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DEVELOPMENT OF A MICRO PULSED PLASMA THRUSTER FOR THE DAWGSTAR NANOSATELLITE

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Abstract

The design and testing of a micro-Pulsed Plasma Thruster (PPT) is presented. Developed for the University of Washington's Dawgstar nanosatellite, the micro-PPT will provide formation keeping, orbit maintenance and attitude control functions to enable the smallest known spacecraft with an active propulsion system. Dawgstar, funded by Defense Advanced Research Projects Agency (DARPA), Air Force Office of Scientific Research (AFOSR), NASA and industry; is a 15 kg satellite which will fly in the ION-F formation to demonstrate nanosatellite formation flying and distributed science. The preliminary design shows that an eight thruster system can be built within a 13.5 W power and 5 kg mass budget. Preliminary testing at Primex Space Systems has provided proof of principle, impulse bit and specific impulse data. Further lifetime testing at the University of Washington validated the propellant feed system and the predicted capacitor lifetime. Data presented shows that a single micro-PPT consuming less than 6.8 W can produce an impulse bit sufficient to enable attitude control and orbit maintenance functions for a small satellite in low Earth orbit, while meeting stringent mass and volume requirements.

Introduction

The pulsed plasma thruster is excellent for numerous future missions that will utilize formation flying within a multiple satellite constellation because of its small impulse bit and high specific impulse. The PPT is a small, self-contained propulsion system that utilizes a solid Teflon™ propellant. Teflon has the advantage of being inert and non-toxic, giving the PPT system an additional benefit of being one of the safest propulsion systems.

The PPT, shown in Figure 1, is operated when a power processing unit (PPU) takes power from the satellite bus and charges a capacitor to several thousand Volts. Teflon is then fed between a pair of electrodes charged by the capacitor. A spark plug igniter, which consists of a semiconductor surrounded by another set of electrodes, is then fired to create a small amount of plasma between the electrodes. This allows the capacitor to discharge and ablate a small amount of the Teflon. This Teflon is then accelerated due to a combination of electromagnetic and gas dynamic forces producing thrust.

With the trend in satellite design being towards small,

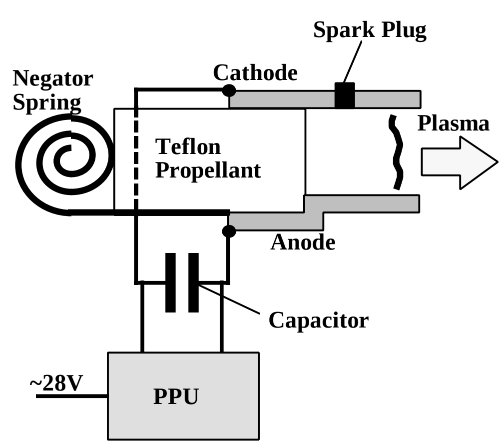

Figure 1 - Major components of a Pulsed Plasma Thruster (PPT).

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low cost, satellite constellations because of the reliability and their enabling science missions, the need for miniaturizing the PPT (and other thruster systems) became apparent. There is work currently at the University of Illinois, Princeton University, and Air Force Research Lab among others, where various techniques are being used to make the PPT smaller. For instance, work at Illinois is attempting to develop a circular PPT thruster in a co-axial form for ease in integration. The work here is to design an integrated micro-PPT system for the small nanosatellite (10-20 kg) class, functioning in satellite modes of formation keeping and maneuvering, orbit maintenance, and attitude control. In order to achieve these performance objectives within the stringent mass, power requirements of the nanosatellite class, the design was examined at the systems level, evaluating issues such as mechanical size, capacitor selection, spark plug design, and miniaturizing the electronics.

The outline of the paper follows the timeline of the design and development of the micro-PPT for the Dawgstar mission. First, a brief overview of the ION-F (and Dawgstar) mission is described, along with a description of the conceptual design of the propulsion system. This is followed by a general overview of the theory of how the PPT works, which is later used to compare to the test results. Then, the preliminary design of the micro-PPT is given, describing attributes such as fuel bar and electrode design. This is followed by the initial set of test results which verified the concept, and allowed several important issues to be evaluated such as capacitor and spark plug selection. A Modular Test Unit (MTU) has been built for the evaluation of the PPU, electrode and fuel bar geometry and the validation of a 45° fuel bar feed angle. Additionally, four different capacitor types have been tested to evaluate their feasibility with a PPT system. Next, the full mechanical and electrical prototypes are described, followed by test results and comparison to theory.

Mission

DARPA, AFOSR, NASA, and industry are jointly sponsoring the development and launch of ten university nanosatellites to demonstrate miniature bus technologies, formation flying, and distributed satellite capabilities. Utah State University (USU), the University of Washington (UW), and Virginia Polytechnic Institute and State University (VT) are designing and developing a system of three 15 kg spacecraft to investigate satellite coordination and management technologies, and perform distributed ionospheric measurements. The three universities are coordinating in the areas of satellite design, formation flying, mission development, and scientific instrumentation.

Tasks are to be performed jointly by the three-team members as well as individually by each spacecraft. The stated mission objectives are simultaneous, spatially distributed measurements of the ionospheric electron density; formation flying, including in-track and ground–track maneuvering and station keeping; and micro-PPT attitude and orbital control.

Initial micro-PPT System Concept

For Dawgstar, the proposed propulsion system consists of eight thrusters providing direct control in two axes for translation and three axes for rotation. The sixth axis (elevation) can be controlled via orbital dynamics. The placement of thrusters was made to maximize the mobility of Dawgstar. The final configuration has the thrusters mounted in orthogonal pairs at the top and bottom of the satellite, as seen in Figure 2. There are four thruster modules, with each thruster module consisting two micro-PPT’s mounted orthogonally. The two fuel bars within each module have a 45° feed angle which allows orthogonal thrust in each pair, and saves volume. One capacitor is used for each pair, for a total of four capacitors in the system.

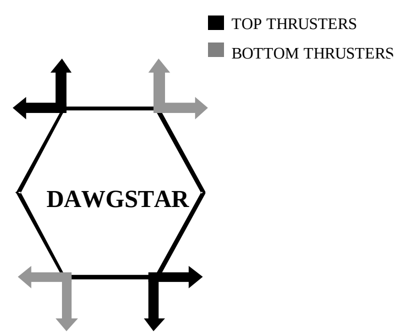

Figure 2 – Thruster Placement on the Dawgstar nanosatellite. Thrusters are mounted in opposite corners on both the top and bottom of the satellite to provide the greatest satellite mobility.

Because of the limited resources on Dawgstar, both the power and mass must be reduced considerably from previous designs. The entire propulsion system is required to operate with less than 13.5 W and have a mass of less than 5 kg. One obvious area of change to achieve this power reduction is the power processing unit. Current designs for two thruster systems consist of a charge converter for one capacitor, and two discharge circuits.[1] For Dawgstar, a two thruster module with one capacitor will again be used. However, one power processing unit and high voltage electronic switching circuits will used to charge the capacitor within the correct thruster module.

The capacitors used in previous PPT designs have been oil based, which are relatively large and have restrictive temperature limitations. By examining new capacitor technologies in cooperation with the NASA GRC/Unison Industries/CU Aerospace/Primex PPT Components Development Program, a new lighter system has been developed with more favorable thermal characteristics.

Theory

The derivation of analytical formulas that accurately predict the behavior of PPT's in general has proved difficult. Following Guman, it is possible to derive some simplified equations in a hope of describing the salient features of the thruster.[2]

Considering a breech fed geometry, the nozzle inductance can be written

$$L = m_o \frac{h}{w} l \qquad (1)$$

with m_o the magnetic permeability, h the distance between the electrode, w the width of the electrodes, and l the length of the electrodes. One can then determine the impulse by finding the instantaneous magnetic pressure on the back wall, and integrating this with respect to time over the length of the pulse. The result for the force produced due to magnetic pressure alone is

$$T_{MHD} = f \frac{m}{2} \frac{h}{w} \int_0^t i^2 dt \qquad (2)$$

where f is the pulse frequency, and i is the total current. Evaluating i through circuit analysis of an RLC circuit, the specific power is found to be directly proportional to h/w and inversly proportional to both l and the RLC circuit resistance, R.

Experience indicates that PPTs also experience a thrust contribution from gasdynamic expansion. Again following Guman, a volume element, V, that isentropically expands into a vacuum after the addition of some energy, E, is considered. If the element is sufficiently thin (such that non-steady pressure waves rapidly transverse the element), the mass averaged flow velocity is equal to the sonic velocity, and the initial temperature is small compared to the final temperature, then

$$T_{GAS} = f \left[\frac{8(g-1)}{g^2(g+1)} \cdot m_d \cdot E \right]^{\frac{1}{2}} \qquad (3)$$

where for Teflon, g= 1.3, m_d is the mass ablated per discharge, and E is the total energy.

Based on equations (2) and (3)

$$\frac{T}{P} \propto \frac{h/w}{l \cdot R \cdot \sqrt{E/m_d}} \qquad (4)$$

The results in (4) then form the basis for the optimization of the Dawgstar PPT.

Initial Design

Based on the maximum satellite power generation, an upper limit of 13.5 W was set for the propulsion system (firing two thrusters simultaneously). Assuming 80% electronics effeciency, 5.2 J is available for each discharge cycle. Further assuming that the discharge initiation circuit can be operated for 0.2 J. The energy available per pulse is 5 J.

Examining equation (4) shows that it is desirable from a specific power point of view to decrease the ratio of energy to discharge mass. Data presented by Guman shows that the ratio of energy to ablated mass is directly proportional to the ratio of energy to area.[2] Further, specific impulse is almost directly proportional to the square root of the energy to area ratio. Estimates for I_{sp} and I_{bit} based on the semiemperical PPT design equations presented by Burton and Turchi are shown in shown in Figure 3 for an energy of 5 J.[3]

Based on attitude control requirements and the formation flying mission, a target I_{bit} and I_{sp} were set at 65 µN-s and 500 s respectively. These requirements suggest a fuel bar face area of 2.32 cm^2.

Height to Width Ratio

Equation (4) shows that in order to maximize the nozzle inductance, the height to width ratio (h/w) should be maximized. Testing completed by Kuang on a breech fed PPT further supports this conclusion.[4] It was shown that for a 4.26 J thruster with a propellant area of 2.0 cm^2 (Corresponding to an E/A within 6% of Dawgstar), that increasing h/w from 2.0 to 3.13 translated to a 57%

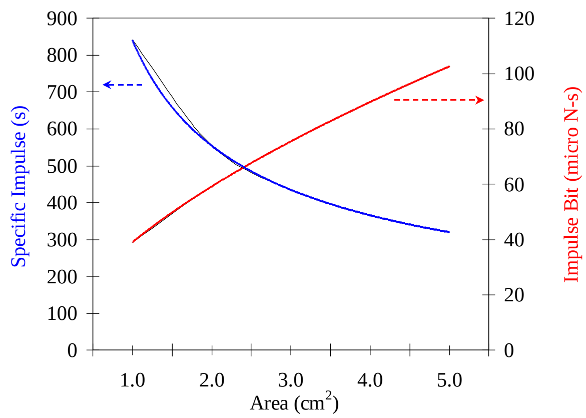

Figure 3 – Fuel area selection criteria based on desired specific impulse and impulse bit at 5 J. Adopted from equations presented by Burton and Turchi[2]

increase in I_{bit} and a 26% increase in I_{sp} with no increase the power consumption. Further, improvements of this magnitude were also achieved at several different h/w ratios, areas and energy levels.

Work by Palumbo and Guman shows that the spacing between electrodes needs to be tuned in order to optimize the performance of the PPT.[5] In work on a side fed PPTs, they show that as the electrode spacing continues to increase beyond some optimal point, efficiency begins to decrease. Because the data presented was for a significantly different geometry and further study of this parameter was not within the scope of the Dawgstar project, it was decided to choose a h/w ratio that was within the range for which data was presented by Kuang. The h/w ratio selected was 4.0.

Electrode Length

The desired h/w ratio fixes two of the electrode dimensions. Examining equation (4) implies decreasing the length of the electrodes will increase performance. Work by Guman provides further experimental data to support this conclusion. It was found that for a 5 J thruster, decreasing the electrode length from 7.6 cm (3") to 0.66 cm (0.25") increased the specific thrust by 100% and the specific impulse by 160%.

The limits on electrode length are then the electrode erosion rate and space available for integral mounting of the igniter. Based on these two concerns, an electrode length of 1.3 cm was chosen.

Predicted Results

The results predicted based on theory and experimental results of previous pulsed plasma thrusters are included in Table 1.

Table 1 – Predicted results for the Dawgstar PPTs

Parameter	**Value**
I_{bit}	65 μN-s
I_{sp}	500 s
Area	2.3 cm^2
Energy per pulse	5.0 J
Power*	6.5 W
h/w ratio	4
Electrode Length	1.3 cm

* Represents power per thruster. Normally two thrusters will be operated simultaneously

Developmental Testing Results

Fall 1999 testing was completed using the Moduluar Test Unit (MTU) developed by Primex and modified at the University of Washington. Data collected includes basic functionality testing, specific impulse and impulse bit measurements and capacitor testing. Testing completed in Winter 2000 includes lifetime tests, Teflon feed system testing, and igniter (spark plug) testing, again with the MTU. Further testing underway during the summer of 2000 is presented in Future Work.

Fall 1999 Testing

The MTU used for testing is shown in Figure 4. This test unit allows three different electrode geometries to be tested without removing the test unit from the vacuum chamber. Shown on the bottom left is the baseline EO-1 electrode configuration. On the right is a scaled down version of the EO-1 configuration. The center configuration is that of the Dawgstar micro-PPT, characterized by the small propellant area, high h/w ratio and 45° Teflon feed angle.

A closer view of the Dawgstar configuration during a thruster firing is shown in Figure 5.

Thrust data was obtained using a pendular thrust stand at Primex. The effectiveness of this type of stand is evident by the confirmation of the original LES 8/9 impulse bit measurements at NASA GRC using a torsion thrust stand.

The method used by Vondra et al. involves increasing the amplitude of the thrust stand in order to determine thrust. Because of the extremely small impulse bit produced by the Dawgstar PPT, this method would not have been successful. Instead, the thrust was used to overcome the natural damping in the thrust stand.

In this scenario, the thrust stand is set into motion, and the PPT is synchronized to fire as the stand passes through the center point of its swing. As the motion of the thrust stand is damped due to friction, there will be a point at which the thrust produced by the PPT has the same magnitude as the friction of the system.

In a method described by Haag, the thrust can then be measured by discontinuing the PPT firing and observing the decay that occurs. The displacement of the thrust stand is measured with a linear variable displacement transducer (LVDT).[6] Figure 6 shows the data taken for the Dawgstar PPT. The zero time point was then adjusted to reflect the first peak after the final discharge of the thruster. Data before time zero represents the equilibrium condition where the PPT thrust is balanced by system friction.

The impulse bit can then be calculated from the following equation

$$I_{bit} = \frac{\Delta x \cdot k}{w_n} \qquad (5)$$

where *Dx* represents the amplitude deficiency after one period, *k* is a thrust stand spring constant which converts LVDT signal into displacement, and w_n is the natural frequency. The displacement in the first period can be calculated by fitting a exponential curve to the absolute value peaks of the LVD. The calculated damping contsant, *b*, can then be used in the equation

$$\Delta x = A\left(1 - e^{-bt}\right) \qquad (6)$$

where *A* is the initial amplitude of the signal and *t* is the natural period. The only remaining unknown is then the spring constant, which can be determined by applying a known force to the thrust stand and measuring the displacement. Through this method, the experimentally measured impulse bit was calculated at 56.1 ± 9.9 µN-s (95% confidence)

With a known thrust, specific impulse can then be determined by recording the change in mass of the Teflon bar. Data presented represents a sample size of ~27,000 pulses. Results show a mass flow rate of 11.8 µg/pulse for a specific impulse of *485* ± 85 *s.* (95% confidence)

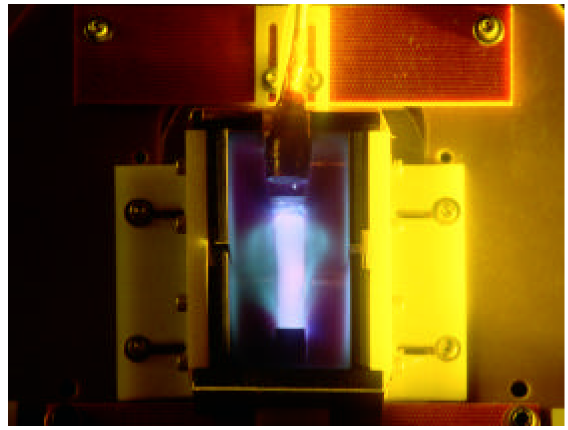

Figure 4 – PPT firing during tests conducted at Primex Aerospace Company.

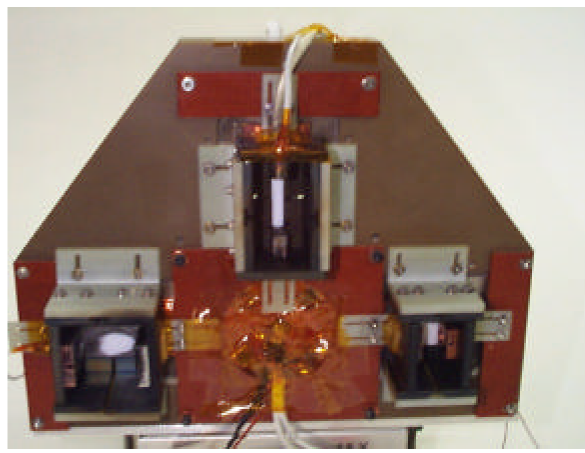

Figure 5 – The Modular Test Unit (MTU). Shown is a modular unit used for testing different PPT concepts. During this test, the MTU was configured for EO-1 (left), EO-1 scaled to 5 J (right) and the Dawgstar configuration (center).

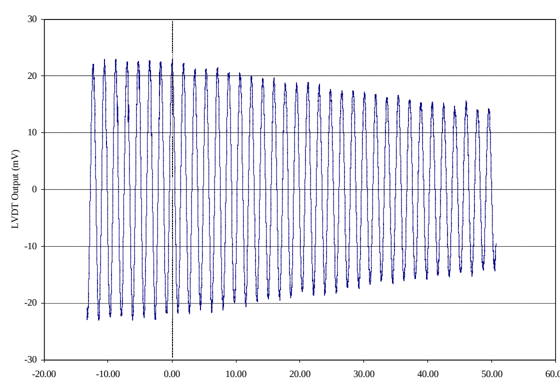

Figure 6 – Thrust stand data. The thrust stand is in equilibrium until time zero, when the thruster is turned off. The original thrust is calculated by measuring decay caused by friction.

Based on voltage measurements taken with 1000:1 high voltage probes connected to the capacitor, the voltage prior to discharge was 1466 V. Based on a measured capacitance of 5.0 ± 0.25 µF, this represents an energy, E of 5.23 ± 0.26 J. Based on previous testing which show the electronics to be 80% efficient, the total power of the system, P = 13.1 ± 0.6 W (2 Hz firing rate)

Table 2 – Measured Results for the Dawgstar PPT

Parameter	**Value**
E (J)	5.23 ± 0.26
I_{bit} (µN-s) †	56.1 ± 9.9
I_{sp} (s) †	483 ± 85
m_d (µg/pulse)	11.8
T/W (µN/W)	8.74
η (%) ‡	2.60

† Represents 95% confidence level
‡ Represents thruster efficiency and does not include electronics.

The other purpose of the Fall 1999 testing was to evaluate several different capacitor technologies in coordination with a concurrent effort undertaken by the PPT Components Program. Figure 8 shows the four capacitors tested. They included (from left to right) the baseline 40 µF Maxwell oil-filled capacitor, a commercial-of-the-shelf 39 µF AVX stacked ceramic capacitor, a 5 µF Unison Industries mica-paper/foil capacitor, and a 45 µF metallized film capacitor provided by Auburn University.

Large variations in the size of the tested capacitors were due to the capacitor sizes and voltage ratings readily available, and are not indicative of relative energy density. The intent of the test was to evaluate the behavior of different types of capacitors with the same thruster load. Tests were completed at a consistent energy level so that direct comparison was possible.

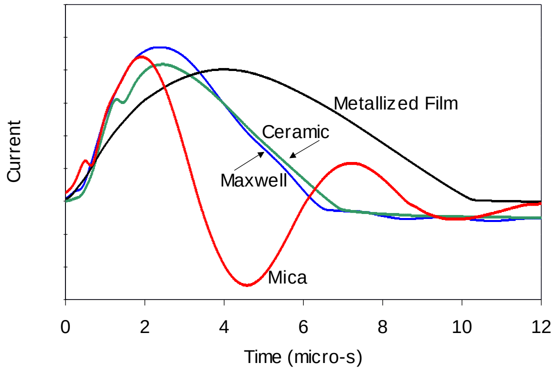

Figure 7 - Comparative Discharge Current Waveforms

The PPT requires very short duration, high amplitude current pulses to achieve the Lorentz electromagnetic and concentrated electrothermal acceleration. Only one of the four capacitor types tested had ever been used to discharge into a PPT load before. Significantly, all four capacitors tested provided sufficient discharge current for the PPT. Preliminary discharge current waveforms for each of the capacitors tested are shown in Figure 7.

These waveforms are the integrated outputs of the Rogowski coil used to measure current from the storage capacitor to the electrodes. While these traces do not have a calibration, they used the same Rogowski coil and are on the same scale, providing a relative comparison. Note that the increased ringing with the mica capacitor is due to the smaller capacitance of the available mica capacitor and is predicted by standard RLC circuit analysis.

Of the four capacitors tested, three survived the 10,000 to 30,000 pulse duration of the tests, with the stacked ceramic capacitor failing after 4450 pulses. In this case, a fracture of the ceramic at the lead was characterized, most likely due to piezoelectric effects from the many pulses. Of the remaining capacitors, the mica-paper/foil capacitor had the highest energy storage density and was chosen for further testing.

The results of these tests and the conclusions of the concurrent study by the PPT Components Program led to the selection of mica as the capacitor technology for Dawgstar. The combination of acceptable energy density, dry impregnant, flat geometry, and easy scaling to the smaller size made mica the best choice.

<u>Winter 2000 Testing</u>

Testing in the winter of 2000 was conducted at the University of Washington in conjunction with Primex and Unison Industries. The purpose of these tests was to characterize a new, smaller 0.250" diameter igniter developed by Unison, evaluate different igniter

Figure 8 – Capacitors tested during Fall 1999 Testing. From left to right: oil-filled, stacked ceramic, mica-paper/foil, and a metallized film capacitors.

materials, verify the 45° propellant feed angle over several million pulses, obtain another measure of Teflon flow rates, and perform capacitor lifetime testing on the mica-paper/foil capacitor. Again, these tests were conducted in close coordination with the PPT Components Program.

Two major MTU modifications were completed prior to testing. The first was moving the sidewalls towards the center of the thruster. As expected, doing so caused higher wall temperatures, which unfortunately caused the Torlon insulator to decompose and swell. As a result of this, a second major modification was to replace the Torlon with a Boron Nitride (BN) liner. Winter 2000 testing also differed in energy level from both previous tests with the expected flight conditions being the focus. In order to couple data with the PPT Components Program and to accelerate possible wear on the igniters, all of the Winter 2000 testing was conducted at 10 J.

Three igniters with different materials were successfully tested to 1 million pulses each, while the breakdown voltage and igniter resistance was measured. Misfire rates for all three igniters were low (1 - 2%), after an initial period of conditioning that can last as long as several hundred thousand pulses.

The minimum igniter voltage required to trigger a complete discharge was also measured periodically. By varying the discharge initiation capacitor charge time , the voltage available to the igniter could be varied, while success at initiating the discharge was observed. The voltages presented represent a 50% probability of firing (10 shots). In general, both 0% and 100% were

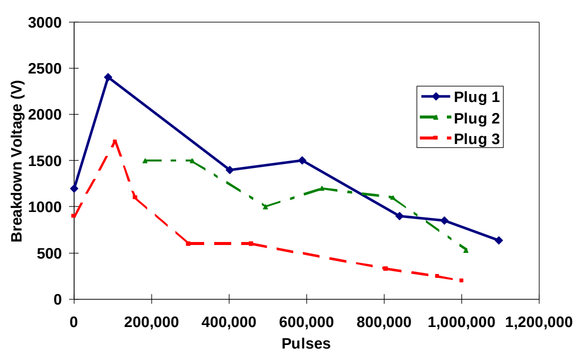

Figure 9 – Minimum igniter voltage required to trigger a discharge.

reached within 100 V. Results, shown in Figure 9, illustrate the conditioning period that takes place for a typical igniter.

Another significant result of these tests was the validation of the Teflon feed system with its 45 degree feed angle. Approximately 4 inches of propellant was fed for each of the 1 million pulse tests without any indication of cold flow or significantly uneven ablation. Therefore, a similar feed design was adopted for the flight unit.

Comparison of Results

The results of the testing completed in the fall of 1999 can be compared to both the predicted results and to those of Kuang, whose configuration was similar. This comparison is shown in Table 3.

The predicted results and those produced by Kuang are both within the experimental error margin of the test data.

Further comparisons can also be made to other thrusters of similar geometry and energy levels. Figure 10 shows well known PPTs on a plot of specific impulse versus energy per area. As expected from Figure 2, as the energy of the thruster increases at the same energy per area ratio, the specific impulse increases. If energy is held constant, specific impulse increases exponentially as the energy release per unit area increases.

Table 3 – Comparison of test results to predicted results and comparable thrusters.

	Dawgstar Test Results	Predicted Results	Kuang Results
Energy (J)	5.2	5	4.3
Area (cm^2)	2.3	2.4	2.0
h/w ratio	4.00	--	3.13
m_d (μg)	11.8	14.3	10.9
I_{SP} (s)	483	500	453
I_{BIT} (μN-s)	56.1	65	48.1
T/P (μN/W)	8.8	10	11.2
η (%) *	2.60	--	2.38

* Note that this represents propulsive efficiency. Electronics efficiency was calculated at 82% for Dawgstar and 50-70% for the Kuang

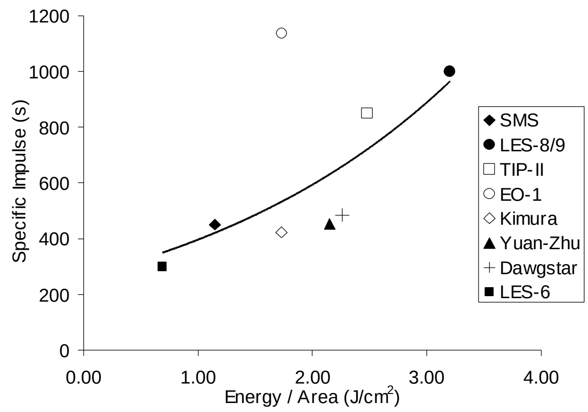

Figure 10 – Comparison of Dawgstar PPT to previous thrusters

Mechanical Design

The flight design of the flight thruster assembly was based primarily on mass and volume constraints. Because the thrusters are placed in pairs, it was desirable package them together.

The flight design is shown in Figure 11. In this design, two fuel bars (white) are fed into the thruster (gold) with constant force springs (red). The entire thruster assembly is then placed on top of the capacitor (orange), sandwiching it between the thruster and the satellite's bottom structural panel.

The thruster assembly is made primarily of Ultem™ 2300, a glass filled thermoplastic. Ultem was selected because it was found to have the highest strength to weight ratio of materials that met the melting point and dielectric strength requirement. The thruster housing was made in two parts to facilitate ease of machining.

Removing the Boron Nitride and the top half of the thruster housing, the internal structure of the thruster is visible (right part of Figure 11). The fuel bar is fed through a hole in the cathode (green) and rests against the anode (blue). An Ultem isolator (orange) attached to the cathode serves to electrically isolate the anode and the cathode. Also visible are two holes in the cathode for the integrally mounted igniters.

Manufacturing

Two thruster prototypes have been manufactured using a combination of 2-Dimensional CNC milling and conventional milling. The first was completed in high-density foam and was meant as a test of both the CNC code as well as to identify and correct manufacturing problem areas.

In June 2000, a fully operational prototype was completed for use with further testing.

The total propulsion system mass is 3.8 kg and includes 8 thrusters (4 pairs), all associated electronics, the electronics box, capacitors, and hardware. A

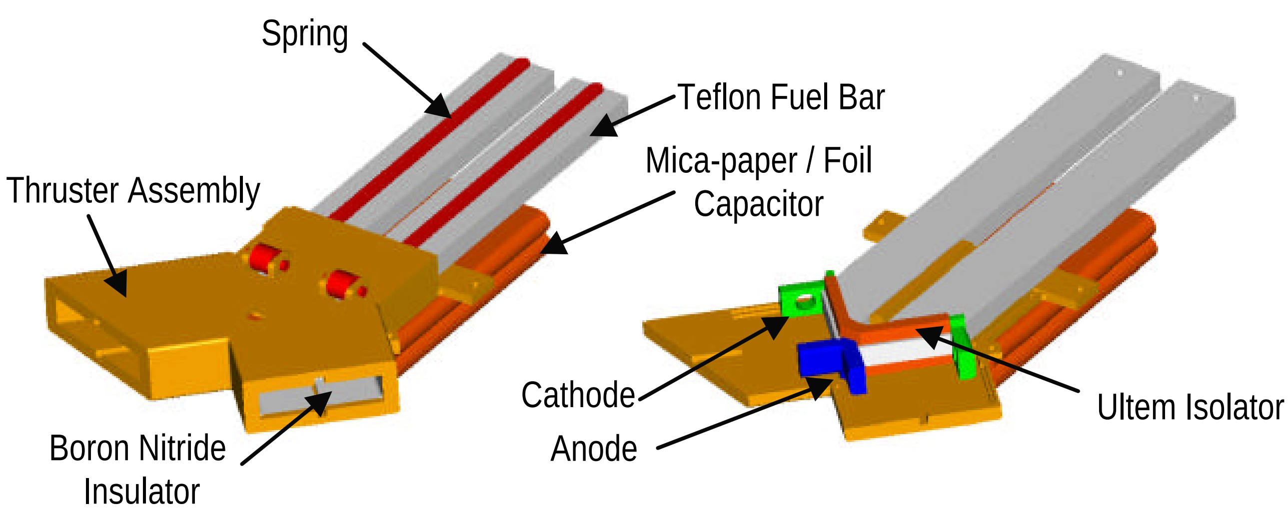

Figure 11 – Mechanical design of the Dawgstar micro PPT flight unit

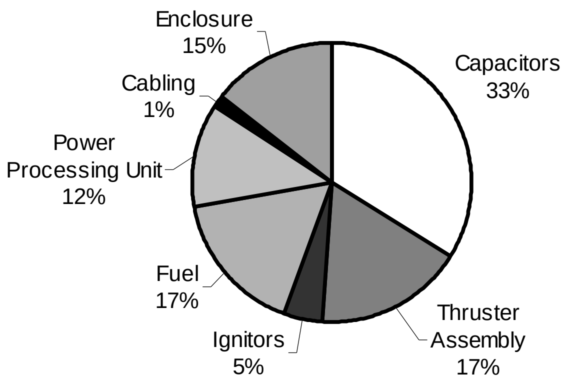

Figure 12 – Mass breakdown of the PPT including electronics. The entire system of eight thrusters has a mass of 3.8 kg.

breakdown of this is shown in Figure 12.

Electrical System Design

The electrical design for the Power Processing Unit (PPU) is based around a flyback converter which takes the bus voltage (nominally 28 V) and uses it both to charge the main capacitor and to power the discharge initiation circuit.

The flyback converter topology is ideal for this type of application because it does not require secondary output inductors. Inductors able to sustain the large voltages produced would impose a substantial mass penalty. Flyback converters also have the advantage of drawing constant power, reducing the impact on the satellite power system design.

Power taken from the bus is first passed through a filter and then continues to the primary of the main transformer. When the pulse width modulator (PWM) commands the switch open, all of the rectifier diodes (denoted **A** in Figure 13) are reverse-biased and the output capacitors (denoted **B** in Figure 13) provide the load currents. At the same time, the primary side of the main transformer acts like a pure inductor and current rises to a peak. When the PWM turns the switch off, the energy stored in the primary ($\frac{1}{2}LI_P^2$) is delivered to the secondary circuits. The feedback circuit continues this process until the desired voltage (2770 V) is met.

Because of the multiple thrusters being flown on Dawgstar, the Central Processing Unit (CPU) is required to select both a capacitor and spark plug to fire. A significant feature of the electronics design is the development of a relatively low mass, high voltage switch to allow this sharing of the converter. When commanded by the computer, the energy stored in the discharge initiation circuit is sent to the igniter which precipitates a discharge across the face of the Teflon fuel bar.

Additionally, the ability exists to 'fast charge' the capacitors in the event that high firing rates became desirable (end of satellite life). In this case, the thruster would be capable of firing two thrusters at 2 Hz.

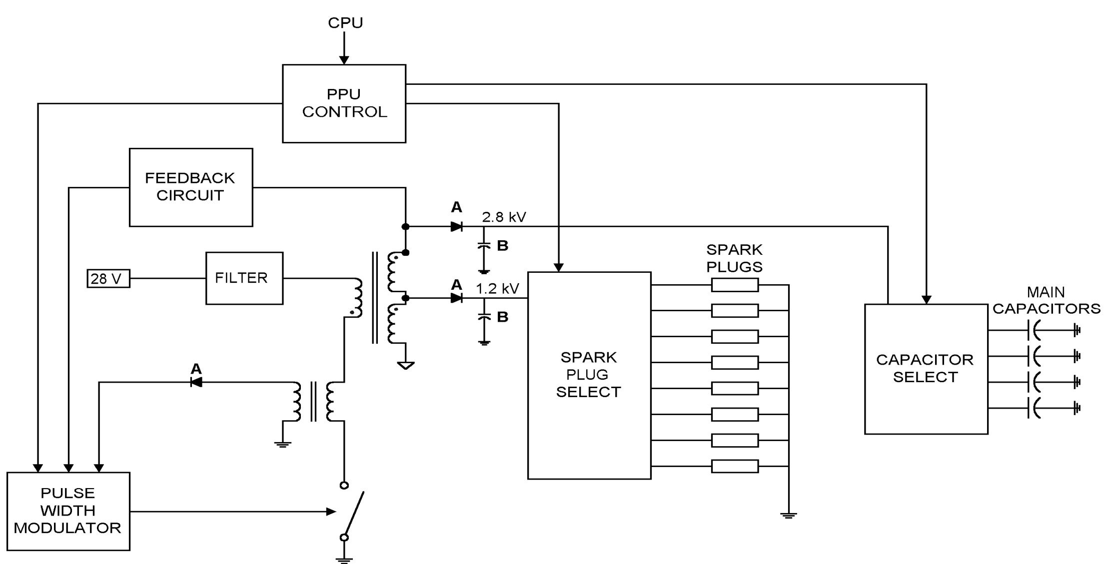

Figure 11 – Functional block diagram of the power processing unit. Bus power supplied is passed through a flyback converter to drive the discharge initiation and main capacitor charge circuits.

Future Work

Testing underway in the summer of 2000 is intended to provide proof of principle for the flight thruster mechanical and electrical design using the built prototype

Specifically, past designs have operated the igniter at voltages of up to 3 kV. Because this involves the use of multiple heavy transformers (Dawgstar would require 8) to increase the voltage, further igniter voltage testing is being undertaken in an attempt to remove these transformers

Additionally, testing beyond one million pulses will be conducted to evaluate end of life performance and likely failure points.

Electromagnetic Interference (EMI), Thermal Cycling and Random Vibration testing will be completed in the fall of 2000.

Conclusions

Testing undertaken at Primex and the University of Washington has provided proof that micro pulsed plasma thrusters are feasible within the limitations of micro- and nanosatellites. Specifically, a system was developed for the Dawgstar nanosatellite that had a total of 8 thrusters and associated electronics for a mass of 3.8 kg. This system can maintain satellite attitude with as little as 3.3 W (2 thrusters at 0.25 Hz) orbit averaged power (based on the nominal Dawgstar orbit of 375 km and a max difference in CP and CG of 5 cm). In the event of thrust maneuvers and formation flying, the system could two thrusters at a rate of up to 2 Hz each.

Capacitor testing identified mica-paper/foil as the most feasible capacitor technology for micro pulsed plasma thrusters. This was not only due to high energy density, but also the robustness of the capacitor itself.

Impulse bit and specific impulse data show that the thruster operated as designed. Data also showed agreement with theory and thrusters of similar design.

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