High Performance, Low Power Ion Propulsion System Design Concept*

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Abstract

A design concept for a low power ion propulsion system was developed. The system, which includes a thruster, power processor, and propellant feed system, will operate in the 100 Watt to 500 W range, and will deliver over 18 mN of continuous thrust. The first phase of the project was a survey of both government and commercial propulsion system users to determine their needs for low power propulsion. The results of this survey show that, in order of priority, the design drivers for these systems are low mass, high thrust/power and low cost. These results were used as guidance for the conceptual design effort. The target performance of the design has a thrust/power that is comparable to that of the NSTAR 30 cm ion propulsion system. The mass of the system (thruster and power processor) is 3 kg, which results in a specific mass (kg/kW) which is 32% lower than the NSTAR system. The system is also designed to have a low overall parts count, and to take advantage of low-cost manufacturing methods.

Introduction

Low power (<0.5 kW) ion propulsion systems have been shown to provide distinct propulsion system mass advantages for both earth-orbit and deep space missions[1]. To develop a system that would provide these system advantages, General Dynamics Space Propulsion Systems (GD-SPS) developed the design concept for a 500 W class ion propulsion system. This system design includes a thruster, power processing unit, and xenon feed system. The system design is based on the low power ion propulsion system development work that is currently on-going at NASA Glenn Research Center (NASA GRC)[2], and was funded through NASA program number NAS3-00035. The objective for both GD-SPS and NASA GRC was to evolve the NASA GRC design to a more flight-ready, low mass, manufacturable design that would meet the broadest range of propulsion system users.

To assure that the design would meet the needs of the propulsion user community, a user survey was also conducted. The results of this survey were used to develop a system specification that for an 8-cm design that maximizes thrust/power while minimizing system mass and manufacturing cost was base-lined as a design goal.

The design concept that was generated included a solid model of the thruster, a block diagram of the power processor, and a schematic of the xenon feed system.

User Survey

A user survey was created and distributed to ensure the design resulting from this effort would meet the needs of the spacecraft integrator community. The survey was sent to 47 individuals, and responses were received from over 30% of the recipients. The results of this survey, which are shown below, thus reflect the responses from a representative sample of spacecraft integrators. Slightly over half of the respondents were from government agencies, with the remaining responses coming from the spacecraft prime contractors. The results of the user survey are shown in table 1.

Table 1: Propulsion System Priorities from the User Survey

Group	Priorities			
	Low Cost	**Low Mass**	**High Th/P**	**High I_{sp}**
All Respondents	**3**	**1**	**2**	**4**
Votes/points	6/22	10/33	9/33	2/7
Commercial Respondents	**1**	**3**	**2**	
Votes/points	5/19	3/9	3/11	
Government Respondents	**4**	**1**	**2**	**3**
Votes/points	1/3	7/24	6/22	1/4
Deep Space Missions	**3**	**2**	**1**	**4**
Votes/points	5/19	7/22	7/26	2/7
Little LEO Missions	**2**	**1**	**3**	
Votes/points	4/15	6/20	3/11	
Earth Science/ Remote Sensing	**1**	**2**	**3**	
Votes/points	4/15	4/12	3/11	

The priorities were calculated using nominal group technique†. Low system mass is the highest priority of all users. Commercial users point to low cost being their main priority. High thrust/power is of sufficient importance to both Government and Commercial users that it ranks second overall as ranked using nominal group technique (very closely trailing mass).

† "Nominal Group Technique" is a method for achieving consensus in a large group that must establish a ranking among options. In this technique, each participant 'votes' for their top options (typically the top half of the options available to them), and then ranks their votes. Each vote is given points associated with their rank. In this application, I considered a 1st or 2nd place ranking a 'vote', and I gave a 1st place ranking 4 points, and a 2nd place ranking 3 points. Each option then gets a total score that includes the number of votes and the number of points. The ranking was then established by the highest combinations

The missions identified by the survey recipients as those they are considering that would require propulsion with the target performance characteristics for this program is shown in figure 1. Two missions were written in by respondents (µSat formation flying, and Space Science Missions at L1 & L2), but the most commonly identified mission by far was Deep Space, followed by Little LEO.

These results indicate that, of the three top priorities of users, it is cost that is perhaps the slightly lower priority. This could indicate that it may be justified to move towards using more expensive materials and manufacturing methods to achieve a system that provided both high thrust/power and low mass.

From these data, some conclusions can be made on the overall priorities of the propulsion-specifying community, and those are the following (in order):

1. Reducing system mass will make ion propulsion significantly more attractive for almost all missions under consideration

2. Increasing thrust/power will expand the number of missions for which ion propulsion would be considered

3. Reducing system cost is important to all missions

These priorities formed the design goals for the low power ion propulsion system.

System Overview

A conceptual design for an improved low power ion propulsion system was developed. The objectives of this effort were to develop a system that improved performance and reduced system mass compared to existing state of the art systems. The resulting design has been tailored to the meet the needs of the satellite and spacecraft integration community as identified in an extensive user survey. The conceptual design for the thruster of the low power ion propulsion system is shown in figures 2 – 4. The basic performance capabilities are as follows:

- Up to 18.8 mNewtons Thrust
- 100-500 Watts input power
- 1600-3500 seconds Thruster Isp
- 10,000 hour operational system life
- Thruster and Cathode mass: 0.95 kg
- Power Processing Unit (PPU) mass: 2.0 kg
- Xenon Feed System (XFS) mass: 3.1 kg (excluding tank)

To maximize thrust/power performance, the following strategies were employed:

- A performance analysis was conducted to determine which physical features of the thruster had the greatest impact on thrust to power so that the physical requirements of the thruster could be determined and implemented
- The cathode designs were optimized for low power consumption
- The power processor was designed for high efficiency operation to reduce power dissipation

These efforts are each described in detail in later sections of this paper.

Achieving a low cost design for the system was a major focus of this effort. To accomplish this goal, efforts were made to minimize parts count and component complexity. In addition, innovative manufacturing methods were specified where a cost benefit would be realized. The following are specific examples within this design where a low parts count was achieved:

- Optics: 36 parts, including rivets & fasteners (76% less than NSTAR)[3]
- Discharge Chamber/Plasma Screen: 11 parts, including fasteners, excluding magnets (NSTAR comparison > 30, excluding fasteners)
- Power Processor: Number of total converters reduced from six to four[4]

A minimum parts count minimizes the number of components that must be fabricated and reduces assembly time, thus reducing overall system cost. Trades are made, however, to assure that the cost of fabrication of a single more complex part is not more than the fabrication and assembly of several simple parts. Often, because of assembly time, the trade favors the single part.

Mass was minimized in two ways, through parts count minimization (as discussed above), and, where possible, through minimizing the mass and volume individual parts and assemblies. The discharge chamber/plasma screen provides the most dramatic example of mass and volume minimization. The combined mass of the discharge chamber/plasma screen assembly is less than 480 gr. (including magnets). This value is 40% less than the same subassembly in the NASA GRC 8 cm 'Functional Thruster'[5]. Similar efforts were made through-out the design.

The thruster mounting interface is integral to the plasma screen. Four mounting holes, shown in figure 3, provide this interface. All propellant lines (main, discharge cathode, and neutralizer cathode) are $^{1}/_{8}{}^{th}$ inch tubing.

The basic functional block diagram of the system is illustrated below, in figure 5. The PPU will receive 28 +/-6 VDC unregulated power from the spacecraft bus. Input-Output isolation is base-lined for proper engine operation and to conform to anticipated single point grounding schemes. The system provides twelve analog telemetry channels, which provide current and voltage data for the beam supply, accelerator supply, and both cathode and heater supplies. The PPU is controlled by four bits of input logic.

The masses of the major components of the low power ion thruster system are as shown in table 2, below.

Table 2: System Masses

Component	Mass (grams)
Thruster	830
Neutralizer Cathode	125
PPU	1999
XFS (excluding tanks)	3100
TOTAL	6054
Total (excluding XFS)	2954

Performance Trades

To determine basic the basic thruster configuration that would maximize overall thrust/power a trade study was conducted to determine the effect of increased perveance, decreased discharge losses, and increased discharge chamber & optics diameter. Each of these parameters was improved by 10% (see discussion below for details), and the impact on system performance was evaluated. These results are shown below. The combined effects of a 10% increase of all three parameters is also included.

Approach

A spreadsheet was developed that predicted ion thruster performance based on the following governing equations:

Conservation of Energy:

$$eV = ½\, m_+ v_+^2$$

Conservation of Charge:

$$J_B = n_+\, e\, v_+\, A_B$$

Child-Langmuir Law:

$$J_B = (4\,\varepsilon_o/9)\,(2\,e/m_+)^{½}\,(V_T^{3/2} / l_e^2)\,A$$

Thruster Efficiency:

$$\eta = ½\,(g\,Th\,I_{sp})/(P)$$

Where:

e = electron charge, 1.60×10^{-19} Coulombs
V = Screen Voltage, Volts
m_+ = Ion Mass, Xenon, 2.18×10^{-25} Kilograms
v_+= Velocity of ions, meters/second
J_B = Beam Current, Amps
n_+ = Number density of ions (m^{-3})
A_B = Beam current Cross Sectional Area, meters^2
ε_o = Permittivity of Space, 8.85×10^{-12} Coulombs^2/Newton Meter^2
V_T = Total Voltage (Screen Grid Voltage – Accelerator Grid Voltage), Volts
l_e = equivalent distance, meters,

$$l_e = [(l_g + t_s)2 + d_s^2/4]^{½}$$

l_g = Gap between the grids, meters

t_s = Thickness of the screen grid, meters
d_s = Diameter of the accelerator grid apertures, meters
A = Beam or Beamlet cross sectional area, meters^2
Th = Thrust, Newtons
I_{sp} = Specific Impulse, Seconds
P = Power, Watts

The performance was predicted in a spreadsheet as a function of system I_{sp} and propellant flow rate. It is assumed that the maximum thrust/power operating condition for a given configuration occurs at the point where the beam current is maximized. This point was determined by lowering the thruster I_{sp} at a given power level (between 100 & 500 Watts) until the beam current was the same as the perveance limited current for that optics configuration and total voltage. The perveance limited beam current was calculated as the current given by the Child-Langmuir law, multiplied by a factor to account for the non-uniform distribution of the beamlet current density across the face of the optics (a current density distribution as published in ref. [6] is assumed). This maximum thrust/power performance condition was determined for each of the configuration variations.

The performance model relies on a number of assumptions. The intent of the model was to determine trends and global effects, and thus neglects some effects. The assumptions and limitations are listed below:

- Maximum thrust is achieved when the beam current is equal to the perveance limited beam current (adjusted for current density profiles), and that beam current includes all of the ions produced within the chamber. This current is also limited by life requirements of the grids.
- The cosine losses due to off-axis thrusting are neglected
- Discharge power was estimated based on the NASA GRC 8 cm thruster
- The voltage on the accelerator grid was maintained at -200 for all cases, which is consistent with electron backstreaming requirements

The NASA GRC 8 cm Laboratory thruster performance [3] was used as a baseline condition which was used to validate the spreadsheet. A secondary baseline condition used the same 8 cm geometry, but varied the performance parameters to give the maximum beam current.

Varying Perveance

Perveance is defined as the space charge limited current divided by the accelerating voltage to the three-halves power[7]. The perveance, or ion extraction capability, is a function of the optics geometry (including screen grid orifice diameter, screen grid thickness, and distance between the grids) and the potential difference between the grids. Perveance can thus be increased through the appropriate modification of any of these variables. To maintain as much of the baseline optics geometry as possible (baseline defined as what was used on the NASA GRC 8 cm optics), the distance between the grids was selected as the variable that would be adjusted. The distance between the grids was reduced to a 10% increase in perveance. Voltage on the accelerator grid was fixed at -200 Volts for all cases.

Effect of Increase in Perveance

Increasing the perveance of the optics by 10% results in an increase in the achievable thrust/power of at least 7% depending on the power level. The improvement in ion extraction capability enables a higher beam current at the same screen and accel grid voltages. Thus, thrust/power is increased at the expense of I_{sp}. This effect was similar across all the power levels. The effect on thruster efficiency is negligible.

The R ratio (and performance parameter [8]) was calculated and compared against accelerator grid life data published by Patterson to ensure the optics would achieve the life requirements stipulated by the User Survey. Grid mass loss was also predicted using a model puplished by Polk, et.al[9], and compared to grid mass loss failure criteria published by Brophy, et.al[10]. In each case, the predicted cases appeared to meet these life requirements.

Varying Propellant Utilization Efficiency

Propellant Utilization efficiency was set at the nominal value (to reflect the NASA GRC Low Power Design) of 0.73, but then also set at 10% more than this, or 0.80. Increasing the voltage of the discharge cathode can increase propellant utilization efficiency. As discharge voltage is increased, two phenomena occur that limit the extent to which propellant utilization efficiency can be increased. Higher discharge voltage leads to a higher percentage of multiply-ionized ions – which increases the discharge losses due to the energy required to accomplish multiple ionization. Higher voltages also give more energy to the ions local to the discharge cathode, thus increasing the rate of cathode erosion, and reducing system life. An increase of this efficiency, however, may be achievable.

Effect of Increase in Propellant Efficiency

Increasing the propellant efficiency reduces the percentage of neutrals (as opposed to ions) that are flowing out of the thruster, thus decreasing the extent to which the ions need to be accelerated to achieve a desired average discharge velocity. Thus, the required screen/discharge chamber voltage for the same acceleration is decreased. Changing propellant efficiency increases the number of ions available for acceleration, but does not have an impact on the thruster's ion extraction capability. Increasing the propellant efficiency by 10% increases thruster efficiency by 6-8% depending on power level, but has a negligible effect on the thrust/power levels (in other words, efficiency increases come from increases in I_{sp} while maintaining thrust/power).

Varying Chamber Size

The frontal area of the optics was increased by 10%, resulting in a grid diameter of 8.5 cm. This increase assumes that that the entire discharge chamber was similarly increased in diameter. An additional assumption here is that the current density profile of a larger thruster and grid has the same basic shape but lower overall amplitude (when operating at low power). The current shape that was assumed was published by Soulas[6].

Effect of increase in Discharge Chamber/Optics Diameter

Increasing the diameter of the optics and discharge chamber serves to distribute the ion generation and beam current across a larger area. This reduces the ion production cost (though the model does not include this effect), and enables a larger beam current at the same ion extraction capability as smaller diameter optics. A 10% increase in optics diameter results in an increase in thrust/power of approximately 2.5%, due to the increased ion extraction capability of these optics. The increase in size has a negligible effect on thruster efficiency. It should be noted that this increase in performance would come at an increase in system mass of as much as 10% (assuming all components scale linearly). Given that both low mass and high thrust/power were priorities identified in the user survey, this trade may weigh towards reducing mass, however, this strategy could be employed if maximized thrust/power were imperative for a specific application.

Performance Conclusions

The great impact of the high perveance optics on the system thrust/power lead to the decision to incorporate high perveance optics into the low power ion thruster design. The impact of this decision is evident when the predicted performance of the system is compared to the published NASA GRC 8-cm ion thruster performance. This comparison is shown in figure 6. In each case, for the same specific impulse, the thrust/power is increased by at least 7%. To be consistent with the published data, these performance numbers do not include the efficiency of the power processor. Thruster specific impulse also hasen't been corrected for neutralizer flow in each case.

Testing of the unit will reveal the feasibility of increasing the propellant utilization efficiency to achieve further performance enhancements, however, since this is still uncertain, the predicted performance assumes a nominal 0.73 for this value. Options to increase the size of the discharge chamber and optics were deemed to add too much mass, and thus were rejected. Thus, this performance prediction assumes an 8-cm thruster.

Future work will focus on reducing discharge losses in the thruster to further improve thrust/power. This parameter was not varied in this study. Recent investigations [11] indicated that improvements in the configuration of the magnet circuit can reduce discharge losses through improved electron containment. These improvements would reduce system power, and thus have a first order impact on improving system efficiency by improving thrust/power. Incorporating the modifications described reference 11, however, substantially increase design & manufacturing complexity, and thus, this work will be incorporated into future designs.

Thruster Design

Optics

To achieve the desired increase in thrust/power of the ion propulsion system the optics must be designed to provide a high ion extraction capability, or, high perveance. The perveance, is a function of the optics geometry (including screen grid orifice diameter, screen grid thickness, and distance between the grids) and the potential difference between the grids. Perveance can thus be increased through the appropriate modification of any of these variables. The NASA GRC 8 cm optics were selected as a baseline The distance between the grids was reduced to provide a 10% increase in perveance.

Perfect alignment of the grid apertures is also critical to the beam extraction capability as well as overall thruster performance. Misaligned grids can cause an increase in current to the accel grid (increasing discharge losses), higher erosion rates of the accel grid and can change the thrust vector of the resulting beam. The optics have been designed to assure alignment. Tight tolarance fit and keying features assure that, once the optics are assembled, the grid apertures are properly aligned. This approach is based on the alignment approach used for the GD-SPS 30 cm optics[12]. Similar tooling and assembly methods are also planned.

Discharge Chamber & Plasma Screen

The discharge chamber provides the ion-producing environment for the ion thruster. A circuit of magnets around the discharge chamber produces a static magnetic field that contains the electrons within the discharge chamber to improve the residence time of the electrons, and thus improve propellant ionization. The plasma screen shields the high voltage of the discharge chamber and the optics from space plasma. The plasma screen resides at spacecraft ground potential, thus the discharge chamber and plasma screen must be electrically isolated from each other. These components together also form the primary structure for the thruster. The optics and cathodes are both mounted on the discharge chamber/plasma screen, and the mounting of the thruster itself is a part of the plasma screen/discharge chamber assembly. Thus, this assembly must fulfill the following objectives:

- Provide a structure for the entire thruster
- Provide electrical isolation between the discharge chamber, plasma screen, optics, and cathodes
- House the magnets that create the magnetic field internal to the discharge chamber

The conceptual design achieves all these objectives. To reduce mass, the plasma screen/discharge chamber assembly was made as small as possible. In addition, many components and subassemblies serve multiple functions. This enabled the very large reduction in parts count for in the discharge chamber/plasma screen as compared to the NSTAR design.

Discharge & Neutralizer Cathode

The to meet design objectives set by the user survey, the cathode design has been optimized for low mass, low power (such that overall system thrust/power is optimized) and low cost. To reduce recurring system cost, the design of the discharge and neutralizer cathodes are as similar as possible, differing only in the mounting mechanisms. The design for these cathodes is based on four existing cathode designs: an improved low flow cathode developed by Domonkos[13] called the SK.012, the Long Life cathode developed at NASA GRC[14], the NASA GRC low flow cathode which developed for use with the NASA GRC 8 cm ion thruster[15], and the GD-SPS 'designed-for-manufacturing' (DFM) cathode[16]. The design approach for the cathodes was to evaluate the design features of each of these cathodes, and decide which option was most appropriate for this application, given the design objectives above. For the most part, the cathode design upstream of the emitter is very similar to the GD-SPS DFM cathode. This enables a low cost, low mass design. Specific trades for reducing cathode power consumption are also addressed in the next paragraph.

The cathode orifice size and shape have a first-order impact on the power consumed by the cathode. This effect was shown both by Domonkos[17], and by Patterson and Domonkos[18]. Both investigations showed reducing orifice diameter reduced power required by the cathode to produce a given discharge current. Reducing the length of the orifice diameter had a similar effect. Thus, to reduce power consumption, both of these dimensions are minimized in the low power cathode designs. The minimum orifice diameter is defined by the Kaufman criterion, and is a function of the maximum anticipated discharge current. Kaufman[19] showed that orifices smaller than this limit would erode over tens of hours of performance to the orifice diameter defined by this criterion. Based on this criterion, the cathode orifice used in the NASA GRC low flow cathode is a good baseline for this conceptual design.

The cathode designs are shown in figures 7 and 8. The design of the low power cathode is based on a design-for-manufacture approach that results in a low overall parts count, minimizes the number of threaded fasteners, and protects the emitter from the risk of poisoning during manufacturing.

Power Processor Design

The design philosophy for the power processor was based on the results of the user survey. These results provided the following priorities for the design:

- High system efficiency (to enable high thrust/power)
- Low system mass
- Low recurring cost
- High reliability (for long life)
- Low EMI (all missions)
- High radiation tolerance (for deep space missions)

Technology trades were conducted on a number of topologies for both the beam supply as well as the entire power processor to optimize a design with the above priorities. Because most of the power for the system is processed by the beam supply, that supply became the primary focus of the design effort. Subsequent efforts focused on reducing system mass by combining the functions of some of the converters.

Several basic converter topologies were evaluated for the beam supply. All of these topologies were evaluated based on the design priorities listed above. A low mass, low cost, low EMI current-fed topology was selected for the beam supply converter. The topology has a low parts count and requires only a very simple power transformer (thus improving reliability). The power transformer also provides isolation between the spacecraft bus and thruster, thus reducing conducted EMI.

To reduce the mass of the power processor, the cathode supplies are combined with their respective heaters supplies. The converters demand similar voltage, current, and power levels, and the heaters are turned off once each cathode discharge is initiated. The cathode and heater converters both need to operate in a controlled current mode, thus, the designs for these converters would be similar or the same even if they weren't combined. The function of the converter can be alternated with a single switch in each converter. This option results in four total converters for the power processor. The configuration for this option is shown in the system block diagram in figure 5.

Xenon Feed System

A Xenon Feed System (XFS) is required for every ion thruster to store the propellant, isolate the propellant when the thruster is not in use, accurately control the propellant to the thruster and its Hollow Cathode Assemblies (HCAs), and maintain and/or improve the purity of the delivered propellant. Several technologies and types of hardware are available to perform these functions, and the key to optimizing their selection is understanding the key requirements of the thruster and its system. Some of these are summarized in table 3.

Table 3: Xenon Feed System Requirements

Function	Requirement
Flowrates	Main Flow: Throttled from ~ 0.3 to 0.7 mg/s Cathodes: ~ 0.03 to 0.05 mg/s
Flowrate Accuracy (Goal)	± 2% for overall flow uncertainty
Throttle Rate	No rapid throttling required
Xenon Purity	Similar to requirements for NASA GRC Long Life Cathode
System Redundancy	Typical of low-cost, long-duration missions

Many feed system technologies are available to meet the needs of the ion thruster system, and several were considered for this study. The key technologies studied in detail were Flow Control Devices (FCDs), fixed and variable pressure regulators, and pressure transducers. The schematic is shown in the system block diagram, figure 5. System mass and complexity are minimized by the featured variable pressure regulator and thermal throttle FCDs, though considerable system volume is still required by the need to weld all of the components together. Careful handling can result in adequate system purity. A variable pressure regulator allows convenience in any required system purging and/or purity sampling.

Technologies not yet available which would improve the proposed XFS include a reliable flight Xe flowmeter for flows of this magnitude, smaller components and/or system manifolds which include many functions within one compact package (e.g. the Vacco Chemically-Etched Miniature Systems[20]) which minimize the space required for tube welding within the XFS.

Future Work

The design concepts that were developed under this effort are planned to be used for the development of future low power ion propulsion systems. These concepts will also be leveraged for the development of high power ion propulsion systems.

Conclusion

The design concept for a 500 W-class ion propulsion system was developed. This design showed substantial mass reductions from the existing lab designs, as well as a greater power density than the NSTAR thruster. To maximize the number of missions for which this thruster would be applicable, the thrust/power capability was maximized by improving the perveance of the optics. The mass reducing features that were developed in this effort will be implemented in this and future designs.

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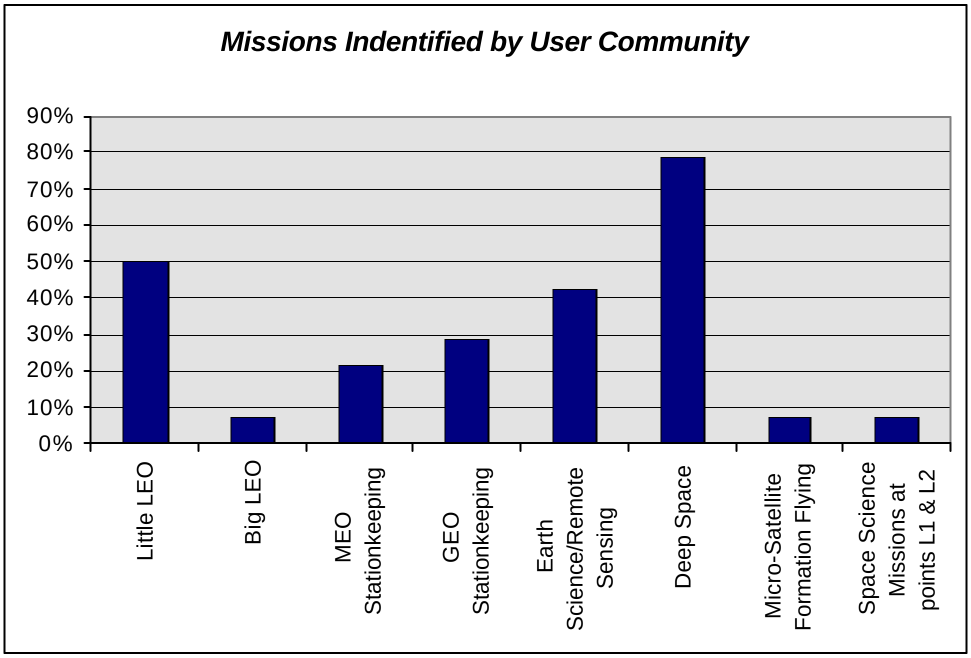

Figure 1, Missions Identified by the User Community in the User Survey

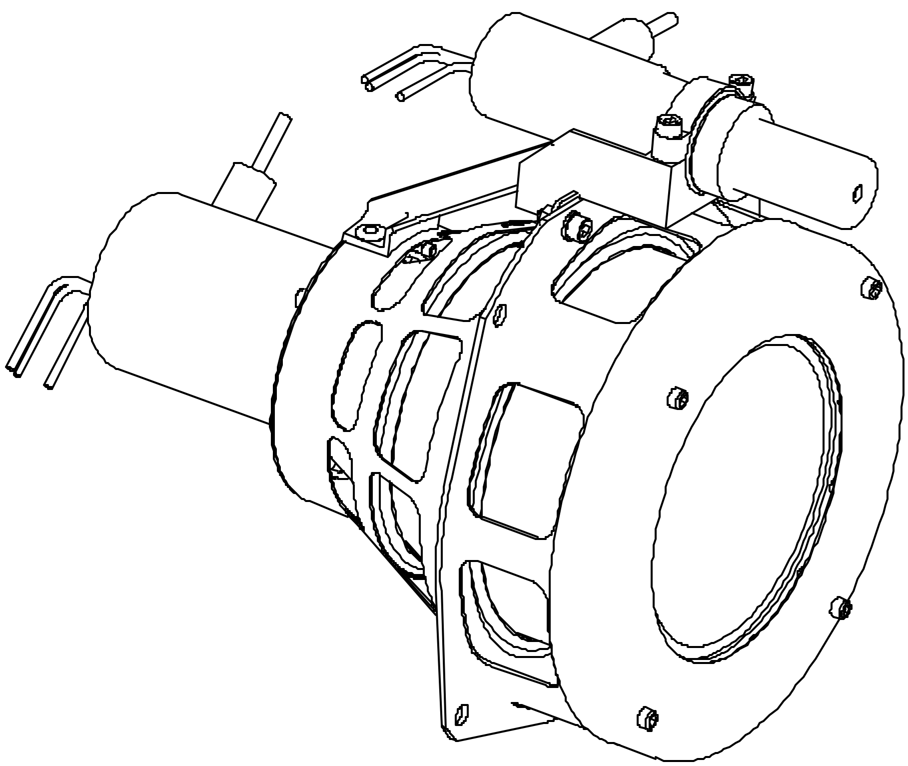

Figure 2, The GD-SPS Low Power Ion Thruster

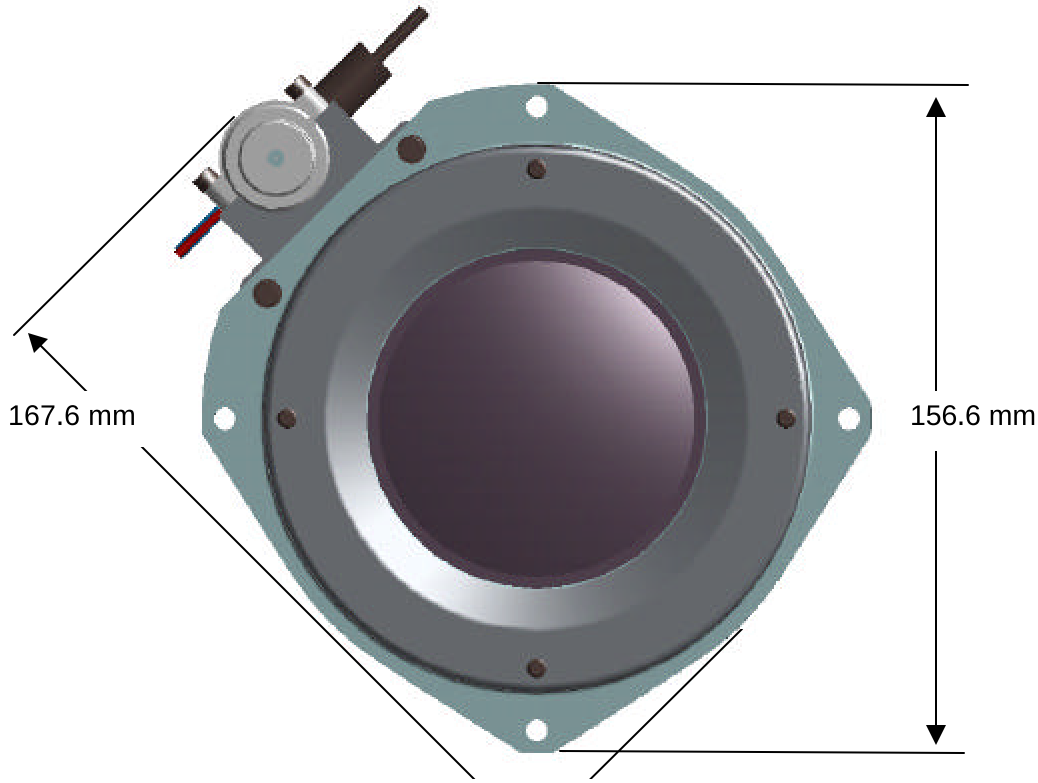

Figure 3, Low Power Ion Thruster, front view

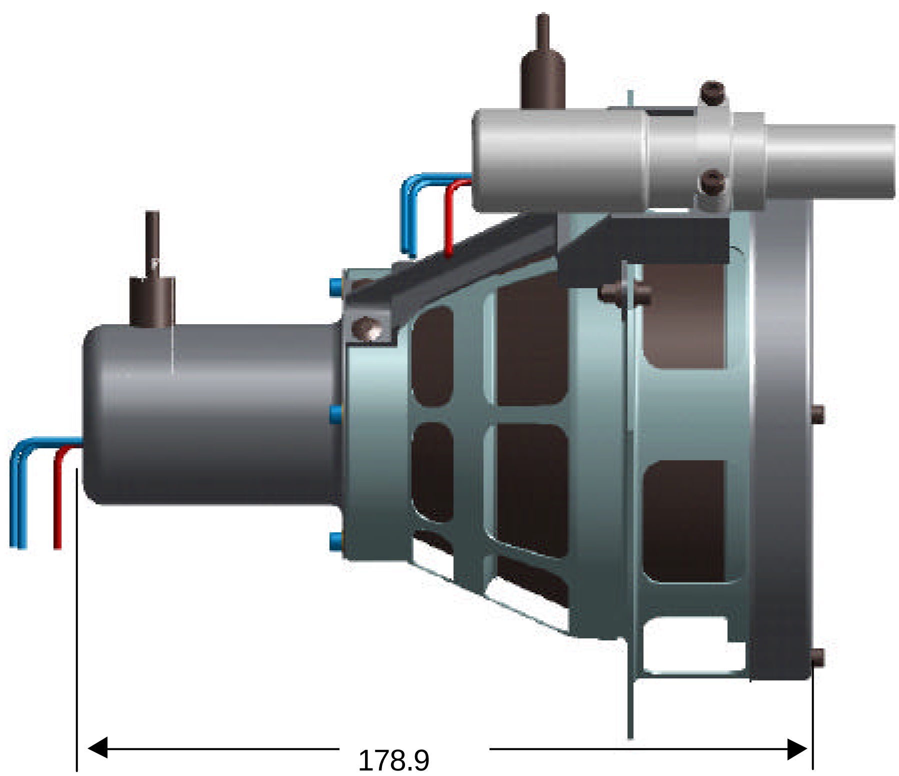

Figure 4, Low Power Ion Thruster, Side View

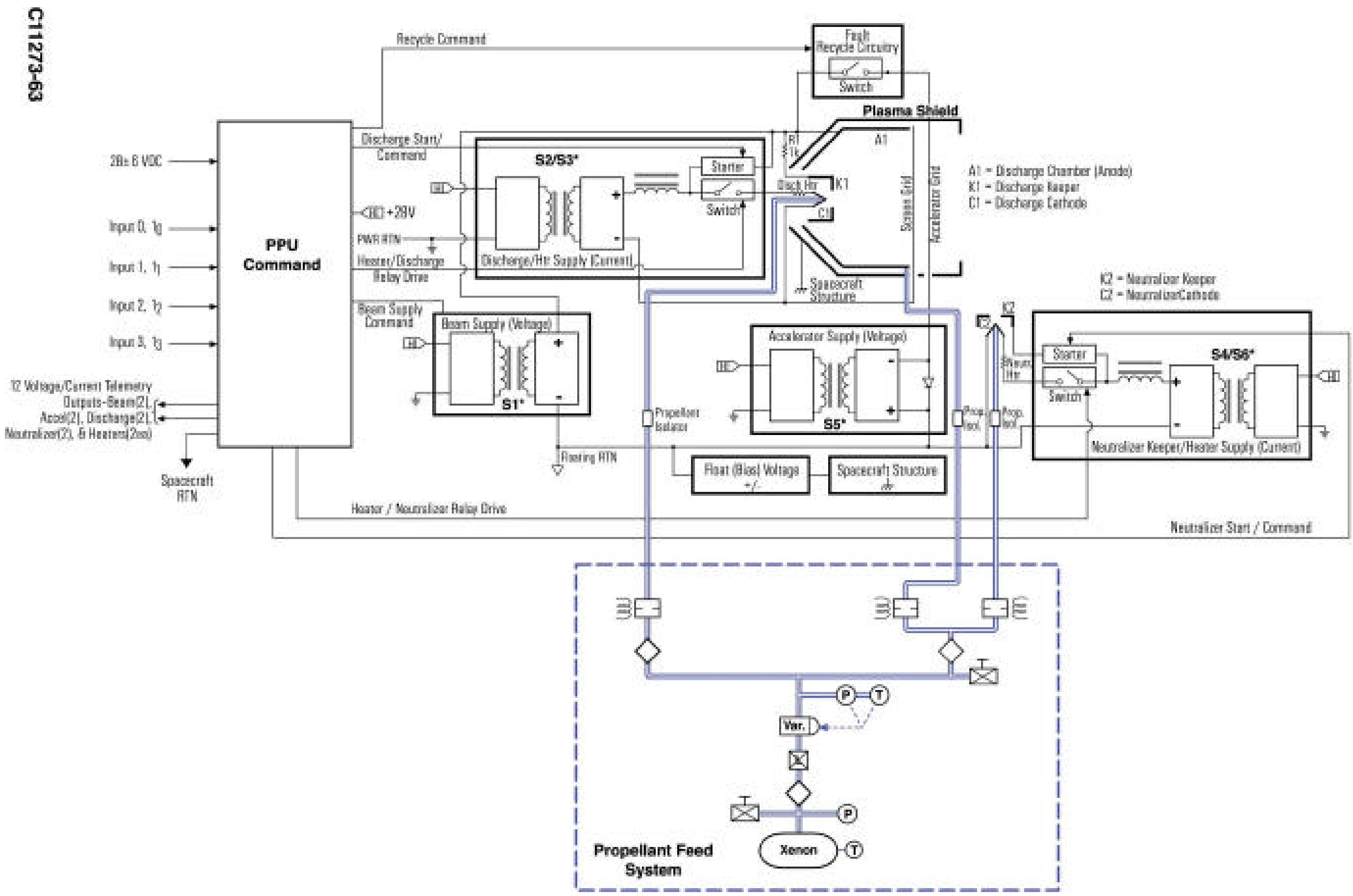

Figure 5, Low Power Ion Propulsion System Block Diagram

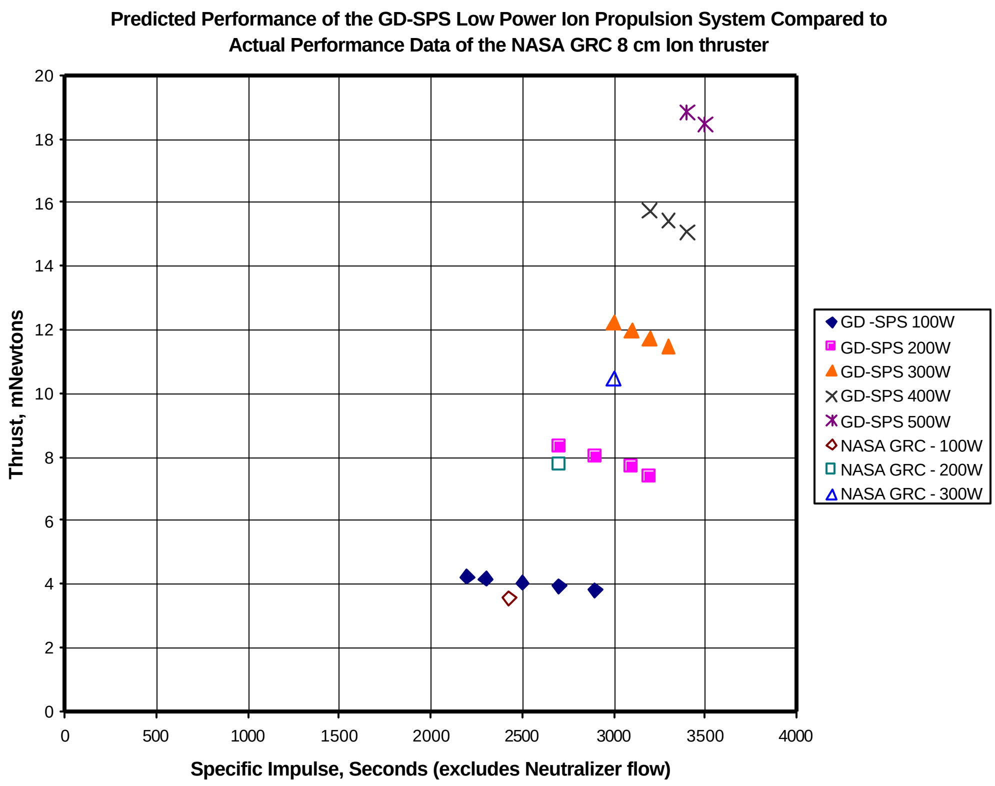

Figure 6, Predicted Low Power Ion Propulsion System Performance

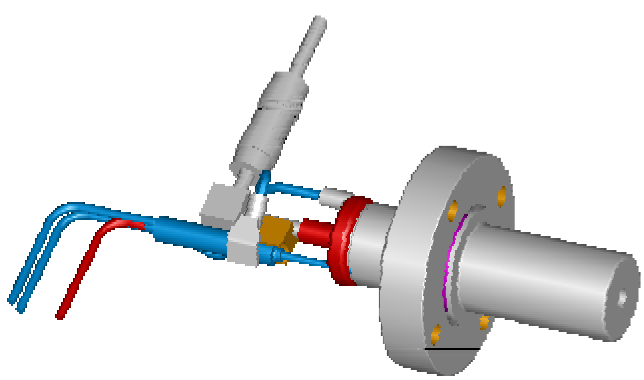

Figure 7, Discharge Cathode

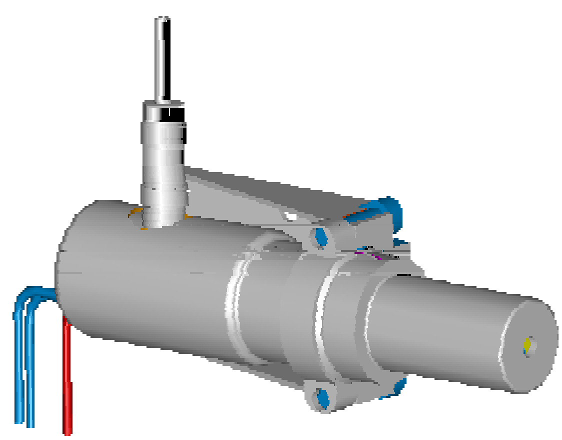

Figure 8, Neutralizer Cathode