Advanced Pulsed Plasma Thruster Demonstration on MightySat Flight II.1

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Abstract

This paper describes the progress associated with a joint effort to demonstrate an advanced pulsed plasma thruster (PPT) on MightySat Flight II.1 to be launched in January, 1999. The PPT currently being developed for this flight represents a significant leap in technology compared to previous flight models. Although the MightySat II.1 launch vehicle is yet to be determined, the Space Shuttle Hitchhiker Eject System is the primary option under consideration. With this launch option, the PPT will be used to extend MightySat II.1 life from about 1-3 months to over one year by raising its operational orbit. The PPT is an ideal propulsion system for extending small satellite life because of its high specific impulse (>1000 sec), low system wet mass (<5 kg), and inert nature when unpowered (thus minimizing Shuttle integration issues). In addition to the life enhancement mission, the on-orbit operations have been specifically designed to rigorously test the PPT and to demonstrate its compatibility with the MightySat II.1 spacecraft in order to validate it for future DoD, NASA, and commercial satellites.

Introduction

This paper describes the progress associated with a joint government and industry effort to demonstrate an advanced pulsed plasma thruster (PPT) on the MightySat II.1 space flight to be launched in January, 1999. MightySat II.1 is a 275 lb. satellite to be manufactured by Spectrum Astro, Inc. of Gilbert, AZ under contract with the Space Experiments Directorate of the Air Force Phillips Laboratory at Kirtland AFB, NM.[1,2] Participants in the joint PPT flight demonstration effort include the Propulsion Directorate of the Phillips Laboratory, the NASA Lewis Research Center (NASA-LeRC), the Jet Propulsion Laboratory (JPL), and Olin Aerospace Company (OAC).

The PPT is an electric propulsion device which uses electric power to ionize and electromagnetically accelerate a plasma to high exhaust velocities, attaining a specific impulse in the 1000-2000 second range. The PPT is ideally suited to the propulsion needs of small satellites because it is compact, uses an inert solid propellant (Teflon™), is easily integrated to a spacecraft, and has a low system wet mass (<5 kg). Although PPTs have performed flawlessly on several satellites, PPT research and development essentially stopped in the 1970's. The PPT to be demonstrated on Flight II.1 represents a dramatic leap in capability compared to previous flight qualified models, and is being developed by OAC under a contract with NASA LeRC.[3,4]

Due to its efficient fuel consumption and low power requirements (1-150 W), the PPT can significantly enhance small satellite maneuvering capabilities. Potential applications include attitude control (including the complete replacement of a reaction wheel/momentum dumping system),[3,5] orbit maintenance, and orbit raising/repositioning.[6] The Phillips Laboratory's MightySat II Program Office has identified the Space Shuttle Hitchhiker Eject System (HES)[7] as the primary launcher for its small satellites.[1,2] With this launch option, the PPT on MightySat Flight II.1 will perform an orbit raising mission to significantly increase on-orbit life from about 1-3 months to over one year. The advanced PPT enables the use of the Shuttle HES as an affordable and reliable launcher for long-design-life small satellites.

In addition to the actual use of the PPT for extending the life of Flight II.1, the objectives of the MightySat II.1 demonstration are twofold. First, this flight will demonstrate advanced PPT performance and on-orbit life on a viable spacecraft. The performance and lifetime of the thruster will be demonstrated during a 1-3 month duration orbit raising maneuver at the beginning of the MightySat II.1 mission, and potentially during a second orbit raising maneuver near the end of the mission. The second objective is to demonstrate compatibility of the PPT with the spacecraft and optical sensor payloads. Potential integration issues include electromagnetic interference (EMI), thermal loading, and contamination of spacecraft surfaces. It should be emphasized that previous space flights have shown complete compatibility of the PPT system with the host spacecraft after many years of operational use. However, to demonstrate and characterize PPT plume compatibility with optically sensitive payloads and thermal surfaces, two quartz crystal microbalance (QCM) / calorimeter sensor packages will be used for measuring spacecraft surface deposition from the PPT exhaust plume. Additionally, this mission will serve as a pathfinder for demonstrating PPT compatibility with Shuttle integration requirements.

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In addition to leading the PPT flight demonstration effort, the Phillips Laboratory Propulsion Directorate is primarily responsible for spacecraft integration and test, flight operations, and flight data analysis associated with the PPT. NASA-LeRC is leading the flight PPT development effort, and will be performing many of the ground based PPT performance, plume contamination, and flight qualification tests. OAC is responsible for developing, qualifying, and fabricating the flight PPT, under a contract with the NASA-LeRC. JPL will provide the flight contamination sensors and will lead the associated flight operations and data analysis efforts.

The Pulsed Plasma Thruster

The PPT is an electric propulsion device which uses electrical power to ionize and electromagnetically accelerate a plasma to high exhaust velocities (10-20 km/sec).[8-13] Its high specific impulse enables significant reduction in propellant mass requirements compared to monopropellant and cold gas systems.

A schematic of the PPT is shown in Figure 1. The thruster consists of a bar of Teflon™ propellant pressed against a lip between two electrodes by a negator spring (which is the only moving part). The negator spring serves to continually replenish the propellant as it is consumed. A power processing unit (PPU) charges a capacitor to voltages in the 1000-2000 V range using unregulated power from the spacecraft bus. The PPU also supplies a high voltage pulse to a spark plug which is used to ignite the discharge. Once the discharge is ignited, the energy stored in the capacitor (~40 J) powers a high current / short duration plasma discharge (~20 kA, ~5-10 microseconds). This discharge ablates and ionizes a small amount of Teflon™ from the face of the propellant bar and accelerates it to high exhaust velocities using the Lorentz force. The pulsed operation of the PPT allows it to function over an extremely wide range of input power levels with the same per-pulse performance. Average spacecraft bus power supplied to the PPT dictates the pulse rate, which is typically not more than 1-3 Hz.

PPTs have flown on LES 6,[8-10] TIP II & III,[11,12] NOVA I, II, III,[13,14] as well as on Japanese[15] and Chinese[16] spacecraft. PPTs have also been flight qualified for the LES 8/9[17,18] and SMS spacecraft.[19] These PPTs have performed flawlessly and would benefit the new generation of small satellites even at their low performance levels. Unfortunately for small satellite designers, these models are no longer available. Furthermore, the performance of previous flight-qualified models, even if they were available, is not well suited for the more ambitious life extension missions discussed in this paper, especially for >100 kg satellites. The absence of an off-the-shelf flight qualified PPT has recently spurred R&D programs at the Phillips Laboratory,[20] NASA-LeRC,[3,4,21] and OAC,[3,4,21] with goals to significantly increase performance and decrease system wet mass while maintaining flight heritage of previous designs.

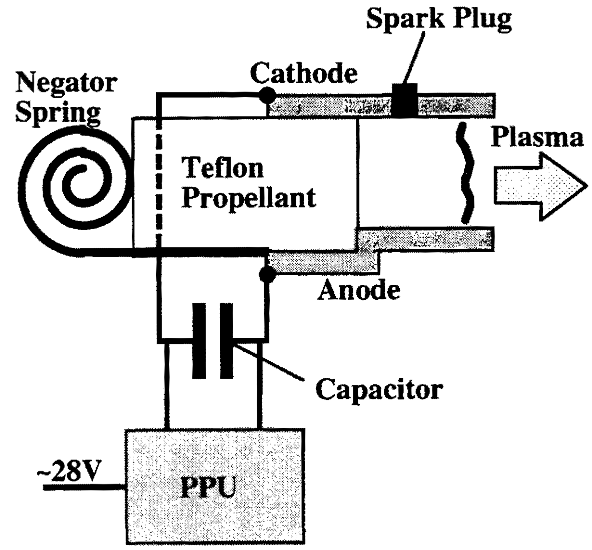

Figure 1: The pulsed plasma thruster

MightySat II.1 Spacecraft

MightySat II.1 is the first of five satellites (two of which are options) to be manufactured by Spectrum Astro, Inc. of Gilbert, AZ under contract with the Space Experiments Directorate of the Air Force Phillips Laboratory at Kirtland AFB, NM.[2] The primary objective of the MightySat II program is to provide timely and affordable access to space for Phillips Laboratory developed technologies. The planned launch date for Flight II.1 is in January, 1999, with the launch of each additional satellite following every 18-24 months.[2] The satellite bus is designed for one year of total on-orbit life. Due to uncertainty in the launch vehicle, the satellite will be designed for deployment from the Space Shuttle-HES as well as a variety of expendable launch vehicles (ELV's) that are being considered.

Figure 2 shows a scale drawing of the MightySat II.1 spacecraft, along with preliminary characteristics. It is a class D satellite[22] with a total power of approximately 325 W and a mass of 275 lb. The spacecraft is 3-axis stabilized, utilizes a UHF communication system, and has electrical power provided by two 2-D articulated silicon arrays (26 ft^2) on a 28 V unregulated bus. The command and data handling system centers around a VME (Versa Module Eurocard) specificiation computer card backplane with two sets of 21-slot card cages housing all of the spacecraft electronics. Payloads are mounted either inside or on top of the spacecraft bus, which has approximate dimensions of 20 x 24 x 12 inches.

There are eleven payloads on MightySat II.1 including the PPT and its own diagnostic package, the Plume Diagnostic Experiment (PDE), which is designed to measure PPT plume effects on the spacecraft. MightySat II.1 has two optical sensor payloads, the Fourier Transform Hyperspectral Imager (HSI), and the Total and Ultraviolet Irradiance Radiometer (TUVIR), which may also be used at the end of life to observe PPT plume effects.

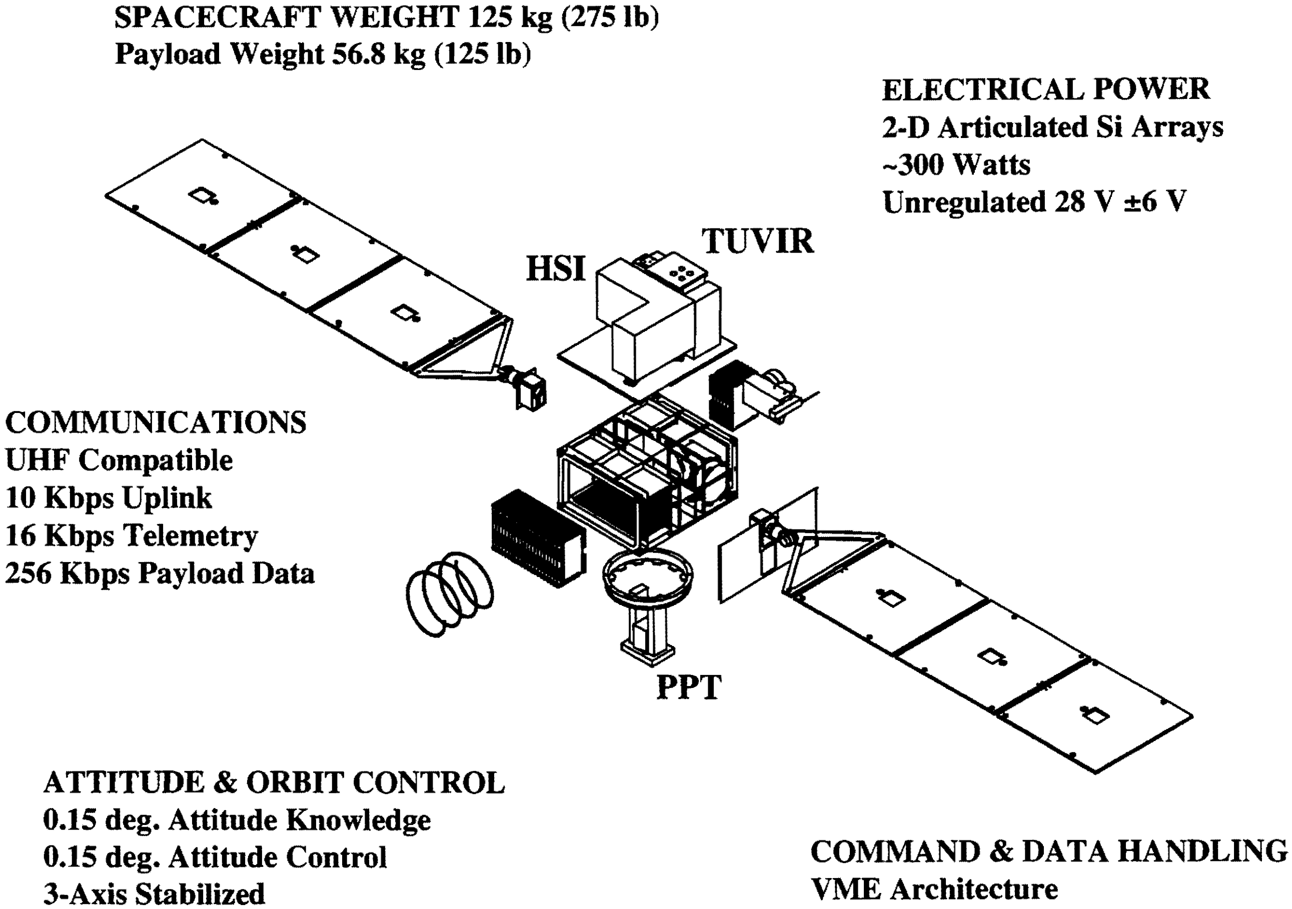

Figure 2: Exploded view of the MightySat II.1 spacecraft showing experiment locations and general spacecraft information

General design requirements of the PPT system on the MightySat II.1 spacecraft design are threefold. First, in order to minimize impacts on the attitude control system, the PPT is aligned with the spacecraft center of mass. Second, to maximize the ability to raise the satellite orbit, articulated solar arrays are required in order to decouple the need to align the arrays with the sun and align the PPT with the spacecraft velocity vector. Finally, the PPT system requires the flight software to have the sophistication necessary to autonomously operate the PPT while in sunlight for many orbits between ground contacts. Additionally, the impact of the PPT on the spacecraft thermal design is currently being assessed.

Mission Analysis

The primary mission of the PPT is to extend MightySat II.1 on-orbit life to greater than one year. Although the launch vehicle has yet to be determined, the worst-case scenario in terms of on-orbit life is the use of the Space Shuttle HES. On-orbit life without propulsion, at Shuttle-deployed altitudes, is less than 100 days for typical small satellites. For many Shuttle-deployed small satellites, the PPT is well suited for extending satellite life to 1-2 years.[6] Unfortunately, the large cross-sectional area and mass of MightySat II.1, in conjunction with the fact that it will be launched near solar maximum, presents an extremely demanding mission for the PPT. PPT power handling capability, total impulse, and performance are required to be much greater than that ever flown or flight qualified before.

Orbital analysis was performed to determine the most efficient PPT thrusting strategy for extending the on-orbit life of Shuttle-deployed satellites.[6] Three primary strategies were identified, with the appellations of Hold, Lift & Coast, and Lift & Hold. The Hold strategy consists of using the PPT at the Shuttle-deployed altitude to provide an orbit-averaged thrust to exactly compensate for the drag force. The disadvantage of the Hold strategy is that the power requirements at Shuttle-deployed altitudes are typically too high. An alternative strategy, Lift & Coast, requires that all payload power be devoted to the PPT at the beginning of the mission to raise the satellite to a higher altitude. Lift & Coast requires the least amount of propellant (and thus total impulse) of all strategies considered. Additionally, it requires no power once the orbit raising mission is complete. The disadvantage of Lift & Coast is the inability to operate the payload(s) during this orbit raising mission, which typically has a duration of 1-3 months. Lift & Hold consists of using the PPT at full power to raise the satellite to an altitude where the Hold power requirements are much more manageable. This strategy provides a compromise between Hold and Lift & Coast by reducing the trip time during the Lift phase and reducing the power requirements for the subsequent Hold phase.

Shown in Figure 3 is a comparison of the three PPT thrusting strategies for one MightySat II.1 conceptual design at nominal solar conditions. Without propulsion, at the Shuttle-deployed altitude of 190 nm, satellite life is only 37 days. At 190 nm, the Hold mission requires 50 W of power in the sunlight and 2.4 kg of propellant. Since the PPT is continually fighting the maximum satellite drag force, the PPT total impulse requirement for this strategy is excessively high (24,000 N-sec). Lift & Hold allows the Hold power to be reduced to 30 watts of power in the sunlight. However, the total impulse requirement for the PPT increases to 16,000 N-sec. The Lift & Coast option, using 100 W in the daylight, requires 60 days of transfer time (TT) to raise the satellite to 250 nm, which is the altitude corresponding to a 1-year natural decay life. For the case shown in Figure 3, the propellant mass is less than half that of Hold, with a total impulse of 10,000 N-sec.

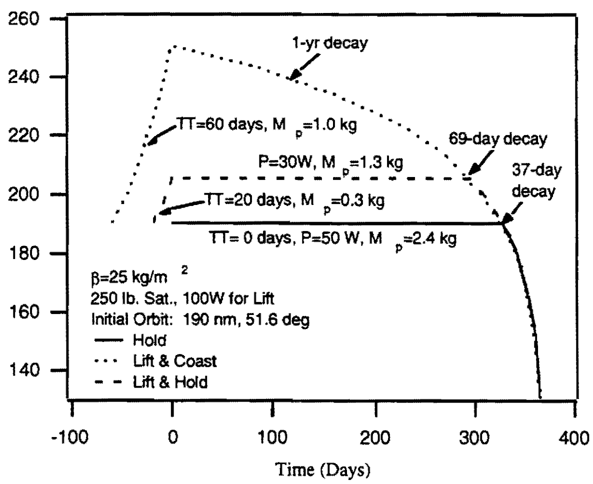

Figure 3: A comparison of Hold, Lift & Coast, and Lift & Hold strategies.

Accepting a PPT transfer time of 1-3 months, the Lift & Coast strategy is most attractive for the MightySat II.1 application because it minimizes the PPT total impulse and payload-on power requirements. Furthermore, it should be noted that the power and propellant mass requirements in Figure 3 for the Hold and Lift & Hold strategies are minimum values as the Hold strategies for power-limited satellites are inherently unstable to long term atmospheric density fluctuations.[6] Additional propellant mass and power must be budgeted to the PPT to account for thermospheric density variations due to solar activity and geomagnetic storms.[6,23,24]

An additional advantage of Lift & Coast was the elimination of simultaneous PPT and payload operation for the MightySat II.1 mission. Contamination concerns for the optical payloads are satisfied for this mission by using shutters on exposed optics that are opened after the lift phase is over and initial PPT operation is complete. Confirmation that the PPT does not produce contamination detrimental to optical sensors will be accomplished on this flight through the use of a dedicated sensor package operational during the lift phase. This sensor package, called the Plume Diagnostic Experiment (PDE), will be discussed in a later section of this paper.

The effects of PPT power input, performance, initial altitude, and solar conditions were also investigated in this orbital analysis effort. The details of the model are described in detail in reference 6, which also includes results that are generalized to all small satellite designs (mass, power, cross-sectional area) and operational scenarios. The MSIS-86 thermospheric model[25] was used, which is accurate to about 10% except at the highest latitudes.[26] For the results shown below, an eclipse fraction of 33.3% was assumed in order to eliminate launch date, orbit inclination, and initial right ascension of the ascending node from the trade space. All of these parameters have a small impact on the PPT's ability to extend satellite life.[6] Initial on-orbit check-out time was also neglected in this preliminary analysis.

Shown in Figures 4-7 are the effects of various parameters on the PPT's ability to increase MightySat II.1 lifetime. As a baseline for comparison, PPT performance was assumed to have: a thrust efficiency of 9%, a specific impulse of 1150 seconds, and a PPU efficiency of 85%. These performance figures represent the minimum values measured for the optimum PPT configuration, described in the next section. Also used as the baseline was an initial altitude of 215 nm, and a solar flux index, $F_{10.7}$, of 160 x 10^{-22} W m^{-2} Hz^{-1}. This value of $F_{10.7}$ represents the average value at solar maximum, over the last 21 cycles.[27]

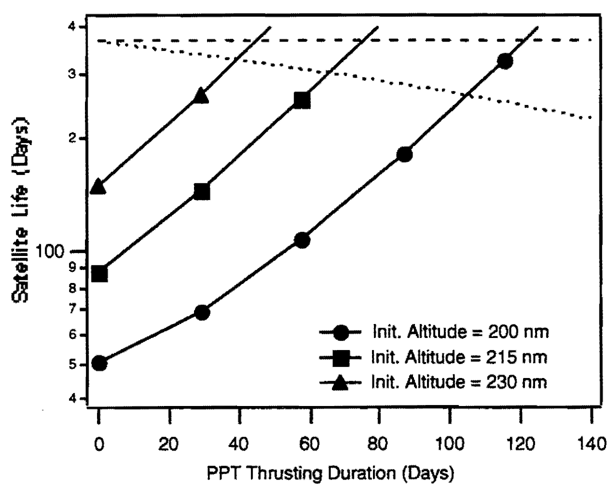

Figure 4: Satellite life versus PPT thrusting duration. PPT daylight power = 125 W, $F_{10.7}$ = 160x10^{-22} W m^{-2} Hz^{-1}.

Shown in Figure 4 is the dramatic effect of the initial altitude on satellite life (after PPT firing is complete) and PPT thrusting duration. The horizontal dashed line represents the goal of a having a 1-year life orbit decay after PPT operation is complete. A dotted line, sloping down to the right, takes into account that the total design life of the satellite is 1 year and shows 1-year life including the duration of PPT operation. The satellite life corresponding to a PPT thrusting duration of zero is the natural decay life of the satellite without propulsion. In addition, the symbols represent 0.25 kg propellant mass increments, allowing for easy comparison of propellant requirements for each condition. For instance, the baseline configuration at a 215 nm initial altitude requires about 0.6 kg of propellant to obtain a 1-year life (including PPT thrusting).

Shown in Figure 5 is the effect of power input to the PPT on MightySat II.1 life. Under nominal conditions, the PPT requires about 70-80 days to enhance on-orbit life to one year. Since the average PPT thrust is proportional to the power input, a higher power corresponds to reduced transfer durations.

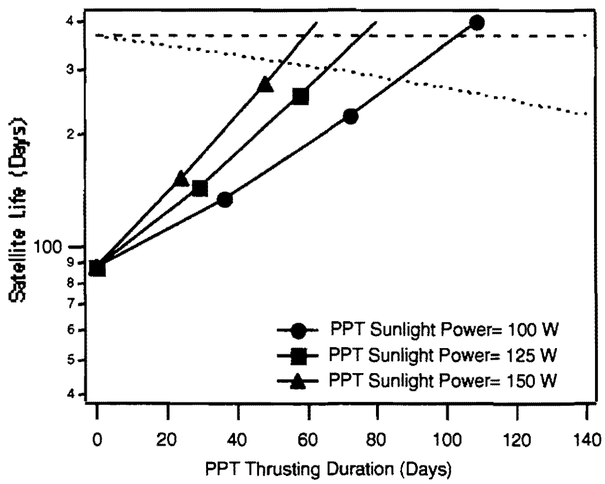

Figure 5: Satellite life versus PPT thrusting duration. Initial altitude = 215 nm, $F_{10.7}$ = 160 x 10^{-22} $Wm^{-2}Hz^{-1}$.

Shown in Figure 6 is the effect of solar activity on the PPT's ability to enhance MightySat II.1 life. The value of $F_{10.7}$ = 80 x 10^{-22} W m^{-2} Hz^{-1} represents the two-sigma worst case value at solar minimum.[27] Although the natural life of MightySat II.1 is greater than 300 days, the PPT is more than capable of extending satellite life at solar minimum well beyond 1 year. The value of $F_{10.7}$ = 240 x 10^{-22} W m^{-2} Hz^{-1} represents the two-sigma worst case value at solar maximum.[27]

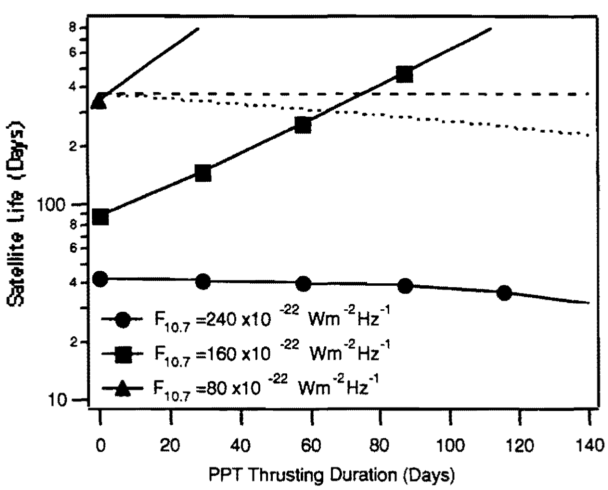

Figure 6: Satellite life versus PPT thrusting duration. PPT daylight power = 125 W, initial altitude = 215 nm

Figure 6 suggests that the PPT thrust is slightly less than the drag force at this condition and consequently unable to raise the orbit of the satellite for the baseline configuration. Since the two-sigma worst case situation at solar maximum is the desired design point (MightySat II.1 will be launched near solar maximum) alternate baselines are currently being explored. These changes include requiring a starting altitude to be above 215 nm, boosting the power input to the PPT beyond 125 W, and improving the efficiency of the PPT beyond 9%.

Figure 7 shows the effect of improving PPT performance. Significant satellite lifetime enhancement has been realized by the performance improvements of the current NASA/OAC design (square symbols) when compared to a PPT with LES 8/9 performance (circles). Note that actual use of a LES 8/9 PPT design is prohibitively heavy because the maximum power capability of the LES 8/9 PPT was only 50 W, thus the need for three LES 8/9 PPTs operating simultaneously. The triangle symbols show the benefits of further performance improvements (12% thrust efficiency, 1000 sec) beyond that of the current NASA/OAC design. Such performance is likely to be obtained within the near future, possibly for the MightySat II.1 flight.

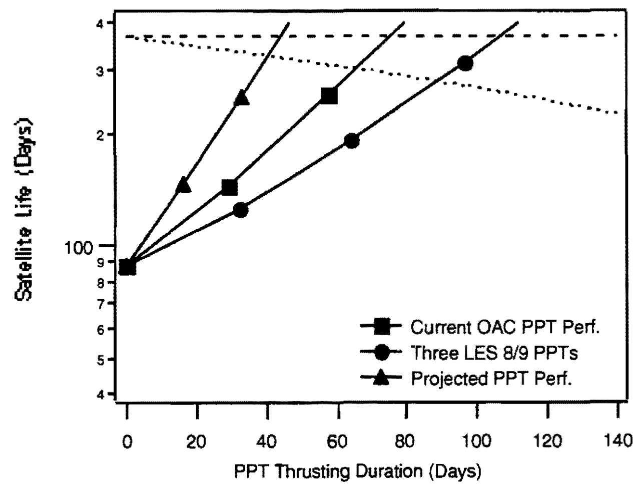

Figure 7: Satellite life versus PPT thrusting duration. Initial altitude = 215 nm, PPT sunlight power = 125 W, $F_{10.7}$ = 160 x 10^{-22} $Wm^{-2}Hz^{-1}$.

The PPT design for MightySat II.1 will also be applicable to other satellites with widely varying masses and power capabilities. The scaling parameters associated with small satellite life extension are discussed in reference 6. For instance, a satellite with one half the mass of MightySat II.1 will require only half the power to perform the same life extension mission. Satellites with a ballistic coefficient which is much higher than MightySat II.1's (~30 kg/m^2) will also require less power to boost the life of the satellite.

Design of the High Power PPT for MightySat II.1

The NASA PPT Program is completing its development phase and has begun initial flight design efforts at OAC. While the original goals for the NASA program called for doubling the total impulse while halving the mass of the PPT relative to the

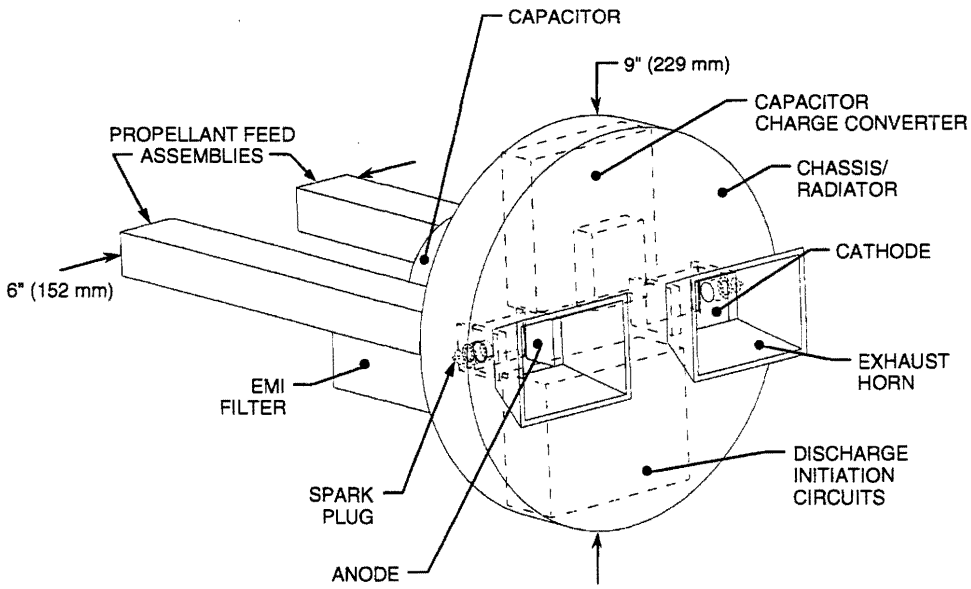

Figure 8: Outline drawing of the NASA/OAC High Power PPT (model PRS-101)

LES 8/9 configuration,[3] several design challenges for the MightySat II.1 mission have required a shift in focus for the flight design. The high total impulse required for the MightySat II.1 mission mandates long component life and increased propellant mass, even at higher specific impulse values. However, even more significant than increasing specific impulse (Isp), maximizing thrust to minimize trip time has emerged as a critical mission requirement.

To maximize thrust for the MightySat II.1 mission, the PPT will operate at over twice the power of the LES 8/9 PPT, as well as utilize improvements achieved in the thrust to power ratio realized during the development phase. This increase in power handling necessitates proportionally greater thermal dissipation requirements. However, unlike the large GEO satellite that was to carry the LES 8/9 PPT, MightySat II.1 cannot absorb large thermal loads from the PPT, and requires the PPT to do much more of its own thermal management. These MightySat II.1-related challenges represent a very aggressive design goal. For this reason, the MightySat II.1 flight PPT design is designated as the High Power PPT. This thruster system will be available in mid-1997 from OAC, having model number PRS-101.

High Power PPT Configuration

The PPT system for the MightySat II.1 spacecraft will be designed to mount in a 4 inch (102 mm) tall by 6 inch (152 mm) wide rectangular opening centered inside the spacecraft/HES interface ring (Marmon ring) on the face opposite the velocity vector. This opening extends through the main spacecraft body, which is 12 inches (305 mm) deep. In addition, the PPT envelope will use part of a cylindrical volume up to 4 inches (102 mm) high outside of the spacecraft and centered within the 9 inch (229 mm) inner diameter of the Marmon ring.

An outline drawing in Figure 8 shows the arrangement of the main components of the flight PPT. A circular chassis structure mounted over the spacecraft surface inside the Marmon ring will serve as a combination mounting structure and radiating surface, as well as electrical and thermal conductor for the PPT. Additional radiator surfaces may be thermally connected to the outer edges of the chassis plate as permitted by the Marmon ring. The main structural mount to the spacecraft will be through thermally isolated fasteners mounted through the outside face of the plate into the spacecraft at the edges of the 4 inch by 6 inch opening. Multilayer insulation will be used to blanket the internal components of the PPT to minimize heat transfer to the spacecraft.

Two electrode assemblies will be located symmetrically to either side of the main storage capacitor on the downstream face of the chassis plate. Each electrode assembly includes an anode, cathode, strip line, spark plug, insulator, and expansion horn. The final configuration of the electrode assemblies will be determined from performance optimization testing conducted at OAC and being completed at NASA LeRC. However, the dimensions of the flat plate electrodes are expected to be on the order of 1 inch (25.4 mm) long and wide with a gap of approximately 1.5 inches (38 mm) between electrodes. To save weight in the discharge initiation electronics, only one spark plug per electrode assembly will be used, as was demonstrated on the TIP/NOVA missions.[13]

Two identical propellant feed assemblies house the Teflon™ propellant rods and are mounted directly behind the electrode assemblies inside the spacecraft. The rods will extend fully into the spacecraft to allow the maximum rod length possible. This is necessary to meet the total impulse requirement for the MightySat II.1 mission during solar maximum. The propellant rods will be fed into the electrode assemblies with negator springs in a fashion similar to that of the LES 8/9 units. The main storage capacitor will be centered on the internal side of the chassis plate between the fuel rod feed assemblies. The design must provide high thermal conduction between the capacitor body and the chassis plate to effectively dissipate heat generated in the capacitor. Finally, the PPU will be housed integrally inside an EMI shielded enclosure located behind the chassis and around the fuel feed assemblies.

The present system mass budget for the major components of the PPT is given in Table 1. The fuel weight, a 0.5 kg increase over the LES 8/9 PPT assumes the demonstrated specific impulse of 1200 sec and a total required impulse of 15,000 N-sec in order to meet the orbit raising requirement with a two-sigma worst case atmosphere at solar maximum. The capacitor mass corresponds to a unit designed to provide up to 20 million pulses at 40 J/pulse, a significant mass reduction compared to the LES 8/9 PPT capacitor, especially considering the energy level required to achieve a high thrust to power ratio, and the significant thermal loads imparted by the higher power processing requirements. The weight of the electronics is the actual weight of the developmental electronics, representing a factor of 2 reduction in weight from the LES 8/9 PPT electronics and at least a factor of 4 increase in power density. Other weights are estimates based on the current design concept.

Component	*Mass (Kg)*
Teflon™ Fuel	1.27
Electrode Assemblies	0.5
Electronics	0.8
Capacitor	1.2
Structure	1.0
TOTAL	**4.77**

Table 1: High Power PPT Mass Budget.

The high power PPT system mass is significantly reduced from the LES 8/9 PPT mass of 7 kg, while the total impulse delivered has been doubled, the power capability has more than doubled, and the maximum thrust has increased by a factor of 2.5. Designs for lighter weight, lower power units for other missions are also in work.[21]

The PPT system dissipates thermal energy from three main components: the PPU, the storage capacitor, and the electrodes. The PPU is expected to dissipate 15% of the input power as heat, which is a reduction from the 20% dissipated by the LES 8/9 PPT charge circuit.[17] Most of this power is dissipated from the high voltage transformer and the MOSFET switches. The capacitor dissipates heat due to its effective series resistance. Vondra[29] measured the LES 8/9 capacitor losses calorimetrically and from the integral of the resistive losses during the discharge, and found the capacitor losses to be equivalent to 19 to 21% of the power input into the PPT. Improvements to the present capacitor design are expected to reduce this loss, and preliminary measurements of a new development capacitor have shown this loss to be 17%. Finally, the power loss due to the current attachment with the electrodes is expected to be on the order of 10% of the power into the PPT. Together the three thermal dissipation sources account for approximately 42% of the input power to the PPT. To provide increased thrust for the MightySat II.1 mission, the steady state power levels of the High Power PPT are expected to be over 100 W. The thermal loads to the spacecraft from the PPT are targeted to be less than 5 W. Additionally, it is critical to keep the capacitor temperature below 40°C to preserve its life capability. All exposed surfaces of the PPT will be designed to be radiating surfaces, while minimizing absorption of solar energy. Additional radiating surfaces will be provided on the spacecraft in a location to be determined. Thermal management may be aided by the fact that the PPT will only fire for the portion of the orbit for which MightySat II.1 is in sunlight and will radiate while not operating for the remainder of the orbit.

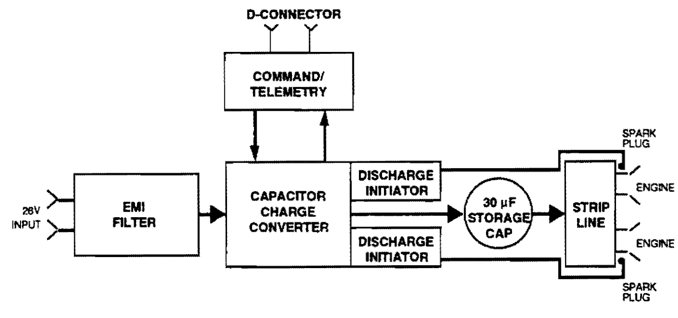

Figure 9: PPT Electronics Block Diagram

Power Processing Unit

The PPU consists of five sections (Figure 9): the EMI filter, the capacitor charge converter, two discharge initiation (DI) circuits and the command/telemetry circuit. The EMI circuit is designed to limit conducted EMI from the charge converter to the spacecraft power supply to levels compliant with MIL-STD-461C. The power interface characteristics are listed in Table 2. The capacitor charge converter is an inductive flyback circuit that steps the input 28 V DC up to as high as 2 kV DC through the use of a pulse width modulator oscillating at 20 kHz. The rectified output of the transformer is directly connected to the capacitor and is short circuit protected. The circuit also allows for complete transformer isolation between the capacitor and the 28 V power supply.

Command/Telemetry Interface

Description	*Voltage*	*Comments*
Capacitor Charge Cmd.	+ 5V	~ 290 ms @ 5 V = 40 J
DI #1 Command	+ 5V	≥40 ms @ 5V
DI #2 Command	+ 5V	fires when go to 0 V
Capacitor Voltage Tlm.	≤ 2V	1000:1
DI #1 Voltage Tlm.	≤ 5 V	200:1
DI #2 Voltage Tlm.	≤ 5 V	200:1
Capacitor Temp. Tlm.	TBD	RTD device TBD
Transformer Temp. Tlm	TBD	RTD device TBD
Propellant Rod #1 Tlm.	TBD	Potentiometer
Propellant Rod #2 Tlm.	TBD	Potentiometer

Power Interface

Description	*Voltage*	*Comments*
Input Power	28V ±4V DC nom.	100-135 W max avg

Table 2: PPT Electrical Interface

The DI circuits provide current pulses to the spark plugs individually. They are powered by a 1 kV tap from the high voltage transformer that charges small storage capacitors. The spark is triggered when one of the Insulated Gate Bipolar

Transistors (IGBT) switch the energy stored in the small capacitors through a transformer creating a high voltage pulse at one of the semiconductor spark plugs embedded in each of the cathodes. By creating a small amount of charged particles between the charged electrodes, the selected spark plug triggers the main discharge across the face of the Teflon™ in the corresponding electrode set.

The analog command and telemetry interface characteristics are listed in Table 2. The charging of the main capacitor is controlled by applying + 5V (high) to the capacitor charge command line. The length of time that the command is high determines the total charge energy. A +5 V discharge initiator command signal to either discharge initiator for at least 40 ms sets that circuit for firing. When the discharge initiator command returns to 0 V (low), it fires the selected spark plug, thereby firing the corresponding electrode set. The telemetry characteristics (Tlm) are also listed in the table.

	Develop. Testing at OAC	Develop. Testing at LeRC	Validation Testing at OAC	Validation Testing at LeRC
Performance	C	C	X	X
Thermal	C		X	
Vibration			X	
Life		X		X
Contamination		X		X
EMI		X		X

C = Completed; **X** = Yet to be done

Table 3: High Power PPT Development Program Testing

PPT Ground Testing and Plume Modeling

Program testing is summarized in Table 3. Testing to date has focused on evaluating possible performance gains. NASA-LeRC is conducting a life test of the development unit and is performing preliminary contamination[28] and EMI studies as well. The qualification effort of the flight design unit will be conducted jointly at OAC and NASA-LeRC, with OAC providing vibration and thermal tests, and NASA providing performance verification, a life test, simultaneous radiated and conducted EMI tests, and contamination evaluations.

Ground tests will be used to evaluate the thruster performance and lifetime, and identify spacecraft integration issues. Thruster performance, including measurements of thrust, propellant usage rate, and system input power, is evaluated at both OAC and at NASA-LeRC to ensure accuracy. Similar procedures will be followed to establish the flight system performance. Thruster lifetime, limited by the energy storage capacitor cycle life, is evaluated using a combination of component level and system level tests. A 1 million pulse capacitor life test has been successfully completed at OAC, and preparations are underway for a 20 million pulse test of the breadboard PPT system. Once the flight hardware is complete it will also undergo a 20 million pulse life test for flight qualification.
Spacecraft integration issues include thermal, mechanical, and electrical interfaces, electromagnetic interference / compatibility (EMI/EMC), and plume impacts. The spacecraft interfaces will be validated at OAC using standard propulsion system qualification approaches. EMI/EMC will be evaluated against MIL-STD-461C at NASA-LeRC using multiple facilities. An array of antennas arranged in a 1 m radius semicircle behind the PPT will be used to characterize the radiated emissions both for a general database which can be used for a wide range of spacecraft and to scrutinize the emissions in selected bands of interest for near-term flight opportunities like MightySat II.1. Radiated electric fields will be measured from 30 Hz to 18 GHz. AC magnetic fields will be measured from 30 Hz to 2 MHz with a near-field loop antenna. Compatibility with the GPS system was previously established using a LES 8/9 PPT by firing the thruster while simultaneously receiving the GPS signals.

Thruster contamination and plume impacts are being evaluated at NASA-LeRC using a combination of direct measurement and modeling. Preliminary measurements have been made using a LES 8/9 thruster to validate the measurement techniques.[28] Collimated quartz samples were used to measure contamination effects in the plume and backflow regions of the thruster. Each collimator had two apertures to limit the impact of contaminants bouncing off the collimator walls, resulting in a sample field-of-view of 22.4°. The collimators were mounted throughout the test facility, with several probes in the plume at different angles to the thrust vector, in the backflow region behind the thruster, and pointing at the wall to measure backscatter from the facility walls. To reduce the effects of the plume scattering from the facility walls, a baffle was placed at the end of the facility opposite the PPT. Future tests include a detailed assessment of contamination from both the development unit and the flight unit.

The ground test measurements of plume impacts will also be used to validate a computational model of the plume to ensure broad utility of the data. The PPT plume model is based on a novel combination of particle methodologies. Neutrals are modeled with a multiple-weight direct simulation monte carlo (DSMC) scheme in order to account for the presence of trace species that may be important for contamination purposes. The plasma is treated via a hybrid electrostatic particle in a cell (PIC) method with fluid electrons and particle ions. The model includes most of the neutral and ion species found in PPT plumes as well as elastic and inelastic collisions between them. The simulation domain includes a large region upstream of the thruster exit in order to assess the backflow fluxes. Inputs to the code at the thruster exit are taken from experimental data and/or internal PPT modeling.

High Power PPT Performance

Performance testing at OAC and NASA LeRC has evaluated the effects of pulse energy and electrode configuration on the Impulse Bit (Ibit), Isp and efficiency of the PPT. Due to the large size of the parameter space and limited test time, Taguchi methods were employed to reduce the number of tests. After an initial series of eight test points, trends were identified and a second set of follow-on configurations based on the trends were tested. Two configurations were then tested on the NASA LeRC thrust stand to verify the performance measurements.

Electrode Length, inches	Electrode Spacing, inches	Flare Angle, deg.	Capacitor Energy, Joules	Energy/ Propellant Area, Joules/ in.^{2}	Impulse Bit, μNewton-Seconds	Specific Impulse, seconds	Average Mass Ablated/ Pulse, grams	Efficiency percent
Data from original Taguchi matrix								
1	1	0	22.0	22.0	300	1000	3.0×10^{-5}	6.5
2	1	0	43.2	43.2	630	1260	5.1×10^{-5}	8.9
1	2	0	43.2	21.6	770	1090	7.3×10^{-5}	9.3
2	2	0	22.0	11.0	320	760	4.3×10^{-5}	5.2
1	1	20	43.2	43.2	690	1040	5.5×10^{-5}	8.5
2	1	20	21.6	21.6	290	920	2.6×10^{-5}	6.1
1	2	20	22.0	11.0	330	1060	4.2×10^{-5}	7.6
2	2	20	43.2	21.6	690	1130	6.7×10^{-5}	8.8
Data from follow-up testing:								
1	2	20	43.2	21.6	710	990	7.3×10^{-5}	7.8
1.5	1	0	43.2	43.2	640	1300	5.0×10^{-5}	9.3
1	1.5	0	43.2	28.8	790	1240	6.5×10^{-5}	11.0
1.5	1	20	43.2	43.2	710	1430	5.1×10^{-5}	11.5
1	1.5	0	43.2	28.8	820	1300	6.4×10^{-5}	12.0
1	1	0	43.2	43.2	650	1180	5.6×10^{-5}	8.5
1	1.5	0	43.2	28.8	760	1140	6.7×10^{-5}	9.1
Data from testing at NASA LeRC								
1	1.5	0	43.2	28.8	770	1180	6.7×10^{-5}	10.3
1	1	0	22.0	22.0	310	n/a	n/a	n/a

Table 4: Results to date from PPT Breadboard Testing

The data from the electrode configuration with a 1 inch length, 1.5 inch electrode spacing, and 0 degree flare angle provides the basis for the MightySat II.1 design. The pulse energy is 43.2 joules. The results for all configurations are summarized in Table 4.

Primarily, the results show that the key performance driver is pulse energy, which has a significant impact on the design of the capacitor and charging circuit. While the effect of the geometric parameters is more complex, and other combinations are still awaiting evaluation, the configuration described above gives the best combination of Isp and high Ibit for the MightySat II.1 mission. The most important parameter for optimization is the thrust to power ratio (T/P) given the limited power available and the mission impact of thrust level. The results to date show promise with a 33% increase in T/P from 12 μN/W for the LES 8/9 PPT[17] to 16 μN/W for the high power PPT.

PPT Flight Operations

In addition to the primary objective of using the PPT to extend the life of MightySat II.1, the flight operations plan has been designed to rigorously test the PPT on-orbit in order to validate the technology for DoD, NASA, and commercial satellites. Although there are certainly challenges to be addressed during the definition of the interface and the resulting flight design, it must be emphasized that all of the issues raised in this section are not expected to be problems. It is anticipated that the MightySat II.1 flight will be a demonstration of PPT compatibility with nominal small satellite design and operations. Specifically the objectives of the PPT flight are to demonstrate PPT performance and lifetime on-orbit, assess the compatibility of the PPT with the spacecraft (i.e. characterize the thermal and EMI environments), and demonstrate compatibility of the PPT plume with general spacecraft surfaces and optical payloads.

In order to accomplish the objectives outlined above, a preliminary flight operations plan is being developed to maximize flight data return. An example flight plan, which assumes a Shuttle launch, is summarized in Figure 10 and consists of several phases of PPT operation. Once the spacecraft is launched and powered-on, it will go through an initialization and checkout process to ensure the deployment was successful. Likewise, the PPT will be powered-on shortly thereafter and checked-out to ensure the thruster survived the launch environment. Once all of the spacecraft systems and the PPT have been checked out, the PPT will perform an initial phase of firings to verify the compatibility of the thruster with the spacecraft bus.

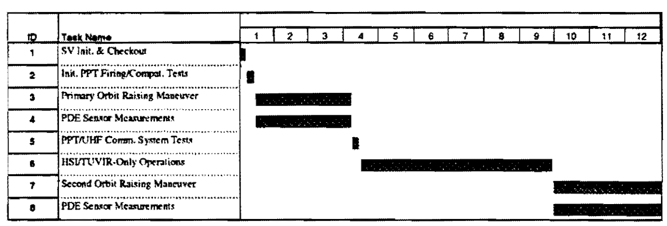

Figure 10: A 12-Month Flight operations schedule

In this example, the PPT will then perform the primary orbit raising mission for MightySat II.1 using the Lift & Coast strategy from the Shuttle altitude. The plan is to complete this phase within 90 days to allow the other payloads time to complete their missions within the one year design life of the spacecraft. Following the orbit raising maneuver, a compatibility test of the PPT and the UHF communications system will be performed. This test is described in more detail below and will consist of a bit error rate test, and some form of uplink and/or downlink integrity test. The PPT is then placed into a low power mode while the optical payloads perform their normal operations. TUVIR will now open its shutter to take data and HSI activates its one-time shutter at this time. Data acquisition phases for these instruments should last 6-8 months. Once they have completed their primary missions, a second PPT maneuver at a much lower power (<50 W) will be performed concurrent with TUVIR and HSI flight operations for a duration of up to 3 months. This operation will further help to characterize the PPT's effect on the optical payloads.

Throughout the MightySat II.1 mission, the PDE will be gathering plume/spacecraft compatibility data as well, starting when the PPT is first powered-on and continuing through the end of satellite life.

PPT Performance and Lifetime Measurements

The PPT orbit raising maneuver on MightySat II.1 provides a unique opportunity to measure the on-orbit performance of the thruster. During this phase of the mission, MightySat II.1 will be positioned such that the PPT thrust axis will be aligned with the spacecraft velocity vector, and the thruster will be fired at a rate of 2-3 Hz. The MightySat II.1 attitude determination and control system (ADACS) will control the firing of each of the two electrode assemblies, to ensure that the average thrust is through the spacecraft center of mass. The need to raise the orbit as fast as possible necessitates the requirement for autonomous operation of the PPT for up to 18 hours between ground contacts.

As a result of the orbit-raising mission, this demonstration will verify that all components of the PPT perform as expected, and do so over a mission life comparable to that of future PPT missions. The flight data will include control signals, temperatures, capacitor voltages, and fuel bar lengths as described in Table 2. PPT thrust performance will be determined by the change in spacecraft orbit. Based on PPT telemetry and satellite tracking data, orbital analysis codes will be used to determine the average thrust. Once the average thrust is known, it can be combined with fuel usage measurements to determine the average Isp and thrust efficiency. The average Ibit and mass per pulse will be determined from the known/measured time history of PPT pulses throughout the maneuver. An examination of the ADACS data, and also the relative number of pulses fired at each electrode assembly, may also yield data on the thrust vector and/or Ibit variation. The operational life will be determined from the total number of pulses fired by the PPT. If the PPT continues to operate throughout the MightySat II.1 operational life as is expected, this flight demonstration will yield a lower bound on PPT on-orbit life. The feasibility of firing only one of the two electrode assemblies in order to determine thruster performance from the response of the ADACS is also being examined. Such a test would serve to determine PPT thrust performance in a fraction of the time compared to the method of altitude change.

PPT/MightySat II.1 Compatibility

Issues of concern to potential users of the PPT system include plume contamination effects, EMI/communication system impacts, and thermal loading on the spacecraft. Although all of these integration issues have been resolved on previous successful flights of PPT systems, a flight demonstration of the advanced PPT on a viable spacecraft is critical for demonstrating this compatibility.

A characterization of the PPT plume effects on all types of spacecraft surfaces, such as optical surfaces, solar arrays, and thermal control surfaces is critical for this demonstration. Flight data is essential for this assessment because facility effects are always present in tests performed in vacuum chambers. The PDE will provide measurements to compare with ground test data and with results from PPT plume models. Analysis of the data will be used to validate the PPT plume model and to verify and/or identify all facility effects.

The high power, short duration plasma discharge associated with PPT operation generates an electromagnetic environment that may impact other spacecraft systems. In addition to monitoring the effects of PPT operation on the spacecraft bus, it is desired to specifically examine the impact of the PPT on the UHF communication system. For instance, the reflection, refraction, and absorption of the carrier wave through the plasma plume may increase the bit error rate (BER) associated with the communication link and/or cause the link to be temporarily lost. Due to the common use of UHF communication systems on small LEO satellites, an assessment of the EMI/communication system compatibility is highly desired. A BER test will be performed on the links between the helical antenna, the omni whip antenna and the ground stations. By altering parameters such as viewing angle and broadcast power, it may be possible to distinguish between the effects of EMI noise input to the transponder and the effect of the plume, and isolate the effects of the PPT on the uplink and downlink. For instance, the downlink can be tested by broadcasting an identical data block twice and comparing the BER with and without PPT operation.

Waste heat generated by a 100-150 W PPT during an orbit raising mission may significantly impact the thermal design of a small satellite. In addition, the temperature sensitivity of PPT components, in particular the capacitor, suggest that the thermal design of the PPT and its interface with the spacecraft are non-trivial. The PPT to be flown on the MightySat II.1 spacecraft will be capable of operating at a steady state power level of >100 W which is greater than twice that ever flight qualified[18], and greater than ten times that demonstrated on-orbit.[15] The demonstration of thermal compatibility with the MightySat II.1 spacecraft will be performed by comparing temperature measurements of critical PPT system components with ground test results and thermal modeling.

Plume Diagnostic Experiment

The PDE is a stand alone experiment on MightySat II.1, operated in conjunction with the PPT, to measure the impact of the PPT plume on the MightySat II.1 spacecraft surfaces. The objectives of the PDE are fivefold; 1. to demonstrate the compatibility of PPTs from a contamination standpoint with current and future DoD, NASA, and commercial small satellite missions, 2. to provide an unambiguous assessment of PPT effects on optical systems, 3. to provide correlation with ground-based PPT plume effects measurements, 4. to provide validation of numerical simulations currently under development, and 5. to develop a low-cost, easily integrated contamination monitoring package.

The PDE consists of two sensor packages each containing a quartz crystal microbalance (QCM) and a calorimeter. A PDE sensor assembly is shown in Figure 11. The two sensors are mounted on a single plate which is mounted to the spacecraft structure with four cylindrical standoffs. These two sensors will collectively provide valuable information regarding contamination and effects of the PPT plume on spacecraft surfaces in terms of material deposited per unit time, as well as the cumulative effect of the deposited material on surface absorptivity and emissivity. The information gathered will be used in conjunction with ground tests to be conducted at NASA-LeRC evaluating the effects of the PPT plume on materials representative of optical surfaces planned for use in the third deep space mission of the New Millennium Program.

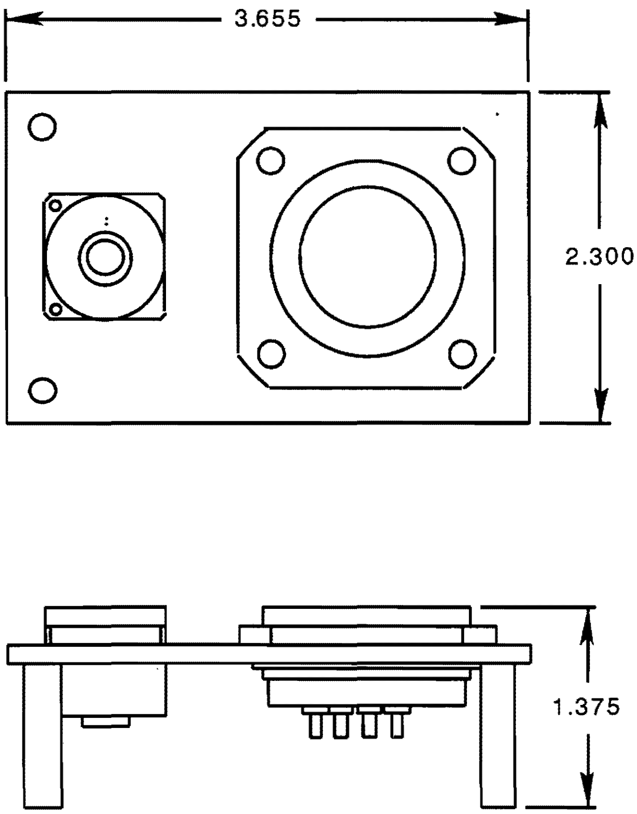

Figure 11. PDE Sensor Panel Assembly (Dimensions in inches)

One sensor panel will be located on the same spacecraft surface as the PPT to characterize the worst-case contamination environment. The other sensor will be located on the opposite side of the spacecraft. Electronics to support signal conditioning, temperature control, analog to digital conversion and serial communication with the spacecraft will be located on a single card. This card is mounted in the MightySat II.1 card cage on which it depends for regulated ±15 V and 5 V. The mass of the PDE including the two sensor panels and electronics is 2 kg.

QCM Description

The operation of a QCM is governed by two piezoelectric quartz crystals which are excited by an external circuit to their resonant frequency ranging from 10 MHz up to 25 MHz. One of these crystals, referred to as the "sense" crystal is exposed to the potential contamination source while a "reference" crystal is enclosed in the housing and protected from any contamination (see Figure 12). The frequency of the crystal is dependent on its mass and hence any coating which accumulates on its surface. As material is deposited on the sense crystal, its frequency decreases and the resulting beat frequency between the two crystals is measured by an external counter. This frequency can then be correlated to the deposited mass.

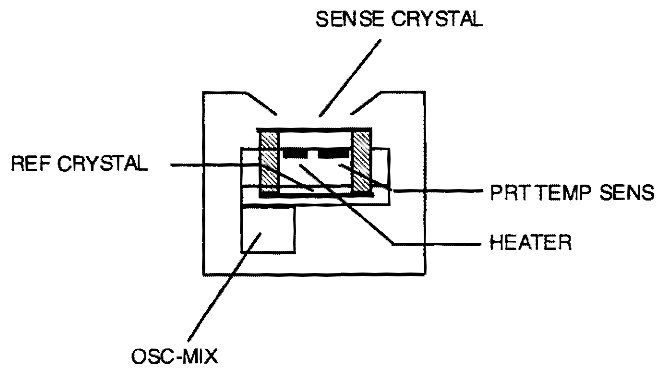

Figure 12: Quartz Crystal Microbalance (QCM)

The QCM is used to correlate mass accumulation as a function of time with any specific events of interest in the mission timeline such as thruster firings. In addition, the sense crystal can be heated to bake off material and clean its surface. This is useful in the unlikely event the crystal is saturated, but can also be used to perform thermogravimetric analysis (TGA). In a TGA, the crystal is heated in a controlled manner such that discrete changes in frequency can be correlated to specific constituents with known vapor pressures. As the sense crystal is exposed to the sun or other thermal sources, however, a thermal gradient is generated which will tend to increase the sense crystal frequency and decrease the beat frequency. This effect can be accounted for in the overall uncertainty of the data, or corrected for if the insolation history is well known.

For the PDE, the QCM sensors will have a crystal frequency of 15 MHz with a maximum mass sensitivity of 1.96×10^{-9} g cm^{-2} Hz. Each unit has a mass of approximately 29 grams, and requires approximately 1.5 W of power for the crystal heaters.

Calorimeter Description

A calorimeter is used to determine the cumulative effects of deposited material on surface thermal properties by accurately measuring the temperature of a surface subject to exposure. A simplified schematic is shown in Figure 13. The two detector surfaces identified as the "disk" and "cup" are thermally isolated from the housing by Kapton™ strips. Platinum Resistance Thermometers (PRT) are used to measure the disk and cup temperatures while minimizing any path for heat leakage to the surrounding structure. While the mechanical design of the calorimeter is relatively simple, the analysis and interpretation of the resulting data can be complex.

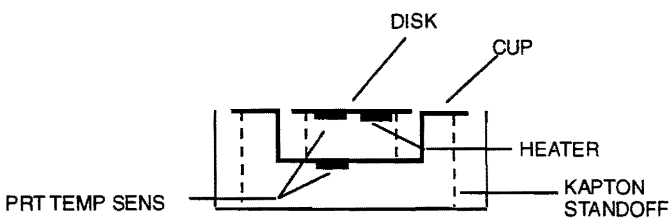

Figure 13. Calorimeter

In order to determine the changes in absorptivity and emissivity of the original coating, it is necessary to have a knowledge of the insolation history throughout the mission. A knowledge of angle with respect to the sun or other warm bodies (such as the earth) within plus or minus one degree is needed. A thermal radiation model for the calorimeter is necessary to relate the measured temperatures to radiant sources as functions of the unknown absorptivity and emissivity and the known physical properties of the disk and cup. In addition, calorimeters will not reach thermal equilibrium during an orbit requiring the analysis models to account for transient effects. To simplify this analysis and reduce uncertainty in the data, it is desirable to have as close to a 2π steradian clear field of view as possible. If this is not feasible, then the thermal radiation model will need to be significantly more complex in order to account for warm spacecraft surfaces with their corresponding view factors before the data can be interpreted. Additionally, if a clear field of view is not available, saturation of the sensor can pose serious problems.

The calorimeter is equipped with a single resistance heater which serves multiple functions. It can be used to bake out the disk and remove deposited material, if necessary, as well as maintain the sensor above the survival temperature of roughly -65 °C. In operation, the heater can be used to heat the disk to a predetermined temperature and then shut off. This is done while the sensor is in the eclipse portion of the orbit. From the rate of decay of the disk temperature it is possible to uniquely determine the emissivity. This information can then be used in conjunction with the heat balance to determine the absorptivity uniquely as well. Each unit has a mass of approximately 40 grams and requires about 1 watt of power for the heater.

Program Status

Shown in Figure 14 is the current (as of August, 1996) schedule associated with the PPT flight demonstration. The flight PPT is scheduled for delivery to Spectrum Astro for

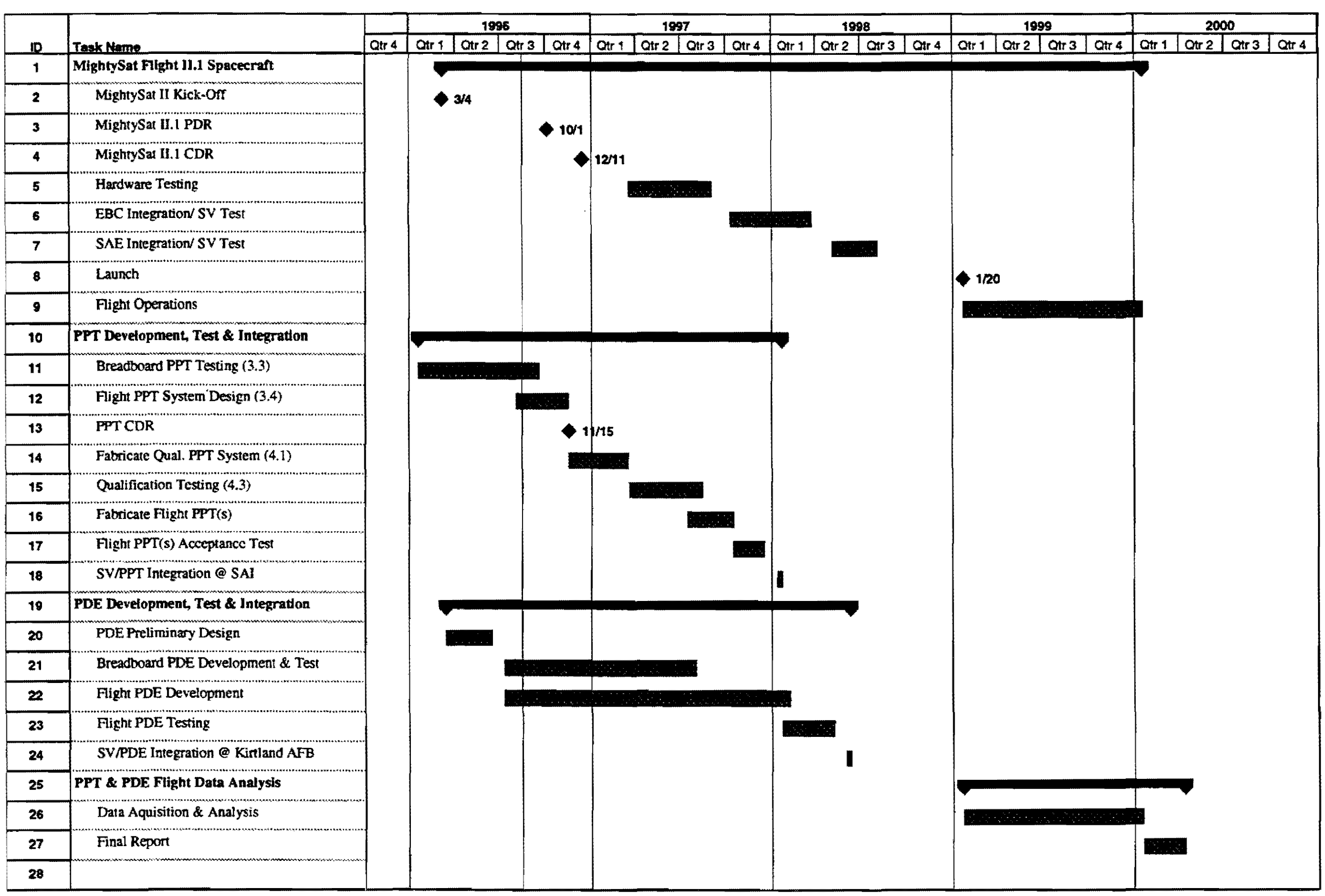

Figure 14: Development schedule

Experimental Bus Component (EBC) integration and test activities in January, 1998. Due to the critical dependence of MightySat II.1 on PPT performance, the current focus of the PPT development effort is to further enhance performance beyond that already achieved. In addition, various aspects of the PPT flight design have been initiated, along with PPT/Spacecraft interface definition activities. The flight PDE is scheduled for delivery to Kirtland AFB for Stand Alone Experiment (SAE) integration and test in May, 1998. The preliminary PDE design is complete, and work has been initiated on the flight design.

Conclusions

The 1999 flight of the advanced NASA/OAC PPT on the Phillips Laboratory's MightySat II.1 satellite represents an ambitious mission to enhance on-orbit life and validate PPT technology for future spacecraft. The advanced PPT to be demonstrated on MightySat II.1 represents a dramatic leap in technology compared to previous flight designs, and will enable the use of the Space Shuttle Hitchhiker Eject System for deployment of long-lived small satellites. The Phillips Laboratory, NASA-Lewis Research Center, and Olin Aerospace Company also have on-going PPT R&D programs to provide even more capable PPT designs for future MightySat II Missions and other small satellites.

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