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Abstract

Pulsed plasma thrusters are currently planned on two small satellite missions and proposed for a third. In these missions, the pulsed plasma thruster's unique characteristics will be used variously to provide propulsive attitude control, orbit raising, translation, and precision positioning. Pulsed plasma thrusters are attractive for small satellite applications because they are essentially stand alone devices which eliminate the need for toxic and/or distributed propellant systems. Pulsed plasma thrusters also operate at low power and over a wide power range without loss of performance. As part of the technical development required for the noted missions, an experimental program to optimize performance with respect to electrode configuration was undertaken. One of the planned missions will use pulsed plasma thrusters for orbit raising requiring relatively high thrust and previously tested configurations did not provide this. Also, higher capacitor energies were tested than previously tried for this mission. Multiple configurations were tested and a final configuration was selected for flight hardware development. This paper describes the results of the electrode optimization in detail.

Introduction

Pulsed plasma thrusters (PPTs) are low thrust electric propulsion devices which can operate at high specific impulse across a broad power range (1-200W). PPTs typically use a fluorocarbon polymer as a solid propellant, and applications for these thrusters on satellites range from precision positioning to orbit raising. In fact, PPTs are scheduled on three upcoming space missions. In 1999, a PPT will fly on Earth Observer 1 (EO-1) to demonstrate propulsive attitude control in NASA's New Millennium program. In 2000, a PPT will be used to demonstrate small satellite orbit raising on the Air Force Mightysat II.1 spacecraft.[1] Finally, PPT technology has been baselined for both spacecraft translation and precision positioning (formation flying) in the proposed New Millennium Deep Space 3 mission.

Compared to conventional propulsion systems, the PPT is attractive in that this technology eliminates the need for distributed and/or toxic propellant systems. PPTs also operate at low power levels and its pulsed nature permits operation over a relatively broad power range without loss of performance. First developed during the 1970's and flown early into the 1980's,[2,3,4] interest in the PPT waned until NASA's On-Board Propulsion (OBP) program began reevaluating the technology for small satellite applications approximately three years ago. Early technical and market assessments led to renewed interest for small satellites, and new generations of PPTs for both near and far term missions are planned.

Most PPTs are ablative devices which accelerate propellant through electromagnetic forces. Simply, the heart of the PPT consists of a pair of electrodes between which a bar of solid propellant is fed. Though a variety of propellants can be used, a fluorocarbon polymer is most typically employed. The electrodes are connected to opposing sides of a high voltage capacitor. Discharge is initiated via a spark plug located in the cathode electrode. As the discharge occurs, propellant is ablated, ionized, and then accelerated via the Lorenz forces created by the discharge. The PPT is typically charged up over a period of milliseconds by a low voltage supply and the energy is released in 1-10 microseconds. Except for the spring which feeds the fuel bar, a typical PPT has no moving parts.

The effect of varying propellant and electrode geometry has been studied in the past. Vondra and Thomassen investigated the variation in impulse bit and mass loss per pulse as a function of fuel face configuration and electrode flaring.[5] Yaun-Zhu showed that increasing the electrode gap increased performance, particularly at higher energy levels, to some limit.[6] A similar study was performed by Polumbo and Guman in which the authors varied both the gap between the electrodes and the included angle between them.[7] The Polumbo and Guman study also showed that angle and gap can be optimized to maximize performance.

More recently, the OBP program has been working toward the development of a new generation of PPTs.[8] As part of this effort, a breadboard PPT was developed by the Primex Aerospace Company (PAC) and this has been tested extensively both at PAC and at the Lewis Research Center (LeRC). As part of this testing, an extensive series of experiments was performed at PAC to optimize performance. In that experimental study,[9] different electrode gaps, lengths, flares, and capacitor energies were tested. The results were to be used to determine configurations for the two near term missions mentioned earlier. Of the two, the planned orbital insertion is the most technically challenging and that study focused on optimizing a configuration for this application. The initial capacitor energy levels studied were 22 and 43 J. Various combinations of electrode spaces of 2.54, 3.81, and 4.08 cm, lengths of 2.54, 3.81, and 4.08 cm, and flares 0° and 20° were tested. It was found that the configuration with 2.54 cm electrodes, a 3.81 cm electrode gap, and no flare provided the best performance with respect to Mightysat mission requirements, i.e. the highest impulse bit and moderately high specific impulse. These results, however, were marginal for the Mightysat insertion orbit at solar maximum and further tests were deemed necessary to to finalize the Mightysat configuration. The breadboard PPT was moved to LeRC where previously untried configurations with potential for performance improvement, along with the most promising configuration from the previous study, were tested. In addition to testing at 43 J, testing at a higher capacitor energy level was included. One configuration was also tested at lower energy levels to demonstrate the PPT could function over a range of energies. This capability was critical for precision positioning of the EO-1 mission. The results of these tests are summarized in this paper.

In addition to the parametric performance tests, a series of experiments were performed to assure that short term test results were not biased due to thermal effects. These effects had been observed recently[10] with other hardware where longer PPT operation resulted in higher propellant temperatures and thus greater mass loss impacting performance. The results of the current study are described and indicate that thermal impacts did not affect the validity of short term tests with the PPT used in this study.

Apparatus

Breadboard PPT

The Breadboard (BB) PPT tested in this study was designed and built by Primex Aerospace Company as part of the On-Board Propulsion PPT development effort. The BB PPT has a modular design (see Figure 1) which allows easy exchange of components for parametric study. As shown in the figure, the BB PPT has two sets of electrode pairs which are parallel to each other and fire in the same direction. The pair of cathode electrodes are attached to a single stripline, as are the anode electrodes. By removing the electrode pairs, and replacing them with electrode pairs of different dimensions, the gap between the electrodes, the electrode lengths, and electrode flare can be changed. The electrodes pairs are surrounded by a horn assembly, and a spark plug is located in each cathode electrode. Energy storage is provided by a 33 μF capacitor located below the horn assembly. The electronics for the spark ignition and the capacitor charging circuit are located behind the capacitor. Fuel bars of fluorocarbon propellant are fed between the electrode pairs by springs held in position by a retaining shoulders built into the electrodes. The fuel bars are removable so that they can be weighed to determine mass loss.

Thrust Measurement

The transient nature of a PPT and the low thrust to weight ratio required the use of a special torsional thrust stand to determine the thrust and impulse bit of a PPT. A precision balance designed and fabricated under the OBP program was used for all thrust measurements taken in the course of this testing. A detailed description of the thrust stand can be found in references 11 and 12, and a photograph of the thrust stand with the PPT installed is shown in Figure 3. Briefly, the stand resembles a swinging arm that rotates around a vertical axis. The thruster is mounted at a fixed radial distance from the axis on the end of the arm with the thrust axis is tangent to the arc formed by motion of the arm. A torsional restoring force is used to resist the motion in the direction of the thrust. Using the principle of angular momentum, the thrust and impulse bit can be determined as a function of the thrust stand deflection, spring stiffness, and natural frequency. The natural frequency is determined by disturbing the thrust stand with the damping circuit deactivated. Since the thrust stand rotates on a frictionless flexure pivot, the disturbed thrust stand sets up a harmonic oscillation from which the natural frequency can be measured. The spring stiffness of the thrust stand can be calculated from the resultant displacement of a known force applied to the stand. When the PPT is operated in single pulse mode the impulse bit can be determined by measuring the displacement of the thrust stand. A single pulse starts the thrust stand from rest and deflects it until the restoring force brings it to rest again and swings it in the other direction, until it overshoots the neutral position. The displacement is measured as half of the peak-to-peak difference of the thrust stand deflection. When the PPT is operated in repetitive mode the thrust can be determined by measuring the average deflection of the critically damped thrust stand over a series of pulses. Since the operating frequency of the PPT is many times greater than the frequency response of the thrust stand it is possible to do this. The restoring force of the stand increases linearly with the displacement and eventually balances with the average thrust of the PPT. The average displacement is compared to the displacements generated by calibrated free hanging weights. From these measurements, the thrust can be calculated.

Vacuum Chambers

Most PPT testing described herein was performed in a medium sized vacuum facility in which ambient pressures were generally in the low 10^{-5} torr range. The PPT was positioned horizontally in the tank to fire along the long axis of the tank. The PPT was fired at a frequency of less than 1 Hz during thrust measurements to eliminate the need for capacitor cooling. The lower energy level tests were conducted at 2 Hz, since overheating the capacitor was unlikely. This also decreased the uncertainties of the thrust measurements at the lower energies. The remaining mass loss testing was performed in a different vacuum facility due to availability. That chamber was a vertical facility that is maintains a vacuum on the low 10^{-5} torr range via cryopumping. Thrust measurements were not feasible in this upright facility. The tests conducted in the upright facility were performed at less than 1 Hz to again prevent capacitor overheating.

Procedure

Experiment

The major objective of this study was to identify a configuration which would provide the performance required by the Mightsat mission, which required performance the testing of different fuel bar face shapes, electrode flare configurations, energy levels and electrode lengths. The different configurations are summarized in Table 1. Previous tests indicated that a 3.81 cm electrode gap was optimal, so this parameter was held constant in this study. All electrodes were 2.54 cm wide. The two fuel face configurations tested are shown in Figure 2. One fuel bar had a flat face perpendicular to the electrode faces. The other had a notched face with a 120° inclusive angle. The apex of the notch was perpendicular to the electrode faces. The perpendicular distance from the center of the spark plug to each face of the notch was the same as the distance of the flat face fuel bar to the center of the spark plug. Both types of fuel bars were approximately 7.62 cm in length. Another condition which was varied was the electrode flare configuration. In one configuration, the electrode faces were parallel to each other, called a 0° or no flare angle. In the other a 20 degree inclusive flare angle was set between the faces. In the main body of the testing pulse energy levels of 43 and 54 J were used. The 43 J testing in one case repeated previous data to provide experimental confidence. The 54 J level was chosen based on spacecraft considerations and the need for higher thrust levels. Other tests at lower energy levels were taken to expand the knowledge base and are included here for completeness. Lower energy levels tested were 5, 10 and 15 J, for one electrode and fuel face configuration, that measured thrust and impulse bit only. These energy levels where tested to demonstrate the PPT could provide thrust at low energy levels. Finally different electrode lengths were tested, where the length is defined as the distance from fuel face to electrode tip. Previous tests at Primex has shown that 3.81 cm electrode gap with the 2.54 cm long electrodes, and no flare showed the most promise for performance increase, so that configuration was repeated.

An additional test was performed to determine if the mass loss per pulse varied with period of operation. For each electrode configuration tested for performance at a specific capacitor energy level, the BB PPT was pulsed a total of approximately 2000 times. Thus, one electrode configuration was chosen to run two tests of different periods of operation at an energy level of 43 J. One test ran for approximately 2000 pulses and the second for 10,000 pulses. The configuration used was the 3.81 cm long electrodes, 3.81 cm gap, 20° flare, and flat fuel face. The mass loss was measured after each test.

Mass Loss Measurements

The typical fuel bar weighed approximately 200g. All mass loss measurements were made on a precision balance capable of weighing up to 1000g. The balance is accurate to +/-0.001g. Prior to weighing each fuel bar, the balance was checked against known calibrated weights. All fuel bars were handled with gloves to avoid contamination.

Performance Calculations

The performance equations for a PPT have been well defined in previous references.[13,14] The impulse bit and thrust are calculated from measurements made on the thrust stand and are defined above. From these parameters the specific impulse, efficiency and power of the PPT can be calculated. The specific impulse, I_{sp}, is defined by the equation:

$$I_{sp} = \frac{T/f}{m\,g}$$

where T is the thrust, m is the mass ablated per pulse, f is the frequency at which the PPT discharges, and g is the gravitational constant. The thruster efficiency, η, is then defined as:

$$\eta = \frac{(T/f)^2}{2mE}$$

where E is the capacitor discharge energy. The power, P, of the PPT is defined as:

$$P = E f.$$

Results and Discussion

As noted in Table 2, several electrode and fuel face configurations were evaluated in a parametric study designed to identify a final configuration which would meet Mightysat requirements. A range of capacitor energies were also evaluated. The results of the electrode configurations showed that increasing the length of the electrodes from 2.54 cm to 3.81 cm electrodes increased the efficiency but not the thrust to power ratio. However, when the electrodes were flared at 20°, both the efficiency and thrust to power ratio improved. Notching the fuel face increased the thrust to power ratio but decreased the efficiency. Increasing the capacitor energy level was found to increase the efficiencies in all cases. Also, the data showed that the impulse bit increased nearly linearly over the range of energies tested, particularly for the 0° flare case. Results are shown in Figure 4, with the highest impulse bits at the 54 J energy level. While the overall results of this study did not show one configuration which is clearly superior in all performance categories (e.g. efficiency, impulse bit, thrust to power, etc.), once plotted with respect to a mission analysis, the selection of a configuration became more apparent.

The impact of the various configurations on the Mightysat mission are shown in Figure 5. In this figure, plots of total specific impulse for a given amount of fuel and an average orbit thrust at various insertion altitudes (nautical miles), are reproduced from Reference 1. The figure also shows various orbit raising trip times, with trip times decreasing to the right. The projected performance of the various configurations examined in this study are located in the upper left hand corner of the figure. Though it is difficult to view on the figure the 3.81 cm long electrodes with the 20° flare at 43 J coincides with the notch fuel face at 54 J. From this it can be seen that the notched fuel face at 43 J would provide the shortest orbit trip time. The notched face feed system, however also required the most rigorous design and no long term testing has been performed to demonstrate unequivocally that the notched face will maintain its shaped (and performance) over the millions of pulses required for small satellite insertion. Thus it was desirable to avoid this configuration until these concerns could be addressed. The 2.81 cm electrodes with the 20° flare produced only a slightly longer trip time and almost as high an average thrust at both 43 and 54 J. This configuration was then deemed the "best". On the component level, higher energies put a greater strain on the capacitor and thus are expected to reduce the maximum mission life. Since both energy levels for the 20° flare configuration met the total impulse requirement in theory, in deference to the life of the capacitor, the lower energy of 43 J was selected for the mission. The final selection for Mightysat was the 3.81 cm long electrodes, 3.81cm gap, and 20° flare with a flat fuel face at 43 J. In an effort to maintain uniformity between missions, the same electrode configuration would have been chosen for EO1. However, dimensional restrictions placed on the PPT by the spacecraft required the use of electrodes with no flare though the same length and gap.

The PPT was also tested at lower energy levels in the 3.81 cm x 3.81 x 0° flare configuration. The PPT was successfully discharged at the 5, 10, and 15 J energy levels. The resultant thrust to power and impulse bit measurements are recorded in Table 2. However, mass measurements were not made in the interests of time. The purpose here was to prove discharge capability across a wide range of capacitor energies, which was the case.

Measurement uncertainties were determined for the performance parameters using standard propagation techniques. The calculated performance parameter of power, specific impulse, impulse bit, and efficiency were calculated from the uncertainties of the measured quantities. The largest contributing factor was the thrust measurement, followed by the uncertainty in the mass loss measurements. The uncertainties are presented in Table 3. Generally for the 43 and 54 J cases, the specific impulse uncertainty was less than 5%, the power uncertainty 1.2%, the impulse bit uncertainty less than 5%, and the efficiency uncertainty less than 9.5%. At the lower energy, mass loss measurement were not available, so only power and impulse bit uncertainties were calculated which were slight higher than at the higher energy levels.

For the mass loss test conducted in vertical facility, the mass loss per pulse was 64.8 μg and 64.1 μg, for the 2000 and 10,000 pulse tests, respectively. The difference between the two measurements is approximately 1.1% which is less than the measurement uncertainty of the weighing device alone (1.5% for the shorter pulse test). Therefore, no mass loss per pulse difference was discernible for this configuration and energy level. This test was performed because there was a concern expressed that the heating of the fuel bar could cause an increase in fuel consumption that would effect the validity of the performance parameters. The longer test was conducted to see if additional heating of the fuel bar increased the mass loss per pulse.

Summary

PPT testing at LeRC showed that a configuration with 3.81 cm long electrodes, a 3.81 cm gap and a 20° flare had the highest efficiencies at the respective energy levels. The same gap and length without a flare and a notched fuel face had the highest thrust to power ratio. Also, the PPT was successfully discharged at lower energy levels, showing the PPT as viable across a wide range of energy levels.

The 3.81 cm x 3.81 cm x 20° flare at configuration 43 J was chosen for the Mightysat II.1 mission because of its high efficiency and good thrust to power ratio, and the ability to best approach the maximum mission requirements for the allotted fuel mass. The lower energy level was chosen to increase the life expectancy of the capacitor. The notched fuel face had better thrust to power ratios, but this particular configuration had the most difficult fuel feed system design. Also, the PPT was not operated over millions of pulses to determine if the fuel face would retain its original notched shape over its life. The same electrode configuration but without the flare was chosen for the EO-1 mission because of the high performance, but the flare was forgone because of dimensional restrictions place on the PPT by the spacecraft.

References

[1] LeDuc, J.R., Bromaghim, D.R., Peterson, T., Pencil, E., Arrington, L., Hoskins, W.A., Meckel, N.J., and Cassady, R.J., "Mission Planning, Hardware Development, and Ground Testing for the Pulsed Plasma Thruster(PPT) Flight Demonstration on Mighty Sat II.1", AIAA 97-2779, July 1997.

[2] Guman, W.J., "Pulsed Plasma Microthruster Propulsion system for Synchronous Orbit Satellite", Journal of Spacecraft and Rockets, Vol. 7, No. 4, April 1970.

[3] Guman, W.J., "Pulsed Plasma Solid Propellant Microthruster for the Synchronous Meteorological Satellite", NAS5-11494, August 1972.

[4] Brill, Y., Eisner, A., and Osborn, L., "The Flight Application of a Pulsed Plasma Microthruster: The NOVA Satellite", AIAA-82-1956, November 1982.

[5] Vondra, R.J. and Thomassen, K.I., "Performance Improvements in Solid Fuel Microthrusters", Journal of Spacecraft and Rockets, Vol. 9, No. 10, October 1972, pp 738-742.

[6] Yuan-Zhu, K., "Effects of Propellant Geometry on PPT Performance", IEPC 84-94, 1984.

[7] Palumbo, D.J., and Guman, W.J., "Effects of Propellant and Electrode Geometry on Pulsed Ablative Plasma Thruster Performance", Journal of Spacecraft and Rockets, Vol. 13, No. 3, March 1976, pp. 163-167.

[8] Curran, F.M., Peterson, T.T., and Pencil, E., "Pulsed Plasma Thruster Technology Directions", AIAA 97-2926, July 1997.

[9] Meckel, N. J., Hoskins, W.A., Cassady, R.J., Myers, R.M., Olsen, S.R., McGuire, M.L., "Improve Pulsed Plasma Thruster System for Satellite Propulsion", AIAA 96-2735, July 1996.

[10] Spanjers, G.G., Malak, J.B., Leiweke, R.J., and Spores, R.A., "The Effect of Propellant Temperature on Efficiency in a Pulsed Plasma Thruster", AIAA 97-2920, July 1997.

[11] Haag, T.W., "PPT Thrust Stand", AIAA Paper 95-2917, July 1995.

[12] Haag, T.W., "Thrust Stand for Pulsed Plasma Thrusters", Rev. Sci. Instrum., Vol. 68. No. 5, May 1997, pp. 2060-2067.

[13] Solbes, A. and Vondra, R.J., "Performance Study of a Solid Fuel-Pulsed Electric Microthruster", Journal of Spacecraft, Vol. 10, No. 6., June 1973.

[14] Guman, W.J., "Designing Solid Propellant Pulsed Plasma Thrusters", AIAA Paper 75-410, March 1975.

Table 1: Electrode Configurations

Configuration	Electrode Length, cm	Flare, degrees	Fuel Face	Energy Levels, joules
1	3.81	0	Flat	54,43,15,10,5
2	3.81	20	Flat	54,43
3	3.81	0	Notch	54,43
4	2.54	0	Flat	43

Table 2: Summary of Primex Breadboard PPT Testing at LeRC

Electrode Length, cm	Fuel Face	Flare Angle, degrees	Capacitor Energy, joules	I_{bit}, μN-sec	I_{sp}, sec	Thrust/ Power, μN-sec/J	Efficiency %
3.81	Flat	0	43	694	1200	16.1	9.5
3.81	Flat	0	54	875	1351	16.2	10.7
3.81	Flat	20	43	734	1228	17.1	10.3
3.81	Flat	20	54	914	1331	16.9	11.0
3.81	Notch	0	43	775	1059	18.0	9.4
3.81	Notch	0	54	950	1168	17.6	10.1
2.54	Flat	0	43	726	1121	16.9	9
3.81	Flat	0	5.2	96	–	18.5	–
3.81	Flat	0	10	97	–	9.4	–
3.81	Flat	0	15	172	–	11.5	–

Table 3: Performance Parameter Uncertainties

Configuration	$U_{impulse\ bit}$, %	$U_{specific\ impulse}$, %	U_{power}, %	$U_{efficiency}$, %
3.81 x 3.81 x 0°, 54J	4.7	4.8	1.2	9.5
3.81 x 3.81 x 0°, 43J	4.4	4.5	1.2	8.9
3.81 x 3.81 x 20°, 54J	3.1	3.2	1.2	6.3
3.81 x 3.81 x 20°, 43J	2.4	2.5	1.2	4.9
3.81 x 3.81 x notch, 54J	2.4	2.5	1.2	4.9
3.81 x 3.81 x notch, 43J	2.1	2.2	1.2	4.3
2.54 x 3.81 x 0°, 43J	3.6	3.6	1.2	7.3
3.81 x 3.81 x 0°, 5.2J	4.4	–	1.8	–
3.81 x 3.81 x 0°, 10J	5.7	–	3.8	–
3.81 x 3.81 x 0°, 15J	6.2	–	2.2	–

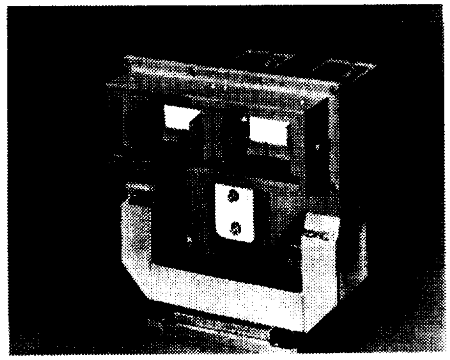

Figure 1: Photograph of Primex Breadboard PPT.

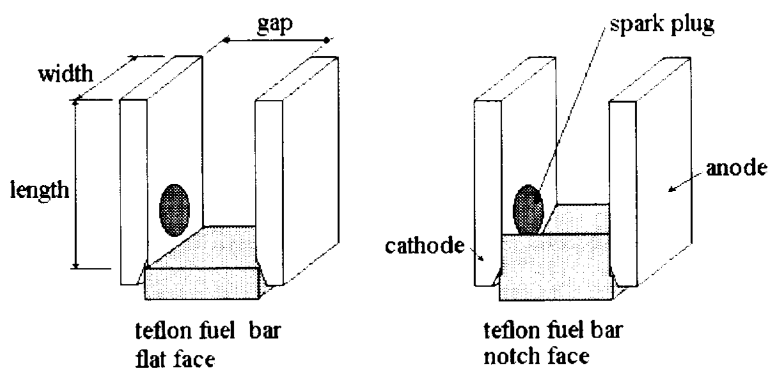

Figure 2: Schematic of Fuel Face and Electrode Configurations for a PPT.

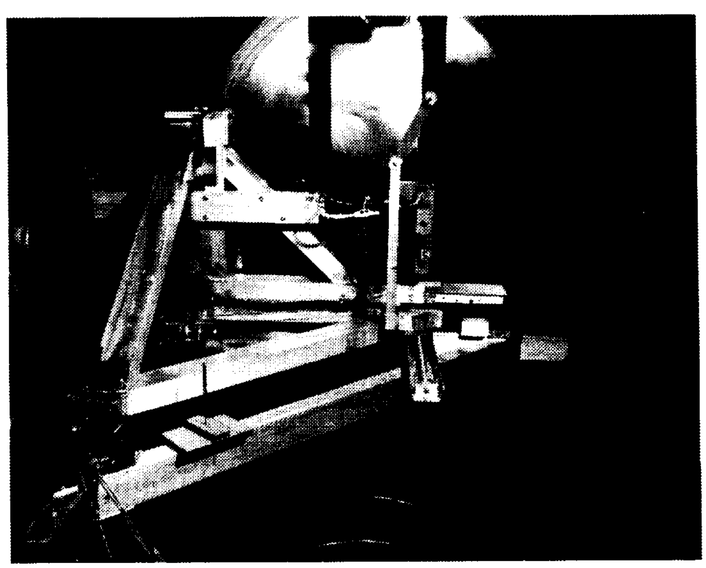

Figure 3: Photograph of Thrust Stand with Primex Breadboard PPT.

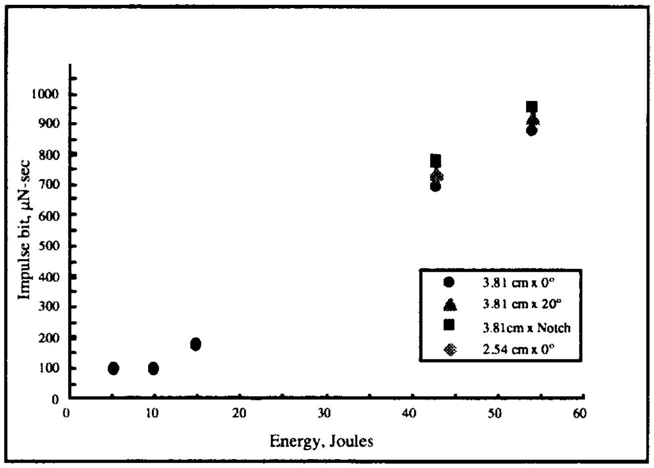

Figure 4: Impulse Bit versus Energy Level for Various Electrode Configurations.

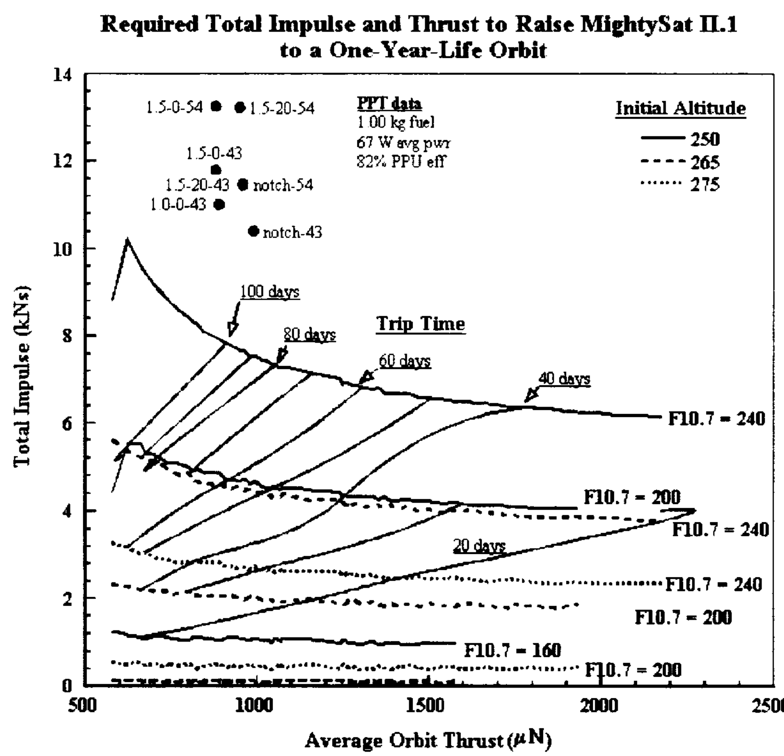

Figure 5: PPT Total Impulse Vs. Average Orbit Thrust Available reproduced from Reference 1.

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13. ABSTRACT (*Maximum 200 words*)

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