. A program to develop efficient, low flow cathodes to support low-power electric propulsion systems is in progress. The performance of one of the first units is shown in Figure 3, a plot of minimum xenon flow rate (to maintain stable spot-mode operation) versus emission current.

Also shown in Figure 3 are data obtained from the NSTAR thruster neutralizer, and the projected (assumed) neutralizer performance used in this analysis. As indicated, the prototype neutralizer operates at approximately 7-8 times lower flow rate for the same emission current, compared to the NSTAR neutralizer. However, additional improvements (factor of 2 reduction in flow rate and emission current) are needed to achieve the performance assumed in this analysis.

At 100 W thruster input power the hollow cathode neutralizer operation can degrade the thruster efficiency and specific impulse by as much as 20 percentage-points and 2000 seconds, respectively, as indicated in Figure 4. In Figure 4, projected thruster performance is given for three cases; a zero-flow rate/zero power-consuming neutralizer, a 0.05 sccm flow rate neutralizer (operating at the same input power as that of the 0.50 sccm neutralizer used in this analysis), and the baseline 0.50 sccm neutralizer. The performance of the 0.05 sccm flow rate neutralizer is comparable to that which was demonstrated previously on an 8 cm mercury ion thruster.[4]

Note for the 0.50 sccm neutralizer curve, the thruster efficiency decreases with increasing specific impulse. This is because at fixed thruster input power a direct trade of beam current for beam voltage is made, for increasing specific impulse. This results in a very rapid decrease in the maximum obtainable propellant efficiency, since the fixed neutralizer flow rate of 0.50 sccm becomes a larger fraction of the total thruster flow rate.

**Lifetime -** For the NASA NSTAR 30 cm thruster, the erosion of the molybdenum accelerator grid due to charge-exchange ions is one of the dominant life limiting wear-mechanisms. If the internal discharge voltage of a small thruster is limited to 28 V to limit internal erosion, then charge-exchange erosion of the accelerator grid is expected to be the overall life limiter.

Examinations of accelerator grid erosion on many different ion thrusters have led to a consensus that the end-of-life of an accelerator grid will be determined by structural failure in the center of the grid where the erosion is the greatest. In particular, erosion occurs around each grid hole as deep pits which are connected together by shallower trenches.[5] Erosion in the trenches is a minimum between adjacent charge-exchange pits. After significant erosion occurs, each grid hole is bridged to its neighbors at these minimum erosion sites.[6] End-of-life of the accelerator grid is thereby defined as the point in time at which these bridges in the center of the grid become unsound and fail.

The relevant local measurement for this accelerator grid end-of-life mechanism is the bridge depth erosion in the grid center. A compilation of the magnitude of accelerator grid erosion experienced during extended-duration tests along with the thruster operating conditions were documented in reference 7. Using these data, a "grid erosion parameter" (or GEP) was proposed (consisting of the product of the accelerator grid impingement current, test time, and grid material sputter yield, divided by the beam area) as a straightforward combination of measured parameters with a high correlation to the magnitude of the charge exchange erosion.[7] The NSTAR lifetest results to date suggest that this approach yields a conservative erosion estimate.

Using the GEP, the accelerator grid lifetime of the NSTAR thruster at 2.3 kW was estimated to be greater than 12,000 hours.[7] This estimate was supported by subsequent post-life test erosion measurements which indicated the grid lifetime was in fact in excess of 12,000 hours.[8,9] The NSTAR thruster lifetime is conservatively-quoted as having a total propellant throughput of 83 kg, which is the propellant expended at a full-power level of 2.3 kW for 8000 hours.

The 8-cm thruster should yield comparable life times to that of the NSTAR thruster when operated at similar voltages and current densities. At equivalent full-power conditions and assuming comparable optics technology, the small thruster should have at least a 10-kg throughput capability.

Using the GEP, estimates of small thruster life times were obtained for the conditions identified in Table II. These data are shown in Figure 5, thruster (accelerator grid) life versus thruster input power. For these calculations, 2-grid molybdenum ion optics was assumed with an initial accelerator grid thickness of 500 microns.

Two curves are shown in Figure 5 corresponding to two different end-of-life criteria. One criterion is erosion of the bridge to a depth of 200 microns, or 40% of the way through the thickness of the electrode. This is presently used in the NSTAR program as a conservative definition of end-of-life. This is quite conservative as the grid still has considerable strength at this erosion level. A second curve is shown in Figure 5, corresponding to a bridge erosion to a depth of 400 microns, or 80% of the way through the thickness of the electrode. In a test-to-failure-test[6] the accelerator grid electrode was eroded to this depth and it was still functional at the completion of the test. As indicated in Figure 5, lifetimes of $\geq$ 8000 hours are anticipated for input power levels up to 300 W assuming 400 microns erosion.

Figure 5 indicates that using the conservative NSTAR end-of-life criterion for accelerator grid erosion may result in thruster lifetimes less than 8000 hours for input power levels greater than about 180 W. At the 300 W power level assumed for the proposed missions, the anticipated life is about 4000 hours. Approaches to enhancing accelerator grid and thruster life are available and these include: changing to a three-grid configuration; application of sputter-resistant coatings to the molybdenum accelerator grid surface; or changing to a carbon-based grid material.

Other approaches to increased life include limiting the maximum power to the 8-cm thruster to a value less than 300 W, or increasing the thruster size at 300 W input. For example, an increase in thruster diameter to 10 cm would be expected to yield at least a 50% increase in grid life.

The thruster total impulse versus input power is shown in Figure 6 for both end-of-life criteria. As indicated, total impulse values ranging from $9.6 \times 10^5$ N-s down to about $3.1 \times 10^5$ N-s are estimated over the power envelope of 85 W to 300 W, assuming 400 microns erosion.

**Physical Characteristics** - A 0.30 kW class electrostatic thruster could implement similar design, materials and fabrication techniques as those employed in the 30-cm NSTAR engineering model ion thruster.[10] These include a partial-conic anode-potential discharge chamber constructed of non-ferromagnetic materials,[10] and a ring-cusp magnetic circuit.[11]

The fabrication techniques and material used in the NSTAR thruster allow for very lightweight thrusters to be built. For example, the 30-cm NSTAR thruster mass is about half that of other engineering model and flight model thrusters of this approximate size.[12] Using this same approach, an 8-cm flight thruster mass of 0.775 kg is estimated.

The overall thruster length, as measured from the tip of the neutralizer, to the rearmost portion of the plasma screen, is estimated to be about 17.6 cm. The outside diameter of the thruster, as defined by a circle which includes the neutralizer assembly, is estimated to be 16.7 cm.

**Power Processing**

The power processor unit (PPU) mass for the ion thruster is estimated to be approximately 2.0 kg, at about 300 W maximum.[13] A PPU topology similar to that implemented in the NSTAR program, with the input bus voltage of 24-32 volts was also assumed.[13] The efficiency of the PPU is assumed to vary linearly with input power, going from about 0.87 to 0.89 over an input power range of 100 W to 300 W.[14]

**Propellant Feed System and Structure**

The tankage in the systems considered is set at 10% of the propellant mass.[15] A gimbal mass equal to 34% of the thruster mass, and a structure mass equal to 31% of the combined thruster, gimbal,and feed system masses, are assumed.[16] Additional mounting structure of 4% of the PPU, propellant, and tankage are also assumed.[17] A thermal radiator mass equal to 31 kg/kW-dissipated was also assumed.[16]

## Mission Analyses

To investigate the relative benefits of developing flight systems based on the 0.30 kW class ion thruster, two mission examples were considered. The first is North-South station keeping (NSSK) of a small (430 kg) geostationary satellite since smaller geostationary satellites designed to serve one customer or provide a single service are currently being considered.[17]

The second mission example is a 65-kg magnetospheric mapping spacecraft. This mission consists of an orbit raise from LEO to GEO to investigate the viability of the small ion propulsion for small science spacecraft. The ion propulsion system component masses used in both mission examples were based on the information presented above. A potential propulsion system configuration was also suggested. Each mission application compared the ion propulsion system to SOA propulsion systems. In both cases, a substantial mass savings was demonstrated as a result of using the ion thruster propulsion system, which could then be allocated to increase the usable payload mass. Conversely, if the baseline payload remained unchanged, the total spacecraft mass and launch mass could be reduced through the use of the ion system.

**Small Geostationary Satellite**

Large geostationary satellites continue to be an important part of the communication industry. Smaller geostationary satellites designed to serve one customer or provide a single service is also being considered. One such example is the planned Indostar 1 spacecraft, shown in Figure 7.[18] With a beginning-of-life (BOL) mass of 430 kilograms and an end-of-life (EOL) power of 0.9 kilowatts, the Indostar 1 is significantly smaller than other planned geostationary satellites.

Using the Indostar 1 as representative of this new class of satellites, the impact of the ion thrusters on reducing the wet system mass was estimated. Reduction in propulsion system wet mass would allow for an increase in the mass of the payload and support systems, a reduction in launch mass, or an increase in the spacecraft life. A 10-year mission with a 45 m/s NSSK budget per year is assumed.[19] State-of-art $N_2H_4$ monopropellant[20] and advanced arcjet systems[21] were used for comparison. The operating parameters and system masses assumed are shown in Table III.

The configuration assumed for the ion and arcjet systems consists of four thrusters, two each on the north and south faces of the satellite and two PPUs. To minimize the effect of plume impingement on the solar arrays the ion thrusters were canted at 301 relative to the optimal thrust direction along the north-south axis and the arcjets were canted at 171.[22,23] Two thrusters are operated at a time. Burns are at one of the orbit nodes once per day, although less periodic burns are possible depending on the orbit inclination tolerance required. The electric thrusters run off of the eclipse batteries while the payload uses the solar array power.[23] While the added cycling may require extra batteries to ensure 10 year payload eclipse operations, this was not included in the analysis.

Each of the electric propulsion systems requires lifetimes less than those currently predicted. For the ion system the two thrusters fire for approximately 45 minutes once a day. For the arcjet system, each of the two thrusters fires for approximately 11 minutes once a day, although longer burns, less often might be tolerated. Slightly longer burns may also be needed to reduce requirements during eclipse period.

The required propellant and propulsion system dry masses are shown in Figure 8. As indicated, all of the system dry masses were below 20 kilograms. However the differences in propulsion system wet masses were significant. The hydrazine monopropellant system was the heaviest, with a fueled mass of 92 kilograms. The arcjet system has a wet mass of 58 kilograms. The SOA ion system has a wet mass of only 23 kilograms. The approximate 75% reduction in propulsion system wet mass for the ion system relative to the hydrazine monopropellant propulsion system, corresponds to a 69-kilogram mass savings on a 430-kilogram spacecraft. This extra mass could be used for more communications payload along with the support systems required.

**Magnetospheric Mapping Constellation**

In this mission, four spacecraft would spiral in a constellation from 600 km to 36,000 km at a 65/ inclination for over a year to obtain spatial data of the Earth=s magnetosphere. This multi-spacecraft mission will allow for continuous spiral exploration of a portion of the magnetosphere and revisits of regions of interest. The four identical spacecraft provide redundancy; two of the four spacecraft could fail and still some spatial data could be attained.

The Orbital Sciences Corporation (OSC) MicroStar bus[24] (shown in Figure 9) was selected as the bus for comparison of the 300 W ion and SOA bipropellant chemical propulsion system performance. MicroStar is a 50-100 kg class satellite with a dry bus mass of ~40 kg and a typical payload of ~50 kg. This spacecraft structure is a 0.981 m diameter x 0.114 m deep ring providing a disc-shaped region which contains the bus subsystems (e.g. the batteries, electronics, and propulsion), as well as the payload.[18,24,25]

For this mission, each of the four spacecraft has either a single small ion system operating at 0.30 kW, or a SOA bipropellant system for propulsion. The mission ªV is ~4700 m/s for both the ion thruster and the chemical thruster since near-circular orbits are to be maintained throughout the mission. The analysis includes shading, degradation, and a 5% coast time during sunlit periods for the ion propelled spacecraft. The bipropellant spacecraft could take more data ans/or complete the mission faster due to the relatively-higher thrust of the propulsion system.

Using the 0.30 kW ion propulsion system, four spacecraft can be launched from a single Pegasus XL. A 570 day transfer is required to transfer the spacecraft to geosynchronous altitude. During the transfer the spacecraft arrays are degraded to about half the original power level due to the Van Allen radiation belt. The power into the thruster PPU drops from about 300 W at BOL to about 160 W by the end of the mission as shown in Figure 10. The propulsion system efficiency (product of the thruster and PPU efficiencies) and the specific impulse decay from about 48% at 2960 seconds to about 41% at 2440 seconds over this power range. The estimated payload is 10 kg, with a total spacecraft launch mass of about 65 kg. The science payload power was assumed to be 25 W.

To perform the same mission using the bipropellant engine would require a 300 kg-class spacecraft, compared to the 65 kg ion propelled spacecraft. A comparison of the spacecraft launch mass for the ion propelled spacecraft versus the equivalent spacecraft using the 290-s bipropellant engine is shown in Figure 11. The fuel mass of the chemically-propelled spacecraft would be around 240 kg. The trip times for the chemically-propelled spacecraft constellation are limited by the time to acquire the spatial data and not the thrusting time.

The 300 kg mass of the bipropellant spacecraft would require a dedicated Pegasus XL launch for each spacecraft. Thus the bipropellant spacecraft would require a total of four Pegasus XL launch vehicles as compared to the single Pegasus XL launch vehicle needed for the ion propelled spacecraft constellation. At around 12 million dollars a launch for the Pegasus XL,[26] the small ion option could save this mission -36 million dollars in launch costs.

While the projected thruster lifetime at 300 W is adequate for the NSSK application, it falls considerably short of the 13,700 hours required for the magnetospheric mapping mission. This is mitigated however by the fact that the mission-average thruster input power is only about 200 W. The total-impulse requirement for the mission is about $3.0x10^5$ N-s which appears feasible based on the data of Figure 6.

## Conclusions

Analyses were conducted which indicate that sub 0.5 kW-class ion thrusters may provide performance benefits for commercial and science missions. Small spacecraft applications with masses ranging from 50 to 500 kg and power levels less than 500 W were considered.

Electron-bombardment xenon ion thruster systems were evaluated for these missions. A low power system was postulated and system characteristics were estimated. Typical projected small thruster performance over the input power envelope of 100-300 W range from approximately 40% to 54% efficiency and approximately 2000 to 3000 seconds specific impulse.

Two potential mission applications for the ion thruster operating at 300 W (BOL) were identified including a geosynchronous north-south station keeping application, and an Earth orbit magnetospheric mapping satellite constellation. Impacts on launch vehicle requirements were quantified for both missions.

The geosynchronous north-south station keeping mission considered the use of the small (430 kg) Indostar 1 spacecraft, and a 10-year mission with a 45 m/s NSSK budget per year. Use of the 300 W ion system yielded an approximate 75% reduction in propulsion system wet mass relative to the hydrazine monopropellant propulsion system.

The Earth orbit magnetospheric mapping satellite constellation mission would use four Orbital Science Corporation (OSC) Microstar-class spacecraft, each propelled by a single 300 W throttleable ion engine. The combination of the OSC satellite bus, low power ion propulsion, and a Pegasus XL launch vehicle, allow for a spiral of the constellation from 600 km to 36,000 km at 65 degrees inclination in approximately 570 days. The use of the small ion thruster enables a single Pegasus XL launch of all four satellites; four Pegasus XL's would be required to perform the mission chemically.

Critical thruster technology areas necessary to achieve the mission-required performance and lifetimes include the development of low-flow rate xenon hollow cathodes, and high-current density long-life ion optics.

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**Table I - Low-Power Ion Thruster Goals**

| Attribute | Value |
|---|---|
| Input Power, W | ≃ 200 W |
| Efficiency | ≥ 46% |
| Mass, kg | ≤ 1 kg |
| Life Time | ≥ 8000 h |

**Table II - 8 cm Thruster Solutions[a]**

| Input Power, W | Specific Impulse, sec | Thrust, mN | Efficiency, % | Beam Current, mA | Screen Voltage, V | Accel Current, mA | Accel Voltage, \|V\| | Discharge Current, A | Discharge Flow Rate, eq. mA |
|---|---|---|---|---|---|---|---|---|---|
| 85 | 1810 | 3.6 | 37.2 | 88.2 | 640 | 0.44 | 160 | 0.94 | 110 |
| 99 | 2030 | 4.0 | 40.1 | " | 800 | " | 200 | " | " |
| 113 | 2230 | 4.4 | 42.2 | " | 960 | " | 240 | " | " |
| 128 | 2410 | 4.8 | 43.9 | " | 1120 | " | 280 | " | " |
| 138 | 2180 | 5.6 | 43.2 | 123 | 800 | 0.62 | 200 | 1.32 | 154 |
| 158 | 2390 | 6.1 | 45.5 | " | 960 | " | 240 | " | " |
| 177 | 2590 | 6.6 | 47.2 | " | 1120 | " | 280 | " | " |
| 187 | 2680 | 6.8 | 48.0 | " | 1200 | " | 300 | " | " |
| 207 | 2510 | 8.1 | 47.9 | 162 | 960 | 0.81 | 240 | 1.73 | 202 |
| 233 | 2720 | 8.7 | 49.7 | " | 1120 | " | 280 | " | " |
| 246 | 2820 | 9.0 | 50.5 | " | 1200 | " | 300 | " | " |
| 286 | 2860 | 10.9 | 53.6 | 204 | 1120 | 1.02 | 280 | 1.94 | 249 |
| 303 | 2960 | 11.3 | 54.3 | " | 1200 | " | 300 | " | " |

[a]Discharge voltage = 28 V. Neutralizer parameters: 0.1 A keeper current, 20 V keeper voltage, 15 V coupling voltage, 36 eq. mA flow rate.

**Table III - Propulsion System Comparison for a 430 kg Geostationary Satellite**

| Propulsion System | $N_2H_4$ Monopropellant | $N_2H_4$ Arcjet | Xenon Ion |
|---|---|---|---|
| Total Spacecraft mass, kilograms | 430 | 430 | 430 |
| Propulsion Dry mass, kilograms | 11.1 | 14.4 | 14.9 |
| Propellant mass, kilograms | 79.9 | 43.6 | 7.6 |
| Propulsion Wet mass, kilograms | 92 | 58 | 23 |
| System Power, Watts | n/a | 2 @ 339 (ea) | 2 @ 339 (ea) |
| [a]V, m/s | 450 | 450 | 450 |
| Thruster Specific Impulse, sec | 223 | 450 | 2960 |
| Gross Engine Thrust, Newtons | 4.45 | 2 @ 0.040 (ea) | 2 @ 0.011 (ea) |
| # of thrusters | 4 | 4 | 4 |
| Cant Angle, degrees | 0 | 17 | 30 |
| Total burn time, hours | 2.7 | 670 | 2700 |
| Daily burn time, minutes | 0.04 | 11 | 45 |

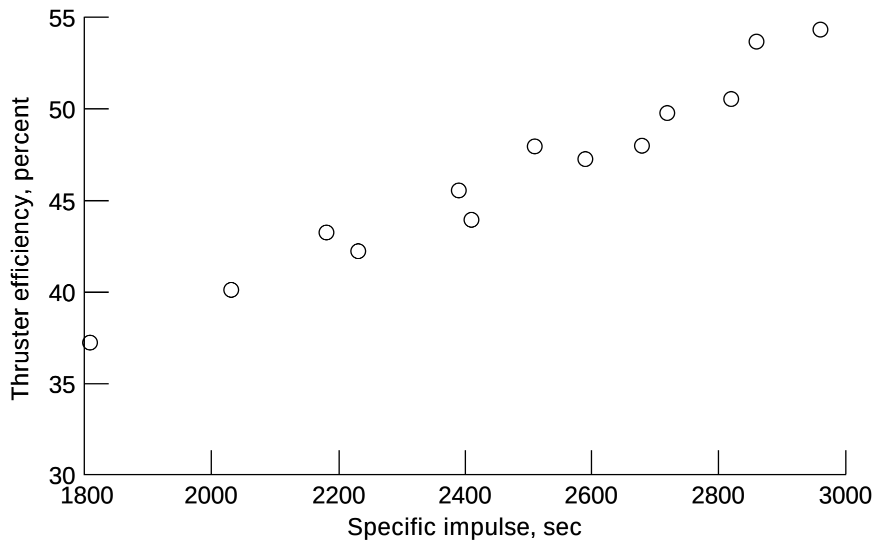


Figure 1.—Projected 8 cm thruster efficiency versus specific impulse.

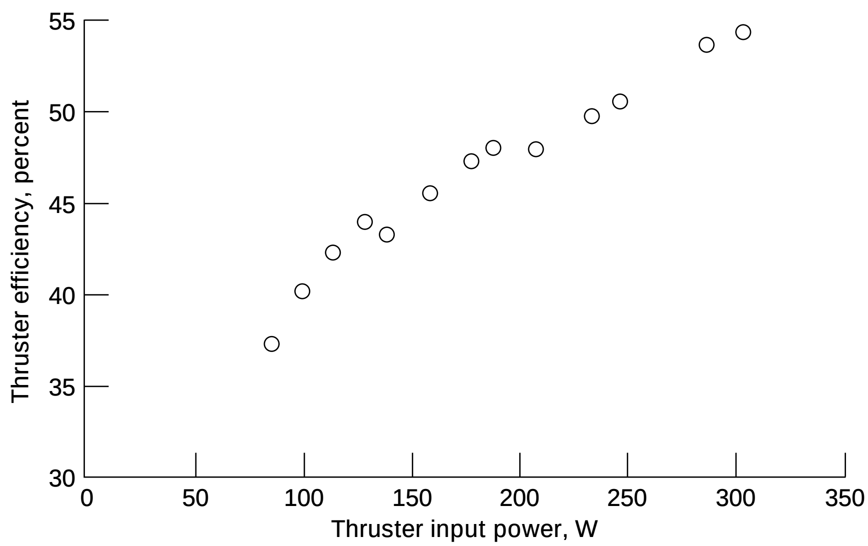


Figure 2.—Projected 8 cm thruster efficiency versus input power.

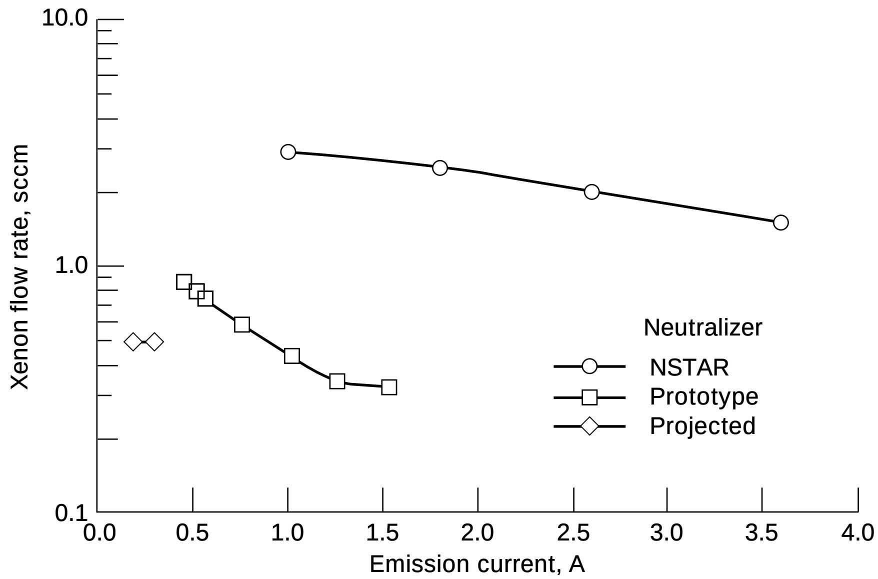


Figure 3.—Neutralizer minimum xenon flow rate versus emission current.

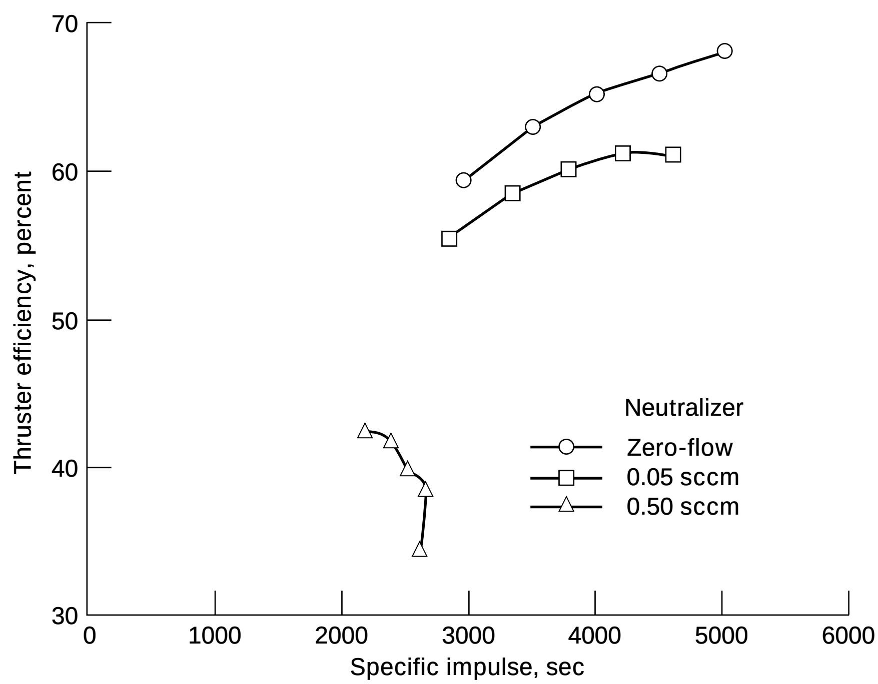


Figure 4.—Thruster efficiency versus specific impulse at 100 W input power; various neutralizers.

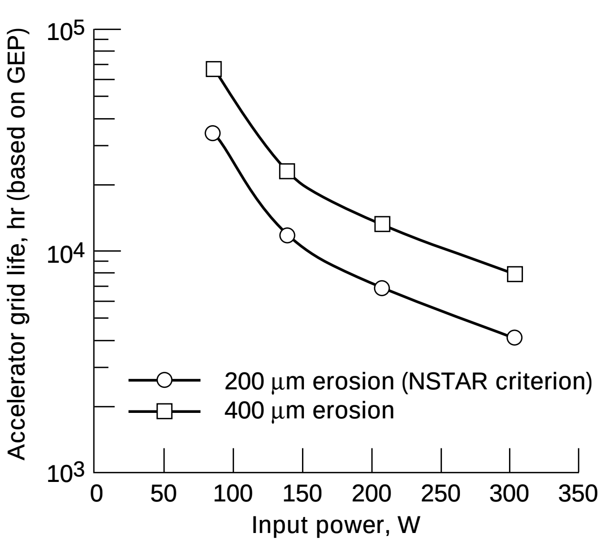


Figure 5.—Thruster life versus input power.

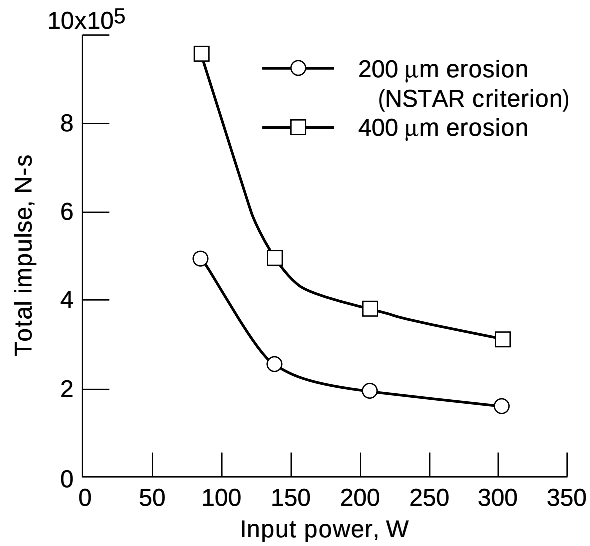


Figure 6.—Total impulse capability versus input power.

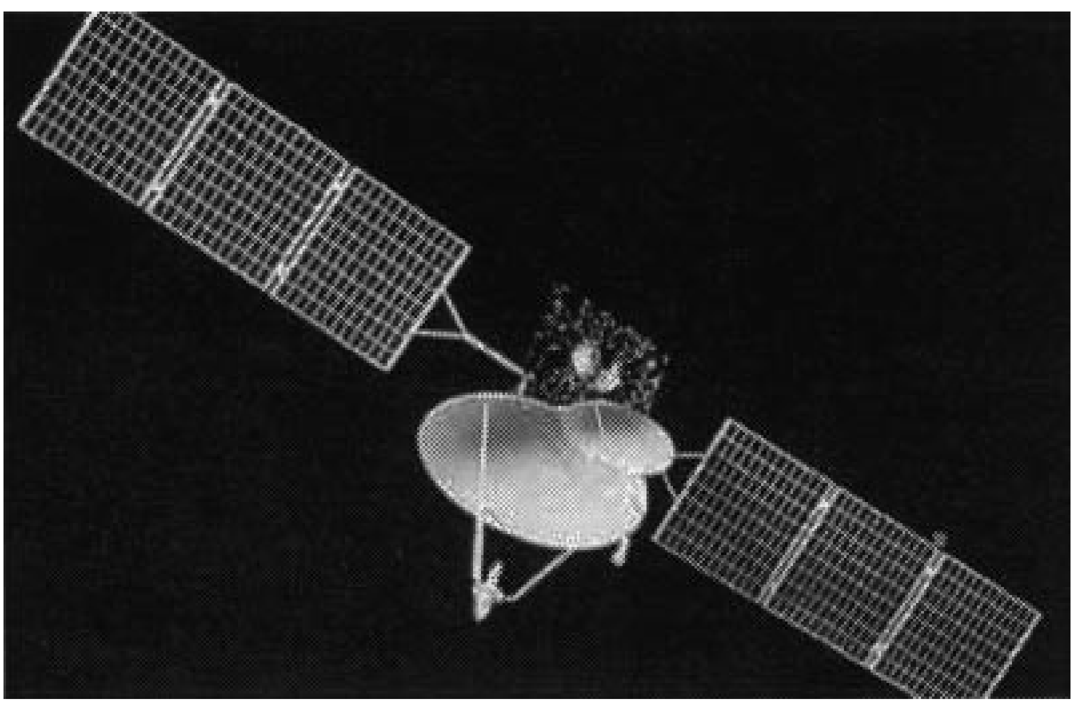

Figure 7.—IndoStar 1 spacecraft.

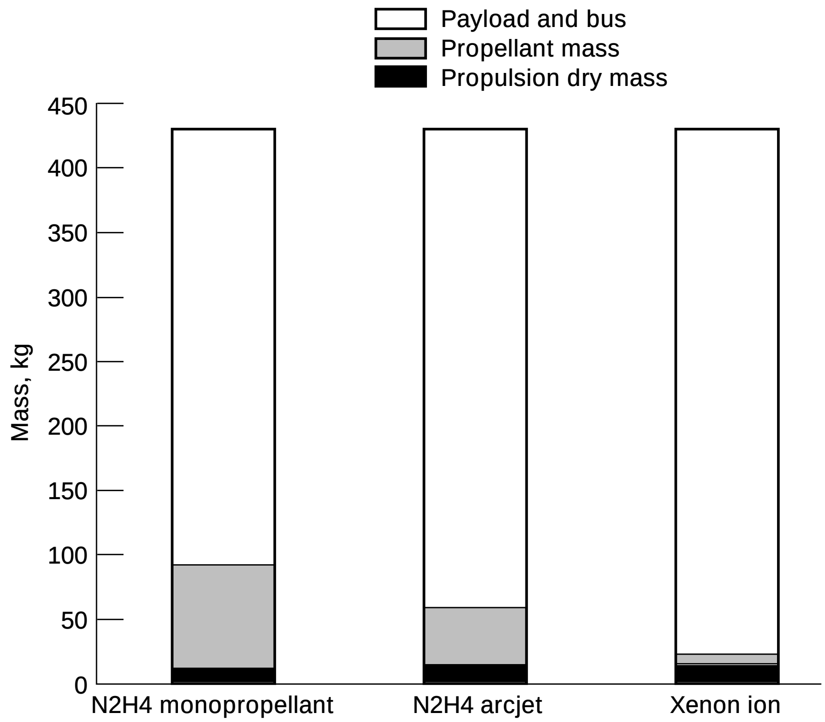


Figure 8.—Spacecraft mass comparison; geosynchronous mission.

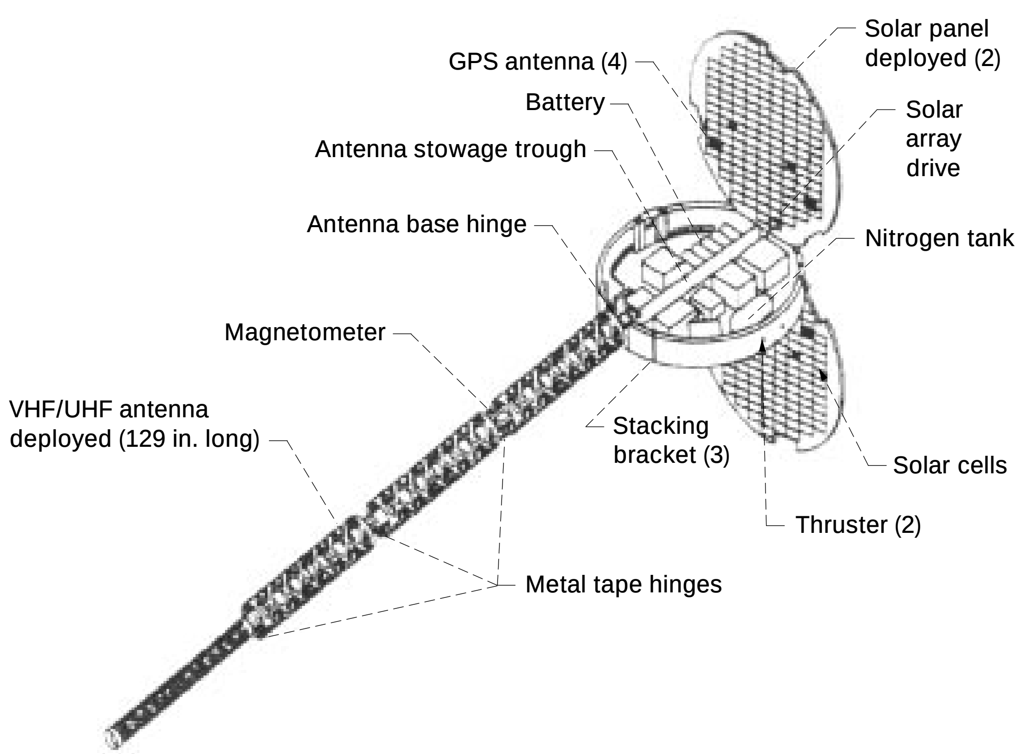


Figure 9.—Orbital Science Corporation's Microstar bus.

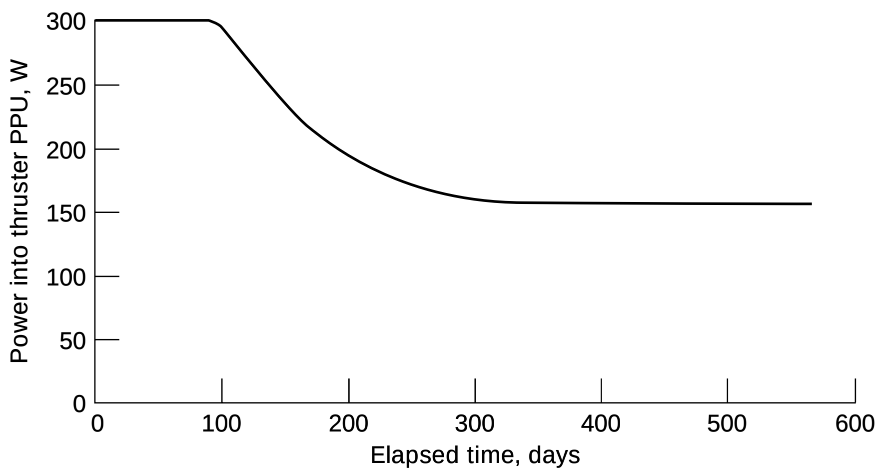


Figure 10.—Available power over mission; Magnetospheric Mapper.

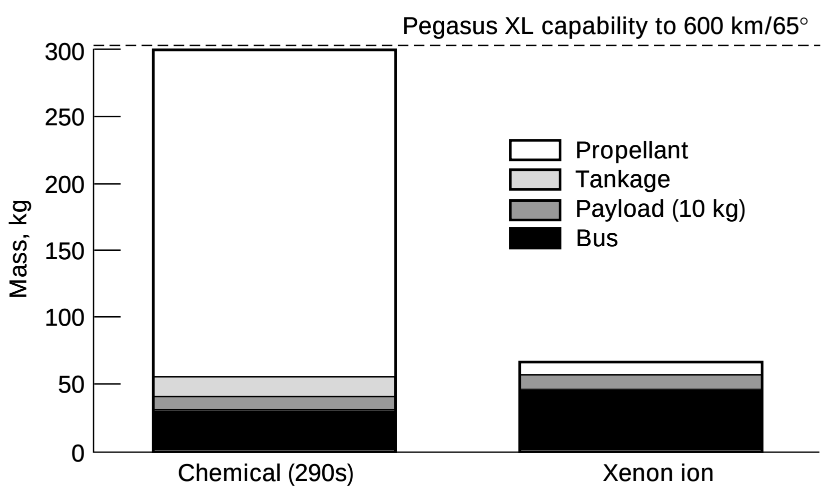


Figure 11.—Spacecraft mass comparison of magnetospheric mapping mission; chemical versus ion.

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**13. ABSTRACT *(Maximum 200 words)***

Analyses were conducted which indicate that sub kW-class ion thrusters may provide performance benefits for near-Earth space commercial and science missions. Small spacecraft applications with masses ranging from 50 to 500 kg and power levels less than 0.5 kW were considered. To demonstrate the efficacy of propulsion systems of this class, two potential missions were chosen as examples; a geosynchronous north-south station keeping application, and an Earth orbit magnetospheric mapping satellite constellation. Xenon ion propulsion system solutions using small thrusters were evaluated for these missions. A payload mass increase of more than 15% is provided by a 300-W ion system for the north-south station keeping mission. A launch vehicle reduction from four to one results from using the ion thruster for the magnetospheric mapping mission. Typical projected thruster performance over the input power envelope of 100-300 W range from approximately 40% to 54% efficiency and approximately 2000 to 3000 seconds specific impulse. Thruster technologies required to achieve the mission-required performance and lifetime are identified.

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