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# Expanding the Capabilities of the Pulsed Plasma Thruster for In-Space and Atmospheric Operation

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**Abstract**

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Of all in-space propulsion systems to date, the Pulsed Plasma Thruster (PPT) is unique in its simplicity and wide range of operational parameters. This study examined multiple uses of the thruster for in-space and atmospheric propulsion, as well as the creation of a CubeSat satellite and atmospheric airship as test beds for the thruster. The PPT was tested as a solid-propellant feed source for the High Power Helicon Thruster, a compact plasma source capable of generating order of magnitude higher plasma densities than comparable power level systems. Replacing the gaseous feed system reduced the thruster size and complexity, as well as allowing for extremely discrete discharges, minimizing the influence of wall effects. Teflon ( $C_2F_4$ ) has been the traditional propellant for PPTs due to a high exhaust velocity and ability to ablate without surface modification over long durations. A number of alternative propellants, including minerals and metallics commonly found on asteroids, were tested for use with the PPT. Compounds with significant fractions of sulfur showed the highest performance increase, with specific thrusts double that of Teflon. A PPT with sulfur propellant designed for CubeSat operation, as well as the subsystems necessary for autonomous operation, was built and tested in the laboratory. The PPT was modified for use at atmospheric pressures where the impulse was well defined as a function of the discharge chamber volume, capacitor energy, and background pressure. To demonstrate that the air-breathing PPT was a viable concept the device was launched on two atmospheric balloon flights.

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## **DEDICATION**

with appreciation for those who inspire...

## Chapter 1 Introduction

Until the middle of the 20<sup>th</sup> century, physical exploration of space could only be accomplished in our imagination. That changed on October 4, 1957, when the Soviet Union successfully launched the first artificial satellite, Sputnik I. Soviet cosmonaut Yuri Gagarin became the first human in space on April 12, 1961, and on July 20, 1969, the United States placed the first humans on the surface of the Moon. There are currently over 1,000 man-made satellites in space; however only a handful have left Earth's orbit. The necessary propellant required for such missions with conventional thrusters is immense. The cost and mass of this propellant limits our ability to explore areas of space where a wealth of scientific discoveries and natural resources are waiting to be collected by those with the technology to reach them.

- Mars has a geological history, and perhaps even a biological history, of which we have only begun to scratch the surface. Even our most advanced rover, Curiosity, is only capable of traveling a few kilometers over its lifetime. The ability to place a larger payload capable of traversing continental distances is required to examine multiple scientific targets.
- The asteroid belt contains vast amounts of iron, nickel, and magnesium, all of which can be used for the building of structures in space or shipped to Earth to augment our own. The Dawn mission, launched in 2007, is the first to enter orbit around two separate planetary bodies, asteroids Vesta and Ceres.
- Titan and Europa, large moons of Saturn and Jupiter, respectively, are currently believed to be the most likely candidates to hold life within our solar system, beyond Earth, due to possible liquid oceans beneath their surfaces. Cassini, launched in 1997, is the only satellite currently orbiting either planet. Juno, launched in 2011, and the Europa Clipper, scheduled for launch in 2025, will be next.
- The edge of the solar system holds numerous objects believed to have remained frozen since the early formation of the sun. These pieces of rock and ice should contain evidence of the processes that were taking place when our sun was

created, evidence that has elsewhere been lost and could shed light on how the early solar system formed. Only a single spacecraft, Voyager, launched in 1977, has left our solar system's magnetosphere, but has long since run out of the fuel required to perform any more orbital maneuvers.

Over the past five years I have researched two electric propulsion thrusters that may one day allow us to unlock the mysteries that our solar system holds. The High Power Helicon thruster is unique in that the electromagnetic wave produced results in a tightly collimated plasma beam with an extremely high ionization fraction. The Pulsed Plasma Thruster is an inherently simple device, allowing it to be easily applied to multiple mission variants for both in-space and atmospheric operation, as well used as a solid propellant source for the High Power Helicon. These thrusters are more mass efficient than conventional chemical rockets and would allow for more regular access at reasonable prices to a wide range of locations within the solar system.

### **1.1 Advantages with Electric Propulsion**

The concepts covered in this dissertation cover three broad regimes of propulsion:

- (1) Interplanetary travel requiring high  $\Delta V$  propulsion (Chapter 4)
- (2) Small satellites requiring low-moderate  $\Delta V$  propulsion (Chapters 5, 8)
- (3) Airships requiring low-moderate  $\Delta V$  propulsion (Chapter 6, 7)

Chemical propulsion has been the mainstay of interplanetary missions to date. However, unless compensated by a large mass bit, which is typically cost prohibitive, or gravitational assists, which complicate orbital trajectories and restrict launch opportunities, such missions are limited to low  $\Delta V$  transfer orbits. This is due to chemical propulsions low exhaust velocity ( $<5\text{km/s}$ ), which is directly proportional to the attainable  $\Delta V$  as shown by the Tsiolkovsky rocket equation

$$\Delta V = C_e \ln \left( \frac{m_0}{m_f} \right) \quad 1.1$$

which relates the spacecraft velocity ( $V$ ), to the exhaust velocity ( $C_e$ ), spacecraft initial mass ( $m_0$ ), and spacecraft final mass ( $m_f$ ).

The limited maximum velocity lengthens mission durations, which adds significant expenses in the form of extra supplies, radiation shielding, and ground support. For example, in 1997 the NASA exploration study team proposed a manned mission to Mars with a transit time of four to six months [1]. The astronauts would then have to stay on Mars for 18-20 months while the planets realign. Another four months is required for the return flight, resulting in a total mission duration of 879 days. The Juno mission to Jupiter, launched in August 2011, has a substantially more complex orbit that takes advantage of a gravitational assist to reach its destination. The spacecraft went outside the orbit of Mars, reignited the engines in 2012, and flew by the Earth in October 2013, which finally placed it on a course for Jupiter [2]. Scheduled to arrive in July 2016, this totals a five-year cruise phase. Missions moving into the solar system under chemical rockets face similar convoluted trajectories to minimize propellant requirements. For example, the Mercury Messenger mission required a  $\Delta V$  of 29km/s [3]. To accomplish this speed, the spacecraft flew by Earth once, Venus twice, Mercury three times before obtaining the correct orbit around Mercury for a total flight time of nearly six and a half years.

Electric propulsion thrusters offer the potential for vastly reducing trip times by improving propellant utilization with increased exhaust speeds. These systems, however, are typically reduced to low thrust levels, per

$$T = \dot{m}C_e + P_e A_e \quad 1.2$$

where  $T$  is the spacecraft's thrust,  $\dot{m}$  the mass of propellant ejected,  $C_e$  the exhaust speed,  $P_e$  the pressure at the nozzle exit, and  $A_e$  the thruster's nozzle exit cross-sectional area. The low thrust is due to a limit on the available spacecraft power, which reduces the mass of propellant which can be ejected per discharge. Although the force these thrusters produce is extremely small, it can be applied continuously over long time periods, resulting in reduced total propellant mass and/or reduced flight times for interplanetary missions.

Electric propulsion inherits its name from using electrical power to ionize and accelerate plasma particles up to extremely high speeds through interactions with electric and

magnetic fields. Plasma is generally referred to as the fourth state of matter due to the presence of free charges. Externally applied electric and magnetic fields, as well as internally generated fields due to the interaction of the charged particles with each other affect the movements of these charges. This allows plasmas to be capable of organized behaviors over a large volume as opposed to a neutral fluid where each finite element can only interact through collision with its immediate neighbors. The composition of the plasma will vary based on the method of ionization, the atom or molecule that is ionized, and the collisions between the charged particles of the plasma and the remaining neutrals (if any). The plasmas created in the experiments detailed here are typically formed from energetic electrons colliding with neutral gas atoms, with the collision imparting enough energy to break an electron free from the nucleus. These collisions result in a plasma consisting of positive ions, free electrons, and neutral atoms.

As an example of electric propulsion, the ion thruster on the Deep Space 1 satellite had a maximum  $I_{SP}$

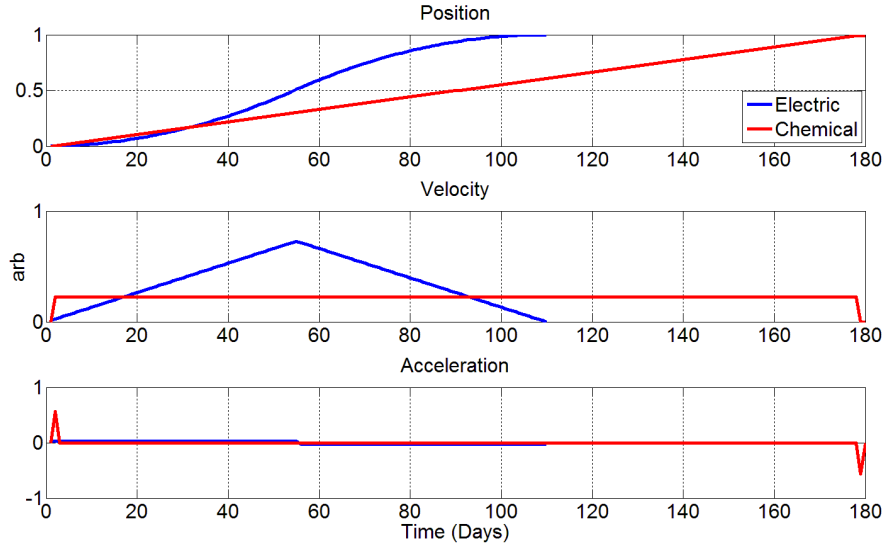
$$I_{SP} = \frac{C_e}{g_0} \quad 1.3$$

of 3200s and a specific thrust of 40mN/kW using Xenon propellant. Solar panels on the craft produced 2.4kWe which enabled the satellite to achieve a  $\Delta V$  of 4.5km/s after a 20-month acceleration, while only using 70kg of propellant over the course of its mission [4]. In 2005, Winglee and Ziemba proposed MagBeam [5], a helicon based beam propulsion device that would allow a manned mission to Mars to be completed in a round-trip of 96 days (50 arrival, 11 stay, 35 return). This 89% reduction in mission time over the Mars mission mentioned above, in addition to the lower propellant mass requirements, would reduce the operational costs by orders of magnitude and significantly increase the likelihood of a successful mission.

### 1.1.1 Chemical and Electric Propulsion 1D Example

As a simplified, 1D example, to explain the difference between chemical and electric propulsion for long duration mission, the position, velocity, and acceleration of a small, but constant acceleration electric thruster and a more conventional high thrust, instantaneous impulse chemical device in a gravity-free, friction-free environment are

shown in Fig. 1.1. The plot was made with an acceleration for the electric propulsion thruster 1% that of the chemical device. As is noticeable, for destinations close to Earth, chemical propulsion provides the more time efficient flight. In contrast, the constant impulse from electric propulsion allows for higher spacecraft velocities if given enough time, making electric propulsion the faster method for long distance missions.



**Figure 1.1. A 1D example of the acceleration, velocity, and position differences between chemical and electrical propulsion.**

With chemical propulsion the specific impulse ( $I_{SP}$ ) is set by the material properties of the chosen fuel. If we define the efficiency of a fixed  $I_{SP}$  thruster as the spacecraft kinetic energy divided by the propellant kinetic energy, and substitute Eq. 1.1 for  $v_f$ , the efficiency is found to be solely dependent on the mass fraction.

$$\varepsilon = \frac{0.5m_f v_f^2}{0.5(m_0 - m_f)C_e^2} = \frac{\ln(m_0/m_f)^2}{m_0/m_f - 1} \quad 1.4$$

Taking the derivative of Eq. 1.4 shows that the maximum efficiency achievable for a chemical thruster is 65% and occurs at a mass fraction of 4.9. Eq. 1.1 states that the maximum velocity possible with a mass fraction of 4.9 is  $\Delta V_{\max} = 1.61C_e$ .

For electric propulsion, the specific impulse is, in principle, a free parameter as the energy in the exhaust comes from electricity that is generated from solar or nuclear

energy. Finding the optimum  $I_{SP}$  at any given time during the mission requires maximizing the energy efficiency by matching the energy in the exhaust to the energy of the spacecraft. There are two energy costs to produce a  $\delta m$  of exhaust, internal energy spent at time of exhaust ( $W_{int} = 0.5\delta m C_e^2$ ), and the cost to bring the propellant up to the speed of the spacecraft ( $W_{speed} = 0.5\delta m v_{s/c}^2$ ). Dividing the impulse produced from the  $\delta m$  of exhaust by the energy required to create that  $\delta m$  results in

$$\frac{I}{W} = \frac{\delta m C_e}{0.5\delta m C_e^2 + 0.5\delta m v_{s/c}^2} \quad 1.5$$

When the derivative with respect to the exhaust velocity is set to zero, the maximum efficiency is found to occur when the exhaust velocity equals the spacecraft velocity at all times, or more formally, when the exhaust is made to be at rest in a stationary reference frame. In this theoretical case, the exhaust is left at rest with no kinetic energy and the thruster achieves 100% efficiency.

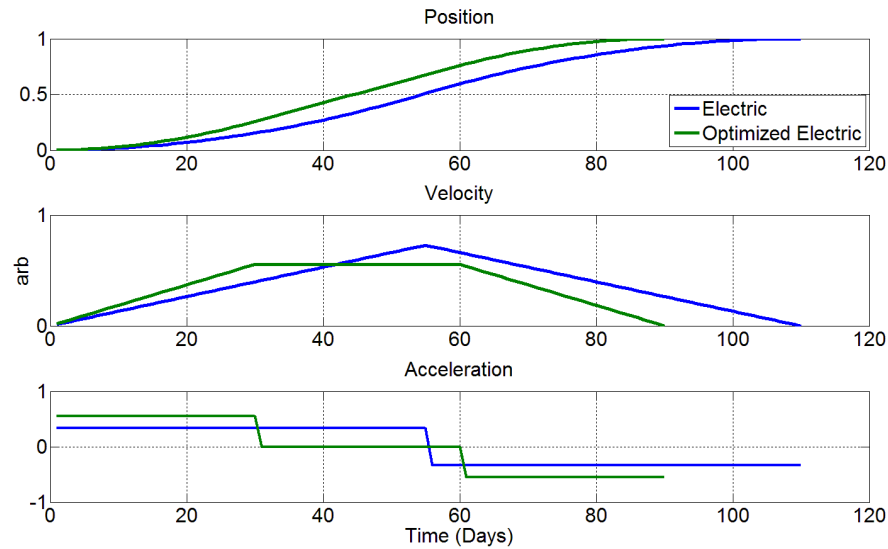
If the  $I_{SP}$  is optimized to match the spacecraft's velocity, the limiting factor with an electric propulsion thruster is the power to mass ratio ( $P/m_f$ ). For the constant acceleration thruster in Fig. 1.1, the duration of a one-way trip can be computed [6] as a function of  $P/m_f$  and the distance ( $z$ )

$$t_{continuous} = \left( \frac{16z^2 m_f}{P} \right)^3 \quad 1.6a$$

In order to minimize the trip time with only a  $P/m_f$  constraint, the use of ejected mass must be maximized. In particular, the acceleration followed immediately by deceleration at the halfway point in Eq. 1.6a is a poor use of propellant. With a set power requirement, higher acceleration is possible when the maximum velocity ( $v_f$ ) is lower. Thus, the optimum use of fuel is to accelerate faster initially ( $t_{acc}$ ), turn the thruster off and drift ( $t_{drift}$ ), and then decelerate at the end. It has been found [6] that the minimum trip duration occurs when  $t_{drift} = t_{acc}$  and is 15% faster than for a continuous burn.

$$t_{optimized} = \left( \frac{27z^2 m_f}{2P} \right)^3 \quad 1.6b$$

The mission profiles for the optimized and constant acceleration electric propulsion thruster for this simplified 1D example are shown in Fig. 1.2.

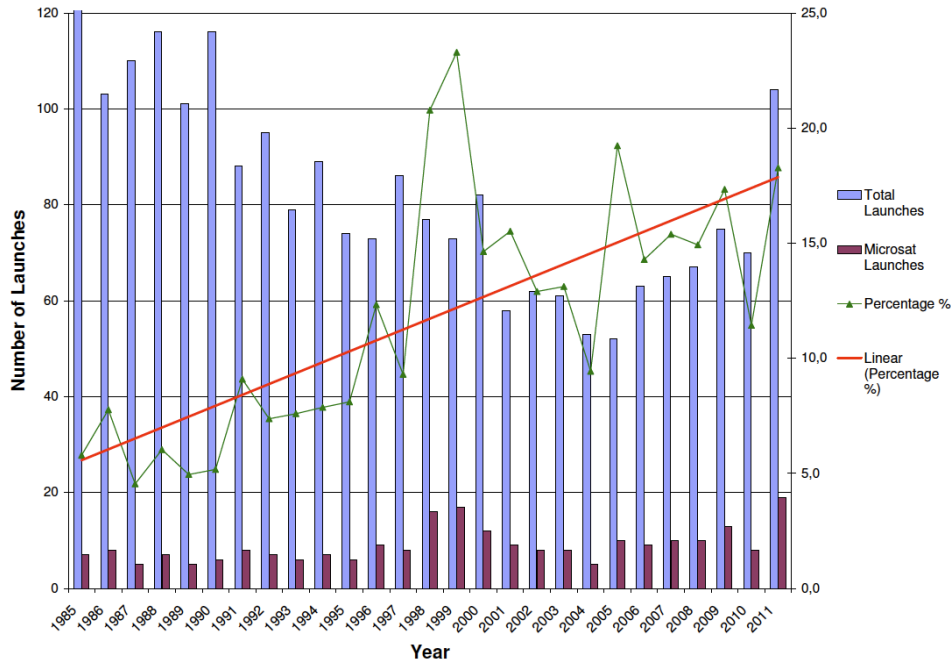


**Figure 1.2. The position, velocity, and acceleration difference for a trip to Mars with a constant and optimized electric propulsion acceleration.**

### 1.1.2 Electric Propulsion for Small Satellites

In addition to being ideally suited for interplanetary trajectories of large satellites due to a high  $I_{SP}$ , electric propulsion's low propellant usage is ideal for long duration position maintenance for miniature satellites. An already increasing trend, enhanced by the failure of the twin Mars probes in 1999 and the Columbia accident, was the implementation of faster, cheaper, and smaller miniature satellites. This increased interest in micro-satellite technologies has caused a departure from the bulky cold gas mono/bipropellant thrusters and conventional reaction-wheel attitude control systems, and replaced them with lighter and more capable electric thrusters serving purposes from attitude-control, drag make-up, to primary propulsion. Fig. 1.3 numerically demonstrates the steady increase of interest in miniature satellites of the last decade. For the purposes of the graph below, miniature satellites are defined as spacecraft with 500kg of mass or less. It is believed to be accurate to within 2 launches every year.



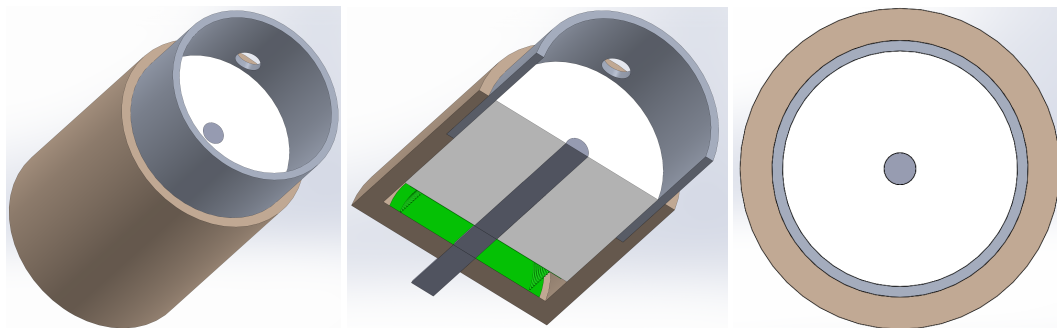


**Figure 1.3. Percentage of miniature satellite (<500kg) launches from 1985-2011 [7].**

## 1.2 The Pulsed Plasma Thruster

The first electric propulsion technology to be flown, the Pulsed Plasma Thruster (PPT), has a successful flight heritage dating back to the 1960's [7,8]. In the last 15 years, severely constrained budgets have driven a wave of satellite miniaturization (Fig. 1.3) that has expanded interest in this technology. The thrusters high specific impulse ( $I_{sp}$ ) per unit power (along with the resulting propellant savings), low mass, simplicity, and robustness make them competitive for low mass and low power applications.

The PPT is a combined electromagnetic and electrothermal device that has traditionally employed a solid Teflon propellant. The thruster is a small, self-contained, inert, and non-toxic propulsion system. PPTs are easily scalable in thrust by varying the discharge energy and firing frequency. The use of solid propellant over pressurized liquids or gasses removes the needs for valves, tubing, and storage tanks. The only moving component of the thruster is a spring to push the propellant rod forward as the front face is ablated away. Simplicity combined with the stability and durability of Teflon makes the PPT an extremely robust and reliable thruster.



**Figure 1.4. The PPTs studied here were all of a coaxial design with parallel electrodes (gray), solid propellant (white), and all encased within a nylon housing (brown). The spring to push the propellant forward is in green.**

There are three primary physics design variants to PPTs, the electrode geometry, propellant feed method, and propellant state of matter.

(1) Electrode Geometry: The electrode geometry can be divided between rail and coaxial designs. For rail electrodes, the propellant is accelerated between two plates, either rectangular or triangular, and are positioned parallel or flared at an angle up to  $20^\circ$ . Coaxial designs are similar to MPD and Arcjet thrusters and can also be parallel or flared.

(2) Propellant Feed Method: Research to optimize the propellant surface area exposed to the discharge has created a number of primary feed mechanisms. The standard is breech-fed where the propellant is injected opposite to the exhaust. This is in addition to oblique, with propellant injected from the sides at an angle  $<90^\circ$ , and side-fed, where the propellant is fed at a right angle to the electrodes. These three methods all use a coiled spring to push the propellant bar forward as it's ablated away and a notch in the electrodes to hold the propellant bar in place. Coaxial PPTs have used an erosion method of propellant injection where the Teflon bar forms part of the walls of the chamber. The Teflon here is not replenished over time but rather eroded over the lifetime of the chamber. In addition to the propellant feed geometries, the surface shape of the solid propellant can vary in three ways. Traditional is a frontal surface, with a single flat surface exposed to the arc. This is in addition to a V-shaped design where two surfaces at an angle between  $<90^\circ$  are exposed to the arc, and a parallel design where two propellant surfaces at right angles to the electrodes are used.

(3) Propellant State of Matter: Both rectangular and coaxial geometries can receive propellant in gas, liquid, or solid forms, with solid being the most common. Solid PPTs have typically used Teflon, polyethylene, and doped combinations of the two; liquid PPTs have been researched with water propellant doped with salts to decrease the plasma resistance, and finally, gas-fed PPTs have typically used ammonia gas [7].

### **1.2.1 PPT Flight and Research History**

PPTs have a flight history spanning over the past five decades. The Soviet Union was the first to develop the PPT as an electric propulsion concept, designing two variations in 1962: one with an increased electromagnetic acceleration and a second with an increased electrothermal acceleration. The latter version was launched on the Zond-2 spacecraft for a Martian flyby mission. These six coaxial, breech-fed PPTs were onboard to provide three-axis attitude control in order to keep the solar arrays facing the sun. Six months after launch, while in transit to Mars, radio communication with the satellite was lost, and with it, control of the thrusters [8].

The first American PPT was designed at MIT's Lincoln Laboratories and flown in 1968 to provide East-West station-keeping capability to the 6<sup>th</sup> Lincoln Experimental Satellite. The LES-6 PPT performed nominally for the entirety of its 10-year mission in orbit [8]. The rectangular breech-fed design to the LES-6 PPT provided the blueprint for the missions throughout the 1970's and 80's. Over the following decades PPT research and funding accelerated due to the growing interest in smaller, lighter, and longer duration missions. The most recent PPT flight was onboard the Earth Observing spacecraft (EO-1), launched in 2001 and still operational today. The thruster provides attitude control for the spacecraft without negatively impacting the spacecraft bus or onboard scientific instruments. EO-1, weighing in at only 5kg, is capable of providing 860 $\mu$ N of thrust with 9.8% efficiency at 56J [9].

In the early 2000's, the Aeronautics Department at the University of Washington, in partnership with the Aerojet corporation, developed the Dawgstar satellite, part of the ION-F (Ionospheric Observation Satellite Formation) nanosatellite program. Using PPTs, the Dawgstar was the only satellite in this program to develop propulsion capability. The Dawgstar PPT was capable of providing a thrust-to-power ratio of 8.8mN/kW and a

specific impulse of 482sec at 5J [10]. Since then the Earth and Space Science Department, in conjunction with Eagle Harbor Technologies, Inc. have investigated a micro pulsed inductive thruster ( $\mu$ PIT). It was tested as a means to increase PPT propellant utilization while maintaining a low overall mass in order to provide a new alternative to the current generation of micro-propulsion technologies.  $\mu$ PIT is capable of providing a specific thrust of 5 mN/kW and specific impulse of 2000s at 400mJ [11].

### 1.2.2 PPT Operation

The operation of the PPT, with only two capacitor banks and no moving components is inherently simple. The basic physics of the PPT consists of three main events: ignition, discharge, and acceleration. A large voltage difference (V) is created between the anode and cathode (the electrodes which form the discharge chamber) with the main discharge capacitor of capacitance C. A small igniter capacitor is charged to a moderate voltage, before a switch is closed allowing current to flow through the primary side of a step-up transformer. The secondary side of the transformer ( $\sim 20\text{kV}$ ) creates a spark that generates an initial seed plasma. This plasma expands between the cathode and anode, completing the main discharge circuit and allowing the energy in the main discharge capacitor ( $E_c = 0.5CV^2$ ) to discharge as an underdamped RLC circuit as through a wire with stray resistance (R) and inductance (L), described commonly as

$$I_c = \sqrt{\frac{2E_c}{L}} \exp\left(\frac{-Rt}{2L}\right) \sin\left(\sqrt{\frac{1}{LC} - \left(\frac{R}{2L}\right)^2} t + \phi\right) \quad 1.7$$

The resulting radial current between the electrodes heats and ablates the propellant surface, creating a high-pressure cloud of neutrals, free electrons, and positively charged ions. This cloud is accelerated both electromagnetically and electrothermally. The radial current sheet (I), flowing over a length l, generates a self-induced azimuth magnetic field (B), which accelerates the plasma particles through the Lorentz force.

$$F_z = I \int dL_r \times B_\theta \quad 1.8$$

The high temperature and pressure heats the gas within the discharge chamber and then expands it out the nozzle to convert its thermal energy into a jet of directed kinetic energy, generating thrust. If a 1D, adiabatic, constant specific heat expansion through the

discharge chamber is assumed, the attainable exhaust speed can be found from a simple energy balance of the exit speed ( $C_e$ ), the chamber speed ( $C_c$ ), the exit temperature ( $T_e$ ), the chamber temperature ( $T_c$ ), and the gas heat capacity ( $C_p$ ).

$$\frac{1}{2}C_e^2 = \frac{1}{2}C_c^2 + C_p(T_c - T_e) \approx C_p T_e \quad 1.9$$

Flight PPTs typically ablate 1-3 $\mu$ g/J of propellant per discharge [8, 12]. However, the ionization fraction for Teflon fuel cell PPTs is typically 10% or less, with the majority of that being singly ionized carbon or fluorine [13]. There is disagreement over what doubly ionized particles exist in statistically important quantities [14, 15, 16, 17]. The entirety of the plasma is produced while the main discharge capacitor discharges ( $\sim 100\mu$ s). However, the hot temperature in the discharge chamber can last for much longer, boiling off neutral particles from the propellant long after the current stops flowing (millisecond time scales) [18, 19]. These late time ablation (LTA) macroparticles have high mass but extremely low velocity, leading to negligible thrust. LTA is the primary reason for the low mass utilization efficiency and hence the low thrust efficiency of the PPT, the primary drawback of the thruster.

### 1.3 Helicon Plasmas

Helicon plasma sources have been an active area of research since the 1980s [20, 21] due to their ability to produce order of magnitude increases in plasma densities over standard inductive methods. Helicons ionize a gas by passing a bounded right-hand circularly polarized electromagnetic wave along an axial magnetic field. They belong to the category of whistler waves, which are magnetically unbounded and commonly found in nature. Helicon experiments operate over a wide range of parameters, including a variety of power levels, magnetic field strengths, neutral pressures, and magnetic field geometries. Helicons usually receive 0.2-4kW from an RF power supply and operate with strong magnetic fields on-axis (10-1000's of Gauss). Typical propellants include hydrogen, helium, nitrogen, and argon, although other gases have been used as well.

Helicon sources for terrestrial applications, such as plasma processing and etching [22, 23], source plasmas for fusion devices [24, 25, 26], and in wave-physics experiments [27, 28] have been an active area of research. Helicon thrusters have also received interest as

in-space propulsion systems [29, 30, 31, 32]. Small size, low magnetic field strengths, high plasma density, and high ionization efficiency are all qualities that make the helicon an ideal thruster candidate. The produced thrust can be increased with either a physical or magnetic nozzle [33, 34] that converts perpendicular into parallel velocity. Recent work [31, 35] has characterized how the plasma detaches from the magnetic fields in such a nozzle. Helicon research for thruster applications have evolved in two distinct directions:

- 1) A helicon source to generate plasma with and then a second mechanism to accelerate the plasma
- 2) A helicon source as the thruster itself

The first category includes the Variable Specific Impulse Magnetoplasma Rocket (VASIMR), which utilizes a helicon source to initially ionize the propellant and then additional heating downstream through ion cyclotron resonance to expel the ions out the thruster nozzle. Recent results show a 72% efficient 200kW thruster capable of 5.8N at an  $I_{SP}$  of 4900sec. A flight test of the 200kW thruster is planned onboard the International Space Station [36, 37].

The High Power Helicon (HPH) experiment of this thesis is one example of where only a helicon source is used. Recent results have shown that altering the magnetic field profile downstream of the antenna can extend the current drive region and result in a denser beam with a higher bulk velocity [38]. A second experiment in this category is the Helicon Double Layer Thruster (HDLT) based on the discovery of a current-free double layer in helicon plasma. Recent experiments have produced a 3mN thruster with only 700W of RF input power [32]. There are two helicon experiments researching low power devices. HPH.com, with funded from the EU, is looking at a 50W micro-Helicon thruster with hydrazine propellant. Current modeling has shown plasma expansion and detachment in a magnetic nozzle and the formation of current-free double layers [31]. While at Michigan, the CAT thruster is a 50W helicon designed for CubeSat operation [39].

To date, research with helicon systems, similar to most high-powered electric thrusters, has involved gas-fed systems. A gaseous propellant requires a complex system of tubes, valves, and tanks that are all under high pressure. A thruster that eliminates the flow of

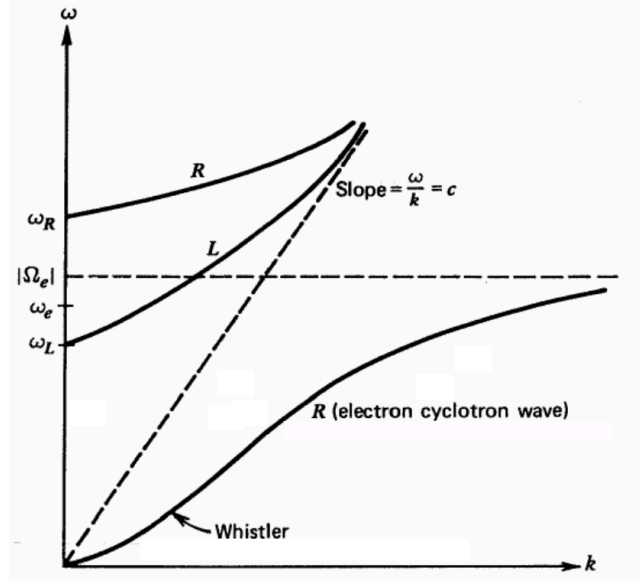
propellant would negate this extra effort, resulting in a device that is lightweight, inexpensive, and reduced in complexity. The slow propagation of gas or liquid through control systems results in an operation that is not well suited for a pulsed system, which is required if wall effects are to be eliminated from the analysis. To date, no literature can be found on helicon thrusters operating on non-gaseous propellants.

### 1.3.1 Helicon Operation

The helicon wave is an electromagnetic wave that propagates along the magnetic field direction in a magnetized plasma. Specifically, they create a whistler wave, which is one of several types of electromagnetic waves that travel parallel to the magnetic field in a magnetized plasma as shown in Fig. 1.5. Helicons operate by oscillating electric and magnetic fields at a time rate ( $\omega_{field}$ ) that is fast compared to the ion cyclotron frequency ( $\Omega_i$ ) but slow compared to the cyclotron frequency of the electrons ( $\Omega_e$ ) and the electron plasma frequency ( $\omega_{pe}$ ).

$$\Omega_i \ll \omega_{field} \ll \Omega_e \ll \omega_{pe} \quad 1.10$$

In this regime the electrons will undergo the whole gyration with an effectively constant background field and will therefore move and drift in a similar fashion as if the field was constant. For the ions, the fields oscillate at a rate that is fast compared to their cyclotron frequency. Here the ions will not have time to move a significant distance in response to the field before the field changes direction and therefore the field will have no net effect on their motion. The ability to inductively drive fields that can accelerate the electrons without affecting the ions and without the use of electrodes or grids is a key characteristic of helicon plasma sources.



**Figure 1.5. Dispersion diagram for parallel electromagnetic waves [40].**

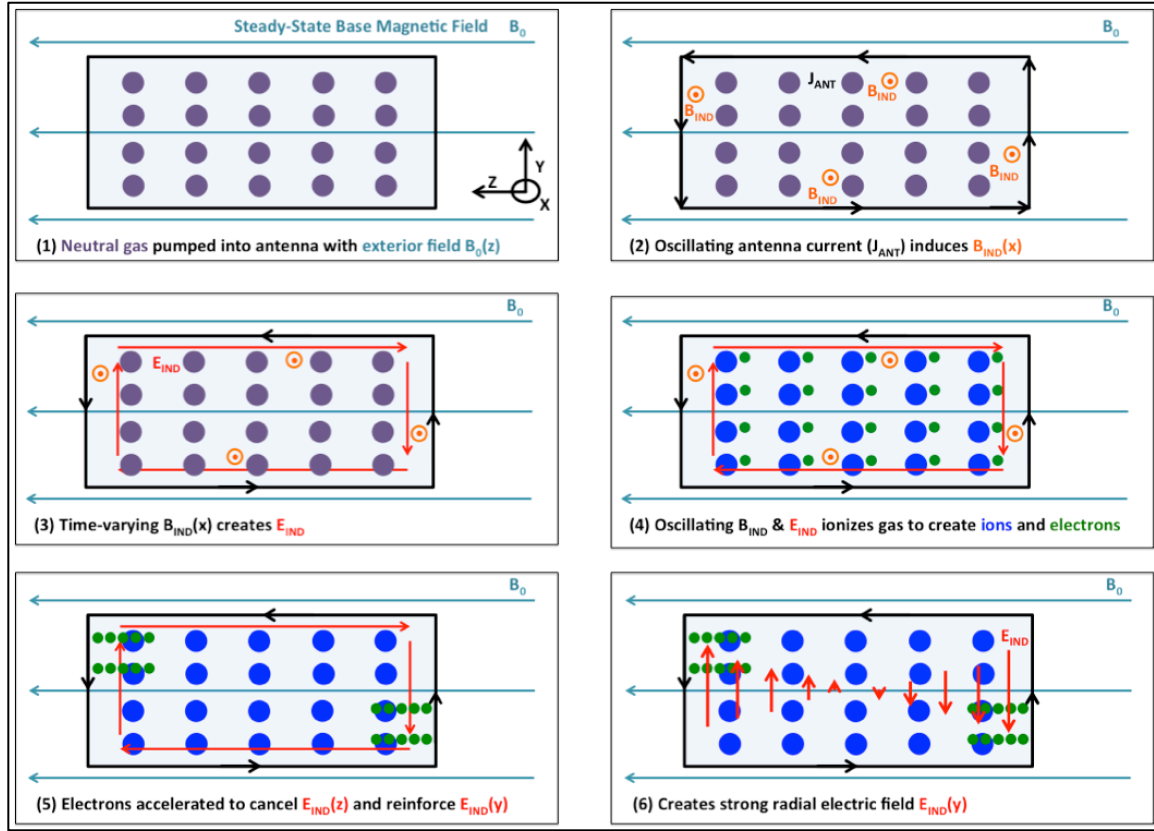
The helicon antenna is designed to generate radial electric and magnetic fields that couple into a wave, which propagates through the plasma. These antenna fields only line up with the plasma wave fields over a limited timeframe and position, but with enough power can overcome the damping mechanisms and generate a helicon wave that can propagate through the source region and downstream for multiple wavelengths before damping out.

In the simplest case, the antenna can be constructed as a single loop of wire that runs along the top of an insulating tube, splits in two to run down opposite sides of the tube, travels back along the bottom, and then splits in two again to run up the sides and meet back at the beginning. A six-step process for this antenna configuration is shown in Fig. 1.6 and 1.7. As current flows through the antenna wires, a magnetic field ( $B_{IND}$ ) perpendicular to the base magnetic field is induced. In this example a coordinate system such that  $x$  points out of the page,  $y$  vertically, and  $z$  axially along the base magnetic field has been chosen.

The induced field ( $x$  direction) rising and falling with the oscillating current in the antenna induces an electric field inside the tube and opposite to the direction of the antenna current. This initial system of the antenna current, induced electric field, and base field is shown in steps (2) and (3) in Fig. 1.6. Due to their heavy mass, the ions remain relatively motionless, while the electrons in the plasma are accelerated by the induced electric field until the charge difference is enough to cancel the axially induced electric

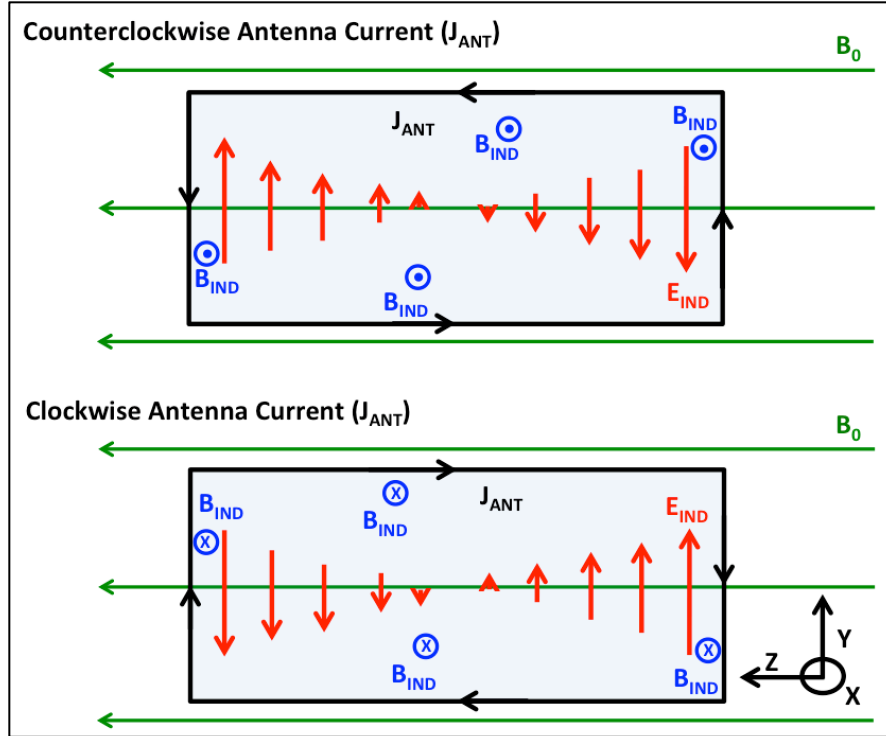


field (step 5). The base magnetic field prevents cross-field movement of the electrons, leading to a vertical electric field that reinforces the induced electric field across the tube.



**Figure 1.6. The six-step process to produce the radial electric and magnetic wave fields from antenna current is shown.**

The end result from the oscillating antenna current is a radial electric field ( $y$ -direction), perpendicular to the base magnetic field, that is strongest near the ends of the antenna and switching sign in the middle. Additionally, there is a radial magnetic field ( $x$ -direction) that does not switch sign in the middle. Both of these fields are shown in Fig. 1.7 for a counterclockwise (top) and clockwise (bottom) antenna current. The loop antenna in this configuration will drive waves along the base magnetic field, in both the positive and negative  $z$ -directions.



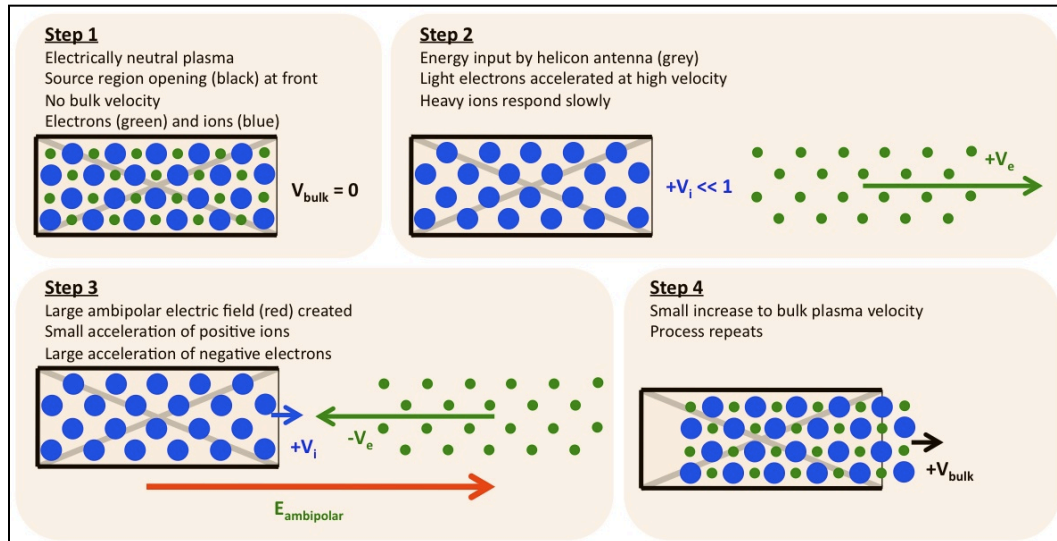
**Figure 1.7. The radial electric and magnetic wave fields generated for a counterclockwise (top) and clockwise (bottom) antenna current.**

In order to better match a wave propagating parallel to the base field, a left-handed, half-turn twist is made in the antenna. This re-aligns the induced electric field to coincide in space with the wave electric field of the left-handed helicon wave. This electric field rotates in space with the same handedness of the helicon wave, but does not rotate in time the same way that the helicon wave does. The wave field of the antenna is able to couple into the helicon wave mode for only a limited time, but can still effectively drive the helicon wave.

Previous work has shown discontinuities in the plasma density as the magnetic field and RF power input are increased [41]. These jumps are believed to be due to how the antenna couples into the plasma. At low power and field strengths the antenna couples capacitively to the gas near the source region walls to form a weakly ionized plasma. The density here is too low to satisfy the helicon dispersion relation and the wave cannot propagate. At moderate power and field strengths the mode of coupling has switched to inductive. Here the changing magnetic fields produce large electric fields within the plasma and the density begins to increase. Once the magnetic field and antenna power have been increased to a point where the density is able to satisfy the helicon dispersion

relation, the source will couple to the helicon wave and the density will jump again as the wave ionizes the remaining neutrals. This last regime is where the HPH thruster is designed to operate.

In general terms, the helicon wave is an electromagnetic perturbation of the base magnetic field  $B_0$ , with wave components  $B_w$  and  $E_w$  perpendicular to  $B_0$ . These wave components twist around  $B_0$  in a helix and also rotate in time at the driven frequency  $\omega_{\text{field}}$ . The wave component is oscillating too fast for the ions to respond, but the electrons are gyrating around the magnetic field ( $B_0 + B_w$ ). Thus, when the wave magnetic field vector rotates in time, the electrons are driven around  $B_0$  as well. This rotational velocity for the electrons is perpendicular to the wave magnetic field  $B_w$ , allowing the Lorentz force to accelerate the electrons along the base magnetic field  $B_0$ . The rotational velocity is also perpendicular to  $B_0$ , so there is an additional component of the accelerating the electrons towards the centerline anti-parallel to  $B_w$ . Since the helicon wave is accelerating the electrons and not the ions due to their mass difference, the plasma build up a strong electric field that acts to accelerate the ions downstream while decelerating the electrons. This is the opposite effect to the plasma leaving an ion or Hall effect thruster. This particular type of electric field is referred to as ambipolar.



**Figure 1.8. Ambipolar electric field acceleration of ions by a helicon source.**

The wave has electric and magnetic components oscillating along the guiding magnetic field, with the magnetic field component of the wave given by

$$B_{wave} = B_{wave,0} \exp[i(k_{\parallel}z + m\theta - \omega t)] \quad 1.11$$

where  $k_{\parallel}$  is the parallel wavenumber of the propagating wave along the z-axis,  $m$  represents the mode of the wave ( $m=1$  for HPH), and  $\omega$  is the angular frequency of the wave. As stated above the angular frequency is chosen so that the ions are relatively motionless over one period of the wave oscillation while the electrons undergo many gyro-orbits in the same timeframe. The parallel wave number is determined by the plasma parameters and boundary conditions imposed on the wave. Downstream, where there are relatively few boundary conditions, the parallel wavenumber is identical to the free propagating whistler wave observed in nature, described by its dispersion relation

$$k_{\parallel} = \frac{\omega_{pe}}{c} \sqrt{\frac{\omega}{\omega_{ce}}} \quad 1.12$$

where  $\omega_{pe}$  is the electron plasma frequency. As additional radial boundary conditions are imposed such as a varying magnetic field, varying plasma density, or a physical boundary, the behavior of the wave deviates from the unbounded whistler wave. In the source region of HPH, where a pyrex tube imposes an insulating boundary, the parallel wavenumber is given by the helicon dispersion relation

$$k_{\parallel} = \frac{\omega_{pe}^2}{c^2} \frac{\omega}{\omega_{ce} k_{\perp}} \quad 1.13$$

where  $k_{\perp}$  is the perpendicular wavenumber established by the radial boundary conditions. Equations (1.12) and (1.13) can be expanded to give relations for the plasma density and phase velocity, defined as the frequency normalized by the parallel wavenumber.

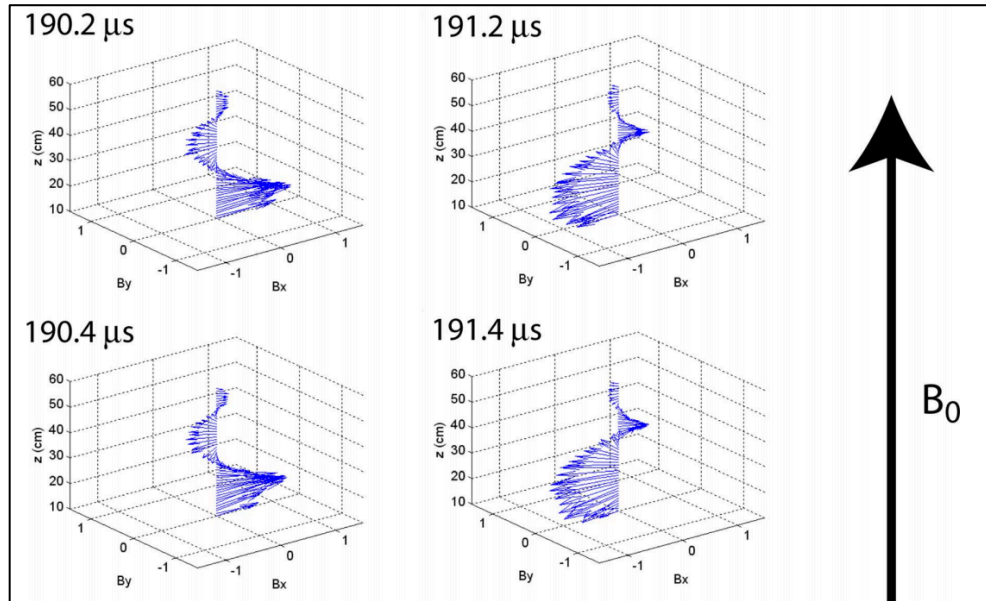
|               | Whistler (a)                                              | Helicon (b)                                                 |      |
|---------------|-----------------------------------------------------------|-------------------------------------------------------------|------|
| Density/Field | $\frac{n_e}{B} = \frac{2\pi}{e\mu_0} \frac{1}{\lambda f}$ | $\frac{n_e}{B} = \frac{3.83}{e\mu_0 r} \frac{1}{\lambda f}$ | 1.14 |

|                |                                                                          |                                                                      |      |
|----------------|--------------------------------------------------------------------------|----------------------------------------------------------------------|------|
| Phase Velocity | $v_{ph,\parallel} = \frac{\omega_{ce} c^2}{\omega_{pe}^2} k_{\parallel}$ | $v_{ph,\parallel} = \frac{\omega_{ce} c^2}{\omega_{pe}^2} k_{\perp}$ | 1.15 |
|----------------|--------------------------------------------------------------------------|----------------------------------------------------------------------|------|

The density over field relation being a constant and inversely proportional to the frequency is a defining property of helicon plasmas. Although a lower antenna frequency would lead to higher densities, the plasma velocity scales linearly with frequency and

would therefore decrease. Higher magnetic fields lead to increases in both the density and velocity. This implies that higher frequencies and higher base magnetic fields are ideal for high thrust operation.

Previous work [42] has determined that the HPH generates a helicon wave within the source region (Eq. 1.13), however downstream in the diverging magnetic field regime the parallel wavenumber more closely matches that of the whistler wave (Eq. 1.12). With propagation of the wave parallel to the base magnetic field  $B_0$  along the  $z$ -axis, Eq. (1.11) suggests the wave component of the field should decrease in  $\theta$  as the axial distance increases, leading to a counter-clockwise (left-handed) rotation around the  $z$ -axis with position. Eq. (1.11) also indicates that as time increases,  $\theta$  should also increase leading to a clockwise (right-handed) rotation about the  $z$ -axis with time. Both of these characteristics were experimentally observed by measuring  $B_{\text{wave}}$  with the results shown below for four separate times in Fig. 1.9. This magnetic component of the helicon wave only exists when there is a base magnetic field and plasma to sustain it.



**Figure 1.9. Measured  $B_x$  and  $B_y$  (arbitrary units) on the axis of the helicon.  $B_0$  points downstream in the  $z$ -direction [42].**

An identifying feature of the helicon is the requirement for an externally imposed guide magnetic field. This field in the source region serves multiple purposes for the production, confinement, and acceleration of the plasma. However, it also raises the concern of plasma detachment, which, without, would lead to zero net thrust as the

exhaust would follow the magnetic field lines back to the spacecraft. Multiple mechanisms for detaching the plasma exhaust from the field lines has been well documented and explained previously [43], a short summary of which will be given here.

The relationship between the plasma particles and the background field is given by the plasma beta ratio, which relates the thermal pressure of particles with density ( $n$ ) and temperature ( $T$ ) to the magnetic pressure of the background field ( $B$ ).

$$\beta = \frac{nkT}{B^2/2\mu_0} \quad 1.16$$

In highly magnetized field regions ( $\beta < 1$ ), the gyroradius of the electrons (and possibly the ions) is small compared to the scale size of the system and the plasma will flow along the magnetic field lines. As the field strength decreases downstream, and  $\beta$  transitions to a value greater than one, the thermal pressure of the plasma will dominate and the plasma will expand outward, dragging the magnetic field along with it. This is accomplished by currents flowing in the plasma inducing a magnetic field, which modifies the existing field. The large diamagnetic perturbations found downstream of the high power helicon creates a high beta plasma, which detaches the plasma particles from the field lines and allows the exhaust to flow away from the spacecraft.

If the thruster only relied on a high electron temperature and density to detach the plasma from the field lines, it would simply result in a thermal expansion of the plasma rather than a directed flow. In this case, most of the exhaust kinetic energy would be angled compared to the direction of motion and not useful in generating directed thrust. The ratio of the directed kinetic energy to the magnetic pressure is defined as the Alfvén mach number

$$M_a^2 = \left(\frac{V_{\parallel}}{V_a}\right)^2 = \frac{n_i m_i V_{\parallel}^2}{B^2/2\mu_0} \quad 1.17$$

which is dominated by the high mass ions. Similarly to the plasma beta ratio, if the directed kinetic energy of the plasma dominates over the background magnetic pressure, the plasma can be expected to detach from the magnetic field lines. For the helicon experiment, exhaust velocities  $>12\text{km/s}$  have been measured, which allows the plasma to

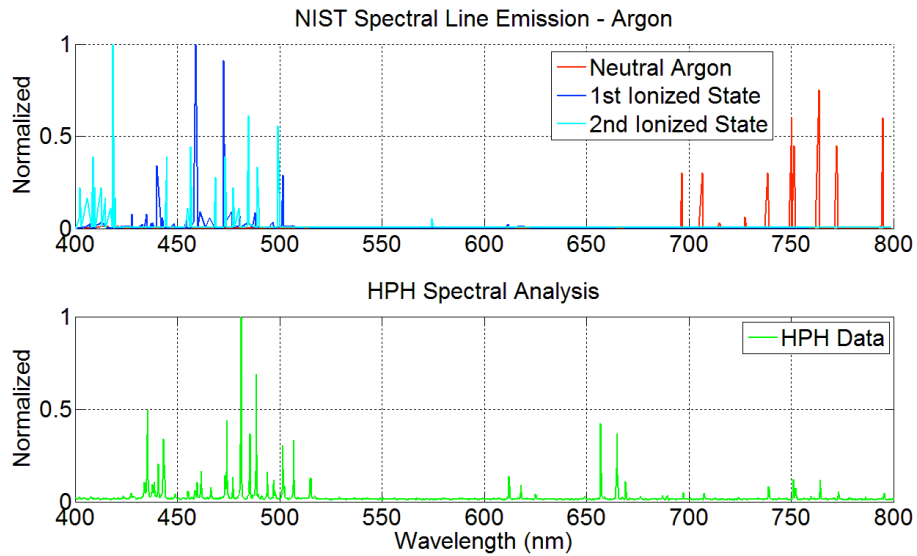
transition to a superalfvenic plasma as it progresses into the lower field strengths downstream.

Lastly, plasma particles can become detached from the magnetic field through collisions. As particles are interrupted in the course of their gyro-orbits, they will be able to move perpendicular to the magnetic field even when the field strength is high enough to keep the orbits small compared to the scale length of the magnetic field. For the HPH thruster, the source region has the highest field strengths, but also is the region with the highest collision frequencies. Looking at the source region, if the ions and electrons are both assumed to be 10eV and existing as a highly ionized gas (ion-electron coulomb collision frequency of  $\sim 90\text{MHz}$ ), the ion gyrofrequency is 100kHz, while the ion-neutral and ion-ion collision frequency are both 200-500kHz. In this regime the ions will be unmagnetized as they are unable to complete a single gyro-orbit without multiple collisions. This is compared to the electron gyrofrequency of 10GHz, which suggests that the electrons undergo multiple gyro-orbits between collisions, resulting in the bulk of the electrons to be magnetized in the source region.

Although it allows the electrons to remain attached to the field lines over a longer distance, extending the magnetic field downstream through the use of magnetic nozzles allows the helicon wave to exist for longer timescales within the plasma and propagate further. Modeling [44] and experimental results [45] have both shown a magnetic nozzle system, positioning downstream of the plasma source such that the plasma transitions to superalfvenic in the throat of the nozzle results in a significant exhaust velocity increase and further collimation of the plume downstream. The increased downstream velocity and self-confinement of the beam are both defining properties of helicon plasmas.

The ionization fraction, a percentage of charged particles to neutral particles, will depend on the plasma energy input against the various loss mechanisms. The primary loss for the helicon thruster is recombination, with the positive ions capturing free electrons and neutralizing. In this experiment, the High Power Helicon has an ionization fraction over 90% for Argon propellant [43], while previous studies have shown the Teflon Pulsed Plasma Thruster to have an extremely low ionization fraction of  $<10\%$  due to large, solid macroparticle creation [19]. The helicon ionization fraction was determined by

comparing relative height of spectrometer readings collected with the HPH thruster to given results for neutral and ionized Argon (Fig. 1.10). The bulk of the emission spectra from the gaseous HPH thruster is between 400-500nm, the same wavelengths expected to be seen for ionized Argon. The only detected neutral argon emission lines were found at 750nm and are an order of magnitude smaller in amplitude than the emission lines found for the ionized species, suggesting an ionization fraction above 90%. The high ionization fraction of the helicon results from the ionizing electrons being directly accelerated by the wave-particle interaction rather than through a random heating process [46].



**Figure 1.10. Spectrometer results for the Argon High Power Helicon (green) compared against NIST references for neutral and ionized Argon [43, 47].**

In summary, the helicon thruster produces thrust through a three-step process: efficient propellant ionization, electromagnetic electron acceleration, and lastly electrostatic ion acceleration. The helicon begins by effectively ionizing the neutral gas through wave-particle interactions. The electrons are accelerated out of the source region electromagnetically, while initially leaving the ions motionless (Eq. 1.10). The electrons remain attached to the magnetic field, building up a large ambipolar electric field (Fig. 1.10), which electrostatically accelerates the ions. Momentum is transferred to the spacecraft as the ions pull on the electrons, which are interacting with the oscillating antenna field and the plasma interaction with the guide magnetic field.



## Chapter 2 Experimental Apparatus

This section briefly describes the experimental apparatus that has been built up as a result of previous research in the laboratory [48, 49, 50, 51] as well as going into depth on the hardware and electronics built specifically for the thrusters researched here. Detailed explanations for the atmospheric launch vehicles and CubeSat satellite design can be found in Chapters 7 and 8, respectively.

### 2.1 The Vacuum Environment

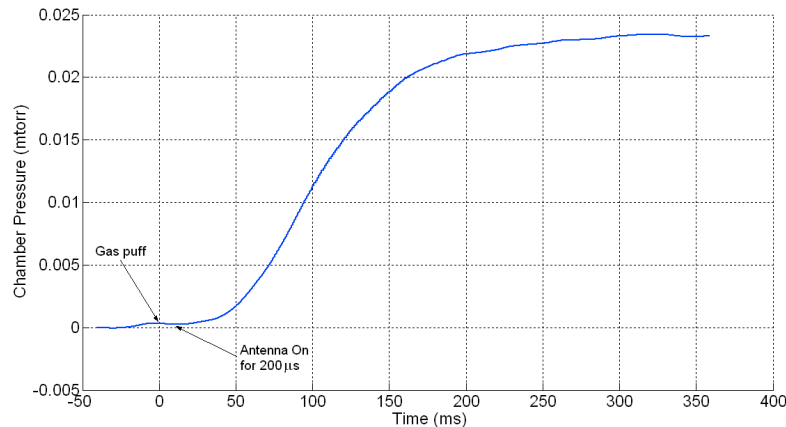
The test facilities at the APL have the ability to simulate pressure environments ranging from Earth ground (760Torr) to space-like (1 $\mu$ Torr). The lab is not equipped to simulate varying temperature environments. Thermal pressure ( $P = nkT$ ) in the interplanetary medium is extremely low as the density in this region is only  $\sim 1$  particles/cm<sup>3</sup>, higher as you move closer to the sun, planet, or other object and lower as you move away. A better representation of the vacuum environment may be the mean free path, defined as the average distance a particle travels between successive collisions with other particles. The mean free path is function of the neutral gas density ( $n$ ) and the average particle cross-sectional area ( $\sigma$ ).

$$l = (n\sigma)^{-1} \quad 2.1$$

Assuming an average molecule diameter of 0.4nm, the mean free path on Earth's surface is 100nm and increases to  $>10^5$ km in the interplanetary medium. No vacuum system on Earth can fully create a true interplanetary vacuum environment. However, when the mean free path is much larger than the size of the chamber, the interior is assumed to be collisionless over the length of the chamber. The results presented in this study were carried out at pressures ranging between 40Torr (approximately 70,000ft altitude in Earth's atmosphere) and 1 $\mu$ Torr. Assuming nominal room temperatures, the mean free path at 40Torr is on the order of a micrometer, much smaller than the thruster's characteristic size and therefore the system is highly collisional. At microTorr background pressures the mean free path is  $>100$ m, much larger than the size of chamber and therefore considered to be space-like with no background particle collisions before the plasma interacts with the wall.

The APL's small bell jar is 44cm in diameter and 100cm in length, capable of measuring high altitude atmospheric pressure (10-40Torr) as well as a vacuum environment ( $1\mu\text{Torr}$ ). There are 10 quick-disconnect openings at the base of the chamber for diagnostic probe access. The exact shapes of these probes vary depending on the orientation of the thruster (horizontally or vertically mounted). The chamber is vacuumed with a roughing and 42krpm turbomolecular pump. A  $2\mu\text{m}$  dust trap was placed between the chamber and the pumping system to prevent PPT ejecta from damaging the turbopump blades.

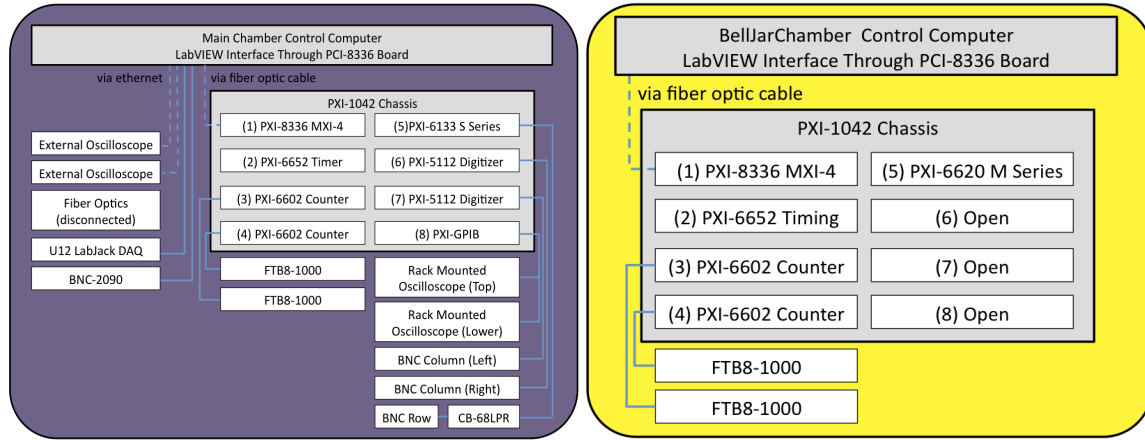
The APL's large vacuum chamber is 4800L in size, with an average diameter of 1.5m and length of 2.5m. The HPH thruster would be mounted at one end and fire towards the other with double Langmuir probes and Retarding Field Analyzers (RFAs) protruding into the chamber radially through quick-disconnects. The chamber is vacuumed by a roughing pump and turbomolecular pump capable of maintaining a base pressure of  $2\pm 1\mu\text{Torr}$ . The chambers pressure change from a gaseous helicon discharge is shown in Fig. 2.1 measured by a capacitive manometer (MKS Baratron) located at the far end of the chamber from the thruster. In this figure the 10ms duration gas puff starts at 0ms while the  $200\mu\text{s}$  duration antenna is triggered 9ms after the gas. The pressure does not begin to rise significantly until  $\sim 20\text{ms}$ , considerably longer than the thruster duration.



**Figure 2.1. The chamber pressure as a function of time after a 10ms gas puff [50].**

The majority of the background hardware in the laboratory has remained similar to previous experiments, with the exception of the LabVIEW interfaces and fiber optic triggering systems. Running multiple experiments in the main vacuum chamber as well

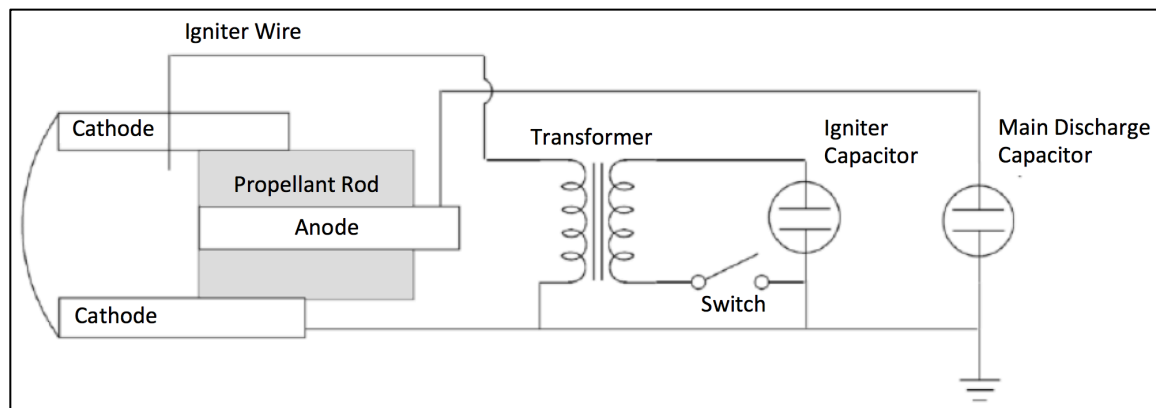
as high frequency repeated discharges in the belljar required new hardware and software. The top-level schematics for the LabVIEW based triggering and data collection hardware built up over the course of these projects are presented in Fig. 2.2 for future reference.



**Figure 2.2. Top-level overview of the fiber optic and data acquisition system for the laboratories main (top) and belljar (lower) vacuum chambers.**

## 2.2 The Pulsed Plasma Thruster

The traditional coaxial pulsed plasma has two electrodes, two discharge circuits, a propellant rod, and a spring (the only moving component). The design follows the traditional setup with the exception of the spring. As we haven't been concerned with long duration testing as of yet and such a small amount of material is ablated with each discharge, there has been no need to advance the propellant rod forward.

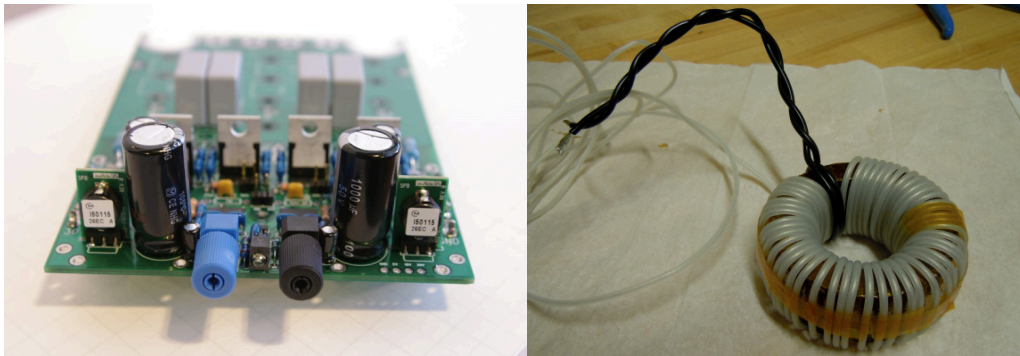


**Figure 2.3. The circuit diagram for the coaxial PPTs tested in this study.**

The outer electrode (cathode) is grounded to the chamber well and the center electrode (anode) is held at a high positive potential. For the alternative propellant designs the electrodes were machined out of Al6062 due to its low copper content in order to prevent

reactions with sulfur. The Teflon fuel cell designs were machined out of copper, except when directly compared to an alternative propellant, in which case they were machined from aluminum. These three components determine the shape and size of what is called the discharge chamber, the area of highest electrothermal and electromagnetic acceleration. The exact shape and size of the electrodes and fuel vary between the different experiments undertaken. These dimensions will be specified in the relevant sections.

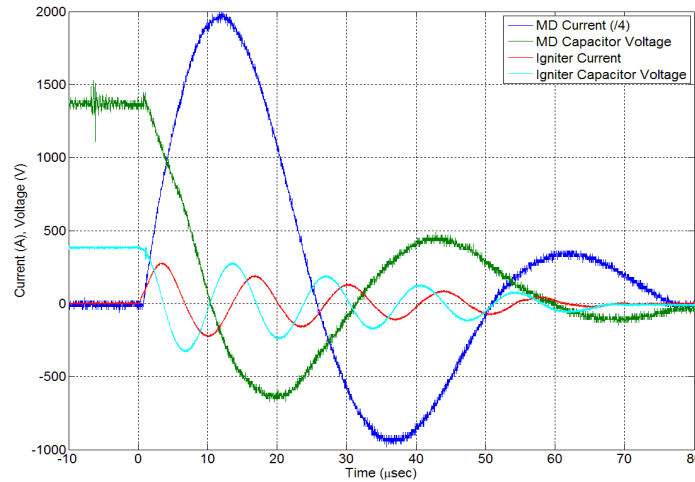
For all designs, a small hole is drilled through the cathode, through which the igniter wire is fed. The laboratory transformer is a 2:100 step-up toroidal design with an iron core. The primary side of the igniter circuit consists of a 1.5 $\mu$ F capacitor charged to 500V and an IGBT circuit board. When the IGBTs are fiber optically triggered, they close for 1ms, allowing the igniter capacitor to fully drain and the current to flow as an underdamped RLC circuit per Eq. (1.10) on both sides of the transformer for  $\sim 60\mu$ s. The current flowing through the primary of the transformer peaks at 400A, while the secondary current is less than 8A.



**Figure 2.4. The four IGBT, fiber optically triggered igniter circuit control board (left) and the 2:100 step-up, iron core toroidal transformer (right).**

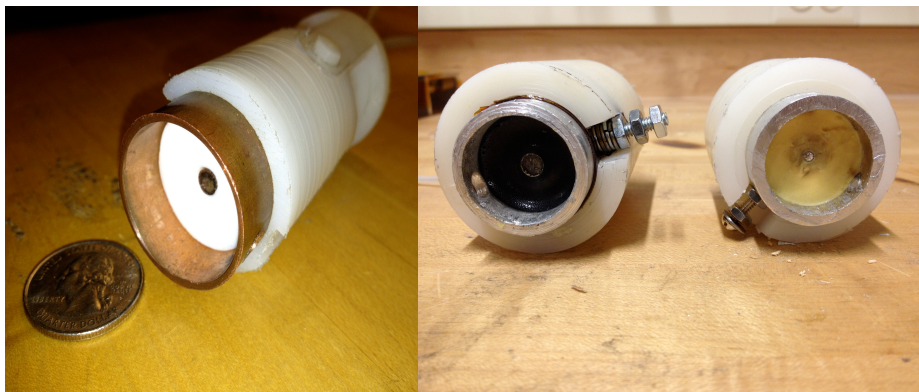
The main discharge circuit is simply a charged capacitor attached to the anode and cathode. Two variations on the capacitor were used for the experiments in this research project, a 60 $\mu$ F / 4kV version and a 33 $\mu$ F / 1.4kV design, with the later typically operated with two capacitors in parallel. Based on the individual configuration, the main discharge circuit had 1.6 - 3 $\mu$ H of stray inductance and 200 – 800m $\Omega$  of stray resistance. The inductance of the system was varied in Chapter 4, while the temperature variations in Chapter 7 modified the resistance of the circuit. The typical laboratory current and

voltage traces are shown in Fig. 2.5 for the main and igniter circuits. The main discharge current peaks at 8kA for a 66J discharge and with nominal inductance and resistance values, damps out in less than 100 $\mu$ s.



**Figure 2.5. PPT current and voltage traces for the main and igniter circuits.**

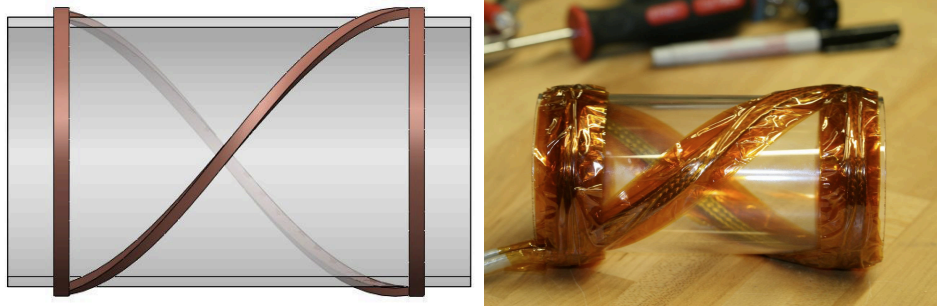
The majority of the electromagnetic acceleration occurs in the first 100 $\mu$ s. This is period when the highest energy plasma particles are created and accelerated out of the discharge chamber. The electrothermal acceleration occurs over a much longer time period, with light emission continuing out for 3ms. This long duration activity is referred to as Late Time Ablation (LTA) and is responsible for boiling off large mass, slowing moving neutral particles, none of which contribute significantly to the thrust in vacuum operation. In atmospheric operation the electrothermal acceleration heats up the ambient air within the discharge chamber to roughly the sound speed. Even with velocities only on the order of hundreds of m/s, the large mass of the ambient air results in high thrust levels.



**Figure 2.6. Teflon / copper (left), Bi<sub>2</sub>S<sub>3</sub> & Sulfur / Al6062 PPT designs (right).**

### 2.3 The Solid Fuel High Power Helicon Thruster (Internal System)

The helicon system is based around a tuned resonant RLC network to flow high currents and high voltages through an antenna. The antenna was constructed by wrapping braided copper wire around the outside of a quartz tube, with the current splitting to run in parallel down the sides of the tube. Each coil was wrapped with two turns and Kapton insulating the layers, in order to increase the antenna inductance, which increases the magnitude of the wave. The antenna is 15cm long and 7cm in diameter.



**Figure 2.7. The left-handed antenna design used in the experiment.**

A tuned resonant RLC network (Fig. 2.8) gets its name from its three components, the inductor (L), capacitor (C), and resistor (R). A network of this type forms a harmonic oscillator which acts to increase the voltage and current flowing through a load. Harmonic oscillators are characterized by the gain (Q factor) of the resonant circuit, given by Eq. (2.2), and the resonant frequency at which the circuit oscillates, Eq. (2.6).

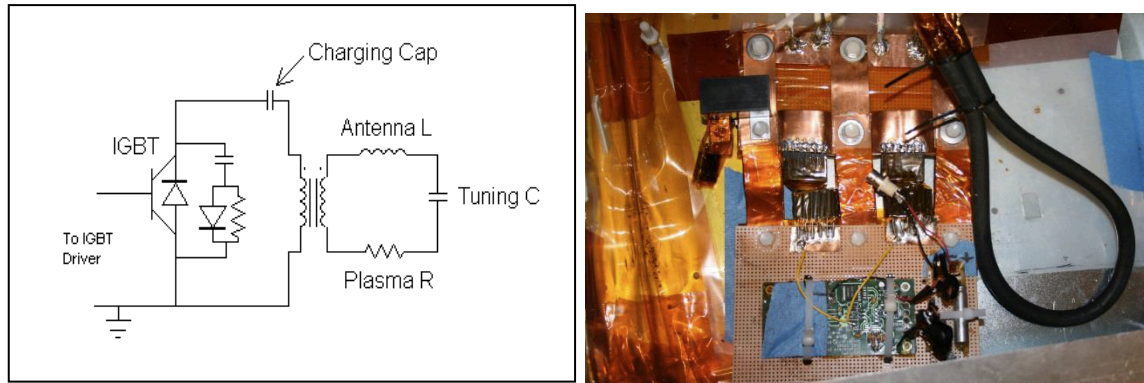
$$Q = \frac{1}{R} \sqrt{\frac{L}{C}} \quad 2.2$$

A higher Q value corresponds to a narrower bandwidth, which leads to higher power levels on the load. Formally, Q is a ratio of the power stored to the power dissipated in the circuit and gives an approximation for how high the voltage from the charging capacitors will be stepped up before flowing over the antenna.

The power supply takes advantage of high power solid state devices called insulated gate bi-polar transistors (IGBTs) that can switch high currents on and off on a timescale of tens of nanoseconds. The RF power supply used is shown in Fig. 2.8. The black chips are the IGBT bank and diodes that behave as the high power switch, with the copper strip-line to carry the current. The green board is the IGBT driver, which receives the fiber



optic pulse and provides the voltage to the gate pin of the IGBT. The large black loop is the 1:1 transformer that connects the primary and secondary sides of the circuit. The power input into the antenna with this supplies was measured to be in the tens of kilowatts [50].



**Figure 2.8.** The circuit schematic of the LC network with the antenna operating as the inductor (left) and the IGBT board composing the power supply (right).

Fiber optic pulses from the control computer triggers the IGBT driver circuit to close/open the bank of 12 IGBTs on the primary side of the transformer. When the IGBTs switch close, current flows out of the charging capacitor bank, through the 1:1 transformer and the IGBT bank. When the IGBTs open again there is a voltage spike across the switches associated with the rapid decrease in current. This spike is kept low with a snubbing circuit (capacitor, resistor, and diode) to allow for higher charging voltages. With the IGBTs open, current then flows in the opposite direction through the diode bank that is in parallel with the IGBT bank, running the opposite direction through the transformer and re-charging the capacitor bank.

On the secondary side of the transformer, the helicon antenna is put in series with a capacitor bank, forming a resonant LC tank network. These two components act as an electrical resonator, storing energy oscillating at the resonant frequency. A capacitor stores energy in its electric field, depending on the voltage across it, while an inductor stores energy in its magnetic field, depending on the current through it. With a charged capacitor connected across an inductor, charge will start to flow through the inductor, building up a magnetic field and reducing the voltage on the capacitor. Eventually all the charge on the capacitor will be gone and the voltage across it will research zero. However, the current will continue to flow as inductors resist change. The energy to keep

the current flowing will be extracted from the magnetic field, which will begin to decline. The current will begin to charge the capacitor with a voltage opposite in polarity to its original charge. When the magnetic field is completely dissipated the current will stop and the charge will again be stored in the capacitor, with the opposite polarity as before, and the process will start over. The charge will flow back and forth between the plates of the capacitor, through the inductor until the stray resistance of the circuit make the oscillations die out. For this research the energy in the circuit is replenished by the charging capacitors, allowing for a constant peak to peak amplitude sinusoidal current waveform.

By Kirchhoff's voltage law the voltage across the capacitor plus the voltage across the inductor must equal zero ( $V_C + V_L = 0$ ) and likewise the currents must be equal ( $I_C = I_L$ ) per Kirchhoff's current law. From the constitutive relations for the circuit elements, it is known that

$$\text{Inductor:} \quad V_L = -L \frac{dI_L}{dt} \quad 2.3$$

$$\text{Capacitor:} \quad I_C = C \frac{dV_C}{dt} \quad 2.4$$

Combining together results in the 2<sup>nd</sup> order differential equation

$$\frac{d^2 I}{dt^2} + \frac{I}{LC} = 0 \quad 2.5$$

The resonant angular frequency can therefore be defined as

$$\omega_0 = \frac{1}{\sqrt{LC}} = 2\pi f_0 \quad 2.6$$

Setting the phase angle to zero, the resulting solution to Eq. (2.5) becomes

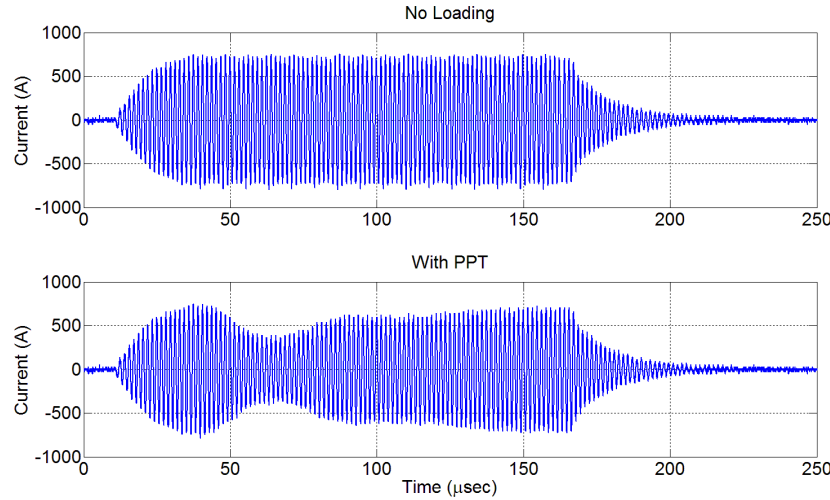
$$I(t) = I_0 \cos(\omega_0 t) \quad 2.7$$

$$V(t) = -\omega_0 L I_0 \sin(\omega_0 t) \quad 2.8$$

This experiment used a tuning capacitance of 36nF and antenna inductance of 1.80μH, resulting in a frequency of 625Hz. The circuits resistance was measured to be 180mΩ, resulting a gain of 439 As the IGBTs are opened and closed, pulses of current are driven through the primary side of the transformer at the resonant frequency of the circuit,

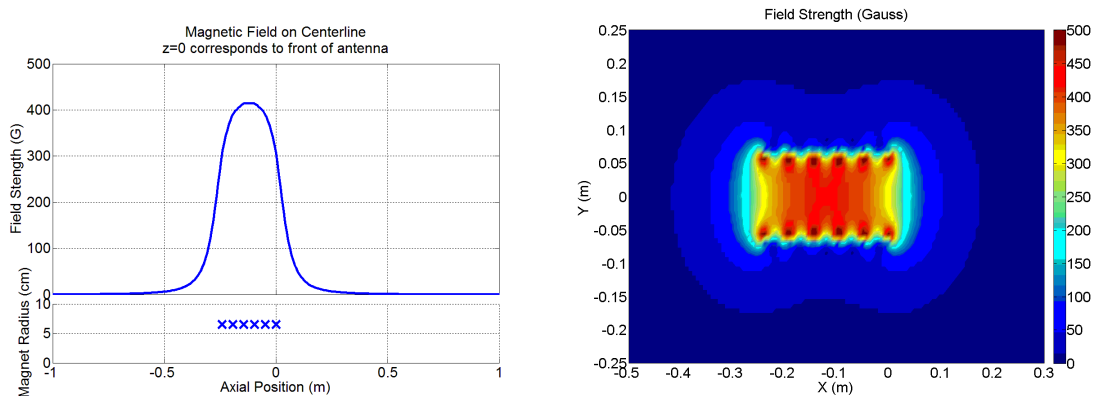


leading to a large oscillating current at this frequency in the secondary side of the circuit. With no plasma present the resistance in the circuit is low and the oscillating current through the antenna builds to  $\sim 1.5\text{kA}$  peak to peak, as shown in Fig. 2.9, and oscillating voltage to  $\sim 12\text{kV}$  peak to peak. The current oscillates sinusoidally at the antenna frequency until the circuit stops being driven on the primary side by the computer.



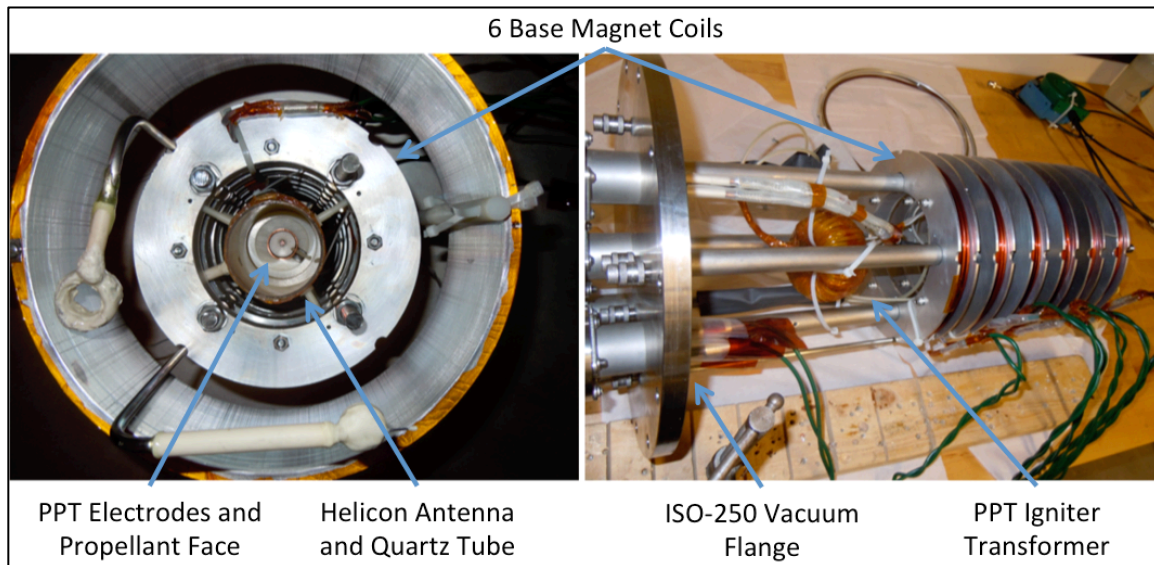
**Figure 2.9.** The oscillating current in the antenna in a vacuum (top) and when it loads over the plasma and neutrals created by the PPT (lower).

When the antenna is fired with the PPT, the antenna will ionize the neutral particles and accelerate them out, drawing energy from the secondary side of the LRC network within the antenna. The plasma acts as an additional resistance in the circuit, placing a load on the antenna. The oscillating current drops as there is more plasma present, followed by increasing again as the plasma is ejected from the source until the antenna is shut off.



**Figure 2.10.** Profile of the centerline axial component of the base magnetic field (left) and the XY plane field strength (right) for the interior magnet system.

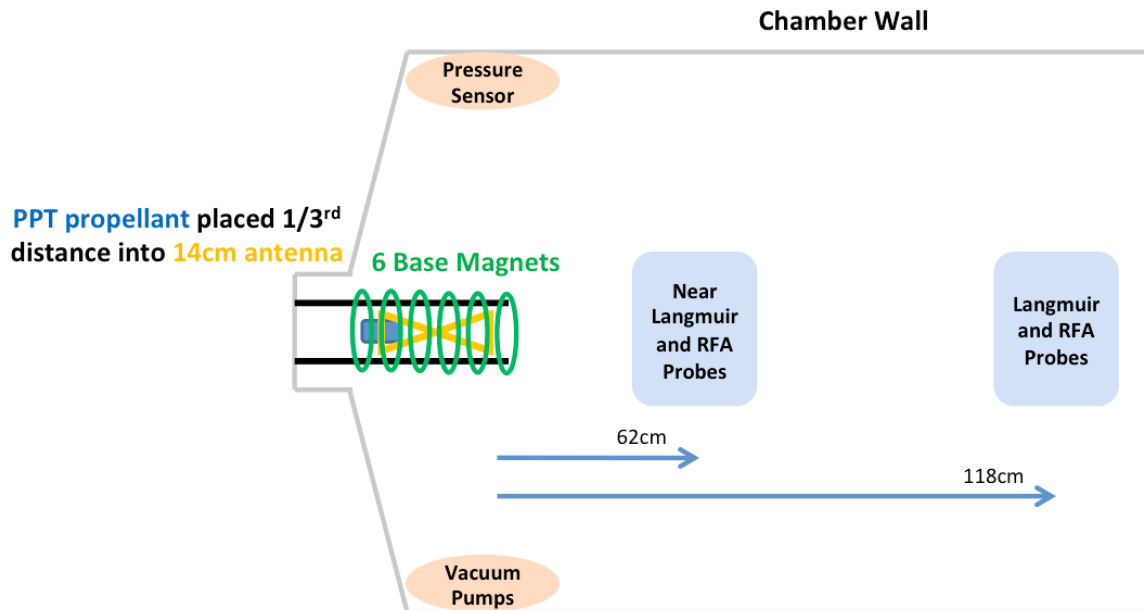
The base magnetic field is generated by six magnetic coils in series (Fig. 2.11), each 15cm in diameter and spaced with 3cm gaps, lined up along the thruster axis. This was done to provide a relatively uniform field through the antenna, while having the field fall off as a dipole once outside the source region. The axial component of the magnetic field is shown in Fig. 2.10. The strength of the field peaks at 400G, roughly 33% of way into the antenna from the back end and varies by less than 20% over its length. While most helicon experiments provide a uniform magnetic field, both under the antenna and downstream, an in-space thruster will use a dipole field as no structure can be placed downstream. Plasma here will experience a decreasing and diverging field as it is ejected away from the spacecraft. The base field falls off to  $\sim 0.5\text{G}$ , the background terrestrial field strength, within 1.5m.



**Figure 2.11. The base magnets, helicon antenna, PPT electrodes and propellant, and igniter transformer positioned in the chamber (left) and on the work bench (right).**

To integrate the PPT within the HPH experiment, all thruster electronics, with the exception of the igniter transformer were housed outside the chamber for ease of access. The igniter transformer was zip-tied to the helicon support structure and all the electrical wires were fed through the flange directly behind the thruster. The coils for the base magnets were bolted directly to the inside of the flange on metal support rods, while the quartz tube which the antenna was wrapped around was supported with nylon bolts from the magnets. The PPT was press-fit to the inside of the quartz tube. The flange had four

custom made electrical feed-throughs to handle the high current of the PPT igniter and main discharges, the helicon antenna, and the base magnetic field.

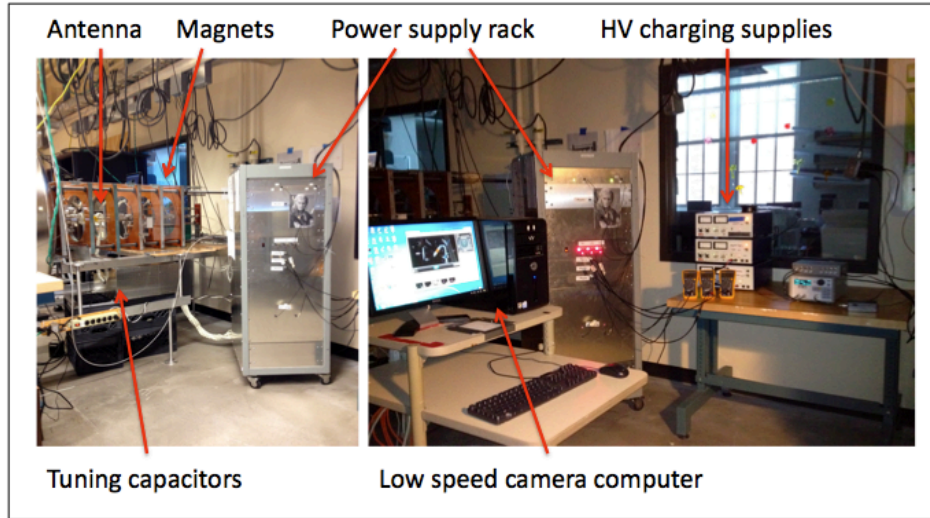


**Figure 2.12. The interior solid fuel helicon thruster in the main vacuum chamber showing the thruster mounted on the left firing towards the right.**

## **2.4 The Solid Fuel High Power Helicon Thruster (External System)**

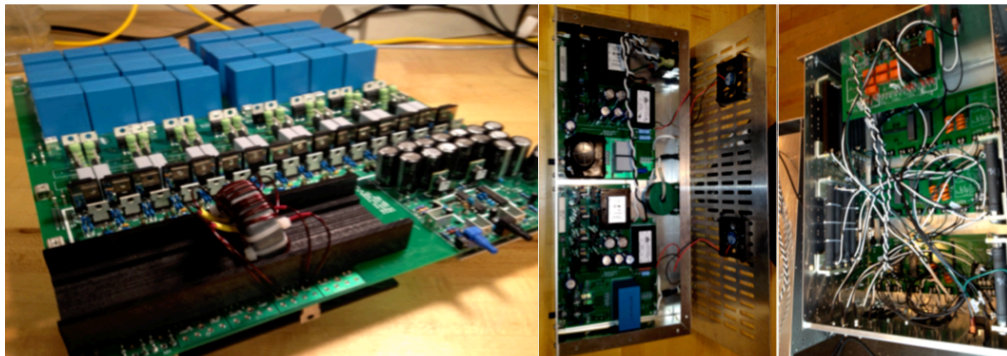
A higher power power-supply, external antenna, and external magnet system (Fig. 2.13) was designed and built in conjunction with Eagle Harbor Technologies, Inc. The system has three primary changes from what the previous system described in Section 2.3.

- 1) Higher energy discharges for the antenna and magnet circuits
- 2) Physical antenna and magnets moved to exterior of chamber
- 3) Increased safety by remotely charging and discharging capacitors



**Figure 2.13. The external HPH system.**

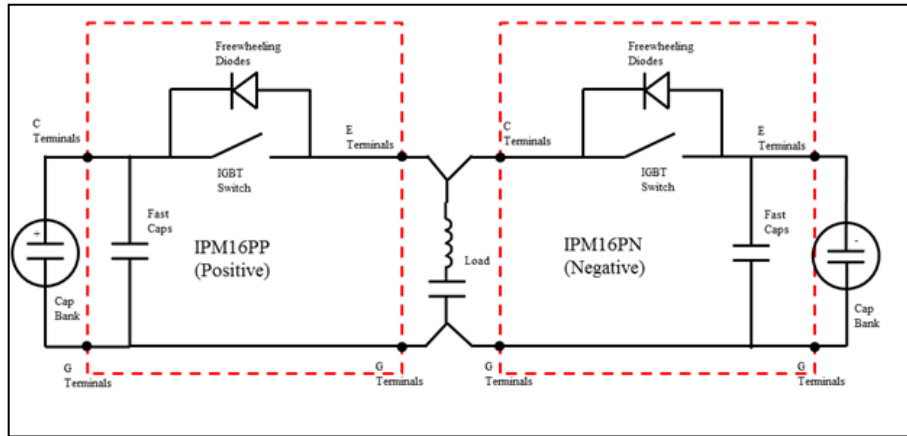
The power supply rack is all-inclusive for the gas-fed system, consisting of nine separate circuit boards. Two boards are specific for the gas propellant version, the control for the puff gate and the power supply for the SPG, both can be seen in Fig. 2.14. The Parker puff valve is opened for  $<3\text{msec}$  with a  $200\text{V}$  voltage pulse, triggered fiber optically, while the SPG to start the initial electron cascade is fired for  $20\mu\text{s}$  with a  $1000\text{V}$  voltage pulse. Each board has two IGBTs.



**Figure 2.14. The antenna positive IGBT board (left), puff gate and SPG circuit boards (center), and the high-voltage charge/discharge system (right).**

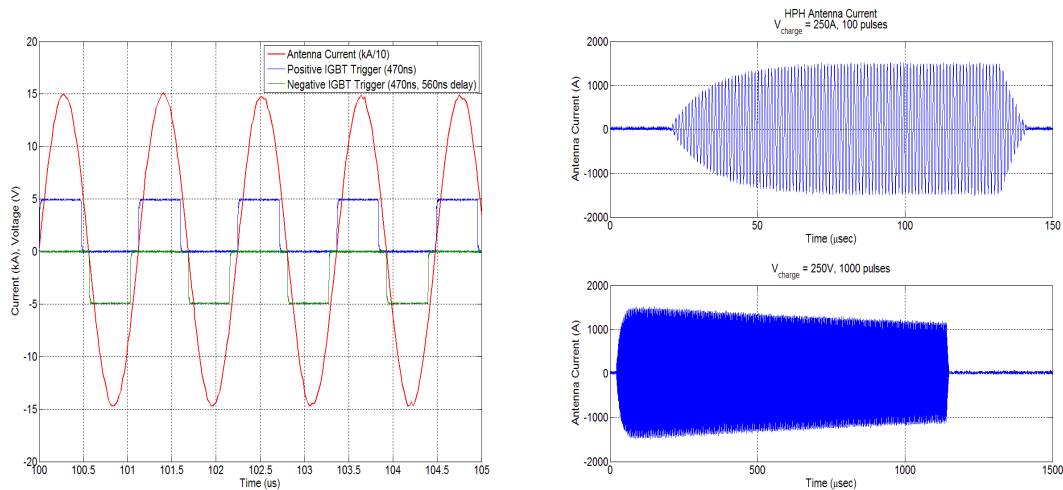
The antenna circuit is similar to the design explained in Section 2.3, except now in a half-bridge configuration (Fig. 2.15) with a positive and negative IGBT board. Each power supply board has 16 IGBTs and an energy storage of  $3.1\text{mF}$ , capable of placing oscillating peak to peak current (Fig. 2.16) and voltage levels of  $3\text{kA}$  and  $30\text{kV}$ , respectively, on the load (antenna and tuning capacitors). The IGBT switches are

triggered fiber optically, allowing current to first flow clockwise through the positive board, and then change direction and flow clockwise through the negative board.



**Figure 2.15. The circuit diagram for the half-bridge HPH tank circuit. The components mounted on the circuit boards are shown inside the red dashes.**

The IGBTs on the positive board are closed for 470ns and then reopened. The negative board IGBTs, after a delay of 560ns, are also given a 470ns pulse. The process is then repeated every 1.12μs, resulting in a frequency of 893kHz. The tuning capacitors have a capacitance of 18nF, the antenna an inductance of 1.7μH, and the stray resistance was measured to be 165mΩ, resulting in a gain of 60.

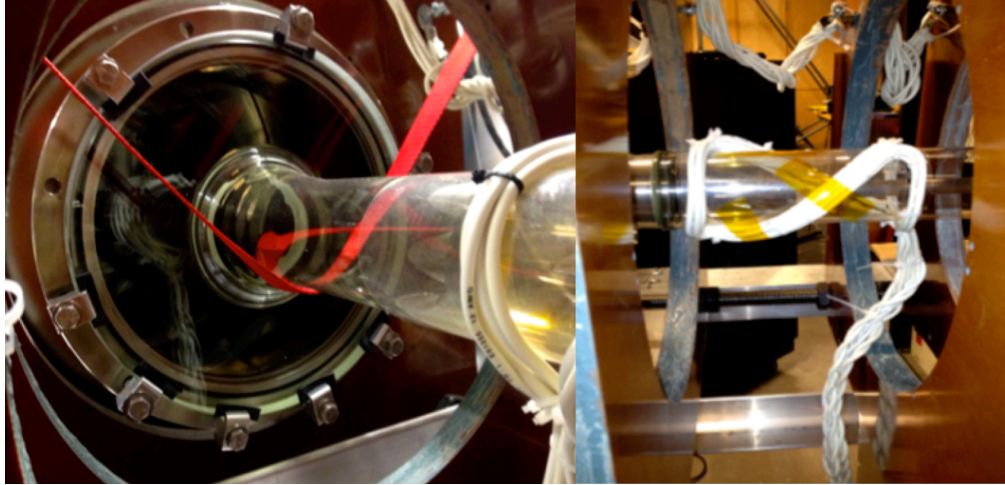


**Figure 2.16. The IGBT triggers with the antenna current over 4.5 oscillations (left) and antenna current with no injected plasma for 100μs and 1.2ms durations (right).**

40kV, 14awg wire with silicon rubber insulation was used to created antenna and electrical connections to the tuning capacitors and IGBT power supply. The antenna was

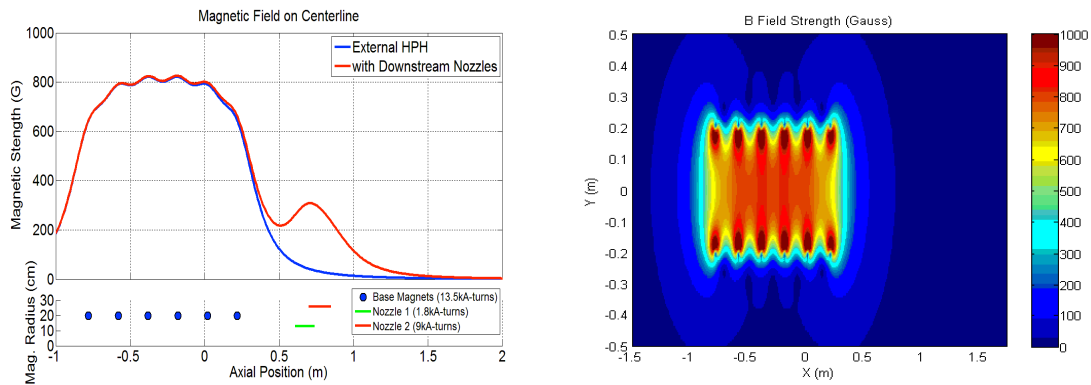


mounted around a pyrex tube which extended to the chamber flange and could be slide over the length of the tube. All results in this thesis were taken with the front of the antenna placed 24cm from the flange. The pyrex tube was vacuum epoxied to the 1" thick glass plate with a 10cm diameter opening cut in the center.



**Figure 2.17.** External helicon 7cm diameter pyrex tube and flange attachment to vacuum chamber (left) and the 14cm length antenna (right).

The magnet system uses a 16 IGBT circuit board. The IGBTs are closed for 250ms, allowing for the complete draining of the magnet capacitors (4.8mF) through the six coils. The coil system (8.1mH) creates a dipole magnetic field configuration of constant strength (up to 780G) through the source region and diverging downstream (matching the terrestrial field strength within 3m, which was the length of the chamber). The magnet was typically triggered 14ms ahead of the PPT and antenna to allow for the helicon discharge to occur at the peak field.



**Figure 2.18.** The axial base field centerline (left) and the XY plane field strength (right) for the exterior magnet system. X=0 corresponds to the front of the antenna.

The magnets were created out of  $0.0625\text{in}^2$  cross-sectional area copper wire that was wrapped within hollowed wooden shells and epoxied sealed, creating an extremely robust coil capable of handling high current and voltage. Each coil was mounted onto four rails, allowing for axial translation. The coils were spaced 20cm apart for all results in this dissertation. 40kV, 14awg wire was used to electrically connect each of the six coils and the magnet power supply together.

The positive, negative, and magnet energy storage capacitors are charged and discharged with a system consisting of a single control board and three separate relay boards (Fig. 2.14). The GIGAVAC relays and boards are rated to 10kV, 500mA, however this application only requires 500V and the charge/discharge resistors were sized appropriately. The control board receives fiber optic inputs allowing for remote charging and discharging of the capacitors. The half-bridge positive, negative, and magnet capacitors are charged with the three variable voltage/current supplies, while the puff-gate and seed plasma generator (SPG) capacitors are charged with 5W constant voltage supplies internal to the power supply rack.

The downstream magnetic nozzles used in the experiment to extend the magnetic field have been detailed previously [43, 44, 51]. Magnetic nozzles convert the random thermal energy of plasma exhaust into directed kinetic energy by creating a diverging magnetic field. This field accelerates charged particles as they travel through the decreasing magnetic gradient by converting parallel gyrovelocity into directed axial velocity. The physical parameters and nominal discharge energy for the nozzles are given in Table 2.1. As shown in Fig. 2.18, the nozzles were positioned downstream such that the peak nozzle field overlaps the location where the base field begins to strongly diverge. This resulted in a magnetic geometry where the magnetic flux diverged after leaving the source, reconverged through the nozzle, then diverged again, slower than before. This has the effect of lengthening the region where the electrons are magnetized.

**Table 2.1. High Power Helicon downstream magnetic nozzle characteristics.**

|                             | First Nozzle                      | Second Nozzle                        |
|-----------------------------|-----------------------------------|--------------------------------------|
| <b>Radius (cm)</b>          | 13                                | 23                                   |
| <b>Width (cm)</b>           | 13                                | 15                                   |
| <b># Wire Turns</b>         | 486                               | 60                                   |
| <b>Discharge Energy (J)</b> | 515                               | 560                                  |
| <b>Power Supply</b>         | High Voltage Xantrex Power Supply | High Current IGBT Pulse Power Supply |

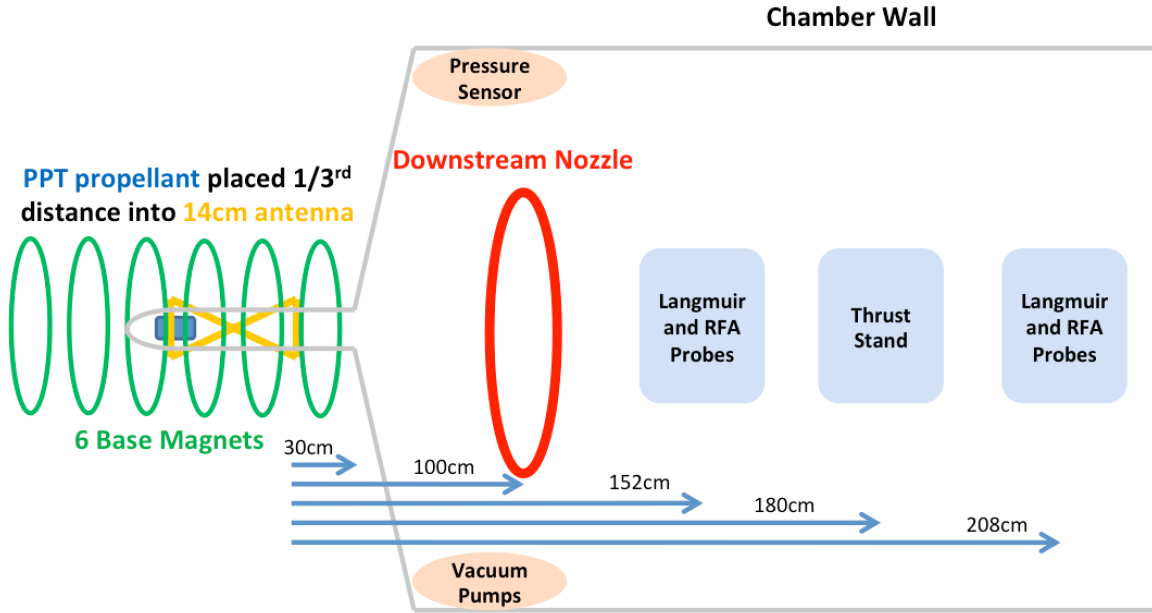
As described by Eq. 1.10, a helicon wave is characterized by operating well above the ion cyclotron frequency, and well below the electron cyclotron and plasma frequencies. Nominal frequency values for the HPH experiment, with an antenna frequency of 890kHz and a carbon/fluorine plasma, are given in Table 2.2.

**Table 2.2. Nominal frequencies in the HPH experiment.**

|                                    | Unit     | Source    | Downstream |
|------------------------------------|----------|-----------|------------|
| Magnetic Field (B)                 | Gauss    | 700       | 10         |
| Density ( $n_e$ )                  | $m^{-3}$ | $10^{21}$ | $10^{17}$  |
| Ion Cyclotron ( $\Omega_i$ )       | kHz      | 500       | 5          |
| Electron Cyclotron ( $\Omega_e$ )  | GHz      | 10        | 0.1        |
| Plasma Frequency ( $\omega_{pe}$ ) | THz      | 1         | 0.01       |

To integrate the coaxial PPT with the system, a 1/4" rod was connected to the back to the PPT (picture) which was run out the back of the pyrex tube. This rod could be rotated and translated axially to move the PPT in relation to the antenna and magnets. Placing the PPT propellant surface 1/3<sup>rd</sup> of the way into the antenna produced the highest downstream densities. The PPT was housed within a nylon doughnut, which was press-fit to the inside of the pyrex tube. This kept the PPT coaxially centered and prevented plasma from within the source region to flow backwards past the PPT.





**Figure 2.19. The exterior solid fuel helicon thruster in the main vacuum chamber showing the thruster mounted on the left firing towards the right.**

As a summary of the internal and external antenna circuits, their primary electrical characteristics are given in Table 2.3.

**Table 2.3. Internal and external system power supply electrical differences.**

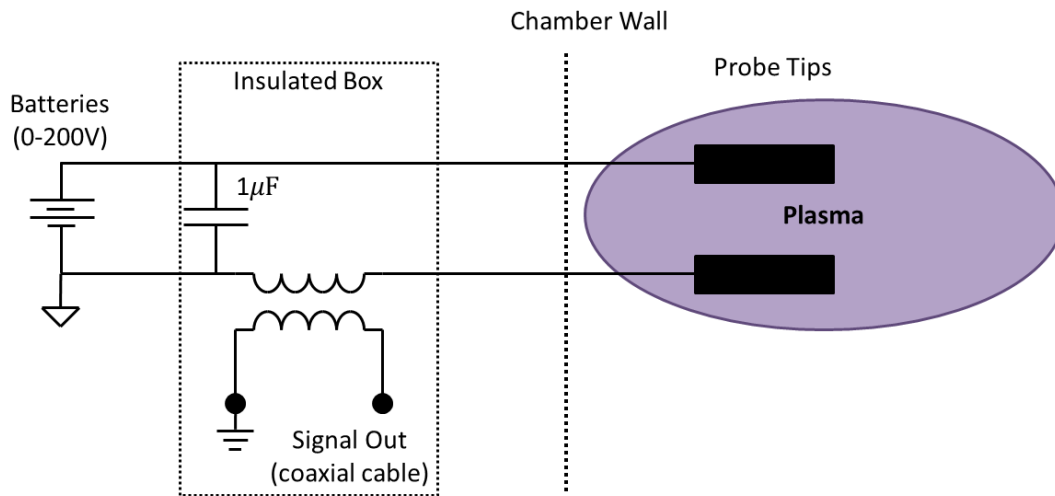
|                    | Internal | External  | Unit          |
|--------------------|----------|-----------|---------------|
| Antenna Inductance | 1.8      | 1.77      | $\mu\text{H}$ |
| Tuning Capacitance | 36       | 18        | nF            |
| Stray Resistance   | 175      | 165       | m $\Omega$    |
| Gain               | 40       | 60        | --            |
| Resonant Frequency | 625      | 890       | Hz            |
| Charging Voltage   | <300     | $\pm 250$ | V             |
| Antenna Current    | 1.5      | 3         | kA            |
| Antenna Voltage    | 12       | 30        | kV            |
| Magnetic Field     | 400      | 780       | G             |

## Chapter 3 Laboratory Plasma Diagnostics

A number of plasma and gas diagnostics were used to understand the thruster dynamics in this study. Double Langmuir probes for density, temperature, and bulk plasma velocity measurements were installed in both the belljar (with the Pulsed Plasma Thrusters) and main vacuum chamber (with the High Power Helicon Thruster). A 5-Langmuir probe array was built for CubeSat testing in the belljar. A retarding field energy analyzer for ion velocity measurements was used with the helicon experiment in the main chamber. The main chamber thrust stand was used with the exterior HPH experiment, while a smaller, modified thrust stand was designed and built for PPT work in the belljar. Cold cathode and hot ion gauge pressure sensors were used on both chambers to estimate the ablated mass differences between varying PPT propellants. Rogowski coils were used to calibrate the thrusters used within the CubeSat and on the atmospheric balloon flights to measured laboratory current waveforms.

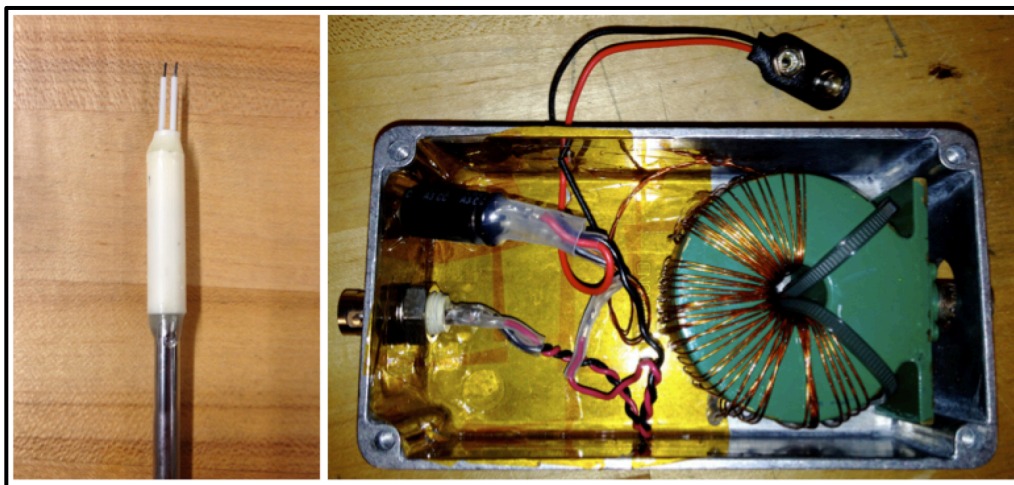
### 3.1 Langmuir Probe

Double Langmuir probes were used to measure the electron density and temperature of the flowing plasma plume. A circuit diagram for the probe is shown in Fig. 3.1. The basic probe concept is that a potential is placed between two metal surfaces, called probe tips, within the plasma. As plasma flows between the probe tips, a conductive path within the probe electronics will be completed and current will begin to flow. This is current is a function of the plasmas density and temperature. A number of assumptions are made to the generic Langmuir circuit, described in [52], to account for our flowing plasma regime. These have been detailed previously [11, 53] for our specific application.



**Figure 3.1. The double Langmuir probe circuit diagram.**

The Tungsten probe tips had a diameter of 1mm, exposed length of 5mm, and were separated by 3mm. The probe tips were placed in the plasma with the potential between them supplied by a series of batteries charging a  $10\mu\text{F}$  capacitor to electrically isolate the circuit. The plasma was typically in contact with the probe for less than a few hundred microseconds. The wires from the probe were completely housed within a  $\frac{1}{4}$ " diameter aluminum tube from the probe tips in contact with the plasma to the electronics box outside the chamber. The probe was isolated from the oscilloscope via a 1:10 isolation current transformer that then passed the voltage signal through an RC filter to eliminate the helicon antenna noise.



**Figure 3.2. A Langmuir probe which is placed within the thruster plume (left) and a Langmuir electronics box which remains outside the chamber (right).**

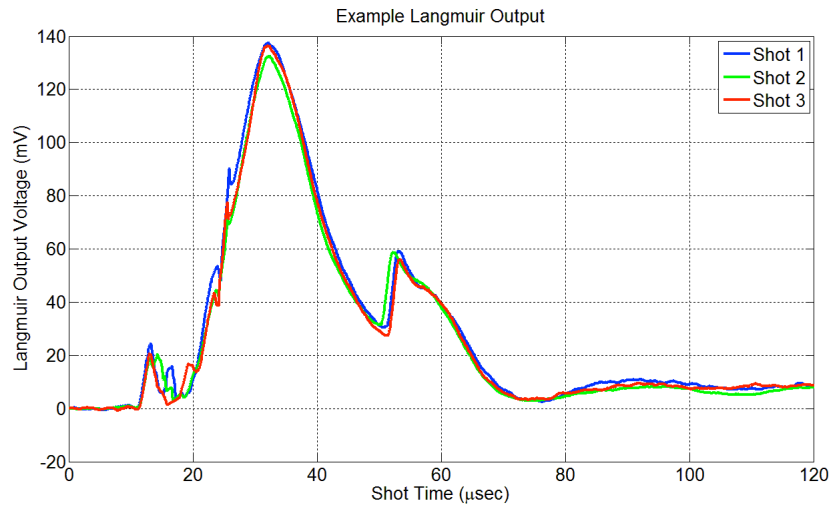
The current output ( $I_p$ ) from the probes can be modeled as a hyperbolic tangent and is dependent on the probe potential ( $\Phi_p$ ), saturation current ( $I_{sat}$ ), and electron temperature ( $T_e$ ),

$$I_p = I_{sat} \tanh\left(\frac{-e\phi_p}{2kT_e}\right) \quad 3.1$$

where then the plasma density ( $n_\infty$ ) can be backed out from the exposed probe tip surface area ( $A_s$ ), potential in saturation ( $\phi_{sat}$ ), electron temperature, and ion mass ( $m_{ion}$ ).

$$I_{sat} = A_s n_s v_s = A_s n_\infty \exp\left(\frac{e\phi_{sat}}{kT_e}\right) \sqrt{\frac{-2e\phi_{sat}}{m_{ion}}} \quad 3.2$$

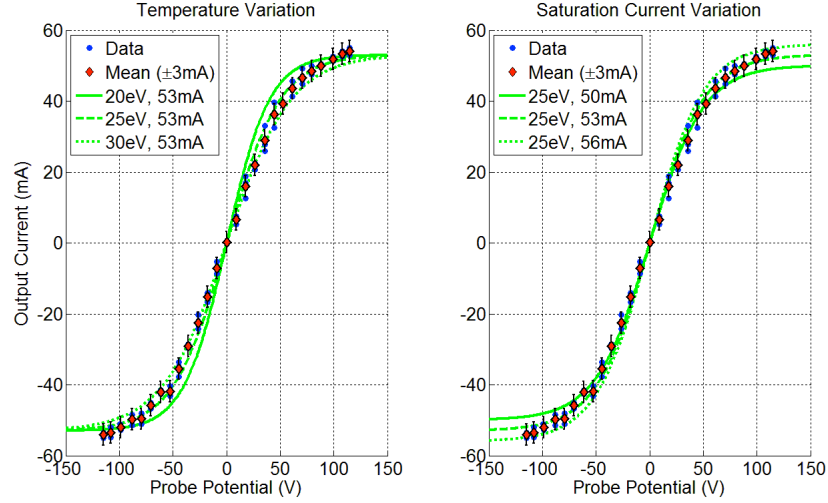
As an example of how to manipulate the output sign into a plasma density, three Langmuir voltage traces at the same battery potential ( $\Phi_p$ ) for a Teflon PPT discharge are shown below. There is some variation between consecutive discharges, which was typical for all thrusters in this dissertation. The peak voltage of each trace was multiplied by the current transformer coefficient and divided by the number of transformer windings to give the peak current output.



**Figure 3.3. Typical Langmuir traces for three consecutive PPT discharges. The voltage was modified by the transformer correction factor to give the current.**

Swinging the potential from negative to positive voltages allowed for Langmuir current curves to be created. The saturation point and electron temperatures in Eq. 3.1 were

varied to find a best fit that matches the data. Eq. 3.1 is fairly accurate to the data, with the exception of potentials above the saturation point, as those tend to float high. This is a non-real effect and ignored while finding the best-fit curves.



**Figure 3.4. Langmuir current vs. potential points with overlaid best fit curves based on Eq. 3.1 with varying electron temperatures (left) and saturation currents (right).**

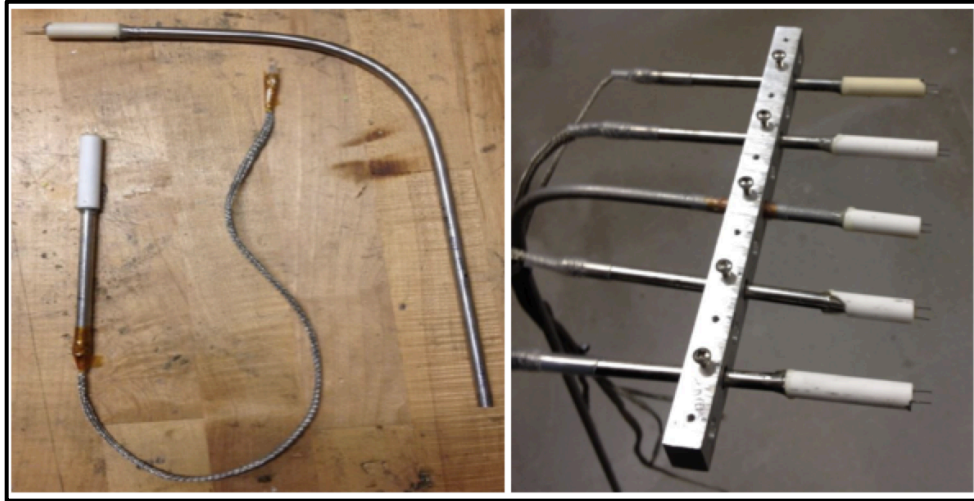
The variations in temperature ( $\pm 5\text{eV}$ ) and saturation point ( $\pm 3\text{mA}$ ) of the best-fit curves Fig. 3.4 resulted in the density values, found with Eq. 3.2, given in Table 3.1. Even with uncertainty to the exact temperature or saturation point, the standard deviation is less than 10% of the calculated densities.

**Table 3.1. Example temperature and saturation current assumptions for the Langmuir results in Fig. 3.3 and 3.4 and the corresponding density.**

| $T_e$ (eV) | $I_{\text{sat}}$ (mA) | $n_{\infty} \times 10^{18}$ (part/m <sup>3</sup> ) |
|------------|-----------------------|----------------------------------------------------|
| 20         | 53                    | 8.6                                                |
| 25         | 53                    | 7.7                                                |
| 30         | 53                    | 7.1                                                |
| 25         | 51                    | 7.3                                                |
| 25         | 56                    | 8.2                                                |

Double Langmuir probes were used in the belljar for the pulsed plasma thruster (Chapter 5) as well the main chamber for the high power helicon (Chapter 4). Additionally a 5-probe array of probes was created to characterize the plume of the low energy ( $< 2\text{J}$ ) discharges of the CubeSat PPT (Chapter 8). The array was simply five standard probes spaced 4cm apart on a rail that could be moved axially over a range of 40cm. The wires

from each of the probes were encircled within a  $\frac{1}{4}$ " diameter flexible braided wire inside and outside the chamber to prevent electromagnetic noise.

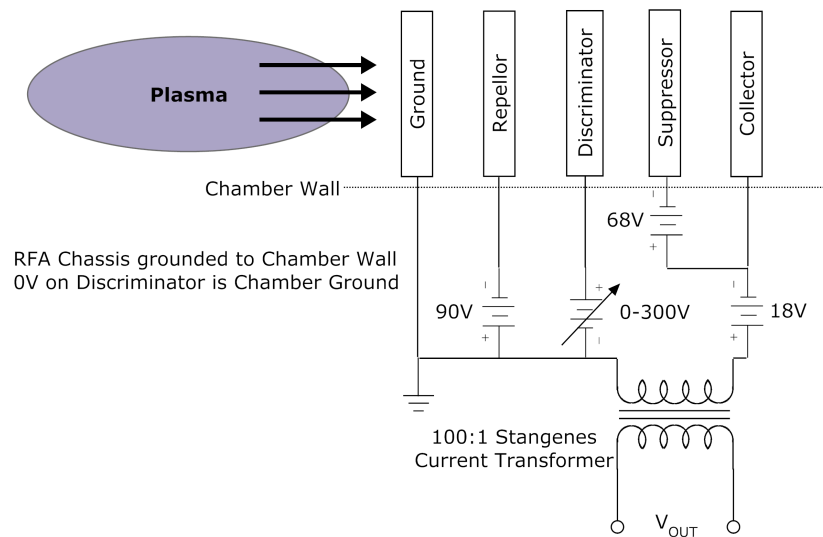


**Figure 3.5. Two of the Langmuir probes (left) built for the 5-probe array (right).**

A bulk plasma velocity could be estimated with a time of flight method by placing two Langmuir probes at different axial distances and measuring when the maximum output signal occurred for each probe.

### 3.2 Retarding Field Analyzer

A retarding field energy analyzer (RFA), the design of which has been described previously in [45], was used to measure the ion velocity distribution function (IVDF).



**Figure 3.6. The RFA probe circuit diagram showing the electrodes inside the chamber and power supplies outside.**

The RFA is composed of stacked grids separated with mylar, all housed within a rectangular steel case that measures 24.9mm long, 19.5mm wide, and 12mm high. The stack begins with a stainless steel orifice plate with a circular hole 1mm in diameter into which the plasma can flow. The next layer is a grounded nickel grid flowed by the repeller, discriminator, and secondary suppressor grids. Finally a nickel plate is used as the collector. The RFA uses the chamber ground as its reference. The three constant voltages are provided by batteries, while the discriminator voltage is provided by a Spellman HV supply that can be varied from 0 to 300V, controlled via a LabVIEW interface. The power to the Spellman supply is given by a 0-30V BK Precision DC regulated power supply. The collector current ( $I_c$ ) is read with a 1:10 Stanges current transformer. The entire power supply is housed in an aluminum box outside the chamber. The ions that hit the collector plate within the probe induce a current described by

$$I_c = Aenv = A_{orifice} e \int_{\sqrt{2eV_D/m_i}}^{\infty} v f_v(v) dv \quad 3.3$$

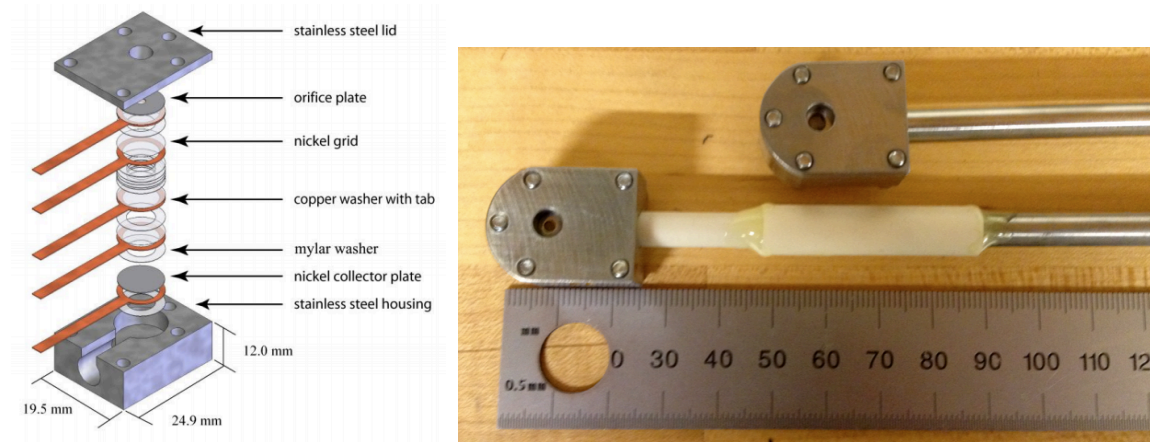
where  $f_v(v)$  is the ion velocity distribution function (IVDF). By varying the discriminator voltage ( $V_D$ ), an I-V curve can be generated. The IVDF as function of  $V_D$  can be computed by taking the derivative of  $I_c$  with respect to  $V_D$

$$IVDF(V_D) = \frac{dI_c}{dV_D} = -A_{orifice} \frac{e^2}{m} f_v \left( \sqrt{\frac{2eV_D}{m_i}} \right) \quad 3.4$$

The derivation of Eq. (3.4) from Eq. (3.3) can be found in [50]. The IVDF (the derivative of the RFA current) yields the directed ion energy distribution in terms of electron volts by comparing the peak value(s) of the IVDF to the corresponding discriminator voltage. With a give ion mass and charge state ( $Z$ ), which we typically assume to be the singly ionized state of the input propellant), the discriminator voltages can be related to the ion velocities

$$V_i = \sqrt{\frac{2eZV_D}{m_i}} \quad 3.5$$

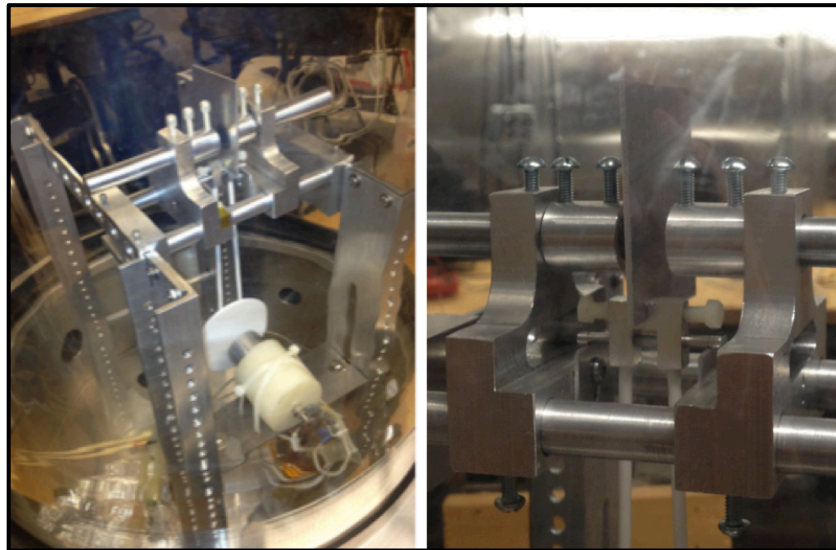




**Figure 3.7. The RFA internal components (left) and the physical RFAs (right).**

### 3.3 Belljar Thrust Stand

A thrust stand was designed and built for use in the bell jar and consists of a circular pendulum made from Teflon supported by a rigid structure. It is a scaled down version of the stand detailed in Section 3.4, where, instead of an optical sensor measuring the displacement distance and outputting to an oscilloscope, a Logitech C260 video camera (30Hz frame rate) viewed the pendulum from the side.



**Figure 3.8. A Teflon PPT mounted on the bell jar thrust stand (left) and the magnetic damping system (right).**

The video from the camera was uploaded into MATLAB where a script was written to detect the brightest frame (which occurred at the PPT discharge) and then the deflection angle for the ten frames immediately following based on a Hough transform. The



maximum angle typically occurred 4-6 frames after the discharge (0.13-0.2sec), which aligns with the quarter period of a simple pendulum with length ( $L_p$ ) of 9.6cm, found by

$$T_{1/4} = \frac{2\pi}{4} \sqrt{\frac{L_p}{g}} \quad 3.6$$

Determination of the maximum deflection angle allowed for the impulse bit to be determined by assuming the pendulum's potential and angular kinetic energy were conserved ( $mg\Delta h = 0.5I\omega^2$ ). Angular momentum ( $L = I\omega$ ) and linear momentum ( $P = mv$ ) are related by the linear momentum equaling the angular momentum divided by the rotation radius. This results in a final equation for the impulse bit as a function of the pendulums mass ( $m_p = 55.6g$ ), distance from centerline of thruster to pivot location ( $L_p = 9.6cm$ ), pendulum moment of inertia ( $I_p = 9.37kg\cdot cm^2$ ), distance from the center of mass of the pendulum to pivot location ( $r_p = 16.5cm$ ), and the pendulums deflection angle ( $\theta$ ).

$$I_{BIT} = \frac{\sqrt{2gm_p L_p I_p (1 - \cos\theta)}}{r_p} \quad 3.7$$

The specific thrust was calculated by normalizing the measured impulse bit by the energy stored in the capacitor bank prior to firing

$$T_{specific} = \frac{I_{BIT}}{E_c} = \frac{2\sqrt{2gm_p L_p I_p (1 - \cos\theta)}}{r_p C V^2} \quad 3.8$$

The MATLAB Hough transform code was dependent on the pendulum moving between pixels in the recorded image. The pixel size for the video camera used in the data analysis resulted in a lower bound for accurate thrust stand measurements of 40 $\mu$ N-sec.



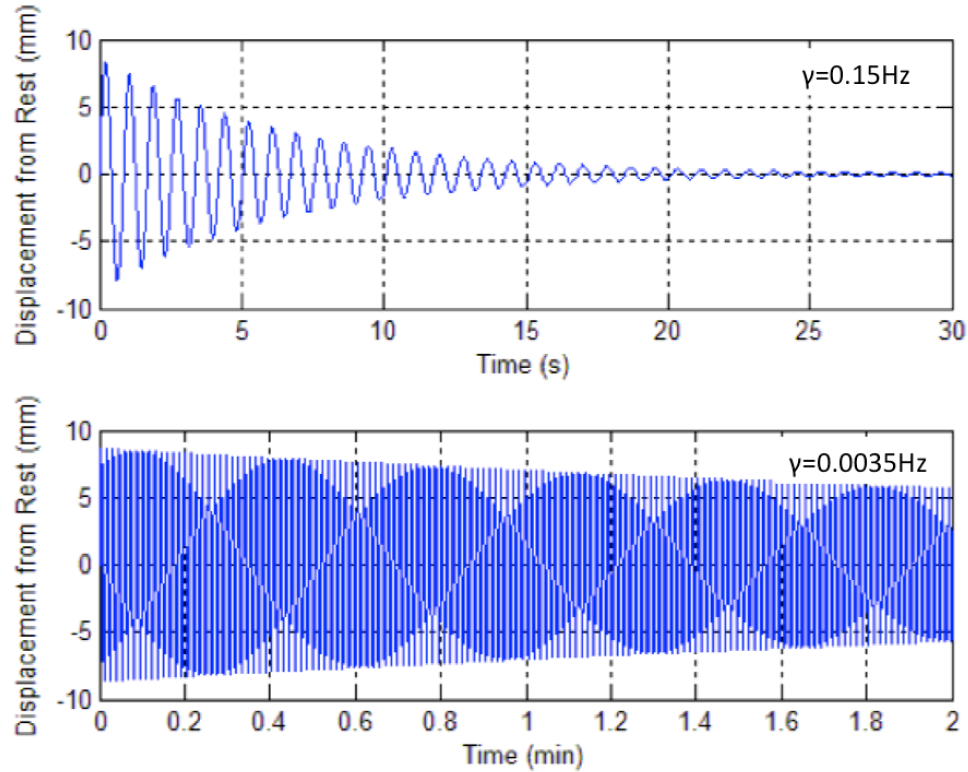
**Figure 3.9. The recorded video of the pendulum before firing, at firing, and at the peak deflection angle which occurred 4-6 frames after the discharge depending on the impulse (left to right).**

There were two primary sources of error in the measurement of the specific thrust. At low deflection angles the pendulum traveled a smaller distance, amplifying camera resolution errors. This error scaled as a decaying power function with increasing deflection angles as  $T_{sm} = 0.87\theta^{-0.77}$ . While at high discharge energies the thruster had a greater variation in shot to shot variation. Two standard deviations of the measured specific thrusts fell within 6.7% of the mean values over all energy and pressure ranges  $T_{sm} = 0.067T_s$ .

A metal plate was attached to the pendulum and placed between two 16lb<sub>force</sub> magnets. As the plate oscillates within the ~3kG magnetic field created by the magnets, small eddy currents are created within the plate (Ampere's Law) that produce a force in the opposite direction to the pendulums movement. This force damps out the oscillations over a time frame of 10's of seconds while negligibly impacting the initial peak displacement of the pendulum (0.02%). Without this magnetic damping effect the pendulum would oscillate for up to an hour between discharges. This initial oscillating would add an additional factor into the analysis as the position and movement of the pendulum when the PPT is fired would be different for every data point.

During calibration testing of the thrust stand the PPT was fired without the magnetic damping system installed, resulting in discharges while the pendulum was in motion. These results showed >50% shot-to-shot variation in the measured impulses. Higher impulses would be measured when the pendulum was swinging away from the thruster, as the pendulums kinetic energy added to the instantaneous impulse of the PPT discharge. While a lower impulse would be measured if the pendulum was swinging towards the thruster as the pendulums kinetic energy subtracted from the instantaneous impulse of the

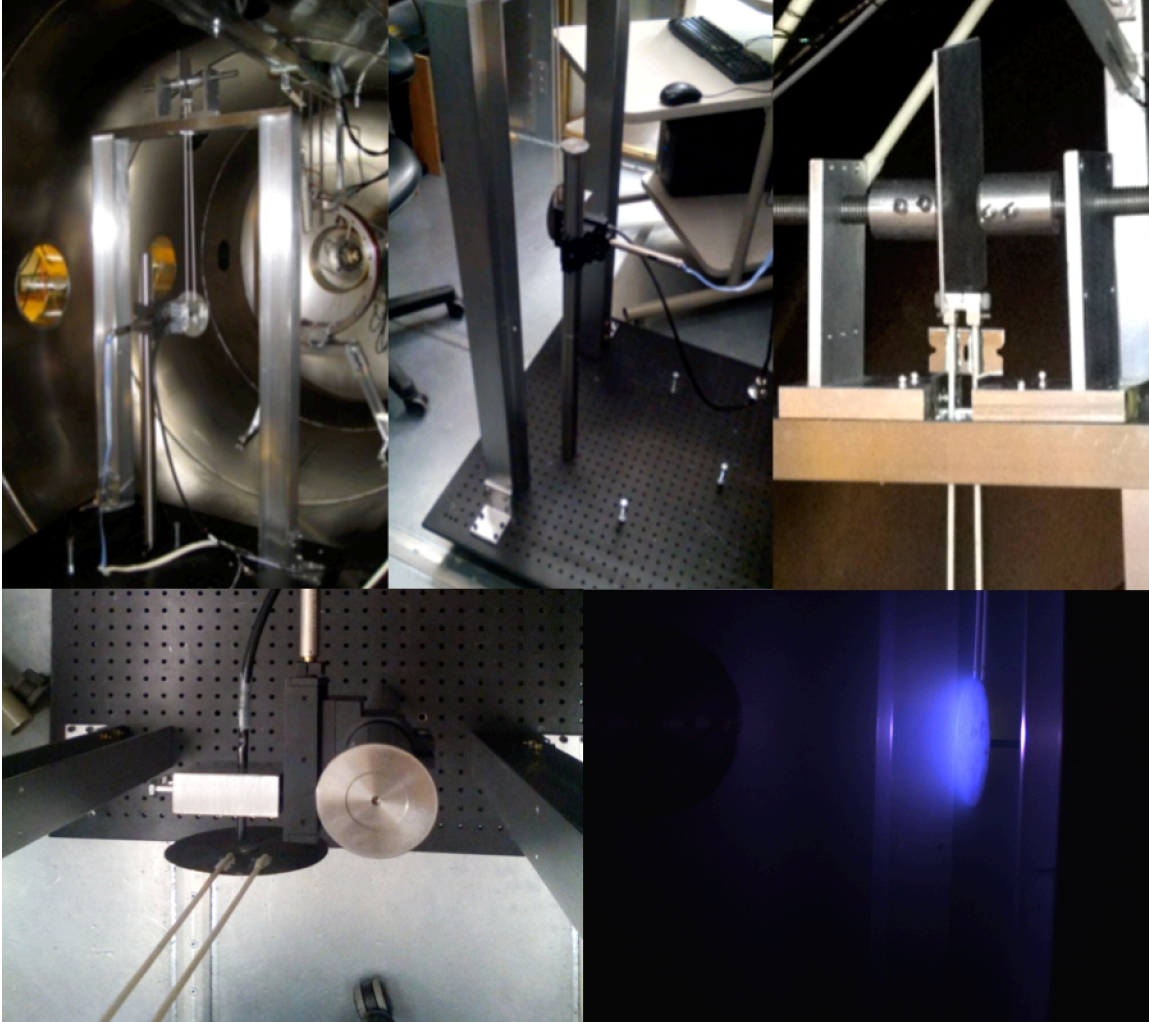
PPT discharge. Computer modeling (Eq. 3.10) showed that the addition of the metallic damping plate increased the pendulums damping coefficient from 0.0035Hz up to 0.15Hz and reduced the motion of the pendulum to zero within 30sec, allowing for higher precision results gathered at a higher frequency.



**Figure 3.10. Computer simulation of the belljar pendulum displacement with (top) and without (bottom) a magnetic damping system.**

### 3.4 Main Chamber Thrust Stand for High Power Helicon

A high-resolution pendulum thrust stand (Fig 3.11) was built for use in the main vacuum chamber for impulse measurements of the laboratories high power thrusters [51] The thrust stand allowed for impulse measurements of 10-500 $\mu$ N-sec by the external solid fuel High Power Helicon thruster (detailed in Section 2.4).

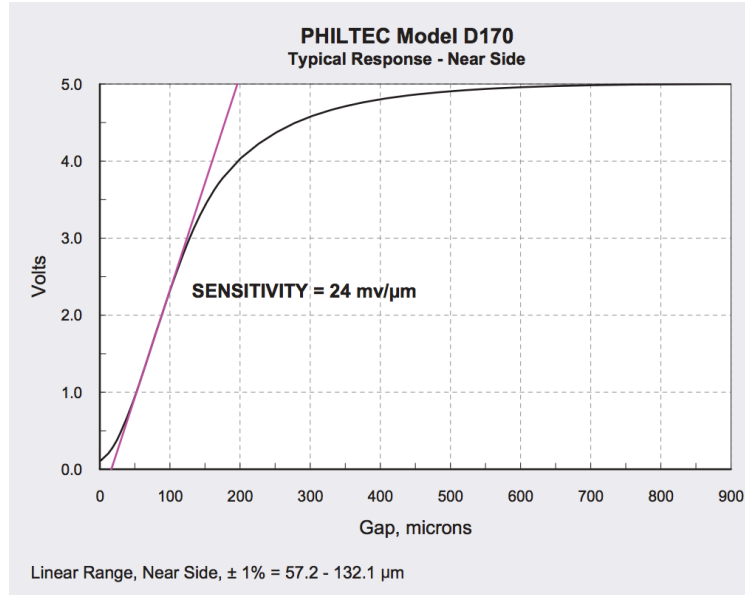


**Figure 3.11. The thrust stand in the chamber, displacement sensor, and magnetic damping system (top, left to right). Top view of pendulum and displacement sensor and light emission from the plasma impacting the pendulum (left to right, bottom).**

The pendulum's target plate is a 10cm diameter aluminum disk placed on the centerline, 180cm downstream of the external High Power Helicon thruster. The pendulum target is supported by two 0.1" diameter ceramic tubes, which are mounted to a razor blade. The pendulum razor blade rests on two rigid frame razor blades to reduce the friction coefficient and create the fulcrum of the pendulum. At the top of the pendulum, a metal plate was attached which oscillated through a  $\sim 3\text{kG}$  magnetic field created by two permanent magnets (Fig. 3.11) to provide passive damping. Eddy currents in the plate imparted a force opposite the direction of movement, increasing the damping coefficient and reducing the pendulum oscillations to a value within the noise of the displacement sensor within 20-40sec. This allowed for negligible initial pendulum motion, which

otherwise would force the inclusion of a phase variable within the sine term in Eq. 3.10 and unnecessarily complicate the analysis.

A small mirror was attached to the back of the pendulum, from which an optical displacement sensor could detect the pendulums oscillations. The output of the sensor was linear over a distance of 57-132 $\mu\text{m}$  as shown in Fig. 3.12.



**Figure 3.12. The displacement sensor output voltage for gap distances 57-132 $\mu\text{m}$ .**

The plasma duration occurs over a few hundred microseconds, allowing for an impulse to the pendulum applied over a timescale much less than the natural oscillation period (1.4sec). The amplitude of the pendulums oscillation can then be related to the force exerted on it by the plasma beam. The pendulums oscillations ( $x$ ) follow the equation of motion for a damped harmonic oscillator

$$\frac{d^2x}{dt^2} = -2\gamma \frac{dx}{dt} - \omega_0^2 x + \frac{F_0(t)R^2}{I} \quad 3.9$$

where  $\gamma$  is the damping coefficient,  $I$  the moment of inertia about the fulcrum,  $\omega_0$  the natural frequency, and  $F_0$  the force applied to the pendulum. For the underdamped case ( $\gamma^2 \ll \omega^2$ ), the pendulum displacement can be solved for as a function of the impulse ( $F_0\Delta t$ ) imparted to it

$$x(t) = \left( \frac{R^2}{\omega I} F_0 \Delta t \right) \exp(-\gamma t) \sin(\omega t + \varphi) \quad 3.10$$

where  $\varphi$  is the angle phase, set to zero for this application. The pendulum oscillation frequency,  $\omega$ , is defined as

$$\omega = \sqrt{\omega_0^2 - \gamma^2} = \sqrt{\frac{mgy}{I} - \gamma^2} \quad 3.11$$

where  $y$  is the vertical distance from the fulcrum to the pendulum center of mass, and  $m$  is the pendulum mass. The voltage from the displacement sensor output can be converted to distance by Fig. 3.12, which can then be compared to Eq. 3.10 to determine the impulse imparted to the pendulum.

**Table 3.2. The measured physical constants of the pendulum thrust stand.**

| Constant                             | Symbol   | Unit              | Value                 |
|--------------------------------------|----------|-------------------|-----------------------|
| Pendulum center of mass to fulcrum   | $y$      | m                 | 0.125                 |
| Pendulum center of target to fulcrum | $R$      | m                 | 0.425                 |
| Pendulum mass                        | $M$      | kg                | 0.097                 |
| Moment of inertia                    | $I$      | kg-m <sup>2</sup> | $6.42 \times 10^{-3}$ |
| Damping coefficient                  | $\gamma$ | Hz                | 0.16                  |
| Oscillation frequency                | $\omega$ | Hz                | 4.30                  |
| Oscillation period                   | $T$      | sec               | 1.46                  |

### 3.5 Ion Gauge Pressure Sensor Measurements

As a means to estimate the amount of propellant input to the thruster, ion gauge pressure sensors were used to measure the pressure change within the chamber per discharge. The pressure of neutral gas is a function of the density and the temperature, expressed as

$$P = nkT \quad 3.12$$

There are two primary gauges used in this study, hot and cold cathode sensors. They both work on the same general principal of generating electrons that then collide with and ionize a small fraction of the chambers neutral gas. The ions created from this ionized gas are accelerated into a collector generating a current. This current is proportional to the neutral density, and therefore to the chamber pressure by Eq. (3.12). Both sensors are composition dependent. A neutral gas with a higher cross-sectional area will skew the

collector current and requires a correction, often called the relative sensitivity factor ( $R_g$ ) as shown in Eq. (3.13).

$$P_{true} = \frac{P_{measured}}{R_g} \quad 3.13$$

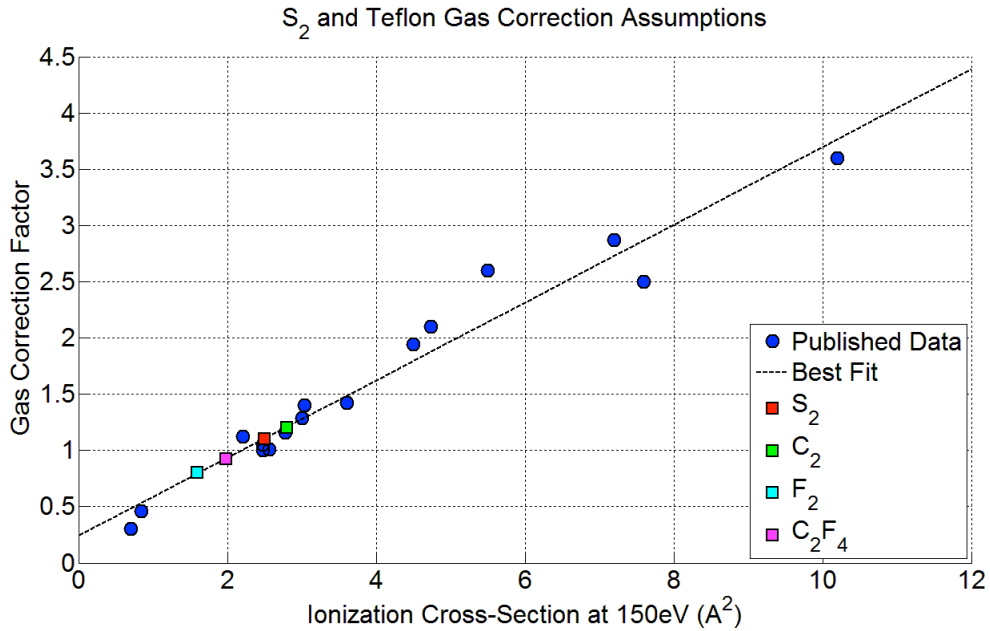
This correction factor is commonly scaled such that a background gas of ambient atmosphere (~nitrogen) has a correction of 1.00. Correction factors for common gases are shown below in Table 3.3 and Fig. 3.13. The correction factors for common gases can be found in the literature and roughly scale as the total cross-section for ionization by electron impact at 150eV [54, 55]. There are differences in the literature, some quite substantial, for experimentally measured ionization cross-section with even the most common gases. As such, the values shown here are those determined with the binary-encounter bethe (BEB) model. These correction factors must be used when comparing pressure results between different propellant compositions, and was used for all pressure graphs in this dissertation.

**Table 3.3. Cross-section for ionization by electron impact at 150eV [54, 55] and ion gauge correction factors for selected gases [56, 57].**

| Species                       | Cross-section ( $\text{\AA}^2$ ) | $R_g$ |
|-------------------------------|----------------------------------|-------|
| Ne                            | 0.70                             | 0.3   |
| H <sub>2</sub>                | 0.84                             | 0.46  |
| N <sub>2</sub>                | 2.48                             | 1     |
| O <sub>2</sub>                | 2.56                             | 1.01  |
| C <sub>2</sub> H <sub>6</sub> | 5.50                             | 2.6   |
| C <sub>4</sub> H <sub>8</sub> | 10.20                            | 3.6   |
| CH <sub>4</sub>               | 3.04                             | 1.4   |
| CO                            | 2.48                             | 1.05  |
| H <sub>2</sub> O              | 2.22                             | 1.12  |
| NO                            | 2.78                             | 1.16  |
| Ar                            | 3                                | 1.29  |
| CO <sub>2</sub>               | 3.6                              | 1.42  |
| Kr                            | 4.5                              | 1.94  |
| SO <sub>2</sub>               | 4.74                             | 2.1   |
| SF <sub>6</sub>               | 7.59                             | 2.5   |
| Xe                            | 7.2                              | 2.87  |

Although the exhaust composition was not measured as part of this study, the propellants tested in Chapter 5 are not expected to produce common gases. Therefore, assumptions

about their composition and gas corrections had to be made. Sulfur was assumed to exist in gaseous form as  $S_2$ , which has a cross-section of  $2.5\text{\AA}^2$ , while Teflon ( $C_2F_4$ ) was assumed to exist as  $C_2$  ( $2.8\text{\AA}^2$ ) and  $F_2$  ( $1.6\text{\AA}^2$ ) gas, in percentages of 33% carbon and 66% fluorine [56, 57, 58]. This gives an assumed gas correction factors of 1.1 for sulfur and 0.9 for Teflon.

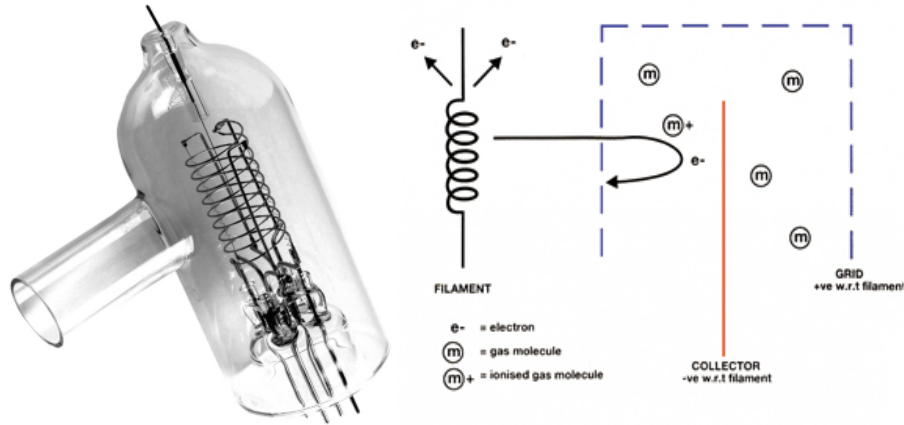


**Figure 3.13. Published correction factors for the species listed in Table 3.3 and assumed values for neutral Teflon and S<sub>2</sub> gas.**

### 3.5.1 Bayard-Alpert Triode Hot Cathode Ion Gauge

The hot cathode ion gauge used in this experiment on the labs main (large) vacuum chamber, has an operational range of  $10^{-12} - 10^{-3}$  Torr. The lower pressure is set by x-ray emissions erroneously increasing the collector current, while the upper limit is due to the mean free path of the neutral gas density becoming smaller than the gauges scale length. The hot cathode ion gauge is shown below in Fig. 3.14 on the left and a generic operation schematic on the right.





**Figure 3.14. The hot cathode ion gauge used in the experiment (left) and the corresponding operational schematic (right).**

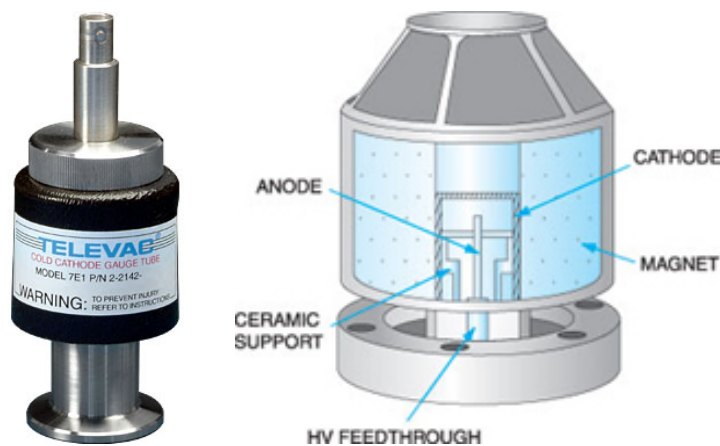
There are three primary electrodes, a grid set to a potential of +180V with a grounded (0V) collector electrode in the center. A negatively charged filament (-30V) is placed directly outside the grid. This filament creates electrons through thermionic heating, boiling off of electrodes by increasing a substances thermal energy to a great value than its binding potential. These electrons are attracted into the volume contained within the grid where they collide with and ionize the neutral gas within the chamber. The ionization of this neutral gas creates positively charged ions that are accelerated towards the grounded collector electrode. This collector current ( $I_c$ )

$$P = \frac{I_c - I_r}{R_g I_e} \quad 3.14$$

scales linearly with the neutral gas pressure ( $P$ ). The pressure is inversely proportional to the composition dependent relative sensitivity ( $R_g$ ) and the current to the filament ( $I_e$ ). As with any system, there are errors that need to be taken into account. The primary of which for the ion gauge is an erroneously high ion current, a residual current ( $I_r$ ), which needs to be subtracted off in Eq. (3.14). This residual current typically results from X-ray induced photoemission of electrons from gas (at low pressures), electron stimulated desorption (ESD) of molecules from anode grid surface, leakage current at electrodes which becomes a larger effect over time, and electrometer offset errors in the current measurement.

### 3.5.2 Cold Cathode Ion Gauge

A Televac 7E double inverted magnetron cold cathode sensor is in operation on the labs small belljar vacuum chamber. The cold cathode ion gauge works on similar principals to the hot cathode version, except that it does not employ a hot filament to produce electrons, which, under typically operation, greatly increases the lifespan of the sensor. Just as with the hot cathode, the cold cathode is based on the ionization probability of the gas with the ionization as a result of a high voltage discharge of electrons. An operational range of  $10^{-8}$  to  $10^{-2}$ Torr is typical for the cold cathode sensor. The upper limit results from a high collector current causing sputtering from the electrodes. An image (left) and schematic (right) of the cold cathode are shown in Fig. 3.15.



**Figure 3.15.** The cold cathode ion gauge used in the experiment (left) and the corresponding operational schematic (right).

The cold cathode has two primary electrodes as well as a permanent magnet housed withinside. Its structure. Electrons are generated from the cathode (high negative voltage) by field emission, photo collisions, or radioactivity and accelerated to the grounded anode (0V). These electrons are trapped by large electric ( $\sim 2\text{kV}$ ) and magnetic (1kG) field and circle the anode, colliding with and ionizing chamber particles. The ions generated from these collisions are accelerated to the anode ion collector, where the generated collector current is proportional to the chambers neutral gas pressure.

### 3.6 Rogowski Coil

A Rogowski coil was used to measure the PPT main and igniter discharge currents during the atmospheric flights and throughout CubeSat testing. The coil is based on Ampere's

law ( $\oint \mathbf{B} \cdot d\mathbf{l} = \mu I_C$ ), which states that a changing current ( $I_C$ ) flowing through a wire will produce a changing magnetic field ( $\mathbf{B}$ ); which will in turn induce a current in a coil wrapped perpendicular to the initial current. The voltage output from the Rogowski coil is proportional to the time derivative of the current passing through the loop, given by

$$V_{ROG} = -\frac{A_{ROG} N \mu_0}{l} \frac{dI_C}{dt} \quad 3.15$$

where  $I_C$  is the total current passing through a winding circumference of length  $l$ , with  $N$  number of loops of cross-sectional area  $A_{ROG}$ .

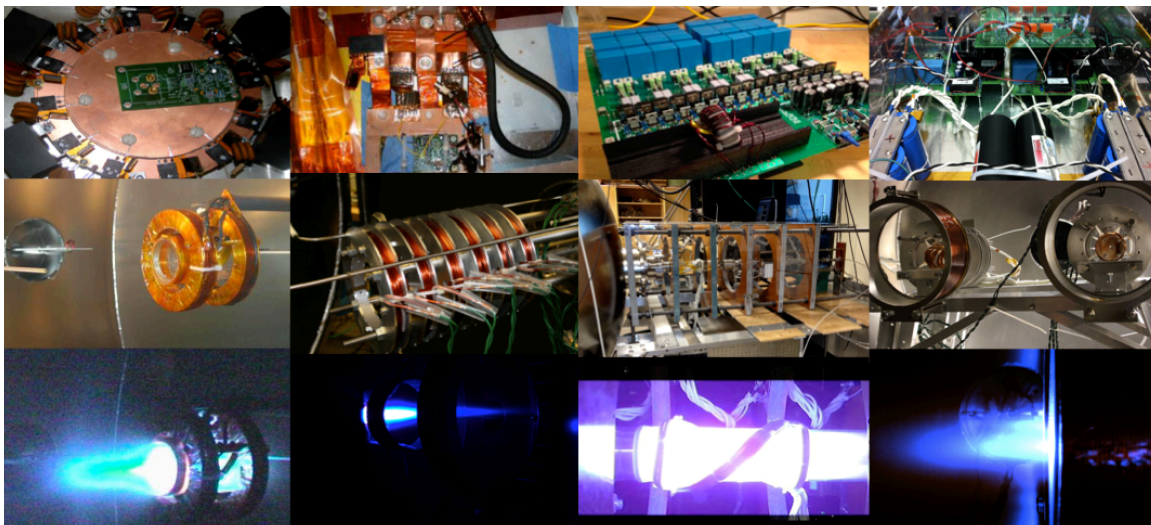
The PPT main discharge had currents of up to 5kA for the atmospheric and CubeSat thrusters. Rogowski coils of varying sizes and number of loops (Fig. 3.16) were created for each thruster in order to produce an output voltage of 4-5V for the highest energy discharge. This was done as the Arduino microprocessors used had maximum voltage levels of 5V for their analog input pins.



**Figure 3.16. The rogowski coil flown on the 2013 flight, 2015 flight, and used onboard the CubeSat (left to right).**

## Chapter 4 The High Power Helicon Thruster

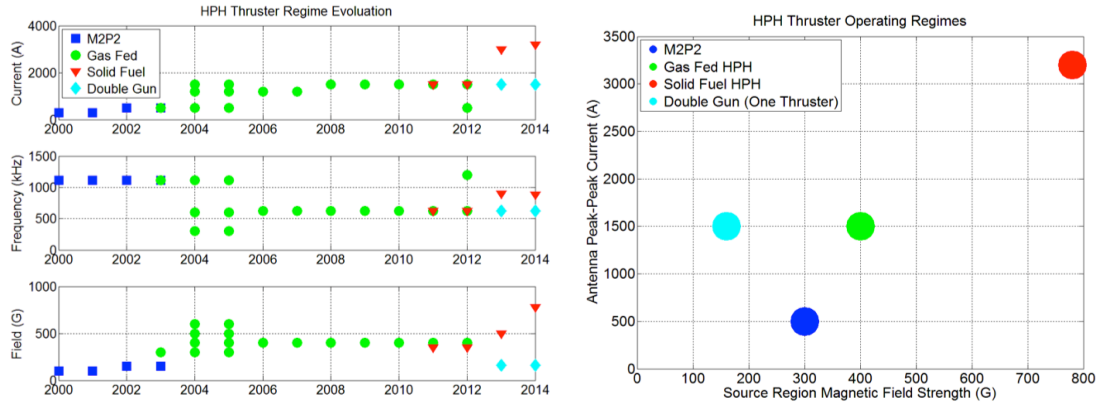
The High Power Helicon experiment at the University of Washington's Advanced Propulsion Laboratory (APL) was originally designed with multiple goals in mind. Initially the helicon was used as a plasma injection source for the Mini-Magnetospheric Plasma Propulsion (M2P2) thruster [48, 49]. Transitioning to a thruster in its own right, work was completed to better understand the energy transfer between the helicon antenna and the plasma [43, 50]. More recently, the addition of downstream magnetic nozzles to increase thrust [51] and an additional downstream antenna in series [43] to increase ion velocity and density was completed. Currently, research into the addition of a 2<sup>nd</sup> source region antenna in parallel is ongoing to increase the thruster power and downstream particle flux without increasing the source region collision frequency [59]. Recently, the lab has been experimenting with solid fuel thrusters [11, 60] and how they can couple into the Helicon experiment [53]. The four primary historical variants of the HPH experiment since its conception in 2000 are shown in Fig. 4.1.



**Figure 4.1. The electronics (top), hardware (center), and plasma optical emissions (bottom) for the four primary phases of the HPH experiment since its conception in 2000. Shown are the M2P2, gas fed, solid fuel, and double antenna HPH thrusters (left to right).**

The helicon experiment at the University of Washington begun with antenna currents of a few hundred amps and has slowly raised by an order of magnitude over the past 15 years as electronics have increased in sophistication. Likewise, the source region magnetic field

has increased from 100G with the M2P2 experiment up to 780G with the current solid fuel experiment.



**Figure 4.2. (Left) Peak-to-peak antenna current (top), antenna frequency (center), and source region magnetic field strength (lower) of the four HPH experiment variants since 2000. (Right) The optimized peak current and magnetic field strengths for the final versions of the four HPH experiment variants.**

Results gathered to date with a solid propellant source for the High Power Helicon Thruster are detailed in the sections below. Initial testing was completed with the same hardware and electronics as previous gas fed HPH work where the antenna and base coils are housed within the chamber, hence the system called the internal HPH (Section 4.2). The attempts at PPT current smoothing and lengthening by varying the wire inductance are detailed in Section 4.3. This was followed by tested on an updated helicon system with components housed exterior the chamber (Section 4.4). A theoretical, continuous operation thruster using gaseous Xenon propellant is proposed in Section 4.5.

#### 4.1 Why Solid Fuel

To date, Xenon and Krypton propellants of choice for electric propulsion in-space thrusters as they're inert, have low ionization potentials, and high atomic mass, leading to high performance for 100kW range thrusters. However, a recent spike in worldwide demand has driven up their costs [61], adding significant development and operational expenses. The slow propagation of a gas or liquid through valves makes them poorly suited for a pulsed system, which is required is wall effects are to be eliminated from the analysis. Any gaseous propellant will require a complicated and heavy system of valves and tubing, including multiple moving components. The density of even a compressed

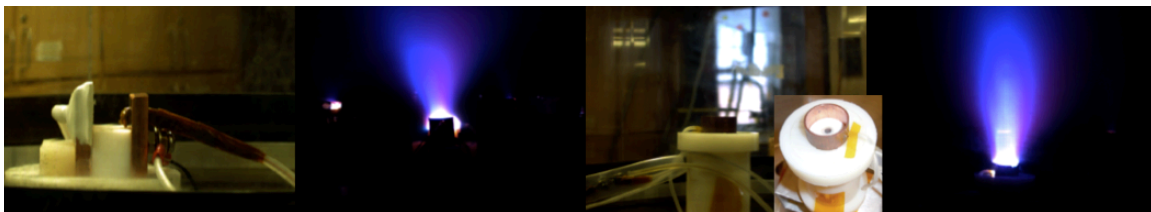


gas is drastically less than a liquid or solid, resulting in less propellant mass for a given volume.

To address the above issues, a solid propellant helicon thruster was developed. The use of a Pulsed Plasma Thruster (PPT) allows for the storage of solid Teflon propellant and the creation of a carbon/fluorine gas input into the helicon source. The PPT, which employs only one moving component (a spring), is fitted onto the back end of the helicon, which then uses high-power RF waves to complete the ionization and acceleration process. This innovation reduces the cost, mass, and volume expenses of the gaseous propellant and tankage. For laboratory experiments, solid propellants opens up the opportunity to have very discrete pulses without the ramp-up time for the flow of gas through various tubing and gate valves.

#### 4.2 The Laboratory Coaxial PPT

The first prototype of the solid fuel high power helicon thruster (SF-HPH) used a 42J coaxial PPT with Teflon propellant as the fuel source. Initial work in the laboratory was with a rectangular PPT which had a 0.5x1” Teflon surface area and 0.5” high electrodes [53]. The rectangular and coaxial variants were tested against each other to determine their density/energy ratios. Both were tested in the belljar vacuum chamber with the same electronics setup using a 60 $\mu$ F main discharge capacitor. The plasma plumes of each thruster are shown in Fig. 4.3 and the on-axis peak density measurement, averaged over 5 discharges, are given in Table 4.1.



**Figure 4.3. The rectangular (left) and coaxial (right) PPTs tested.**

Density measurements taken between the two geometry variants showed an order of magnitude increase in peak plasma density with the coaxial thruster for a discharge energy of 10J at axial distances of 20, 30, and 40cm above the propellant surface as well as discharge energies of 20 and 43J at 40cm. Sputtering of the Langmuir probes at 20 and 30cm distances prevented the probes from being placed in saturation at discharge

energies of 20 and 43J. The results shown in Table 4.1 are an average of 5 discharges with a maximum shot to shot variation of 5%. With the helicon antenna and base magnets both being circularly concentric about the centerline, integrating a coaxial PPT into the helicon experiment maintained that circular symmetry. All data with the solid fuel helicon presented in this thesis was collected with a 1” diameter, coaxial Teflon PPT.

**Table 4.1. Average peak Langmuir density differences with the rectangular and coaxial PPTs.**

| Energy (J) | Height (cm) | Rectangular (part/m <sup>3</sup> ) | Coaxial (part/m <sup>3</sup> ) | % Increase |
|------------|-------------|------------------------------------|--------------------------------|------------|
| 10         | 20          | $5.6 \times 10^{17}$               | $3.7 \times 10^{18}$           | 85         |
| 10         | 30          | $3.0 \times 10^{17}$               | $2.3 \times 10^{18}$           | 87         |
| 10         | 40          | $1.9 \times 10^{17}$               | $1.5 \times 10^{18}$           | 87         |
| 20         | 40          | $5.1 \times 10^{17}$               | $4.7 \times 10^{18}$           | 89         |
| 43         | 40          | $1.0 \times 10^{18}$               | $9.2 \times 10^{18}$           | 89         |

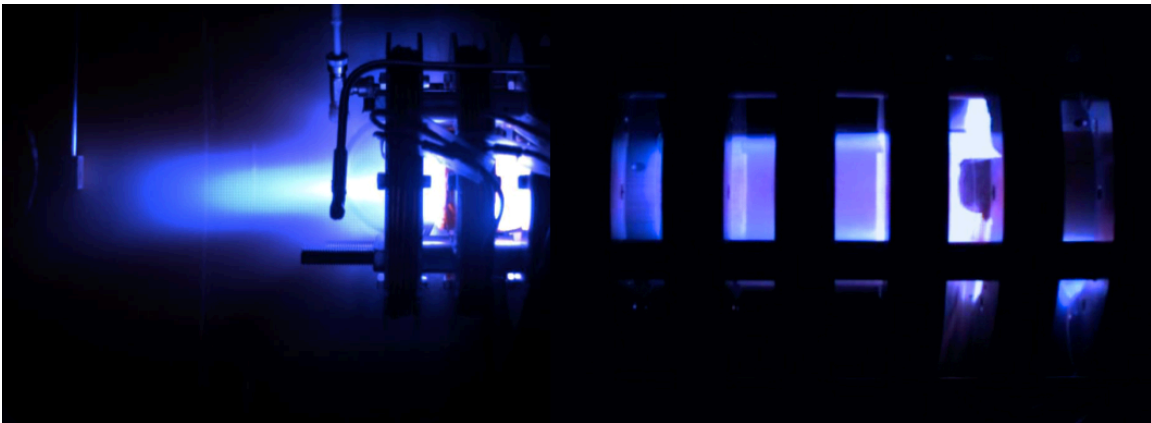
Surface charring has been an active area of research with the PPT for decades [62]. Charring, or redeposition, as a result of an ablative discharge is the accumulation of heated, but unaccelerated material within the thruster. This material has different chemical properties to virgin Teflon propellant and is a leading cause of shot-to-shot variation over the lifetime of a PPT. The addition of the PPT is mean to simplify the propellant injection for the helicon and surfacing charring would add unnecessary complications. Testing of the PPT without the helicon antenna attached at energy levels of 10, 20, and 42J showed drastically less surface char at 42J after 150 shots, implying that higher energy discharges will result in less shot-to-shot variation and increased lifetime of the thruster. These results are similar to what has been found previously with research through the Airforce Research Laboratory (AFRL) on the MicroPPT thruster, where lower energy discharges produce far more charring over long duration testing [63]. The AFRL charring model shows that higher discharge energy for a set capacitance and set propellant area leads to a higher Teflon surface temperature and enhanced Teflon ablation (and carbon char ablation) and therefore prevents char formation.



**Figure 4.4. PPT surface charring after 150 shots at a base pressure of 1 $\mu$ Torr for discharge energies of 10, 20, and 42J (left to right).**

### **4.3 Integration with the Internal High Power Helicon Experiment**

The experimental setup for the solid fuel internal high power helicon experiment is described in Section 2.3. The exposed propellant surface of the PPT was placed such that it was 3.5cm (~25%) from the back of antenna. This 25% position was consistent with the location for the exit of the gas puff for previous Argon testing of the HPH thruster (Race/Jim/Ilia). The PPT electrodes and propellant were housed within a nylon cylinder that was press-fit to the inside walls of the quartz helicon tube, preventing plasma from escaping to the back. The base magnetic field peaked at 400G in the source region and was constant throughout each shot.

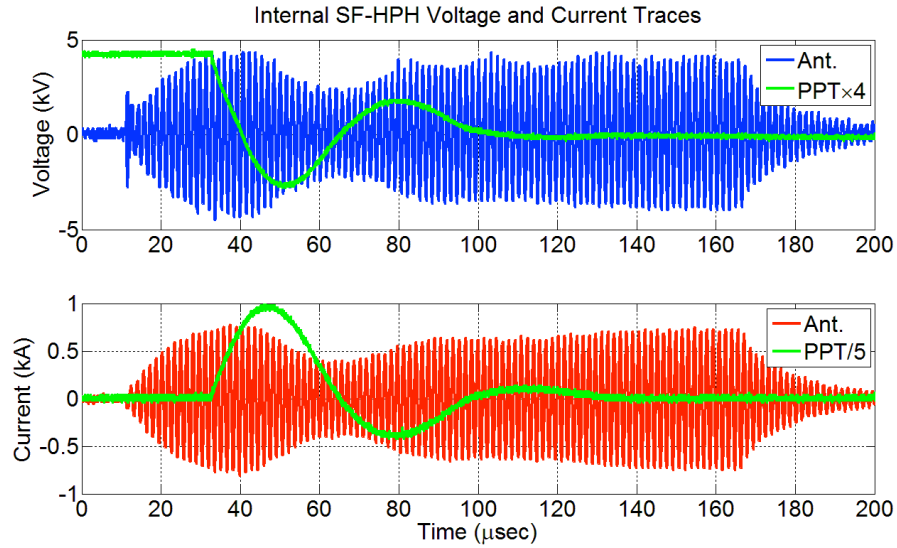


**Figure 4.5. The internal HPH system firing with Argon gas propellant (left) and solid Teflon propellant (right).**

The helicon antenna was operated with a 1.4kA peak-to-peak current and 8kV peak-to-peak voltage, with a duration of 160 $\mu$ sec. The 42J PPT had a 5kA current peak, which decayed in 70 $\mu$ s and was timed to discharge as the helicon antenna reached full strength. The majority of the loading profile is seen to coincide with the PPT's main discharge, with some late time ablation effects of the PPT causing secondary loading curves. It is evident

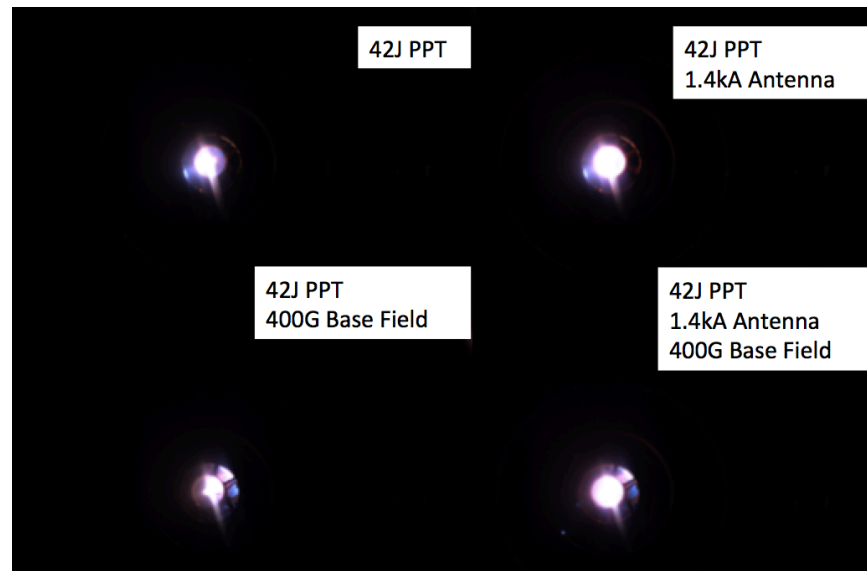


however that the later half of the helicon antenna pulse is being unused as the current profile returns to near the full peak-to-peak height of 1.4kA by 100 $\mu$ sec. This implies that the ablated mass from the PPT that interacts with the helicon wave has left the source region within the first 100 $\mu$ sec.



**Figure 4.6. The HPH antenna (red and blue) and PPT (green) discharge current (bottom) and voltage (top) traces.**

Imaging of the solid fuel HPH thruster was taken from the far end of the chamber, directly opposite of the source region. The images with the antenna operational show higher intensity and more symmetrical light emission.

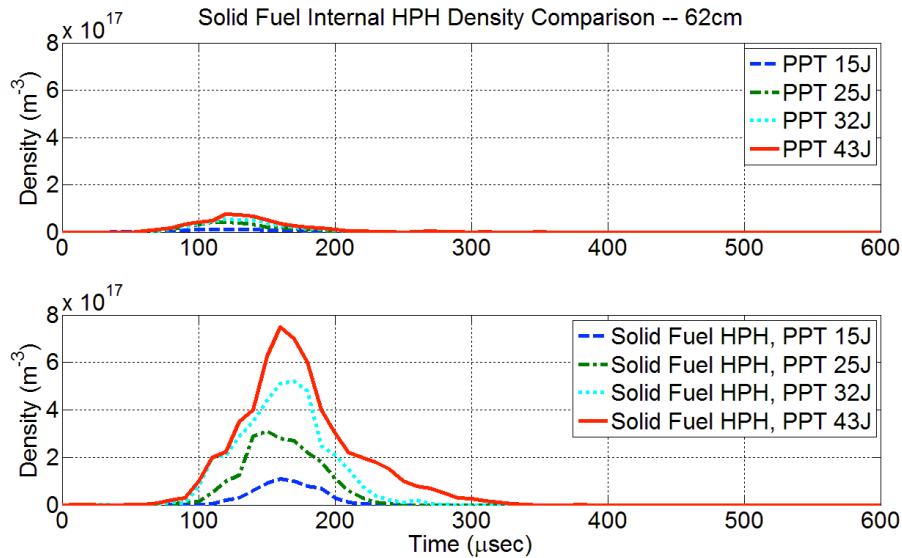


**Figure 4.7. High-speed camera imaging viewing from downstream towards the source region of the PPT, PPT/antenna, PPT/base field, and the solid fuel HPH.**

Langmuir probes were placed in the plasma plume at distances of 62 and 118cm from the front face of the helicon antenna to measure the plasma density. As shown in Fig. 4.8 and Table 4.2 increasing the PPT discharge energy increased the peak density as well as the density flux, defined as

$$\phi_n = \int n_e dt \quad 4.1$$

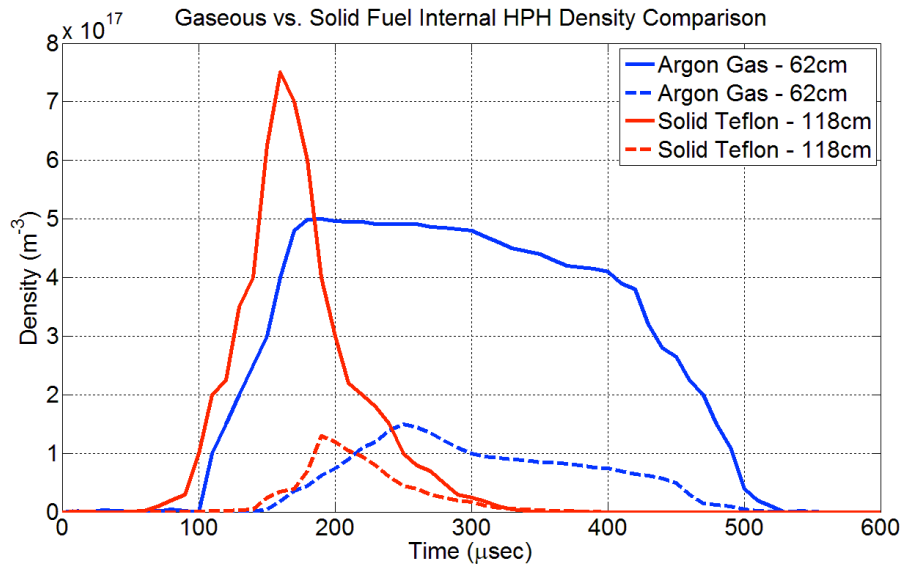
on-axis at a distance of 62cm for both the PPT firing alone and with the helicon system. Both the peak density and density flux increase linearly with PPT discharge energy. The addition of the helicon system increases both the density and flux by, on average, an order of magnitude.



**Figure 4.8. On-axis density measurements taken at a distance of 62cm from the front face of the helicon antenna for the PPT (top) and solid fuel HPH thruster (bottom) at PPT discharge energies of 15-43J.**

A comparison of the HPH thruster with solid fuel and Argon gas was made with Langmuir probes placed 62cm downstream from the thruster. The Argon gas data was taken in 2010 [51] using the same antenna and magnetic field hardware as testing with the solid fuel. The solid fuel thruster showed a 33% higher peak density at a distance of 62cm than the gas propellant, however the total particle flux was a factor of two lower. This is attributed to the fact that the Argon gas is continually applied over the antenna duration, while the PPT is only producing plasma for the first ~50-100μsec (Fig 4.6). At a distance of 118cm the peak densities of the solid and gaseous propellant thrusters are

approximately equal ( $1.3 \pm 0.1 \times 10^{17}$  part/m<sup>3</sup>), however the particle flux difference has increase to a factor of 3. As described in Section 1.3, the helicon wave has self-containing properties (from an azimuthal current and diverging magnetic field). It's believed that the helicon wave is better able to couple into the Argon plasma due to its high density over a longer period time.

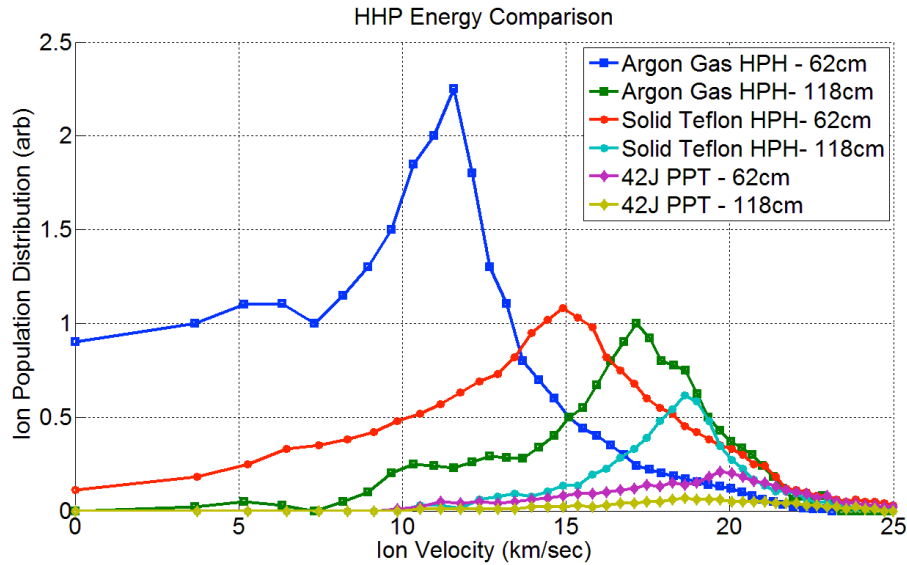


**Figure 4.9. On-axis downstream density measurements of the helicon antenna for the Argon and solid fuel HPH thrusters.**

In addition to the Langmuir probe, a retarding field energy analyzer was placed in the plume on-axis at distances of 62 and 118cm to describe the plasma velocity distribution. The PPT firing alone had the highest percentage of particles traveling at  $19.2 \pm 0.4$  km/sec and no detectable particles traveling under 10 km/sec. No large velocity difference was found between the two probe locations. The Argon gas HPH thruster had a peak velocity of 11.6 km/sec at 62 cm and 17.2 km/sec at 118 cm. This downstream velocity increase as a result of wave energy coupling into the plasma has been well documented previously. A similar velocity increase was found the solid fuel helicon thruster, however the peak velocities occurred at lower speeds than the PPT firing alone. The RFA trace of the solid fuel HPH thruster shows the existence of the high velocity PPT produced particles as well as a larger population of slower particles.

The larger population of slower particles is created by a combination of the helicon ionizing and accelerating the neutral gas produced by the PPT and additional propellant

ablation that does not occur with only the PPT discharge. The addition of the axial magnetic field to the PPT results in increase electron gyromotion above the propellant surface, while the ambipolar electric field created by the helicon system will increase the number of electron collisions with the propellant surface. Both of these will act to increase neutral collision frequency and the temperature of propellant; the result of which is more ablated Teflon.



**Figure 4.10. Downstream RFA velocity results for the Argon gas HPH, the solid fuel HPH, and the PPT.**

These results shown that it is possible to integrate a solid fuel propellant source with the helicon system. There were a number of promising results with this initial testing of the solid fuel HPH thruster:

- (1) order of magnitude peak density and density flux increases between the PPT and solid fuel HPH thruster
- (2) high peak density with the solid fuel HPH thruster
- (3) increased ablation and acceleration of Teflon propellant with the solid fuel HPH thruster
- (4) existence of downstream velocity increase of the bulk plasma with the solid fuel HPH thruster

These results show that the helicon physics do not dominate the dynamics of the PPT, but rather they both exist within the system. The PPT in its current configuration is producing

far fewer particles than the gas propellant system, limiting the helicons influence on the plasma. This implies that increasing the amount of propellant input to the system can further increase the performance of the solid fuel helicon thruster.

**Table 4.2. Peak plasma density and particle flux measurements for the PPT, solid fuel HPH, and Argon gas HPH thrusters.**

| PPT Energy (J) | Probe Location (cm) | Peak Density (part/m <sup>3</sup> ) | Particle Flux (part/m <sup>3</sup> -μsec) | Peak Velocity (km/sec) |
|----------------|---------------------|-------------------------------------|-------------------------------------------|------------------------|
| PPT Only       |                     |                                     |                                           |                        |
| 15             | 62                  | $1.2 \times 10^{16}$                | $1.0 \times 10^{12}$                      | --                     |
| 25             | 62                  | $4.2 \times 10^{16}$                | $3.1 \times 10^{12}$                      | --                     |
| 32             | 62                  | $5.4 \times 10^{16}$                | $4.1 \times 10^{12}$                      | --                     |
| 43             | 62                  | $7.5 \times 10^{16}$                | $5.5 \times 10^{12}$                      | 19.6                   |
| 43             | 118                 | --                                  | --                                        | 18.8                   |
| SF HPH         |                     |                                     |                                           |                        |
| 15             | 62                  | $1.1 \times 10^{17}$                | $0.6 \times 10^{13}$                      | --                     |
| 25             | 62                  | $3.1 \times 10^{17}$                | $2.1 \times 10^{13}$                      | --                     |
| 32             | 62                  | $5.2 \times 10^{17}$                | $3.9 \times 10^{13}$                      | --                     |
| 43             | 62                  | $7.5 \times 10^{17}$                | $5.9 \times 10^{13}$                      | 14.9                   |
| 43             | 118                 | $1.3 \times 10^{17}$                | $1.0 \times 10^{13}$                      | 18.4                   |
| Argon Gas [51] |                     |                                     |                                           |                        |
| --             | 62                  | $5.0 \times 10^{17}$                | $1.5 \times 10^{14}$                      | 11.6                   |
| --             | 118                 | $1.5 \times 10^{17}$                | $2.7 \times 10^{13}$                      | 17.2                   |

#### 4.4 Current Smoothing of the Coaxial Pulsed Plasma Thruster

The PPT current flows for 70μsec, considerably shorter than the antenna is on for. This suggests two areas for more efficient operation:

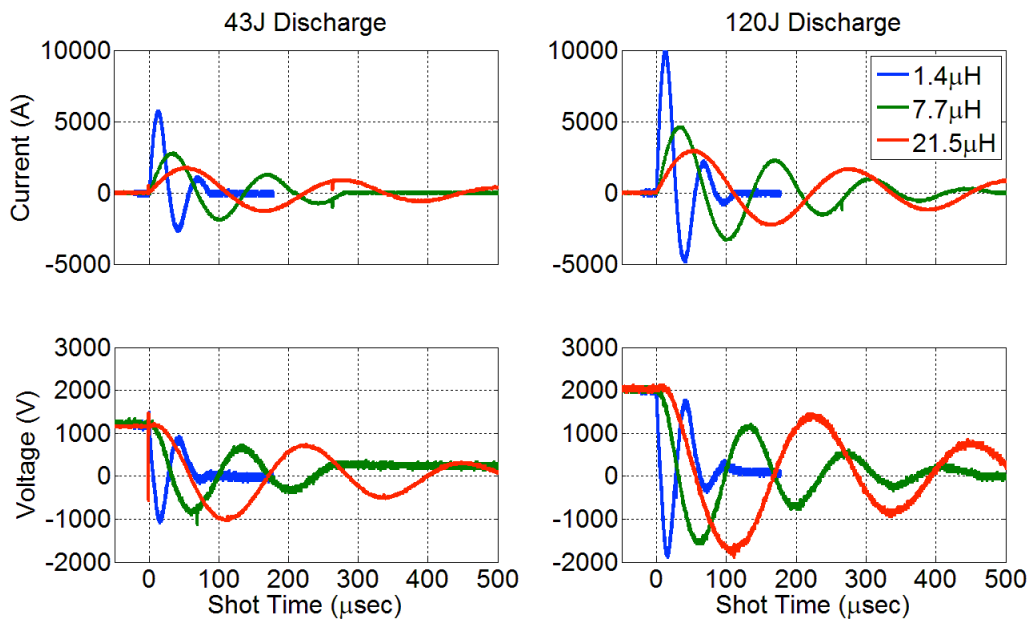
- (1) reduce the length of antenna pulse to reduce the overall power input
- (2) increase the amount of ablated material from the PPT.

The antenna pulse length is easily controlled with the IGBTs, however this won't change the downstream plasma characteristics, only increasing the thrust/power ratio and therefore increasing the overall efficiency of the thruster. Increasing the length of the PPT current pulse is more complicated.

The PPT discharge can be modeled as an RLC circuit with Eq. (1.7). Increasing the inductance will reduce the damping coefficient  $\sim L$  and the natural frequency  $\sim L^{0.5}$ . The result of these two changes will be an increase to the damped frequency and a decrease to the attenuation, both of which should work to lower the peak current and lengthen the

overall current pulse. These two effects should create a smooth ablation of low energy Teflon particles with a number density. The goal being a soft, simmering boil of the Teflon surface compared to an abrupt extremely high temperature ablation.

A typical PPT for in-space propulsion will see increased performance with hot, fast ions. In the case of using the PPT to feed the helicon thruster, that's not the case. A colder, denser, higher number density plasma should provide higher densities downstream of HPH without sacrificing velocity as the acceleration of the particles due to the helicon should have a larger effect than from the PPT.



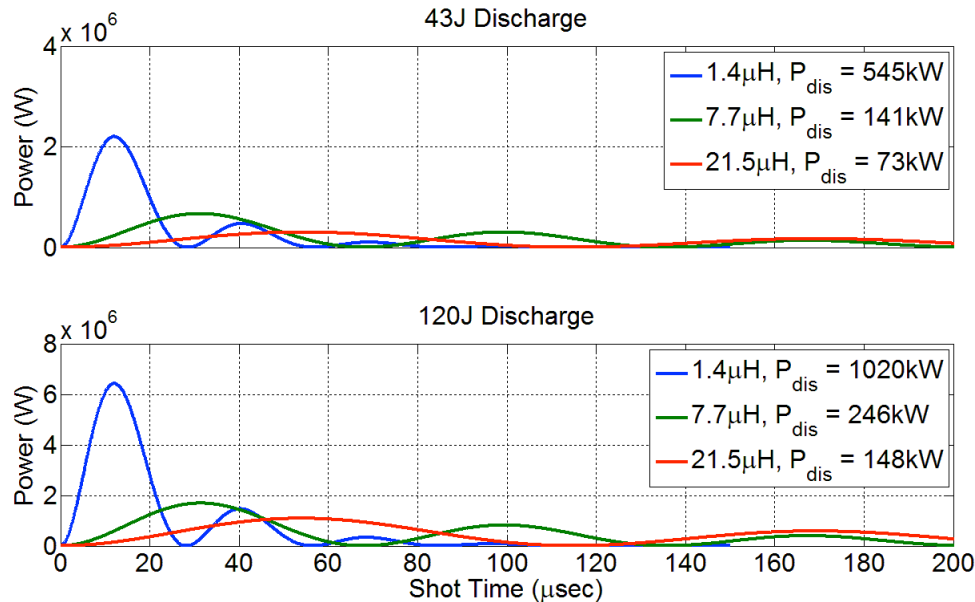
**Figure 4.11. Current and voltage traces of the PPT discharge taken at 43 (left) and 120 J (right) at 1.4 (blue), 7.7 (green), and 21.5 $\mu$ H (red).**

Initial testing of increasing the inductance was completed with the PPT in the bell jar without the helicon thruster attached and the results are shown in Fig. 4.11. Computer simulations show the stray inductance of the wires to be 1.4 $\mu$ H. Inductors of 6.3 and 20.1 $\mu$ H were added for cases 2 and 3, bringing the total inductance for the three cases to 1.4, 7.7, and 21.5 $\mu$ H, respectively. The 6.3 $\mu$ H coil consisted of 9 wraps of high voltage wire at an average diameter of 7cm, while the 20.1 $\mu$ H coil consisted of 22 wraps at an average diameter of 7.5cm. The two large-scale distinguishing features expected with an increase in inductance, a decrease in peak current and a lengthened pulse, are both seen. The peak current decreases 68% and 70%, while the pulse length increases 708% and

571% as the inductance is increased from 1.4 to 21.5 $\mu$ H for the 43 and 120J discharges, respectively. Computer simulations (based on Eq. 1.7) were found to fit the data to within 90% until the current stopped flowing. The simulations fit the data to within 95% for the first half period.

**Table 4.3. The simulation values of the current pulses from the discharge of a 60 $\mu$ F capacitor charged to 43 and 120J at 1.4, 7.7, and 21.5 $\mu$ H.**

| Inductance Added ( $\mu$ H)              | 0     |       | 6.3  |      | 20.1 |      |
|------------------------------------------|-------|-------|------|------|------|------|
| Capacitor Energy (J)                     | 43    | 120   | 43   | 120  | 43   | 120  |
| Inductance ( $\mu$ H)                    | 1.4   | 1.4   | 7.7  | 7.7  | 21.5 | 21.5 |
| Resistance (m $\Omega$ )                 | 80    | 70    | 90   | 85   | 150  | 110  |
| Amplitude (kA)                           | 8.5   | 14.0  | 3.3  | 5.5  | 2.2  | 3.4  |
| Attenuation (kHz)                        | 28.6  | 26.9  | 5.8  | 5.5  | 3.5  | 2.6  |
| Nat. Frequency (kHz)                     | 109.1 | 113.2 | 46.5 | 46.2 | 27.8 | 28.2 |
| Dam. Frequency (kHz)                     | 105.3 | 110.0 | 46.2 | 45.9 | 27.6 | 28.0 |
| Peak Current (kA)                        | 5.75  | 10.05 | 2.82 | 4.66 | 1.82 | 3.02 |
| Pulse Length ( $\mu$ s)                  | 84    | 118   | 281  | 482  | 679  | 792  |
| 1 <sup>st</sup> zero crossing ( $\mu$ s) | 28    | 29    | 70   | 71   | 110  | 111  |



**Figure 4.12. The power ( $I^2R$ ) as a function of time in the discharge.**

The power in each current pulse ( $I^2R$ ) was calculated and plotted for each of the 3 inductances and 2 discharge energies in Fig. 4.12. The expected power

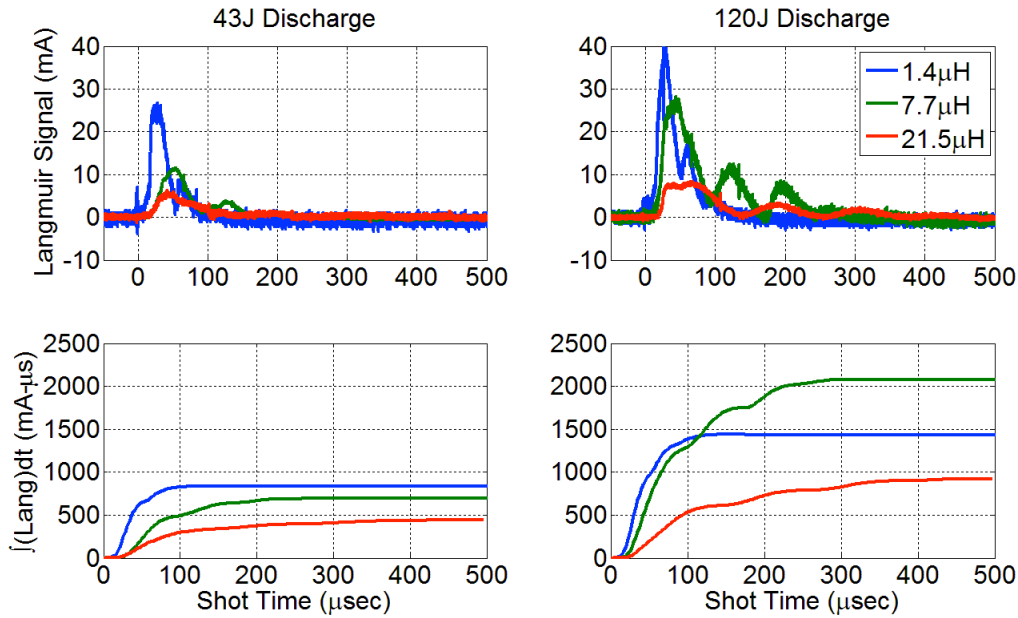
$$P_{exp} = \frac{E}{t} = \frac{CV^2}{2t_f} \quad 4.1$$

as well as the discharge power

$$P_{dis} = \int_0^{t_f} \frac{I^2 R dt}{t_f} \quad 4.2$$

for each discharge are shown in Table 4.4; the shot lengths ( $t_f$ ) in Table 4.3.

The double Langmuir probe described in Section 3.1 were biased to 45V and placed 40cm above the Teflon surface. It is assumed that the Langmuir probe will be in saturation at 45V. Shots were taken at each discharge energy and for each inductance case, the results are shown in Fig. 4.13. As expected, an increase in energy lead to an increases in the peak plasma density, while increases in inductance lead to decreases in peak plasma density but increases in the amount of time the Langmuir probe was in contact with plasma. The area under the Langmuir curve will be proportional to the amount of total plasma produced with each shot. That area is shown on the bottom in Fig. 4.13 as a function of time.



**Figure 4.13. The Langmuir output (top) and cumulative Langmuir output (bottom) for discharge energies of 43 (left) and 120J (right).**



The 120J, 1.4μH case had the maximum peak plasma density, while the 120J, 7.7μH case had the maximum density flux. In order to determine the efficiency of plasma production, a plasma production coefficient can be defined as

$$C_p = \frac{1}{E} \int (Lang) dt \quad 4.3$$

It is the lowest energy and inductance case (43J, 1.4μH) that has the highest  $C_p$ , implying the lower energy discharges have a higher ablation rate. For the 43J case, plasma production decreased 17% with an inductance increase from 1.4 to 7.7μH and again decreased 36% as the inductance was increased to 21.5μH. However at 120J, the plasma production increased 108% as the inductance was raised from 1.4 to 7.7μH and then decreased 57% as the inductance was increased to 21.5μH.

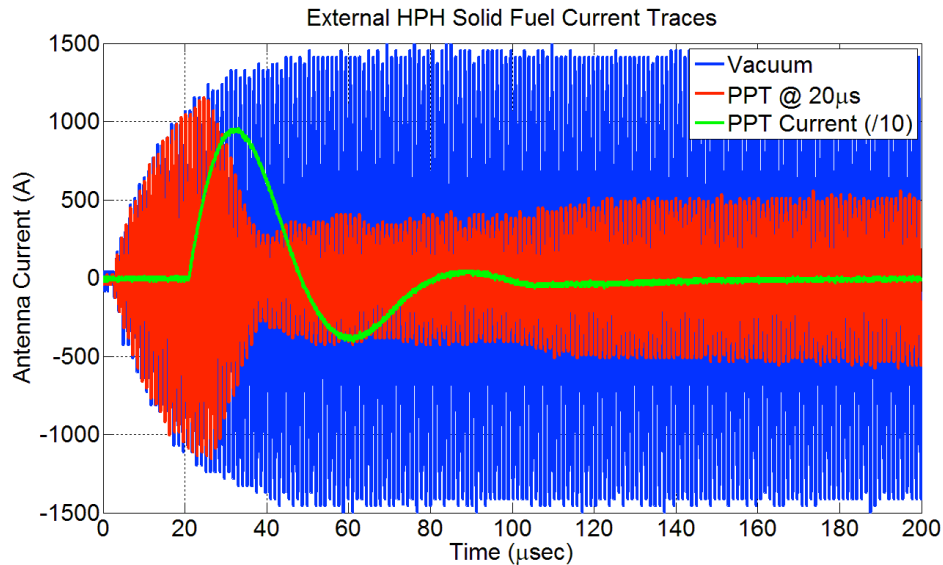
**Table 4.4. The expected and discharge power found from current traces and the Langmuir results at 43 and 120J with inductances of 1.4, 7.7, and 21.5μH.**

| Inductance (μH)             | 1.4   |        | 7.7   |       | 21.5 |       |
|-----------------------------|-------|--------|-------|-------|------|-------|
| Capacitor Energy (J)        | 43    | 120    | 43    | 120   | 43   | 120   |
| Expected Power (kW)         | 511.9 | 1016.9 | 153.0 | 249.0 | 63.3 | 151.5 |
| Discharge Power (kW)        | 545.0 | 1020.0 | 140.8 | 246.2 | 72.8 | 147.7 |
| Peak Density Signal (mA)    | 26.1  | 11.4   | 6.1   | 41.0  | 27.6 | 8.1   |
| Density Signal Flux (mA-μs) | 834   | 689    | 441   | 1438  | 2079 | 914   |
| $C_p$ (mA-μs/J)             | 19.4  | 16.0   | 10.3  | 12.0  | 17.3 | 7.6   |

This analysis does not take into account neutral production from the PPT, which has been measured in other experiments as being approximately 90% of all particles ablated from the Teflon surface. The HPH ionization efficiency has been measured previously to be 90% [43]. Assuming that value of 90% stays constant, higher plasma production is an indication of a larger total population of ablated material. This suggests that a PPT fired at 120J with 7.7μH of inductance will produce a longer and more uniform propellant input to the helicon thruster.

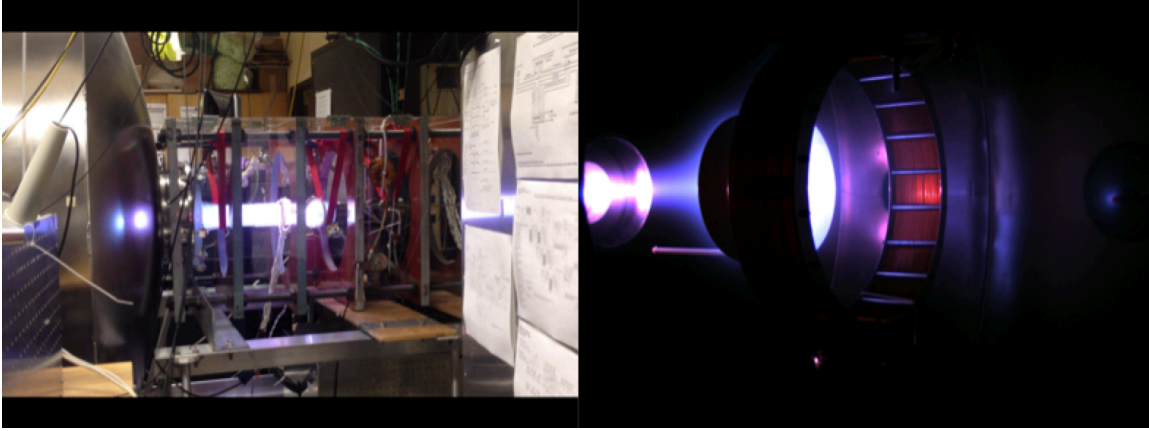
## 4.5 Integration with the External High Power Helicon Experiment

An upgraded helicon power supply (described in Section 2.4) was installed shortly after the results of Section 4.3 were taken. This supply was capable of antenna currents reaching 3.2kA peak-to-peak and voltages of 32kV peak-to-peak with a 7.8mF of energy storage. The magnet supply was also increased in sophistication, allowing for 1kG fields with 4.8mF of energy storage.



**Figure 4.14. The external HPH antenna current with (red) and without (blue) the PPT (green) discharge.**

The existing electrical connections on the chamber were not suited for the high voltages and currents associated with the antenna and magnet power supplies. As such, the new helicon system was designed to be external to the chamber. A pyrex flange was placed over an ISO-250 port on the chamber and the antenna and magnets were placed around it. In addition to easier electrical access, the new setup allowed for changing the position of the antenna, base magnetic field coils, and PPT relative to each other.



**Figure 4.15. Light emission from by external helicon experiment firing with the solid fuel PPT viewed at the source region (left) and downstream (right).**

The downstream nozzles, visible in the right panel of Fig. 4.15) used with the internal helicon experiment were used in the same configuration, position, and current levels with the external helicon. The small nozzle (13cm radius) and large nozzle (23cm radius) extended the downstream magnetic field in addition to creating a converging/diverging effect as shown in Fig. 2.20.

#### **4.5.1 System Calibration**

Using a PPT with no added inductance, the relative position and timing of the experiment for maximum performance was found. As a baseline for the base magnetic field, the magnet supply capacitance bank voltage was varied between 100-550V and the current and magnetic field were measured (shown in Fig. 4.16). As expected, per Eq. 1.7, the current scaled linearly with the capacitor voltage and the magnetic field scaled linearly with the current. The maximum field, in Gauss, was found to be linearly related to the charge voltage by Eq. 4.4.

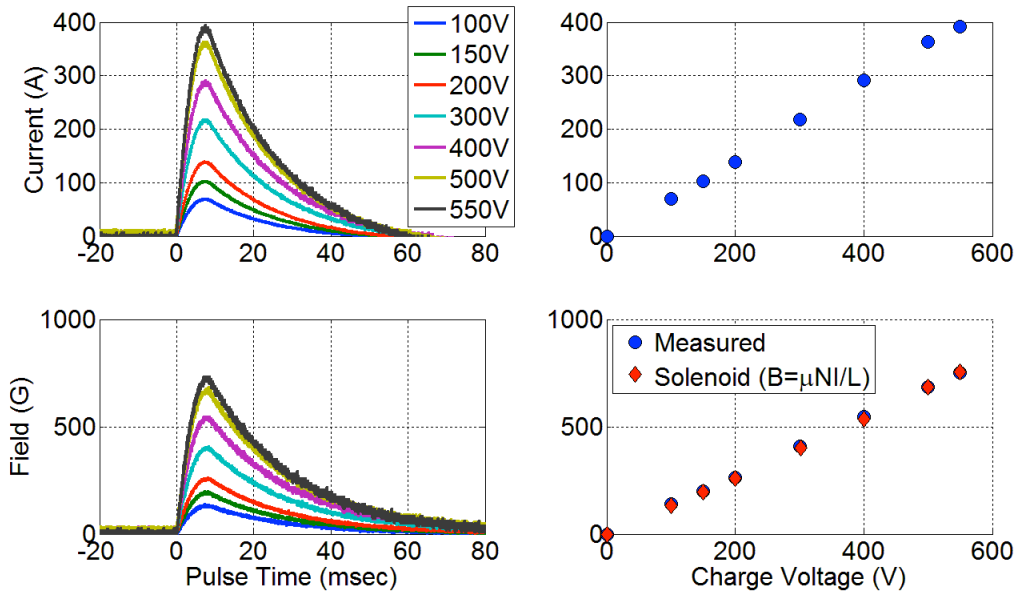
$$B_{max} = 1.35V_{charge} \quad 4.4$$

If the magnets were assumed to be arranged in a solenoid, an expected field would be found with Eq. 4.5, where  $N$  is the number of turns (180 over all six magnets),  $I$  is the measured peak current, and  $l$  is solenoid length (1.2m between first and sixth magnet). Such an approximation matches the measured field data to within 2%.

$$B_{max} = \frac{\mu_0 NI}{l} \quad 4.5$$

The current through an inductor is related to the capacitor voltage by Eq. 2.3, which, when substituted into Eq. 4.5, gives a relationship between the magnetic field and capacitor voltage which matches Eq. 4.4 to within 8%, when the rise time is assumed to be the quarter period of the systems natural frequency.

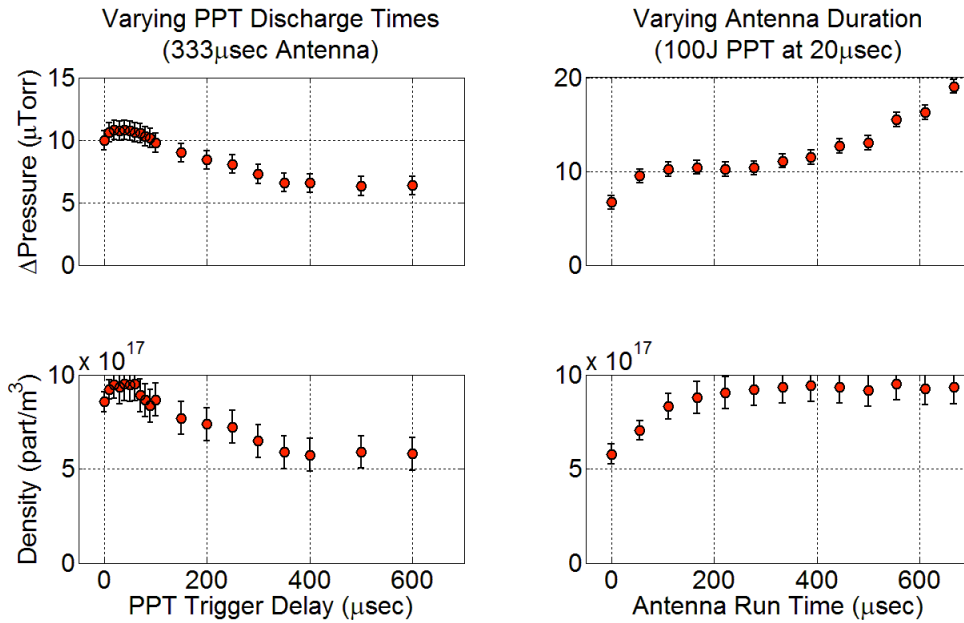
$$B_{max} = \frac{\mu_0 NI}{l} = \frac{\mu_0 N}{l} \left( -\frac{1}{L} \int V dt \right) = \frac{\mu_0 N}{4l} \sqrt{\frac{C}{L}} V_{charge} = 1.46 V_{charge} \quad 4.6$$



**Figure 4.16. The measured current (top left) and magnetic field (bottom left) of the base magnetic field for capacitor voltages of 100-550V. The current peak (top right) is given as well as the maximum field (bottom right) along with a solenoid approximation of the peak field.**

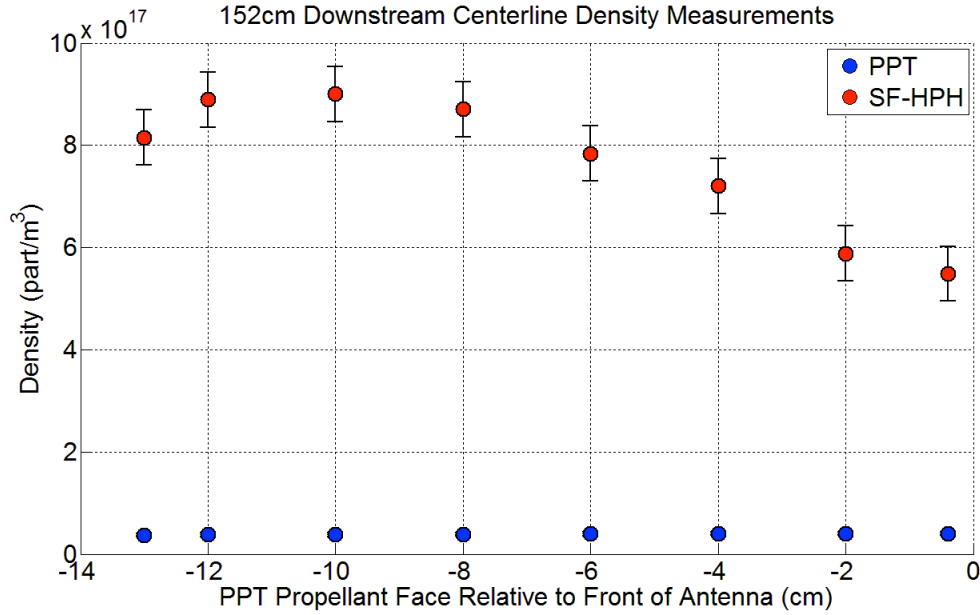
The chamber pressure increase and on-axis Langmuir signal at a downstream distance of 152cm with a constant 780G base field and 100J PPT discharge were measured for varying antenna durations and PPT triggering times and are shown in Fig. 4.17. The chamber pressure increase is a measurement of the total number of particles ablated, while the Langmuir signal is a measurement of the generated plasma density. With an antenna duration of 333μsec, the chamber pressure increase and downstream Langmuir

output both peak at triggering the PPT 20-30 $\mu$ sec after the antenna. Triggering later in time results in a portion of the ablated material not interacting with the helicon wave, hence the decrease in both the Langmuir and pressure measurements. The downstream Langmuir signal increased as the antenna duration was increased was 0-300 $\mu$ sec, while for durations longer than 300 $\mu$ sec, the Langmuir signal remained relatively constant. The chamber pressure increase however, continued increasing for duration of 0-700 $\mu$ sec. This is due to the helicon wave continuing to ablate material from the PPT, but not necessarily ionizing and ejecting it downstream for long antenna durations.



**Figure 4.17. Chamber pressure increase and peak density (probe at 152cm) for the thruster at varying PPT trigger times (left) and varying antenna durations (right).**

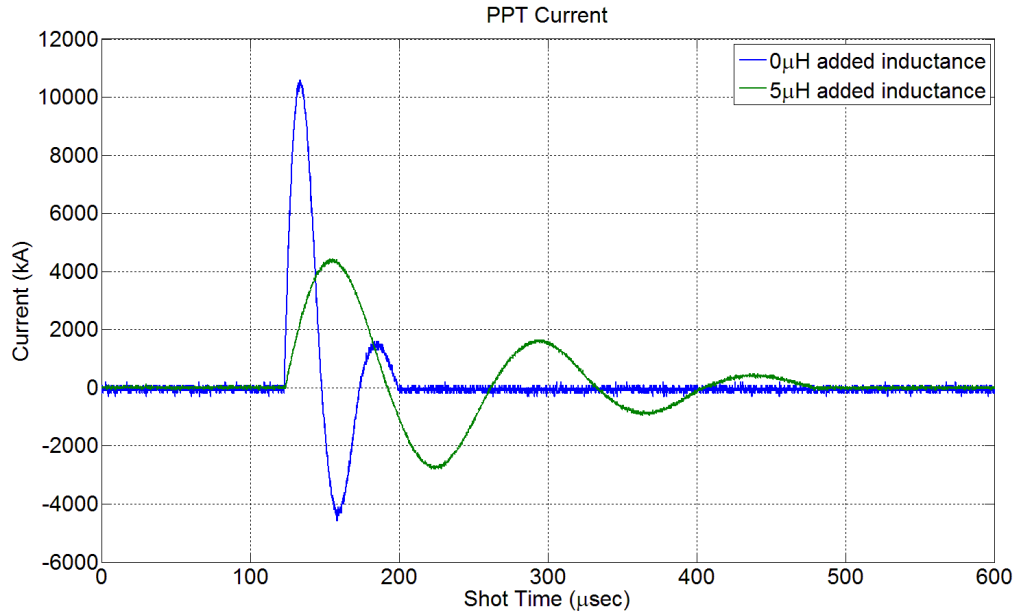
The position of the PPT propellant face relative to the front of the helicon antenna (14cm length) was varied for a 100J PPT discharge triggered 20 $\mu$ sec after the antenna, which was operational for 333 $\mu$ sec. The position of the PPT was varied between 1cm from the back of antenna to 0.5cm from the front. The Langmuir signal was relatively unchanging when the PPT was fired alone, while the helicon Langmuir signal peaked with the PPT positioned 10cm from the front of the antenna. This position (~30% from the back) agreed with the maximum performance position of the gas-puff in the gaseous HPH thruster [43].



**Figure 4.18. Downstream plasma density at 152cm for a 100J PPT and the solid fuel helicon experiment (100J PPT, 580G field, and 2.9kA antenna) as a function of the PPT position relative to the antenna.**

#### 4.5.2 Varying PPT Inductance

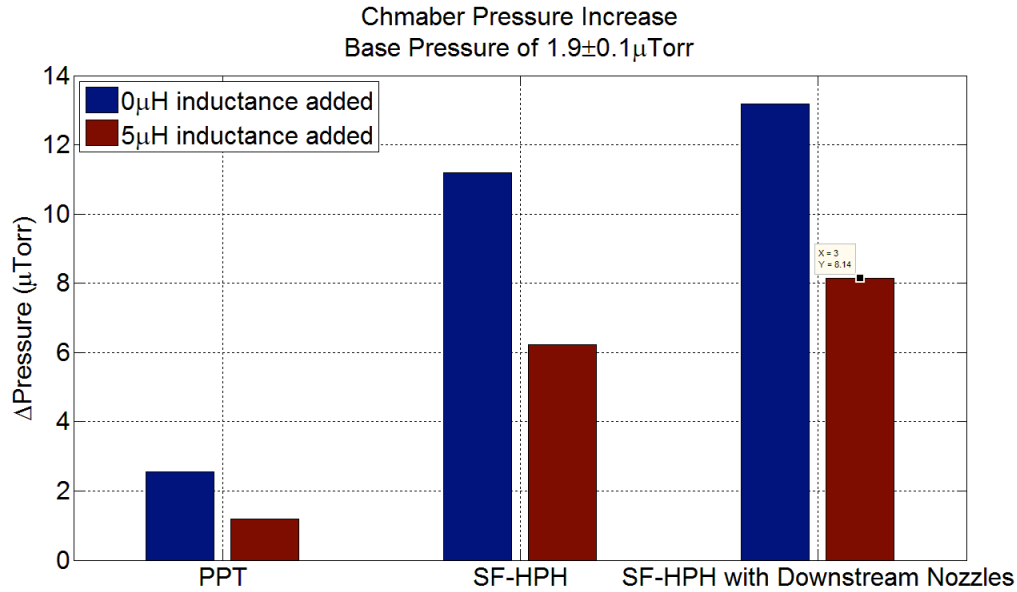
Based on results of Section 4.4, a 5 $\mu$ H coil was added to the main discharge of the PPT to increase the total number of plasma particles produced. The current traces of the standard 100J PPT (no added inductance) and a 5 $\mu$ H added inductance 100J PPT are shown in Fig. 4.19 for the device configured for operation on the main chamber for helicon use. The results shown in Section 4.4 were all taken from within the bell jar vacuum chamber which allowed for shorter electrical wires from outside to inside the chamber, resulting in the minor current waveform differences. Modeling with Eq. 1.7 shows that the PPT electrical leads had 1.7 $\mu$ H of stray inductance, resulting in a total inductance of 1.7 and 6.7 $\mu$ H for the two cases.



**Figure 4.19. Current traces of a 100J PPT with 0 and 5μH of added inductance.**

The maximum pressure increase in the chamber was found for both inductance cases of the PPT. As described in Section 3.5, the chamber pressure increase is proportional to the total number of ablated particles. In Section 4.4 it was found that the peak plasma density occurred with no added inductance, while adding 5μH of inductance created the highest total number of plasma particles. Chamber pressure measurements (Fig. 4.20) showed that although the 5μH PPT produced more total plasma particles, adding no inductance produced more total particles. The same effect was found for the helicon, both with and without downstream nozzles.

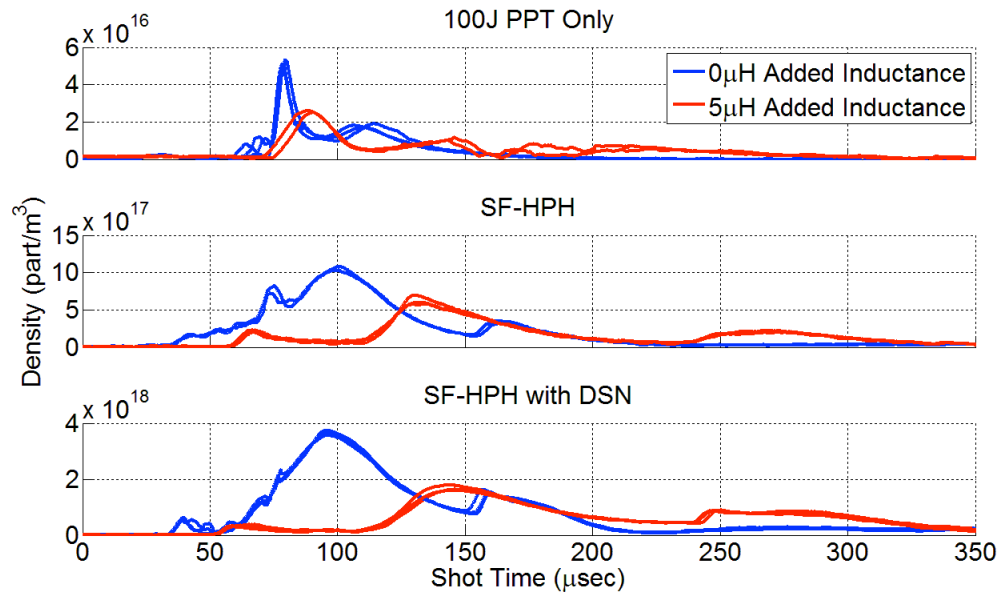
Based on results from the internal HPH, the plasma generated by the PPT is left relatively unaffected by the helicon, while the helicon ionizes and accelerates the neutral produced by the PPT as well as increasing the ablation rate. A higher PPT current pulse will create high temperatures, which should produce more neutral particles.



**Figure 4.20. Chamber pressure increase for the 100J PPT, solid fuel helicon, and solid fuel helicon with downstream nozzles for 0 and 5μH of added PPT inductance.**

Similar to the findings in Section 4.4 for only a PPT, a downstream Langmuir probe placed on-axis of the helicon thruster show a higher peak density with the 0μH PPT while the helicon with the 5μH PPT produces particles over a longer duration. This was found with and without the downstream nozzles included. The production of more total particles as well as higher plasma densities suggests the highest performance for the solid fuel HPH will occur with a low inductance, high current PPT. This will limit the length that the helicon can operate for, eliminating the possibility for steady-state operation in its current configuration. High efficiency can still be achieved with short duration pulses by focusing on increasing the ionization fraction of the ablated material. This is a similar concept to the μPIT thruster concept [11] that attempted to inductively add energy to the plasma plume, although at a drastically lower energy levels.





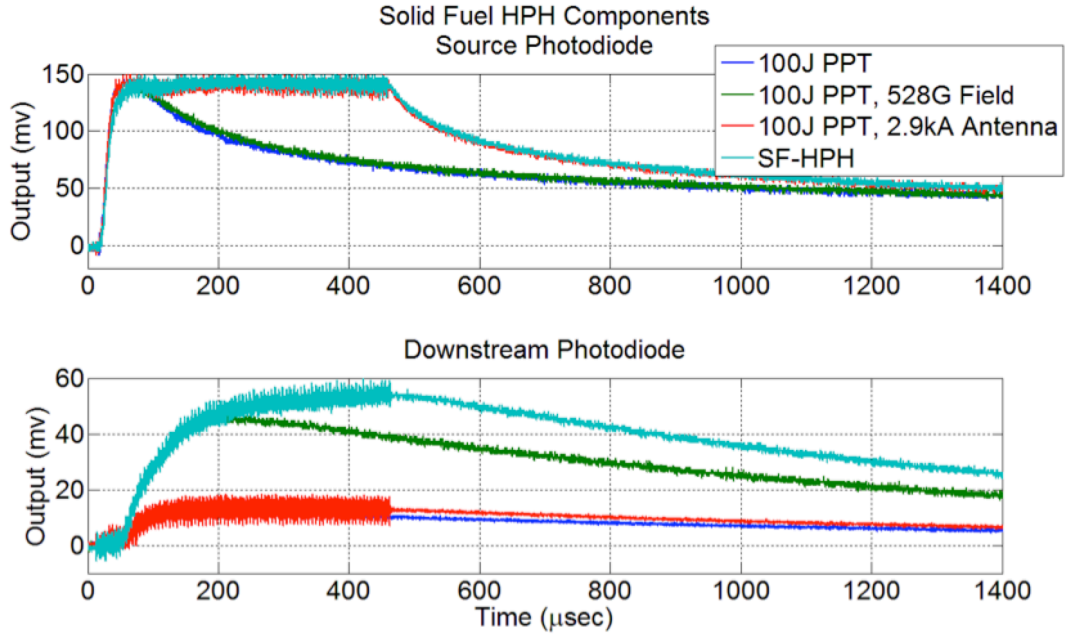
**Figure 4.21. Density measurements at 152cm downstream for the PPT (top), the solid fuel helicon with (center) and without (bottom) downstream nozzles.**

#### 4.5.3 Thruster Components

The solid fuel helicon experiment has four distinct components that all interact together. In addition to the density results in Figs. 4.17, 4.18, and 4.21, photodiodes, chamber pressure measurements, and high speed camera imagery was used to determine each components influence on the plasma. Two photodiodes were placed to view the plasma generated by the experiment, one viewing the source region pyrex tube and the other through a chamber port downstream. Photodiode results for the 100J PPT discharge with the antenna, magnetic field, and both are shown in Fig. 4.22. The photodiodes saturated at high light emission input with an output of 150mV. The source region photodiode showed little difference between adding the base magnetic field to operating with only the PPT, as well as negligible difference between the PPT and antenna firing with and without the base magnetic field. The light emission with the antenna remains constant over the duration the antenna is operating.

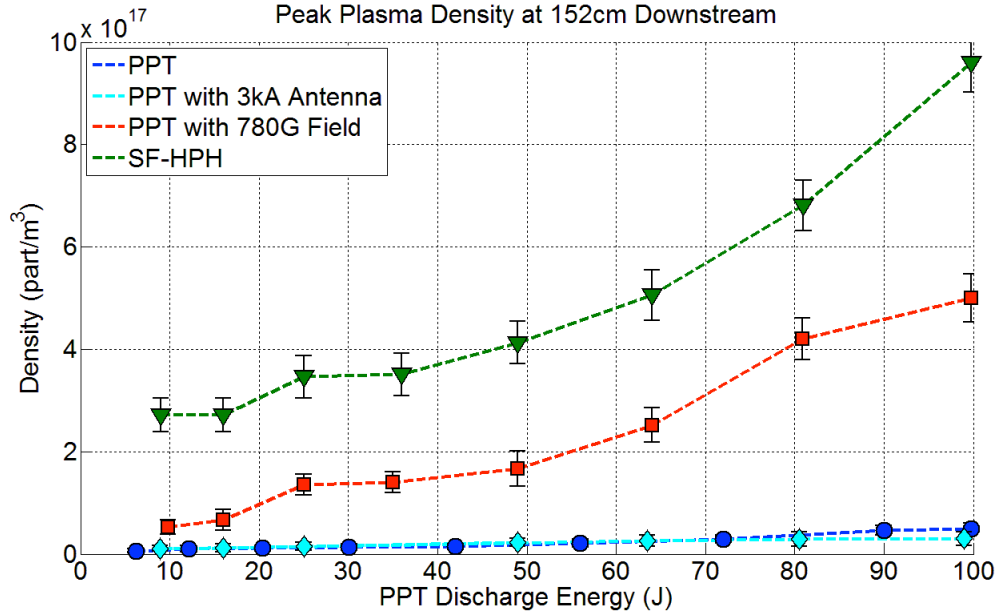
The downstream photodiode shows that the antenna without an axial magnetic field generates 25% more light production (15mV) than just the PPT alone (12mV), implying that without a magnetic field to propagate the helicon wave, the antenna increases ionization in the source region but is unable to accelerate the particles. A 275% increase

in downstream light emission is found with the addition of the magnetic field over the PPT firing alone, as well as with the complete solid fuel experiment (350% light emission increase). The antennas influence is larger on the system with the base field than without.



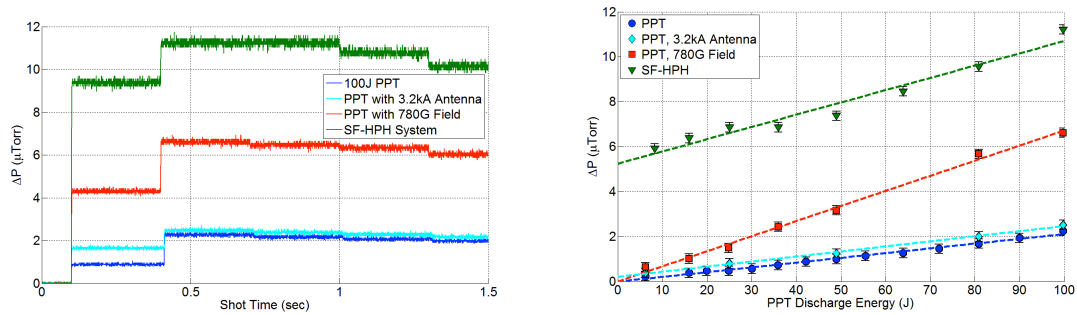
**Figure 4.22. Source (top) and downstream (bottom) photodiode output voltage for a 100J PPT, the PPT with a 528G field, the PPT with the 2.9kA helicon antenna, and the all three components together.**

Downstream density measurements were taken of the system at varying PPT discharge energies; the peak values of which are shown in Fig. 4.23. The density results agree with the conclusions drawn from the photodiode data. An increase from  $5 \times 10^{15}$  to  $5 \times 10^{16}$  part/m<sup>3</sup> was found with the PPT as the discharge energy was varied between 6 and 100J. The 3kA antenna without the base field produced no noticeable increase of on-axis plasma density, while the 780G base field increased the peak density an order of magnitude over the PPT alone. The full helicon system produced a doubling of plasma density as was found without the antenna, reaching a peak density of  $9.5 \times 10^{17}$  part/m<sup>3</sup> with a 100J PPT discharge. The continued increase of the peak density with increasing PPT discharge energy implies that the optimal mass input to the helicon for maximum thrust has not yet been reached. Further increases in discharge energy, or modifications to the PPT geometry to increase the mass ablation rate, is suggested for further performance increase.



**Figure 4.23. 152cm downstream plasma density for variations of the solid-fuel helicon at varying PPT discharge energies.**

Results in the left panel of Fig. 4.24 show the chamber pressure traces for a 100J PPT discharge, while results in the right panel show the peak pressure increase as a function of PPT energy for 10-100J discharges. The chamber had a base pressure of  $1.9 \pm 0.1 \mu\text{Torr}$  for all results. Similar to the photodiode results, the antenna without a base magnetic field has little effect on the total ablation rate (12% increase), while the base field without the antenna is the single largest influence on the amount of ablated material (200% increase). The complete solid fuel helicon ablates 400% more propellant than the PPT alone and 67% more propellant than the PPT firing with the base field. Varying the discharge energy produces linear effects on all four configurations of the experiment. The chamber pressure increase from the PPT firing alone, with the base field, and with the antenna all reduce to zero as the PPT discharge energy is reduced. The solid fuel system however linearly decreases to  $5.6 \mu\text{Torr}$  as the PPT energy approaches 0.



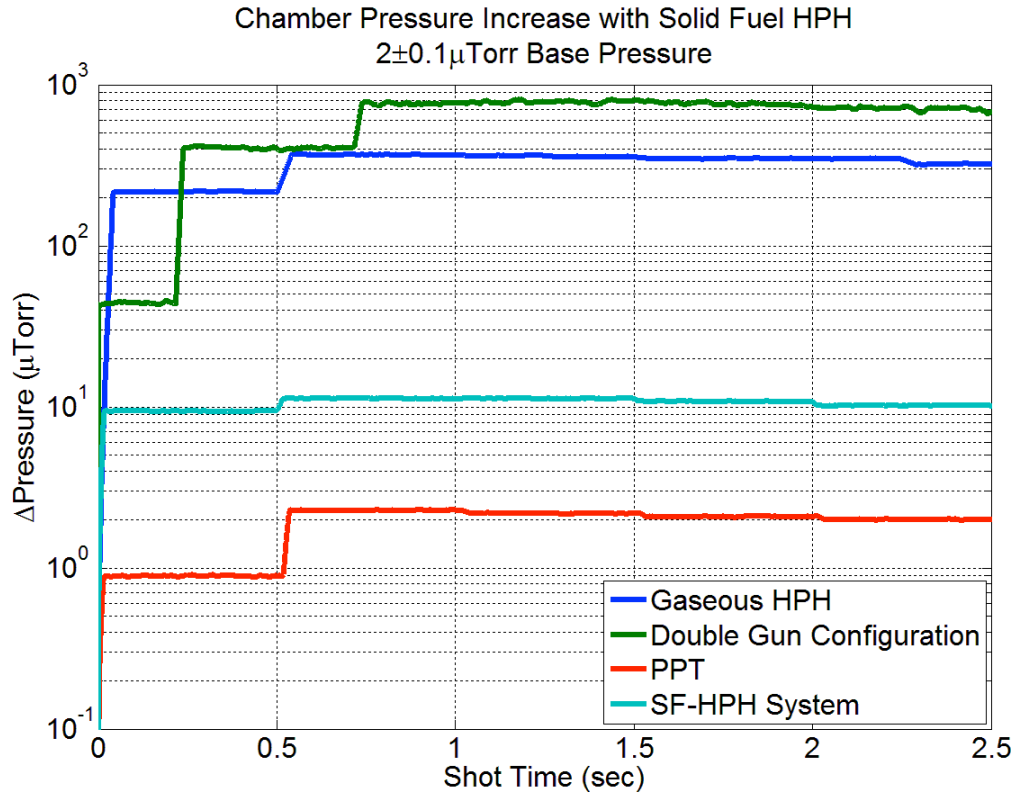
**Figure 4.24. Chamber pressure traces from a PPT, PPT with antenna, PPT with field, and the complete solid fuel HPH system (PPT, field, antenna) as a function of shot time (left) and the peak pressure increase for varying PPT discharge energies (right).**

Comparing the pressure increase from the solid fuel helicon to the gaseous helicon is shown in Fig. 4.25. The gaseous helicon shows a pressure increase of  $400\mu\text{Torr}$ , while the gaseous double gun helicon results in an  $800\mu\text{Torr}$  increase. This two order of magnitude higher pressure change with the gaseous thruster suggests two conclusions:

- (1) the solid fuel helicon has a higher mass efficiency than the gaseous configuration, ionizing more of the total particles input to the antenna, and
- (2) as the performance of the solid fuel helicon continues to increase at the current upper energy limit of the PPT discharge, the optimum fuel input is still above what the 100J PPT is capable of producing

The chamber pressure is an indication of the total amount of ablated material from the PPT. Previous studies have shown that a Teflon PPT ablates material at a rate of  $0.1\text{--}4\mu\text{g/J}$  depending on the electrode and propellant geometry [7, 8]. To convert between pressure and ablated mass, the ideal gas law can be used, which relates the pressure ( $P$ ), volume ( $V$ ), number of moles ( $n$ ), and the temperature ( $T$ ) through a gas constant ( $R=8.314\text{J/mol-K}$ ).

$$PV = nRT \quad 4.7$$



**Figure 4.25. The chamber pressure increase from a 100J PPT, the solid fuel helicon thruster, a single gaseous helicon thruster, and the double gun configuration of the helicon thruster.**

The vacuum chamber has an interior volume of 4800L. At standard atmospheric pressure (101kPa) and temperature (293K), Eq. 4.7 results in 200mol of air within the chamber. Dry air, composed primarily of nitrogen ( $N_2$ ) has a molar mass (MM) of 28.9g/mol, which gives a mass of 5.7kg of gas molecules inside the vacuum chamber while at atmospheric pressures. While at vacuum (2μTorr), the chamber would consist of 15μg of air. The Teflon exhaust is estimated to have a MM of 31g/mol, as fluorine exists as a diatomic molecule with a MM of 37.9g/mol, and while carbon can't exist as  $C_2$  in an oxygen rich environment,  $C_2$  would have a MM of 24g/mol, resulting in an average MM of 31g/mol. The gaseous HPH thruster uses Argon propellant, which exists as a monoatomic gas as it's a noble gas and has a MM of 39.9g/mol. Based on these molar masses, it can be estimated that the gaseous HPH thrusters will input 4-8mg of Argon per discharge, while the PPT alone ablates 20μg of Teflon from a 100J discharge and the solid fuel HPH ablates 80μg. This results in a PPT ablation rate of 0.2μg/J and a solid fuel helicon ablation rate of 0.8μg/J if only the PPT energy is considered. It should be

noted that these ablation rates only account for the material detected by the pressure sensor and may not include ablated macroparticles, which previous studies have shown to be as high as 30% of the total ablated Teflon mass [18]. These macroparticles however are accelerated up to low velocities ( $\sim 500\text{m/s}$ ) and do not contribute substantially to the impulse of a PPT.

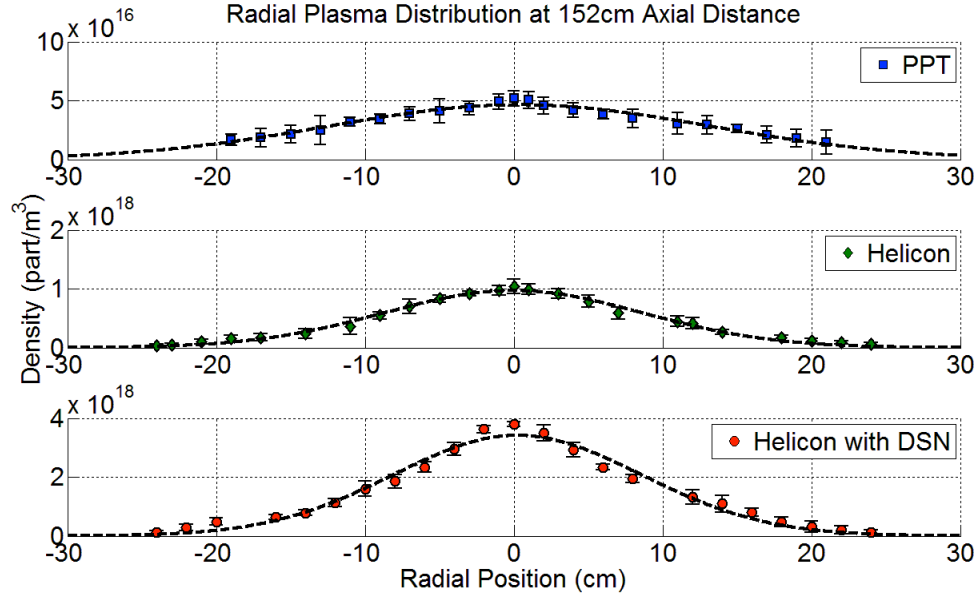
**Table 4.5. HPH propellant masses for the solid fuel and gaseous configurations.**

|             | Composition | $\Delta P$<br>( $\mu\text{Torr}$ ) | MM<br>(g/mol) | Propellant<br>Mass ( $\mu\text{g}$ ) | Ablation<br>Rate ( $\mu\text{g/J}$ ) |
|-------------|-------------|------------------------------------|---------------|--------------------------------------|--------------------------------------|
| Atmosphere  | Air         | 760                                | 28.9          | $5.8 \times 10^9$                    | --                                   |
| Base Vacuum | Air         | 1.9                                | 28.9          | 15                                   | --                                   |
| PPT         | Teflon      | 4.4                                | 31            | 20                                   | 0.2                                  |
| SF HPH      | Teflon      | 11.0                               | 31            | 80                                   | 0.8                                  |
| HPH         | Argon       | 400                                | 39.9          | $4.2 \times 10^3$                    | --                                   |
| DG HPH      | Argon       | 800                                | 39.9          | $8.4 \times 10^3$                    | --                                   |

A radial sweep with the 152cm downstream Langmuir probe was completed to determine the spatial variation to the plume. The Langmuir probe was varied between -25 to +25cm of the centerline for thruster variants of a 100J PPT, the solid fuel HPH (100J PPT with 3.1kA antenna and 780G field), and the solid fuel HPH with the 13cm radius downstream nozzle (nozzle 1) running at 3.3A. The density is shown in Fig. 4.26 with best-fit Gaussian curves overlaid, which matches the data to within 15% for radii within 15cm. The errorbars show the maximum shot-to-shot variation in peak density at each location. Peak centerline densities matched the values given in Fig 4.21 and 4.23,  $5 \times 10^{16}$  part/m<sup>3</sup> for the PPT,  $1 \times 10^{18}$  part/m<sup>3</sup> for the HPH, and  $3.8 \times 10^{18}$  part/m<sup>3</sup> for the HPH with nozzle. The FWHM of the Gaussian distributions are 25cm for the PPT, 18cm for the Helicon, and 16cm for the Helicon with the nozzle, implying that the helicon not only increases the plasma density, but better confines the plasma to the centerline. This is an expected result of the helicon as described in Section 1.3.1, and also matches radial distribution data taken with the gaseous HPH [43] which show beam widths of 10-20cm at an axial distance of 66cm downstream.

The measured FWHM for the PPT of 25cm at an axial distance of 152cm is smaller than would be expected for a truly thermal expansion of the plasma. This is believed to be due to the PPT firing from within the 7cm diameter pyrex tube. The PPT is positioned 35cm

from the front of the tube, resulting in any particles that would normally be ejected at a large scattering angle to neutralize upon interaction with the wall.



**Figure 4.26. Radial plasma density sweep with a 100J PPT (blue), the solid fuel helicon (100J PPT with 3.1kA antenna and 780G field) (green), and the solid fuel helicon with downstream nozzle 1 at 3.3A (red).**

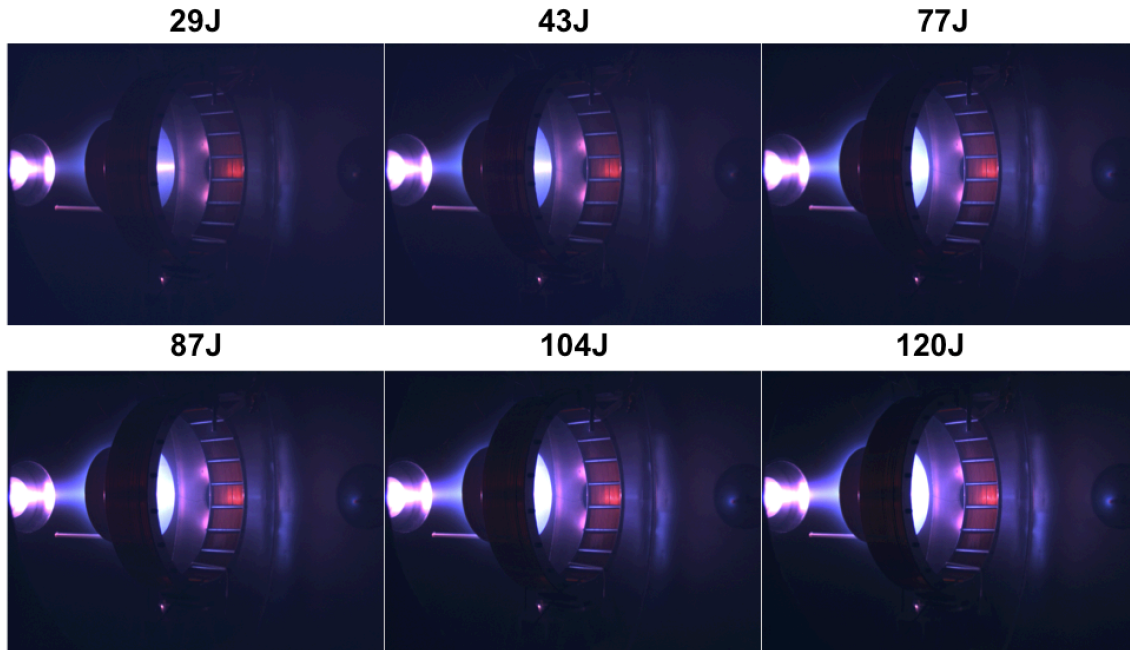
The area under each of the three curves in Fig. 4.26 was calculated, and assuming axisymmetric symmetry about the centerline, the plasma density flux ( $\Phi_n$ ), with units of particles/m, at 152cm downstream was found with

$$\Phi_n = \int_0^{2\pi} d\theta \int_0^{r_\infty} nrdr = \sum_0^{r_\infty} \pi r^2 n \quad 4.8$$

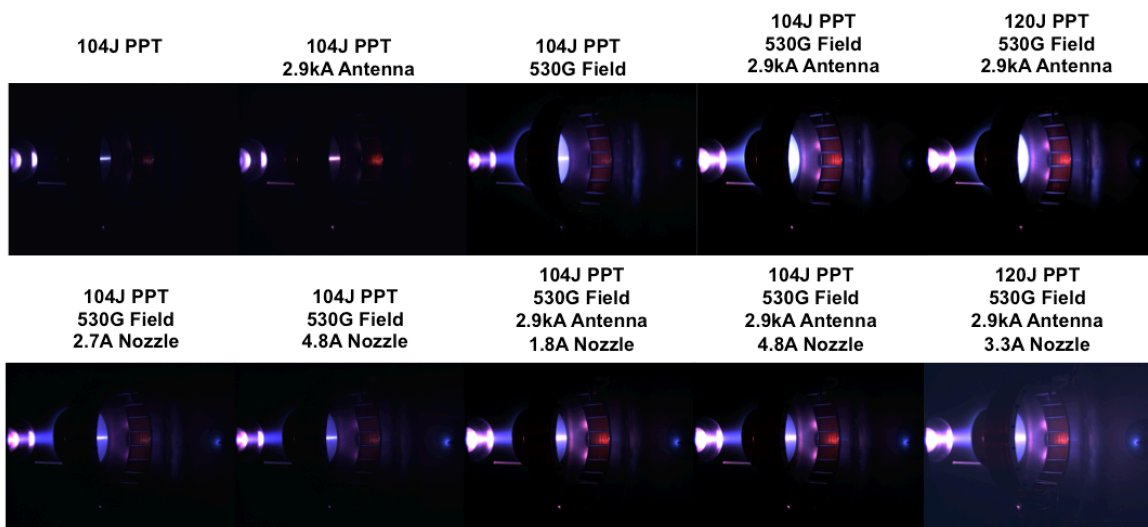
where the density ( $n$ ) at a given radius ( $r$ ) was taken from Fig. 4.26. The density flux results were measured to be  $7.7 \times 10^{20}$  part/m for the PPT,  $5.1 \times 10^{21}$  part/m for the helicon, and  $1.6 \times 10^{22}$  part/m for the helicon with downstream nozzle. If a center column of the exhaust beam at 152cm downstream is arbitrarily taken to be circular with a diameter of 10cm, Eq. 4.8 shows that the PPT has 1.5% of its exhaust in the center, 4.5% of the helicon, and 5.1% for the helicon with nozzle. These results support the FWHM measurements that the helicon acts to confine the plasma as it propagates downstream.

Downstream light emission from varying configurations of the solid fuel high power helicon thruster are shown in Figs. 4.27 and 4.28. At constant antenna current (2.9kA)

and field strength (530G). Increases in the PPT discharge energy results in higher downstream light emission and more collimation of the plasma beam. The addition of the helicon antenna without a base magnetic field results in minimal downstream light emission, while the downstream nozzle leads to increased downstream light emission.



**Figure 4.27. Downstream light emission from the helicon thruster (2.9kA antenna, 530G base field) with varying PPT discharge energies.**



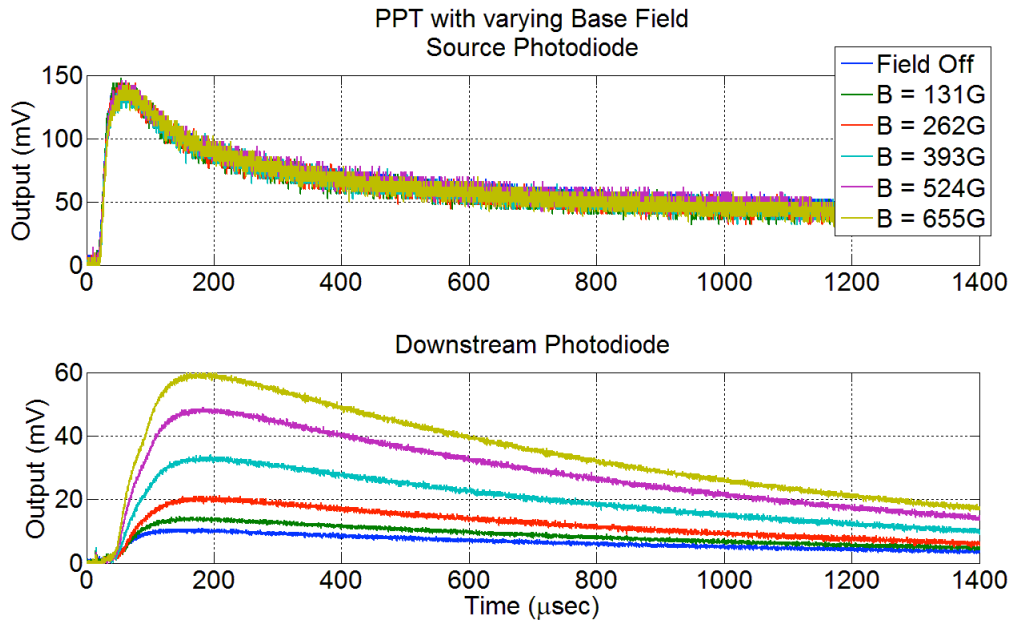
**Figure 4.28. Downstream light emission from varying combination of the solid fuel helicon thruster components.**



#### 4.5.4 Axial Magnetic Field Influence on PPT

The PPT fired with only the base magnetic field is identical to a PPT firing from the center of a dipole magnetic field or within a magnetic nozzle. Magnetic nozzles serve much the same purpose as regular material nozzles, which is to convert the random thermal energy of the exhaust into directed kinetic energy. Multiple previous studies have examined PPTs with applied radial magnetic field to increase the electromagnetic acceleration ( $J_r \times B_\theta$ ). Adding an axial magnetic field will act to increase the perpendicular gyromotion of the particles and downstream velocity as the perpendicular gyromotion is converted to parallel velocity as the magnetic field strength (magnetic pressure) decreases

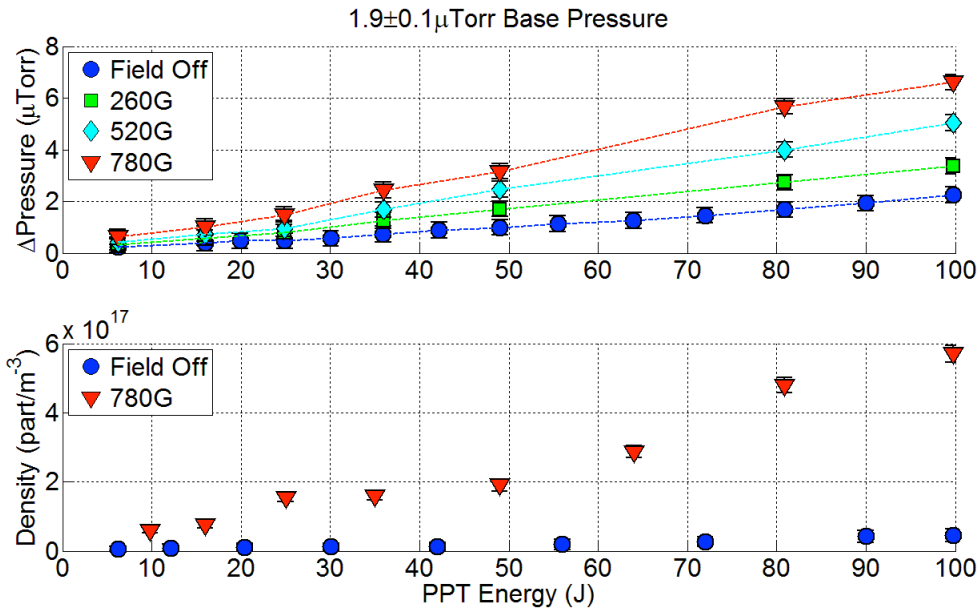
A source region and downstream photodiode were positioned to view the light emission of the plasma. As shown in Fig. 4.29 increasing the magnetic field had minimal effect on the source region light emission, but large increases on the downstream light. Coupled with the chamber pressure increases shown in Fig. 4.30, this implies increasing magnetic fields are not only increasing the ablation rate, but also ionizing the added ablated material.



**Figure 4.29. Source (top) and downstream (bottom) photodiode output voltage for a 100J PPT with applied axial magnetic fields of varying strengths.**

The chamber pressure increase due to varying magnetic field strengths is shown in Fig. 4.30 plotted against the PPT discharge energy. Higher magnetic fields produce larger

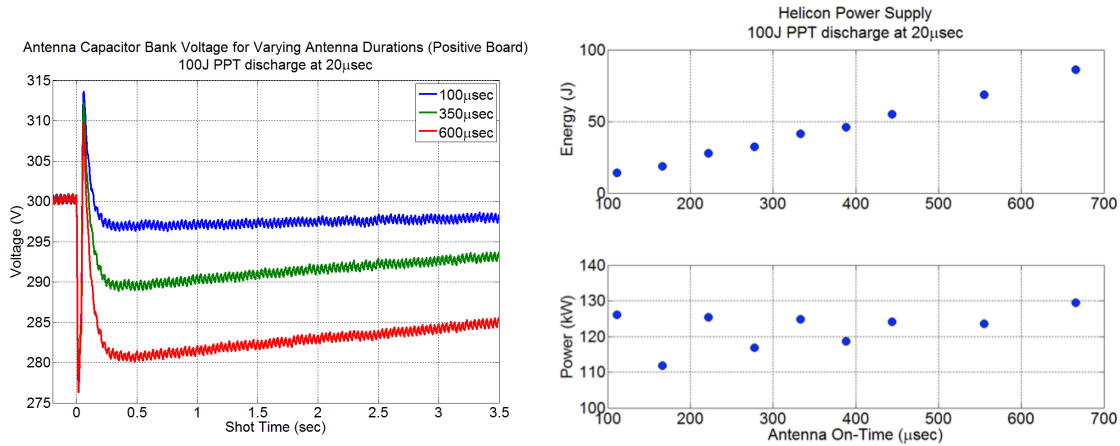
pressure increases, implying that the increased magnetic field increases the mass ablation rate. This is due to the increased electron gyromotion in the source region, increasing the collision frequency and temperature of the solid propellant. The downstream on-axis density was found to increase with higher magnetic field strengths due to the increased mass ablation rate, leading to a larger plasma population.



**Figure 4.30. The chamber pressure increase (top) and peak downstream density at 152cm (bottom) for field strengths of 0-780G and discharge energies of 10-100J.**

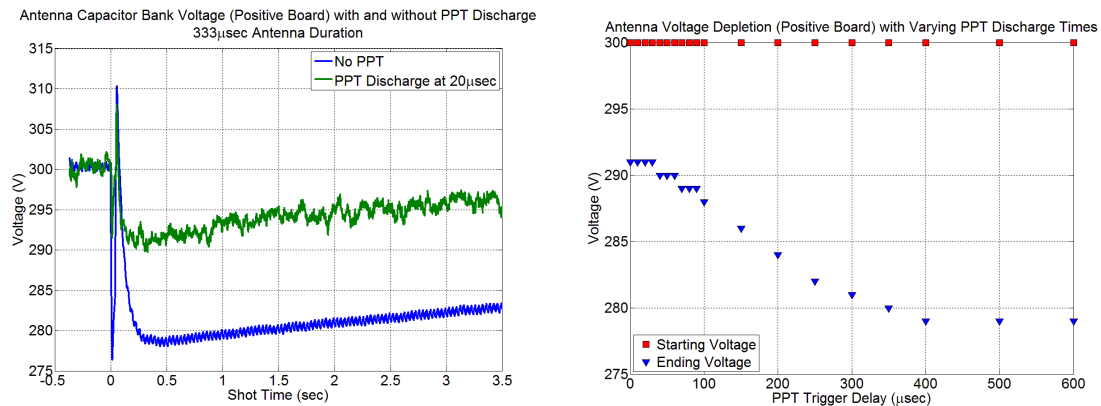
#### 4.5.5 Helicon Antenna Power Draw

The amount of energy drained from antenna capacitors is necessary for thruster efficiency estimates. This isn't a trivial measurement as the antenna power supply is charged to high voltage, while only discharging a small percentage over the course of the shot. A high voltage probe was attached to the positive and negative capacitors. The output of these probes is plotted in Fig. 4.31 for antenna durations of 100, 330, and 600 $\mu\text{sec}$  with no plasma present. As would be expected, longer antenna duration leads to higher capacitor energy depletion. The energy and power (energy/duration) draw from the half-bridge (positive and negative) antenna power supply is plotted in Fig. 4.31. The energy increases linearly with antenna duration while the calculated power remains relatively constant at 125kW.



**Figure 4.31. Positive half-bridge antenna capacitor voltages for 100, 350, and 600μsec antenna durations (left). Total helicon antenna power supply energy and power draw for varying antenna durations (right).**

The antenna drains more energy without plasma present than with. This is shown in Fig. 4.32 for antenna durations of 333μsec with and without a 100J PPT discharge occurring at 20μsec after antenna trigger. This can be seen in the antenna current trace (Fig. 4.14) where the plasma from the PPT produces a loading effect on the antenna, increasing the electrical circuits effective resistance and decreasing the current flowing from the capacitors.

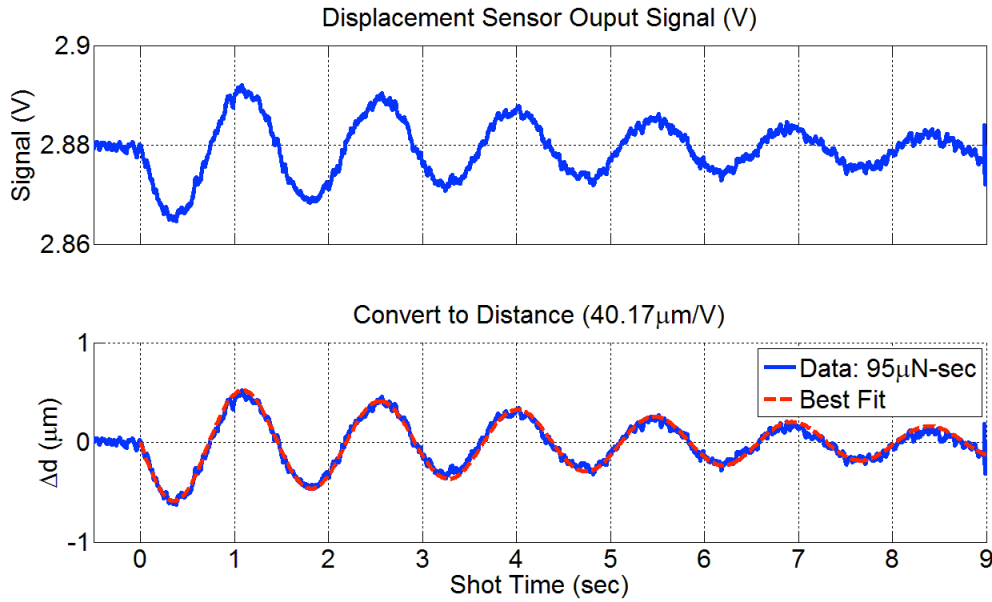


**Figure 4.32. Positive half-bridge antenna capacitor voltage shown with and with a 100J PPT discharge (left) and the starting and ending antenna capacitor voltages for varying PPT discharge times (right).**

#### 4.5.6 Thrust Stand Results

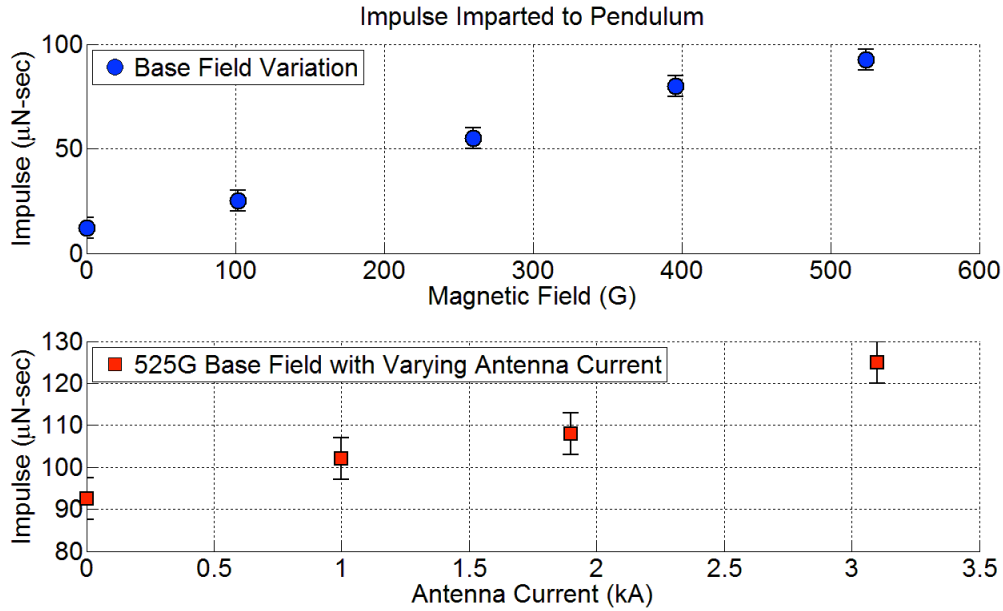
The thrust stand described in Section 3.4 was placed 180cm downstream of the thruster to characterize the thrusters performance. Best-fit curves were fitted to the displacement sensor data per Section 3.4, which allowed for the impulse to be calculated. As an

example, the displacement sensor output for a 100J PPT discharge and 525G base field is converted to distance and overlaid with the best-fit computer simulation in Fig. 4.33. For this case, the simulation gave an impulse imparted to the pendulum of 95 $\mu$ N-sec. For all thrust data presented here, the frequency, damping efficiency, and pendulum physical characteristics were held constant for all simulations. All impulse values shown here are for the impulse imparted to the pendulum and not the impulse from the thruster, which will be higher due to beam dispersion.



**Figure 4.33. Example thrust stand displacement sensor data and best-fit simulation.**

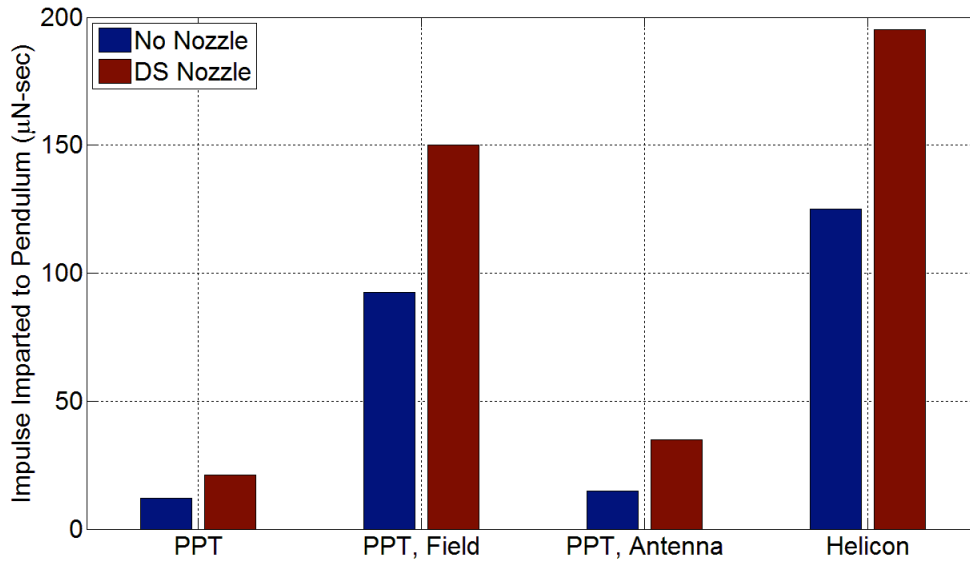
Shown in Fig. 4.34 are the magnetic field and full helicon system influence on the plasma at varying field and antenna strengths. The range of magnetic field (0-525G) increased the impulse 400% over the PPT alone, while the helicon (0-3.1kA) had a 10% increase when used with the 525G field. Linear increases in field strength and antenna current both show linear increases in impulse.



**Figure 4.34. Impulse imparted to the pendulum from a 100J PPT with varying base magnetic field strengths (top) and varying antenna strengths (bottom).**

The impulse from the PPT, PPT with base field, PPT with antenna, and the full helicon thruster are shown in Fig. 4.35, with and without the downstream nozzle. The 525G base field shows an impulse increase of 325% without the nozzle and 170% with the nozzle. The antenna without the base field produced a 30% without the nozzle and a 15% impulse decrease with the nozzle. The full helicon system shows a 10 times increase in impulse over the PPT without the nozzle and a doubling of impulse with the nozzle. The nozzle adds 150% impulse to the PPT and 50% to the helicon.

The addition of the antenna to the base field only increases the impulse by 20%, both with and without the nozzle. This implies the antenna is not being utilized as effectively as with the gaseous experiment, which supports the density, pressure, and light emission results from above.



**Figure 4.35. Impulse imparted to the pendulum from a 100J PPT, PPT with 525G field, PPT with 3.1kA antenna, and all three components together with and without the small downstream nozzle.**

The impulse results given in Figs. 4.34 and 4.35 shows the impulse imparted to the pendulum ( $I_{BIT,p}$ ) and not the true impulse produced by the thruster ( $I_{BIT,t}$ ). However, using the radial profile of the exhaust in Fig 4.26 allows for an estimate of the percentage of exhaust impacting the 10cm diameter pendulum ( $\%_p$ ), which in turn allows for an estimate of the thrusters true impulse bit to be calculated by

$$I_{BIT,t} = \frac{I_{BIT,p}}{\%_p} \quad 4.9$$

It's expected that ions traveling off-axis have a lower velocity than those on-axis, further reducing the overall thrust, however this effect will be ignored here. Estimates of the impulse bit and specific thrust (impulse / discharge energy) are given in Table 4.6 for the PPT, solid fuel helicon, and the solid fuel helicon with downstream nozzle. The specific thrust is given for magnetic fields generated by electromagnets ( $T_{s,EM}$ ), the configuration used for laboratory testing, and from the theoretical use of permanent magnets ( $T_{s,PM}$ ), which wouldn't require electrical energy. The discharge energies for the base magnets and downstream nozzle are given in Section 2.4.

The use of multiple electromagnets offers the capability to test with a wide range of magnetic field strengths and configurations. Thrusters with permanent magnets do not offer this flexibility, however they do have the potential for greatly reduced overall power consumption. The use of permanent magnets simplifies the design by removing the multi-component electromagnetic coils from the thruster. A 7cm ID, 13cm OD, and 4cm thick Neodymium Iron Boron (NdFeB) magnet has been shown to produce a 600G field for helicon operation, while only weighing 2.3kg [64]. Extending the NdFeB magnet over the entire length of the HPH 14cm antenna would result in a mass of 8kg.

**Table 4.6. Estimates of the thruster impulse bit and specific thrust with electromagnets (EM) and assuming permanent magnets (PM).**

| Thruster            | % <sub>op</sub> | I <sub>BIT,p</sub> | I <sub>BIT,t</sub> | T <sub>s,EM</sub> | T <sub>s,PM</sub> |
|---------------------|-----------------|--------------------|--------------------|-------------------|-------------------|
|                     | %               | mN-sec             |                    | mN/kW             |                   |
| PPT                 | 1.5             | 0.010              | 0.6                | 6                 | 6                 |
| Helicon             | 4.5             | 0.125              | 2.7                | 4.2               | 16.9              |
| Helicon with Nozzle | 5.1             | 0.195              | 3.8                | 3.3               | 23.8              |

The results presented above show that a helicon plasma source could be operated from a solid propellant. The PPT hardware and electronics here allowed for discharge energies up to 120J and an assumed mass input of 80μg, two orders of magnitude lower than the gaseous version. It is believed that performance increases can further be achieved by a PPT design allowing for more mass to be ablated and higher base and downstream nozzle magnetic field strengths. The results here also suggest the testing of pulsed plasma thrusters with axially applied magnetic fields increase the thrusters performance. No gaseous results have been taken with the increased power helicon supply to date. This should be completed to make a complete comparison between the internal and external systems.

#### **4.6 Theoretical Steady-State Performance of the High Power Helicon Thruster**

To date the gaseous HPH thruster has been laboratory tested with pulses of up to 1msec as the vacuum chamber is not equipped with the necessary pumping capability to maintain space-like conditions for longer duration. If the necessary power is available, an in-space helicon thruster could run at steady-state operation to achieve high thrust levels. This analysis is a modification from the design of the CAT thruster, a Helicon designed for CubeSat use by the University of Michigan [39]. Current versions of the HPH thruster

use 37 and 125kW, although testing up to 250-300kW in pulsed mode is being considered for the future; therefore a power range of 37-300kW is plotted here. The thruster has a 35mm radius antenna pulsed at 1MHz, no physical nozzle, resulting in plasma temperature between 5-20eV. The mass flow rates required for steady-state operation suggest a gaseous propellant; therefore a tank of Xenon, 131amu and 5.8kg/m<sup>3</sup>, was assumed for the propellant source. This thruster could be used onboard a spacecraft similar in size to the DAWN mission (dry mass of 300kg and 400kg of propellant). To date, Xenon propellant has most often been used with electric propulsion systems as it is, in part, inert, has a low ionization potential, and high atomic mass.

Helicons are capable of producing nearly fully ionized plasmas; doing so requires a set amount of energy per ion (W), called the ion cost, to sustain the discharge. The ion cost depends on propellant, but is typically 60eV/ion for Xenon and is an empirical energy value that accounts for ionization, wall losses, radiation losses, and all other smaller energy sinks. This suggest the power can be expressed as

$$P_{in} = \frac{\dot{m}}{M} N_a W \quad 4.10$$

where  $\dot{m}$  is the mass flow rate, M the molecular weight of the propellant, and  $N_a$  Avogadro's number. Through the ideal gas law (Eq. 4.7) the above power balance can be rewritten as an expression for the volumetric flow where R is the universal gas constant and the propellant tank  $P_n$  and T are assumed to be at STP. The mass flow rate is found by multiplying the volumetric flow rate by the gas density.

$$\dot{V} = \frac{P_{in} R T}{P_n N_a W} \quad 4.11$$

The flow rate into the thruster must be balanced by the output flow. The molecular rate into the thruster for full ionization is simply the power normalized by the ion cost. While the molecular rate out of the thruster is determined by the thruster cross-sectional area ( $A_e$ ) plasma density ( $n_e$ ), and velocity which is assumed to travel at the Bohm speed, a function of the electron temperature ( $T_e$ ) and ion mass ( $m_i$ ). Setting these two expressions equal results in the exhaust plasma density.



$$\dot{N}_{in} = \frac{P_{in}}{W} \quad 4.12$$

$$\dot{N}_{in} = n_e c_s A_e \quad 4.13$$

$$n_e = \frac{P_{in} m_{ion}^{0.5}}{W A_e T_e^{0.5}} \quad 4.14$$

The necessary magnetic field strength can be calculated from the Helicon dispersion relation (Eq. 1.13). As mentioned in Section 1.3, the density being proportional the magnetic field is a defining characteristic of Helicon plasma sources. Higher input power levels can allow for higher plasma densities, however they also will require higher magnetic fields in order to create and propagate the appropriate wave.

The plasma inside the thruster is expected to have temperature of 5-20eV and cold ions. The helicon antenna acts as an electromagnetic accelerator for the electrons, forcing them out of the thruster along field lines. Meanwhile the much heavier ions remain relatively motionless. This establishes an electric field that acts to electrostatically accelerate the ions, known as ambipolar ion acceleration. The electrons cool and are assumed to transfer nearly 100% of their thermal energy to the ions in the form of directed kinetic energy. The initial electron temperature is much larger than the final temperature, so the ion energy can simply be scaled by a constant  $\gamma$ , the adiabatic index of this two-degree of freedom system. The ion velocity and  $I_{SP}$ , assuming 100% ionization, falls out from  $E_{ion} = 0.5mv^2$ , and the  $\Delta V$  imparted to the spacecraft can be calculated from Eq. 1.1.

$$E_{ion} = \gamma T_e \quad 4.15$$

$$I_{SP} = \frac{v_{ion}}{g_0} = \frac{1}{g_0} \sqrt{\frac{2\gamma T_e}{m_{ion}}} \quad 4.16$$

The electrical power into the thruster goes into the ion kinetic energy (ion cost), which by definition includes all energy modes other than ion kinetic energy. Therefore, the maximum possible efficiency of the thruster is

$$\eta_t = \frac{E_{ion}}{W + E_{ion}} \eta_n \quad 4.17$$

where  $\eta_n$  is the nozzle efficiency and is assumed to be ~90%, using the assumption of uniform density as a function of radius. The thrust can be calculated as

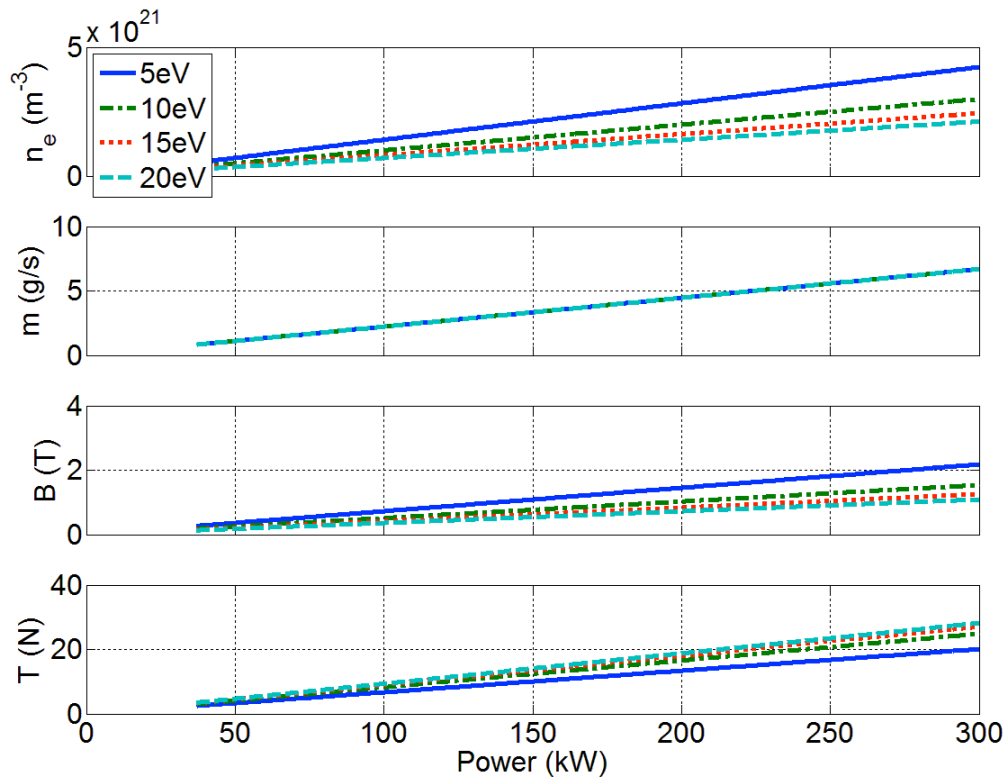
$$T = \frac{2P_{in}\eta_t}{v_{ion}} \quad 4.18$$

The above equations show that the electron density, mass flow rate, field, and thrust are all linear functions of the input power, while all but the mass flow rate vary as the square root of the temperature. For the lower power levels the electron density is  $10^{20} \text{m}^{-3}$  and increases by an order of magnitude at higher powers. The mass flow rate varies from 1-7g/s, while the required fields will vary from 0.1–2T. The thrust is 2N at lower powers, and increases by an order of magnitude at 300kW. It's interesting to note that this analysis gives a specific thrust of 90mN/kW for a 20eV plasma. This is more than double what current state of the art electric propulsion thrusters can provide. The specific thrust for a 5eV plasma is only decreased by 30%, still well above current thrusters.

The ion energy (and corresponding velocity and Isp), thruster efficiency, and total  $\Delta V$  will be irrespective of the input power, only varying with the plasma temperature. The ion energy will scale linearly with the temperature per the adiabatic constant, while the exhaust velocity, and the corresponding specific impulse and  $\Delta V$ , will have a square root relation with the temperature. The thrusters overall efficiency will asymptote towards the nozzle efficiency at large temperatures.

**Table 4.7. The ion energy, Isp, efficiency, and 700kg spacecraft  $\Delta V$  for plasma temperatures of 5-20eV (left) and electron density, mass flow rate, field, and thrust over the same temperature range for power levels of 37-300kW (right).**

| $T_e$ (eV)        | 5   | 10  | 15  | 20  | Power (kW)                               | 37        | 125       | 300       |
|-------------------|-----|-----|-----|-----|------------------------------------------|-----------|-----------|-----------|
| $E_{ion}$ (eV)    | 10  | 20  | 30  | 40  | $n_e$ ( $\text{m}^{-3} \times 10^{21}$ ) | 0.5 – 0.3 | 1.8 – 0.9 | 4.2 – 2.1 |
| $I_{SP}$ (sec)    | 390 | 550 | 675 | 780 | $\dot{m}$ (g/s)                          | 0.8       | 2.8       | 6.7       |
| $\eta_t$ (%)      | 13  | 23  | 30  | 36  | B (T)                                    | 0.3 - 0.1 | 1 – 0.4   | 2 - 1     |
| $\Delta V$ (km/s) | 3.3 | 4.4 | 5.6 | 6.5 | T (N)                                    | 2 – 4     | 8 – 12    | 20 - 28   |



**Figure 4.36.** The electron density, mass flow rate, source region magnetic field, and thrust of a 37-300kW helicon thruster with electron temperatures of 5-20eV.

## **Chapter 5 Alternative Propellants in a Pulsed Plasma Thruster**

### **5.1 Propellant Chemistry**

This study began in 2013 with the testing of bismuth sulfide and sulfur propellants against a Teflon baseline, with the motivation of investigating the effect on efficiency with propellant type for possible in-situ refueling opportunities. The study expanded to ten propellants in 2014 including multiple minerals, metallics, and plastics. Teflon has been the historical standard fuel for ablative PPTs due to a high specific impulse and little surface charring [8] compared with other propellants. Minor improvements to the efficiency and mass bit have been found with Teflon variants and other plastics [7, 8]. In addition to ablative PPTs, substantial research has been invested into gas-fed versions. Higher thrust efficiencies and mass bits have been measured with gas-fed PPTs in the laboratory [65, 66]; however, the added complexity, mass, and size have prevented a gas-fed PPT from being flown.

This study tested minerals found on in-space rocky bodies as well as compounds that are commonly used in spacecraft design. Both of these would allow for an in-situ refuel, increasing mission duration and science objectives without increased initial propellant mass. The use of a derelict spacecraft for propellant is particularly interesting as a means for space debris mitigation. Typical space debris removal options are limited to just a few pieces per mission due to the propellant requirements required to locate one object, deorbit it, and then move to a second. Using the debris as a means to continuously refuel could drastically increase the number of objects that could be deorbited in a single mission. The majority of in-space propulsion systems are not robust enough to allow for any generic propellant. As demonstrated below, this robustness is a major advantage of the Pulsed Plasma Thruster.

#### **5.1.1 Overview of Propellants Tested**

Although not currently used in space operations, sulfur has been previously considered as a propellant for in-space thrusters. Research in the early 1990's into a pulsed plasmoid electric thruster with a specific impulse range of 5,000 - 20,000s and 1 – 5000N of thrust [67] is the first published example that can be found of a laboratory studying a sulfur

propellant. Although less of a concern with other plastic propellants, Teflon, being a heteroatomic fuel composed of carbon and fluorine, will experience a change in surface composition due to carbon charring over longer duration firing which can lead to a change in performance [62]. The use of a homoatomic or elemental fuel cell, such as sulfur, will eliminate carbon charring.

Chalcopyrite, bismuth sulfide, and olivine are common minerals found on earth that may be extremely similar to the composition of asteroids. Chalcopyrite ( $\text{CuFeS}_2$ ) is a copper iron sulfide mineral that crystallizes into a tetragonal shape [68]. Bismuth sulfide occurs in nature as the mineral bismuthinite and is created by reacting a Bismuth-3 salt with hydrogen sulfide [68]. Olivine, easily recognizable due to its olive-green color, is a magnesium-iron silicate with the chemical formula  $\text{Mg}_2\text{SiO}_4$  in forsterite form,  $\text{Fe}_2\text{SiO}_4$  as a fayalite, or a percentage of each. The olivine tested here was primarily magnesium based. Olivine constitutes over 50% of the Earth's upper mantle and has been discovered on meteorites, the Moon, Mars, and asteroid 25143 Itokawa [69].

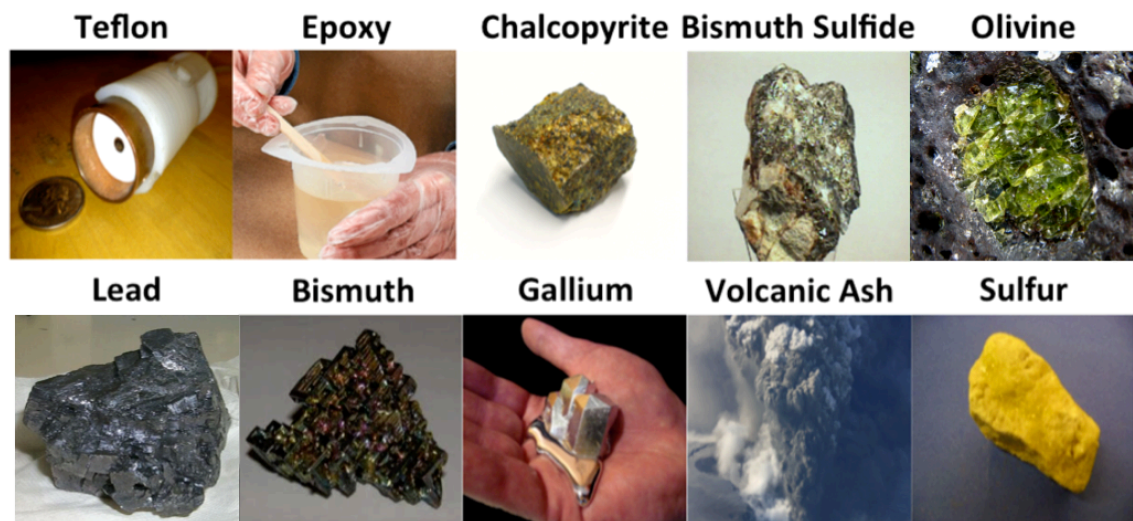
Volcanic ash consists of fragments of pulverized rock and minerals created during explosive eruptions when dissolved gases in magma expand and escape violently into the atmosphere. The particular ash tested here came from the Tungurahua volcano in Ecuador, which has been constantly erupting for the past 15 years. The exact chemical make-up of the ash is unknown, however ash is typically composed of iron, magnesium, and silicon oxide ( $\text{SiO}_2$ ) [70].

Lead is a soft and malleable material as well as being one of the heaviest non-radioactive elements. These properties have resulted in lead being a primary material in radiation protection and mitigation. If ingested, lead is poisonous to humans, and when burnt, releases a poisonous gas; both can cause blood disorders [68]. Lead was chosen for testing here due to its extremely high atomic mass, suggesting a low plasma velocity, but potentially a corresponding high impulse bit.

Gallium is an interesting element in that its melting point is only slightly above room temperature. It's been used in numerous applications including semiconductors, LEDs, and to create alloys that melt at low temperatures [68]. Gallium's a metal that requires

relatively low energy input to turn into a plasma, and coupled with its high mass, makes it an interesting propellant possibility.

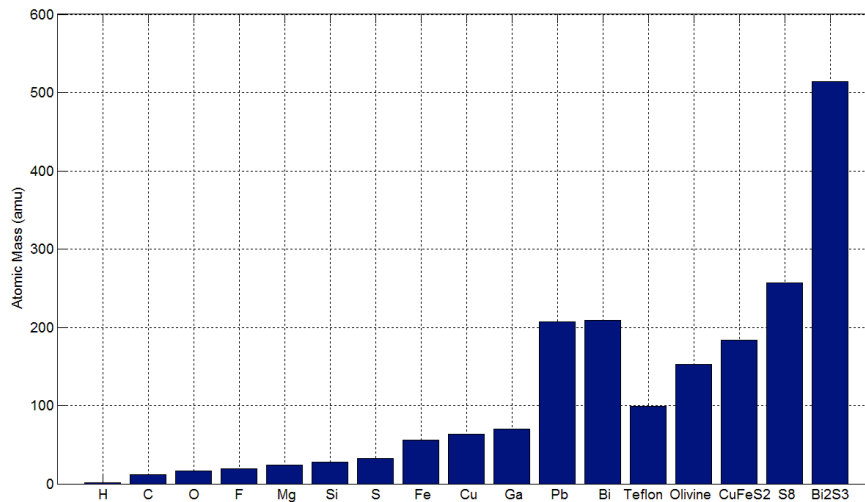
Bismuth has been considered for use in high power Hall thrusters [71] due to its high performance ( $I_{SP}$  of 8,000s at  $> 70\%$  efficiency). At \$75/kg, bismuth is drastically less expensive than xenon (\$2,000/kg), the typical Hall thruster propellant. The use of bismuth also alleviates several logistical Hall thruster concerns. Since its solid at room temperature, vaporized bismuth can be condensed using low cost means. Finally, xenon-fed Hall thrusters have difficulty operating at high voltages ( $>1\text{kV}$ ), while with bismuth, voltages approaching 10kV can be achieved, leading to higher exhaust velocities.



**Figure 5.1. The ten thruster propellants tested in 2014.**

### 5.1.2 The Conversion of a Solid Propellant to Plasma Exhaust

The use of a heavier fuel will increase a thrusters specific thrust (thrust/power) at the expense of lower exhaust velocities. At constant energy ( $E \sim mC_e^2$ ) and with all else being equal, a doubling of the propellant mass will result in a reduction to the velocity by a factor of  $\sqrt{2}$ . As thrust (Eq. 1.2) is proportional to mass times velocity ( $T \sim mC_e$ ), a doubling of the mass, along with a  $\sqrt{2}$  reduction in velocity, will result in a factor of  $\sqrt{2}$  thrust increase. When broken down to its fundamental units, specific thrust is inversely proportional to velocity. Therefore, specific thrust is directly dependent on the exhaust velocity, and consequently the atomic mass, of the ejected material for electric propulsion. Figure 5.2 shows the ion masses for the elements and compounds of interest.

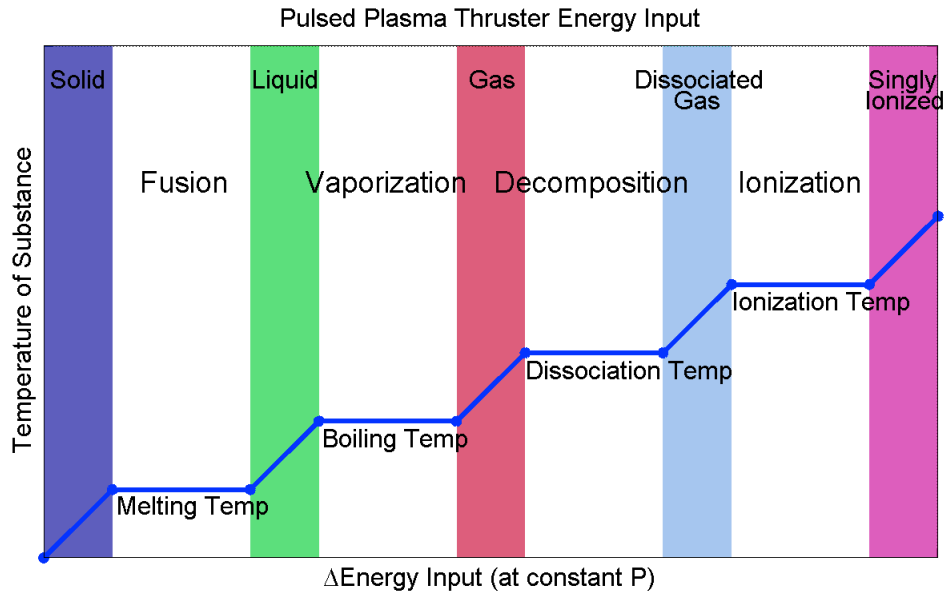


**Figure 5.2. The atomic mass for the atoms and compounds in this study.**

As described in Section 1.1, the optimum exhaust velocity for high efficiency occurs when the exhaust velocity matches the spacecraft  $\Delta V$ . That is, when the exhaust is made to be at rest when viewed from a stationary reference frame. For this reason a variable exhaust velocity PPT, possible through changes in the propellant composition, would allow for increased thrust efficiency by switching from a heavier propellant initially to a lighter propellant once a high spacecraft speed has been reached. If a mission's time constraint is not critical, a lighter propellant is ideal as it will result in a higher  $\Delta V$  for a given fuel mass, due to its higher exhaust velocity. For time limited missions, a heavier fuel, capable of higher thrust levels is required, even with its lower mass efficiency. The alternative fuels examined in this study look at this latter case.

The ablation of the solid propellant to create a gaseous and plasma exhaust will vary based on the propellant chemistry and the energy input to the discharge. The temperature of the solid substance will start relatively cool and then drastically increase as more energy is input. Previous research has shown that ~10% of the total ablated material from a Teflon propellant will become plasma, the majority of which will be singly ionized. The highest temperatures, created during the 10's of microseconds of which the current from the main capacitor is flowing, will create the fast moving plasma as well as dislodge slow-moving solid macroparticles. As temperatures begin to cool once the current flow has stopped, large plumes of cool, neutral gas will be ablated. Figure 5.3 is a generic

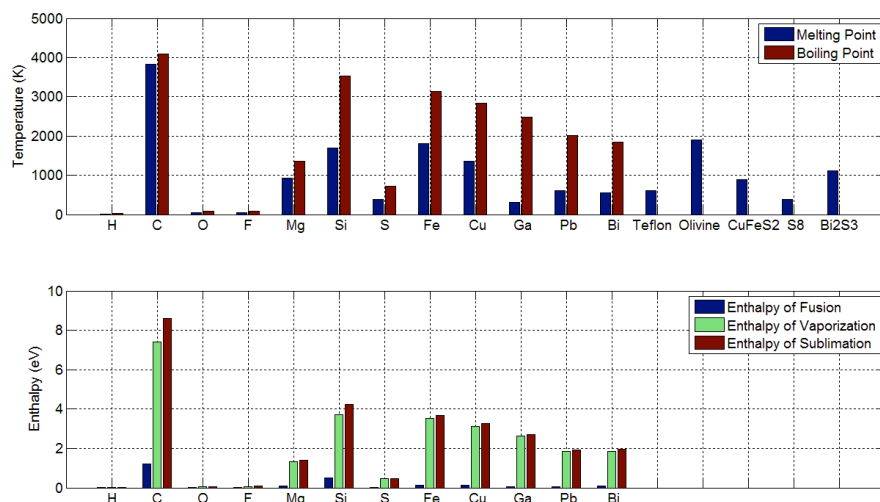
diagram in that not all the steps will be taken for a given energy input and that substances can change composition or dissociate between phases. For example,  $C_2F_2$  cannot exist in gaseous form, as the solid will dissociate into its carbon and fluorine components beforehand.



**Figure 5.3. Generic phase diagram showing the process from a solid to the gas/plasma exhaust of the PPT.**

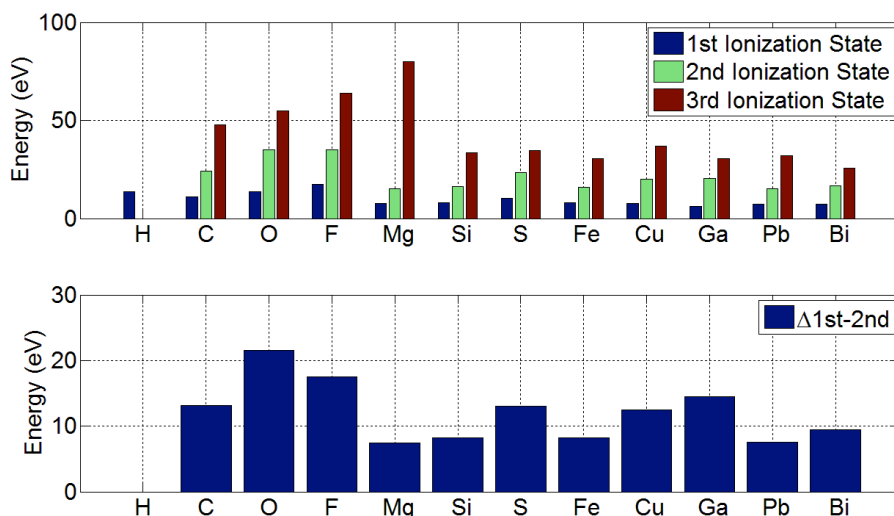
The creation of gaseous material from a solid substance depends on the particles melting and boiling points, as well as sublimation enthalpy. Sublimation enthalpy (solid to gas) is simply the sum of the fusion and vaporization enthalpy's. The enthalpy of fusion is the change in enthalpy resulting from a given quantity of substance changing its state from a solid to a liquid and occurs at the melting temperature. The enthalpy of vaporization is the enthalpy change required to transform a given quantity of substance from a liquid into a gas at atmospheric pressure and occurs at the substances boiling points. A substance with a lower sublimation enthalpy will create more ablated material for the same energy input.





**Figure 5.4. The melting and boiling temperatures (top), and the fusion, vaporization, and sublimation enthalpy's (bottom) for the propellants tested.**

Figure 5.4 shows the melting and boiling points, as well as fusion, vaporization, and sublimation enthalpies for the molecules of interest in this study. Information for carbon, oxygen, fluorine, magnesium, silicon, iron, copper, gallium, and bismuth are included. Additionally, the melting points for Teflon, olivine, chalcopyrite, octo-sulfur (S<sub>8</sub>), and Bismuth Sulfide are shown. Those last five solid substances will disassociate before reaching a gaseous state and therefore do not have a boiling point nor enthalpy values.



**Figure 5.5. The 1<sup>st</sup>, 2<sup>nd</sup>, and 3<sup>rd</sup> ionization energies (top) and energy difference between the 2<sup>nd</sup> and 1<sup>st</sup> states (bottom) for the atoms of interest in this study.**

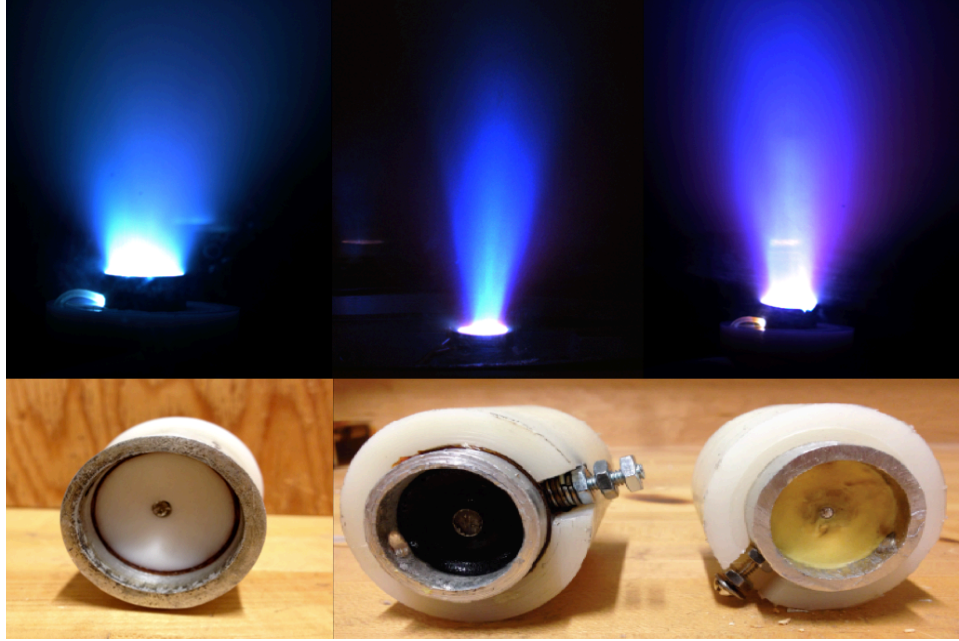
Creation of a plasma ion from a gaseous molecule in the ground electron state requires an energy input to strip away an electron. Lower ionization energies will create a higher plasma fraction, higher ionization state, and/or additional disassociation for a given energy input. Ionization is a necessary, but unrecuperated energy loss for electric propulsion systems. Simply ionizing a particle won't accelerate it axially out of the thruster, but the high velocities associated with electric propulsion are only achievable with plasma. Generally, a particle can be accelerated faster with higher energy states, suggesting that the more electrons that can be removed, the higher the thrusters performance will be. However, any energy that is placed into ionizing a particle isn't placed into accelerating it and generally the energy cost to remove a second electron is higher than any added kinetic energy. The ideal propellant, from an ionization standpoint, is one that requires low energy to singly ionize, but high energy to doubly ionize, as well a high mass such that its kinetic energy is greater than its ionization energy.

**Table 5.1. Chemical properties for the atoms and compounds of interest [72].**

|                 | Atomic Mass | Melting Point | Boiling Point | Sublimation Enthalpy | 1 <sup>st</sup> Ionization | 2 <sup>nd</sup> Ionization |
|-----------------|-------------|---------------|---------------|----------------------|----------------------------|----------------------------|
|                 | (amu)       | (K)           |               | (eV)                 |                            |                            |
| Hydrogen        | 1.01        | 14            | 20            | 0.01                 | 13.6                       | --                         |
| Lithium         | 6.94        | 454           | 1603          | 1.44                 | 5.4                        | 75.6                       |
| Carbon          | 12.01       | 3825          | 5100          | 8.64                 | 11.2                       | 24.4                       |
| Oxygen          | 15.99       | 55            | 90            | 0.04                 | 13.6                       | 35.1                       |
| Fluorine        | 18.99       | 54            | 85            | 0.03                 | 17.4                       | 35.0                       |
| Sodium          | 23          | 371           | 1156          | 1.04                 | 5.1                        | 47.3                       |
| Magnesium       | 24.31       | 922           | 1380          | 1.42                 | 7.6                        | 15.0                       |
| Silicon         | 28.09       | 1683          | 2630          | 4.24                 | 8.2                        | 16.3                       |
| Sulfur          | 32.06       | 392           | 718           | 0.48                 | 10.4                       | 23.3                       |
| Potassium       | 39.1        | 337           | 1032          | 0.82                 | 4.3                        | 31.6                       |
| Iron            | 55.85       | 1808          | 3023          | 3.67                 | 7.9                        | 16.2                       |
| Copper          | 63.55       | 1356          | 2840          | 3.25                 | 7.7                        | 20.3                       |
| Gallium         | 69.7        | 303           | 2478          | 1.77                 | 6.0                        | 20.5                       |
| Rubidium        | 85.5        | 312           | 961           | 0.74                 | 4.2                        | 27.3                       |
| Cesium          | 133         | 302           | 944           | 0.68                 | 3.9                        | 23.2                       |
| Barium          | 137         | 1000          | 2118          | 1.55                 | 5.2                        | 10.0                       |
| Lead            | 207.2       | 600           | 2020          | 1.89                 | 7.4                        | 15.0                       |
| Bismuth         | 208.98      | 545           | 1837          | 2.71                 | 7.3                        | 16.7                       |
| Radium          | 226         | 1233          | 2010          | 1.26                 | 5.3                        | 10.2                       |
| Teflon          | 99.98       | 600           | --            | --                   | --                         | --                         |
| Olivine         | 153         | 1900          | --            | --                   | --                         | --                         |
| Chalcopyrite    | 183.5       | 895           | --            | --                   | --                         | --                         |
| Octo-Sulfur     | 256.48      | 392           | --            | --                   | --                         | --                         |
| Bismuth Sulfide | 514.14      | 1123          | --            | --                   | --                         | --                         |

## 5.2 Alternative Propellant 2013 Testing

Initial alternative propellant testing occurred in 2013 with Teflon, Sulfur, and Bismuth Sulfide ( $\text{Bi}_2\text{S}_3$ ) at a background pressure of  $1\mu\text{Torr}$ . Density, bulk plasma velocity, and specific thrust measurements were taken at discharge energies ranging from 5 – 62J.

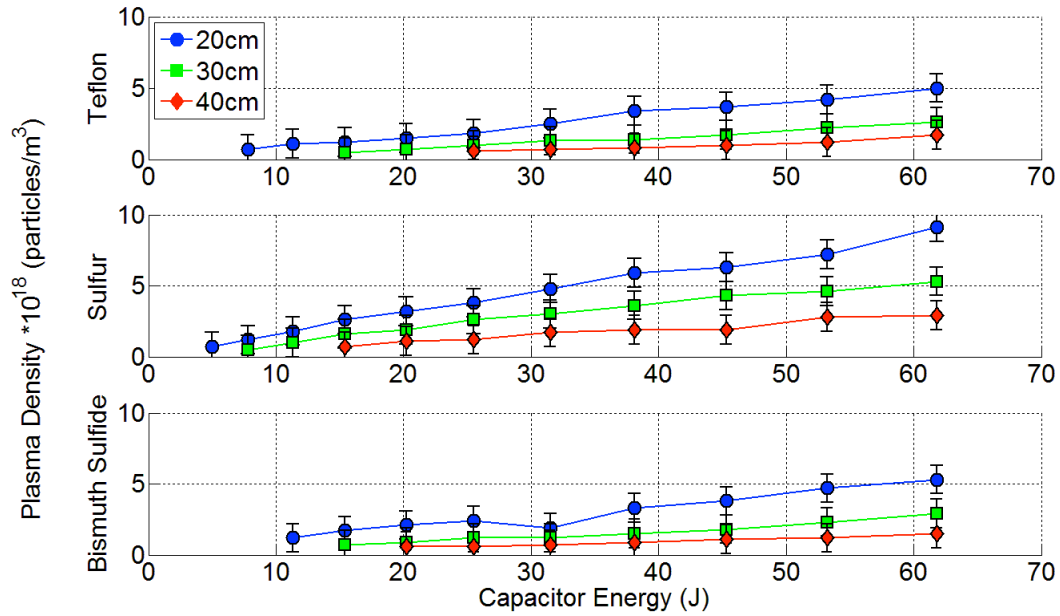


**Figure 5.6. Exhaust plumes (62J discharge energy) and physical thrusters for the Teflon, Bismuth Sulfide, and Sulfur (left to right) propellants.**

Langmuir probes allowed for the calculation of the electron density and an estimate of the bulk plasma velocity. Here the PPT was fired vertically with a Langmuir probe placed on the centerline, perpendicular to the plasma flow, which could be positioned axially above the propellant surface. The axial centerline electron density was measured at 20, 30, and 40cm downstream and is plotted in Fig. 5.7. Generally speaking, Langmuir measurements at distances close to the thruster caused sputtering on the probe, while measurements at distances further away resulted in a signal with lower amplitude than the noise. This electromagnetic noise prevented measurements under  $5 \times 10^{17} \text{ particles/m}^3$ , while shot to shot variation results in errors of  $\sim 1 \times 10^{18} \text{ particles/m}^3$ . The three measurement locations allowed three separate bulk velocities to be calculated (between 20-30cm, 30-40cm, and 20-40cm), the average of which is shown in Fig. 5.8.

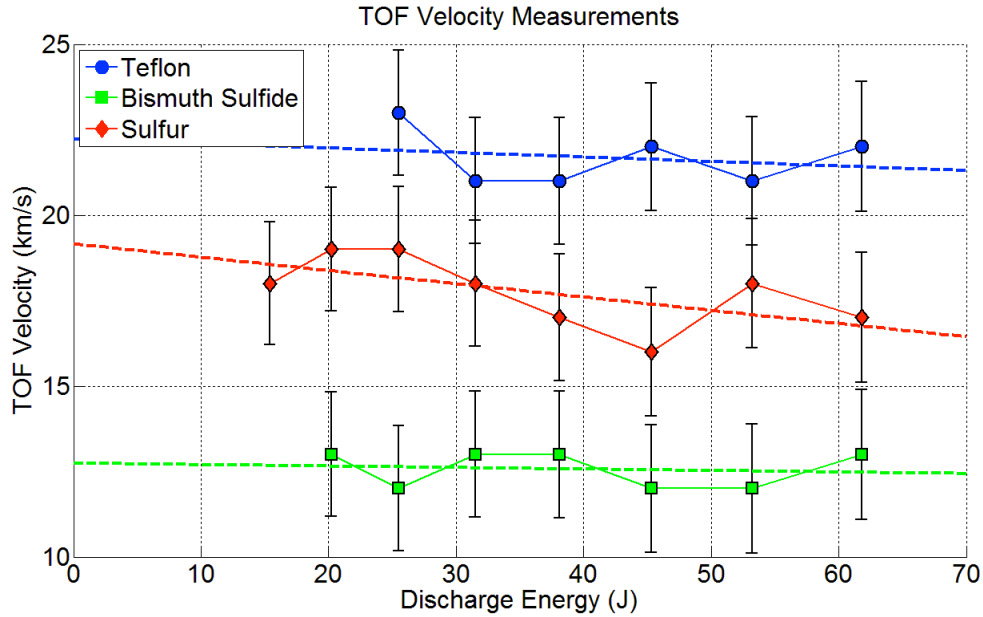
To calculate plasma density from the probes output current (Eq. 3.2), an ion mass must be assumed. For Teflon, the plasma was assumed to be composed of 50% Carbon and 50%

Fluorine, while for bismuth sulfide 60% sulfur and 40% bismuth was assumed. For all three propellants, the plasma density rises linearly with discharge energy at a given height, while decreasing as  $\sim 1/r^2$  with increasing axial distance for a given discharge energy. The bismuth sulfide and Teflon propellants show similar plasma densities, while the sulfur thruster produced densities  $\sim$ twice as high for all discharge energies.



**Figure 5.7. Electron density measurements with Teflon, sulfur, and bismuth sulfide (top to bottom) propellants on the centerline of the thruster at axial distances of 20-40cm for discharge energies of 5-62J.**

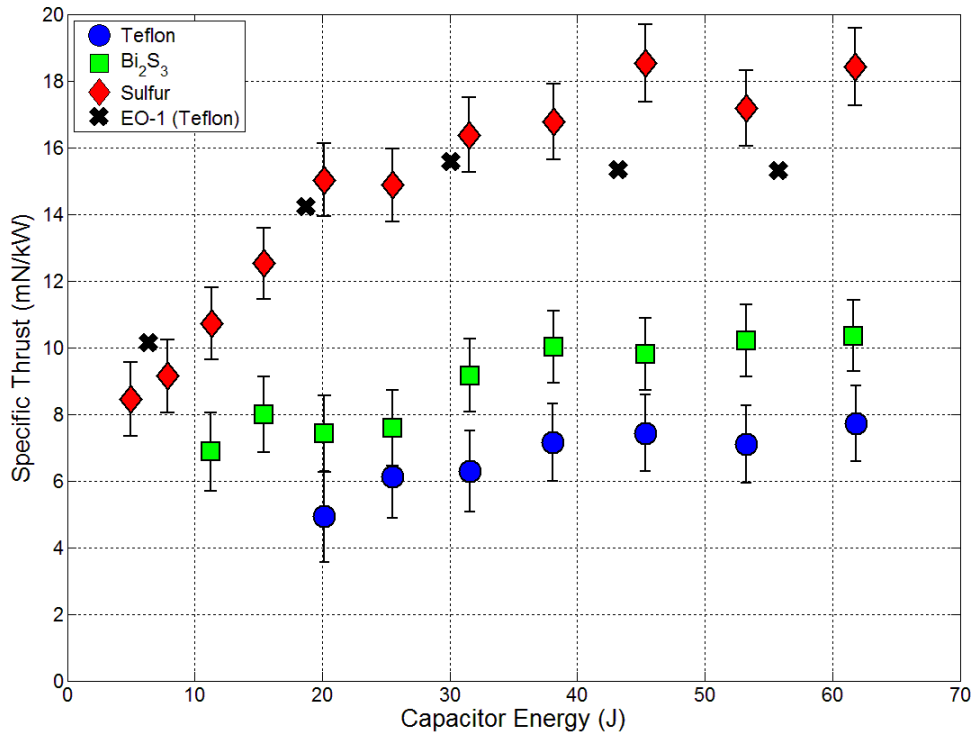
Bulk velocity measurements showed that the Teflon plasma had a velocity between 21-23km/s, the sulfur plasma between 16-19km/s, and the bismuth sulfide 13-14km/s. The shot to shot variation resulted in velocity uncertainties up to 2km/s. Lines of best fit for all three propellants show a trend of slightly decreasing velocity with increasing discharge energies, however the trend lines are well within the margin of error.



**Figure 5.8. Bulk plasma velocity measurements for the three propellants tested.**

The higher exhaust velocity of Teflon can be attributed to the lower ion mass, while the increased plasma density of sulfur is most likely due to a combination of a high vapor pressure, low melting and boiling points, and low ionization energy. These chemical properties suggest it would take less energy to not only ablate, but then ionize the same mass of sulfur than Teflon particles.

Using a pendulum thrust stand a comparison of impulse for the three propellants was made. Shown in Fig. 5.9 are specific thrust measurements, defined as the impulse per discharge normalized by discharge energy, for discharge energies of 5-62J. As explained in Section 3.3, the data analysis method for the thrust stand imposed a lower impulse bound of 40 $\mu$ N-sec, preventing specific thrust data for some of the lower discharge energies.



**Figure 5.9. Specific thrust comparison between Teflon, sulfur, bismuth sulfide, and the EO-1 Teflon [10] PPT for varying capacitor energy levels at 1 $\mu$ Torr.**

For all three propellants, the specific thrust initially rises before leveling off at capacitor energies above 40J. The sulfur PPT leveled off at 18.4mN/kW, 2.3 times higher than with Teflon and 1.8 times higher than bismuth sulfide. A comparison to the specific thrust of the EO-1 PPT is also included in Fig. 5.9; as with the laboratory thrusters, EO-1 showed an initial increase in specific thrust until leveling off at 15.6mN/kW, double that of the UW Teflon thruster and 18% lower than the sulfur version. There were numerous design differences between the EO-1 and UW Teflon thruster. The most significant were the increased propellant surface area (36%), increase in electrode height (83%), and rectangular design of the EO-1 PPT. All of which may contribute to the performance difference between the Teflon thrusters. However, if the increase in performance with sulfur over an identical Teflon thruster transferred to the EO-1 design, then the EO-1 design with sulfur propellant could reach specific thrusts of over 35mN/kW.

One of the drawbacks to testing with sulfur was how it reacted with the copper gaskets used to form the vacuum/atmosphere interface with conflate flanges (CF). The gaskets

used were created from 1/4 hard, high-purity, oxygen-free copper stock. Bolting two flanges together causes the CF's knife-edges to deform the gasket. Under plastic flow, the copper fills in the knife-edges to form the vacuum seal where the inner surface is exposed to vacuum while the outer is exposed to air.

The belljar could achieve a 700nTorr vacuum when left pumping for longer than a week immediately after new gaskets were installed and with no thruster/propellant inside the chamber. During data runs lasting for numerous weeks with sulfur present within the system, that base pressure could rise to  $>10\mu\text{Torr}$ . Replacement of the gaskets occurred on an as needed basis to keep the pressure at  $2\pm 1\mu\text{Torr}$ .

Figure 5.10 shows three gaskets used on the vacuum chamber, replaced after durations of one week, one month, and one year. After only one week exposed to sulfur outgassing there was no noticeable corrosion to the gasket. However, within one month, the copper begins to react with the sulfur, forming copper sulfide ( $\text{Cu}_2\text{S}$ ), more commonly known as chalcocite. This is a common reaction on Earth when a hot sulfur vapor comes in contact with solid copper and produces a black substance. Gaskets used for longer time periods, undergoing multiple pump-downs and up-to-airs, produced copper sulfate ( $\text{CuSO}_4$ ), which has a bluish hue due to its high degree of hydration in its pentahydrate form,  $\text{CuSO}_4\cdot 5\text{H}_2\text{O}$ .



**Figure 5.10. Copper gaskets used while testing with sulfur for durations of (left to right) one week, one month, and one year and a sulfur propellant thruster with copper electrodes after 100 43J discharges (far right).**

Some initial testing with sulfur propellants occurred with copper electrodes. Simply placing the sulfur in direct contact with the copper electrode and bringing the vacuum chamber down to microTorr pressures produced noticeable copper sulfide reactions as shown in Fig. 5.10. Discharging the thruster 100 times at 42J produced a high degree of copper sulfide, including flakes that easily broke off from the electrodes. For this reason,

all electrodes for the alternative propellant testing were machined from aluminum-6061 due to its low copper content (<0.4% by weight).

### **5.3 Alternative Propellant 2014 Testing**

The success with sulfur in 2013 lead to additional testing in 2014 with a wider range of propellants. The goal for this second round of testing was two-fold: (1) to better understand how the chemical properties of the propellant influence the thrusters performance and (2) to determine what types of propellant could be used within a Pulsed Plasma Thruster for the feasibility of in-situ propellant gathering in-space. Ten propellants in three categories were tested in total:

- (1) Plastics: teflon and epoxy
- (2) Minerals: sulfur, olivine, chalcopryrite, bismuth sulfide, and volcanic ash
- (3) Metallics: lead, gallium, and bismuth

A number of small changes between 2013 and 2014 were made to increase consistency between propellants as well as accuracy in the measurement technique. These included reducing the mass of the thrust stand pendulum to create a higher displacement angle. The larger oscillations created a more accurate impulse measurement at the lower discharge discharges, which, in 2013, would only move a couple of pixels on the recorded camera image. The percentage of epoxy by mass in each of the propellants created was reduced from ~20% in 2013, to 6-8%. Additionally, a thin Teflon insulator was included between each of the propellants and the electrodes. This was necessary as metallic propellants would simply short circuit the electrodes without insulation. And finally, a dust trap was added between the chamber and the pumping system. This was made necessary by the small grains composing the propellants. These sand sized particles could become dislodged and damage the turbopump blades. The dust trap increased the size of the vacuum chamber as well as placing a number of filters in front of the turbopump; the combination of which created a higher base pressure within the chamber, 25 $\mu$ Torr up from 1 $\mu$ Torr in 2013.

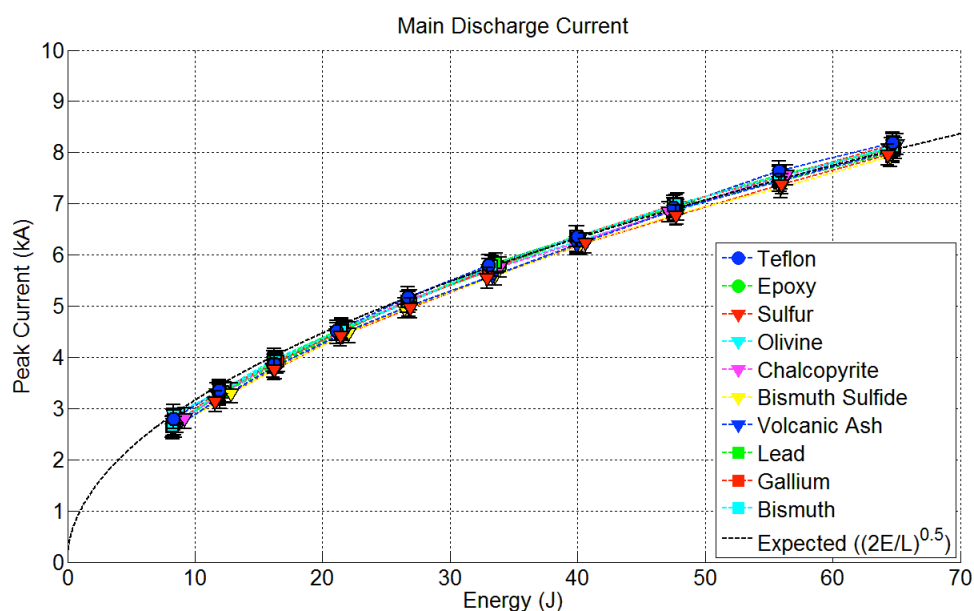




**Figure 5.11.** The ten propellants tested, Teflon and epoxy (left to right, top row), sulfur, olivine, chalcopyrite, bismuth sulfide, and volcanic ash (middle), lead, gallium, and bismuth (bottom). A Teflon insulator and chalcopyrite grains (right).

For the results below, each propellant was fired a few dozen times, after which the thrust stand and pressure data was collected. Shot to shot variations of the pressure and thrust stand measurements were all under 5%. No propellant was fired for long enough to require the use of the spring.

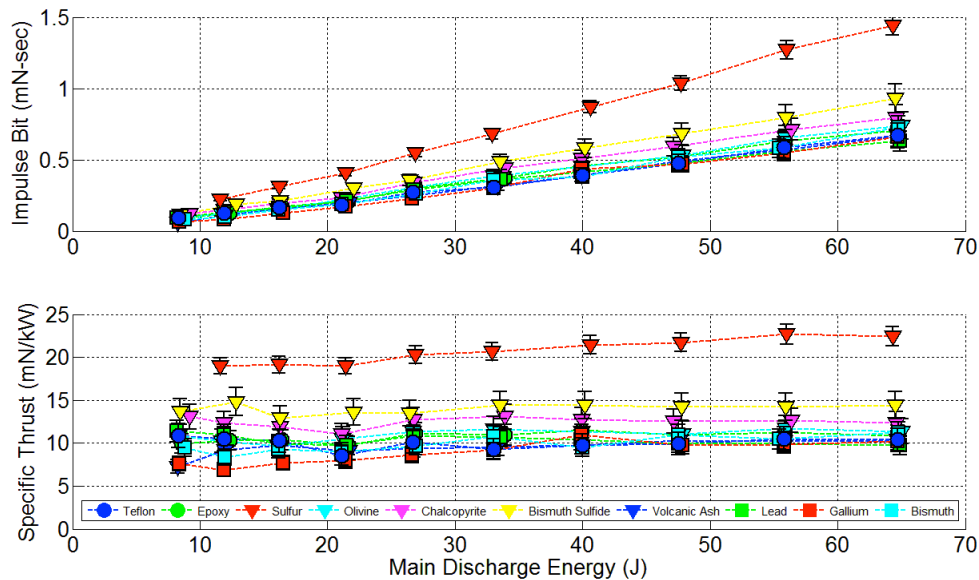
Data runs were taken between discharge energies of 8-65J. Current and voltage traces of the igniter and main discharge circuitry were recorded to ensure electrical consistency between discharges. The peak of the main discharge current is plotted in Fig. 5.12 along with the expected peak, calculated from Eq. 1.6 for the limit of  $t \rightarrow 0$ sec. All discharges were within 5% of each other and within 3% of the expected value. The 5% error is within the uncertainty of the displayed data on the oscilloscope.



**Figure 5.12.** Peak main discharge current for each propellant tested.

### 5.3.1 Specific Thrust Measurements

Using the same technique as the 2013 measurements, the specific thrust, defined as the impulse bit normalized by the discharge energy, was calculated and is shown in Fig. 5.13. Teflon, gallium, and bismuth saw the lowest specific thrust performance, while sulfur once again showed the highest. Teflon ranged between 8-11mN/kW, while sulfur increased from 19-23mN/kW. Chalcopyrite and bismuth sulfide, both with high concentrations of sulfur, resulted in higher specific thrusts than the remainder of the minerals and metals, which were within the margin of error for Teflon.



**Figure 5.13. The measured impulse bit and resulting specific thrust results.**

An attempt was made to empirically derive a relationship between the propellants chemical properties and the measured performance. The thrusters impulse bit was assumed to be a function of the ablated mass, ionization fraction, and plasma velocity, while the effect of neutrals will be assumed to be negligible.

The sublimation enthalpy is simply the energy required to turn a solid substance into a gas (sum of the enthalpies of fusion and vaporization). Previous PPT research with Teflon propellants empirically found a linear relationship between the ablated mass and the discharge energy. To a first order analysis, it's believed that multiplying the energy and sublimation enthalpy together, when each is raised to a separate power, can represent

the ablated mass. The ablated mass is expected be proportional to the energy and inversely proportional to the sublimation enthalpy.

$$m_{abl} \sim E^\beta \left( \sum f_n W_{sub,n} \right)^\gamma \quad 5.1$$

The elemental species fraction ( $f_n$ ) was determined by assuming the neutral gas consists of single elements and that the composition fractions will be proportional to the numerical proportions of the solid propellants chemical formula. The assumed compositions of the exhaust are given in 25.3, while the chemical properties for the elements of interest are listed in Table 5.1.

**Table 5.2. Thruster propellants and their assumed gaseous chemical compositions.**

| Propellant                  | Formula                               | Assumed Percentage of Gas Composition |          |          |          |
|-----------------------------|---------------------------------------|---------------------------------------|----------|----------|----------|
| <b>Teflon</b>               | C <sub>2</sub> F <sub>4</sub>         | F (66%)                               | C (33%)  |          |          |
| <b>Epoxy</b>                | CH <sub>2</sub> -CH-O                 | H (50%)                               | C (33%)  | O (17%)  |          |
| <b>Sulfur</b>               | S <sub>8</sub>                        | S (100%)                              |          |          |          |
| <b>Olivine (Forsterite)</b> | Mg <sub>2</sub> SiO <sub>4</sub>      | Mg (30%)                              | Si (14%) | O (56%)  |          |
| <b>Chalcopyrite</b>         | CuFeS <sub>2</sub>                    | S (50%)                               | Cu (25%) | Fe (25%) |          |
| <b>Bismuth Sulfide</b>      | Bi <sub>2</sub> S <sub>3</sub>        | S (60%)                               | Bi (40%) |          |          |
| <b>Volcanic Ash</b>         | (SiO <sub>2</sub> ) <sub>2</sub> FeMg | O (50%)                               | Mg (13%) | Fe (13%) | Si (25%) |
| <b>Lead</b>                 | Pb <sub>2</sub>                       | Pb (100%)                             |          |          |          |
| <b>Gallium</b>              | Ga <sub>2</sub>                       | Ga (100%)                             |          |          |          |
| <b>Bismuth</b>              | Bi <sub>2</sub>                       | Bi (100%)                             |          |          |          |

The ionization fraction will be primarily dependent on the ionization energy, which is the energy required to remove a single electron from a neutral gas particle

$$\frac{n_e}{n_n} \sim \left( \sum f_n W_{ion,n} \right)^\delta \quad 5.2$$

The ion mass will have two effects, which will be included as one term. Thermal, as well as Lorentz force acceleration of particles will be inversely proportional to the ion mass, while, as has been described previously (Section 5.1.2), heavier particles will carry more momentum.

$$v_{plasma} \sim \left( \sum f_n m_{ion,n} \right)^\epsilon \quad 5.3$$

Multiplying Eqs. 5.1-5.3 together and adding a proportionality constant,  $\alpha$ , results in an empirically derived equation for the impulse bit based entirely on the chemical properties of the propellant.

$$I_{BIT} = \alpha E^{\beta} \left( \sum f_n W_{sub,n} \right)^{\gamma} \left( \sum f_n W_{ion,n} \right)^{\delta} \left( \sum f_n m_{ion,n} \right)^{\varepsilon} \quad 5.4$$

We've defined the specific thrust as the thrusters impulse bit normalized by the discharge energy and will therefore have the form

$$T_{Specific} = \alpha E^{(\beta-1)} \left( \sum f_n W_{sub,n} \right)^{\gamma} \left( \sum f_n W_{ion,n} \right)^{\delta} \left( \sum f_n m_{ion,n} \right)^{\varepsilon} \quad 5.5$$

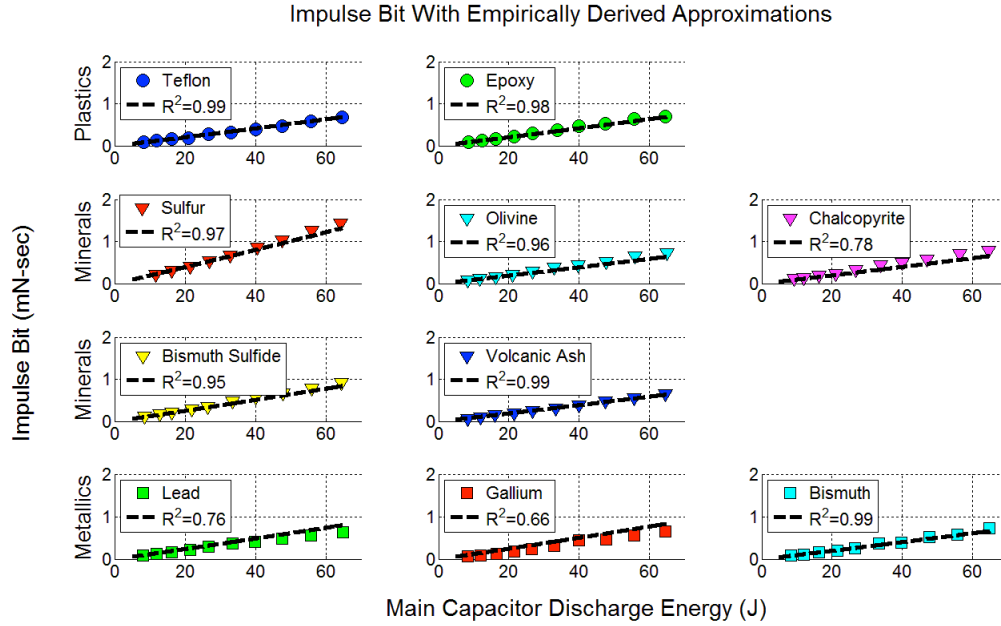
The constants in Eq. 5.5 were varied and compared to the measured results in Fig. 5.13, to find a form that maximized the  $R^2$  coefficient of determination for all 10 propellants tested. Table 5.3 shows the determined values for those five constants that resulted in maximum average  $R^2$  coefficient.

**Table 5.3. The empirically found constants for Eq. 5.5.**

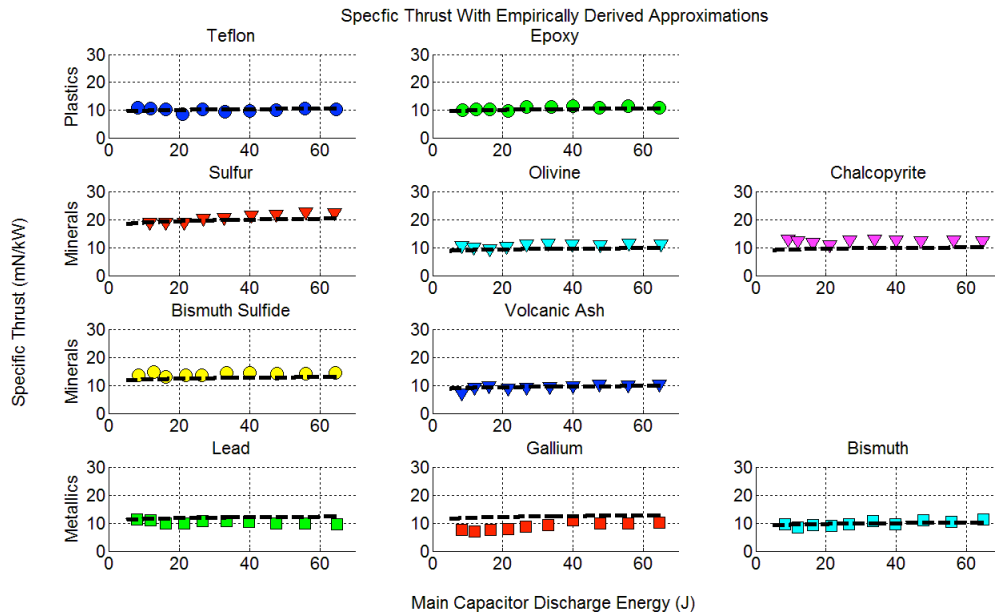
| Constant      | Value                                                                       |
|---------------|-----------------------------------------------------------------------------|
| $\alpha$      | 20.95 mN-sec/(J <sup>1.03</sup> -eV <sup>-0.92</sup> -amu <sup>0.08</sup> ) |
| $\beta$       | 1.04                                                                        |
| $\gamma$      | -0.55                                                                       |
| $\delta$      | -0.36                                                                       |
| $\varepsilon$ | 0.07                                                                        |

The empirically derived equations are shown overlaid across the measured (Fig. 5.13) impulse bit in Fig. 5.14 and specific thrust in Fig. 5.15. The  $R^2$  values between the empirically derived equations and measured are given in Table 5.4. The empirical model matches the measured data reasonably well for all ten propellants, with seven of the propellants having an  $R^2$  coefficient greater than 0.95, while chalcopryrite is 0.78, lead 0.76, and gallium 0.66. Chalcopryrite is a complex mineral and the assumptions made about the composition may be incorrect. As will be shown in Section 5.3.2, lead and gallium show 2<sup>nd</sup> order effects with the chamber pressure increase at the upper discharge energies, implying that the physics of the ablation may be different than for the other eight propellants. The exact discrepancy is unknown at the current time, however the thermal coefficients for lead (35W/m-K) and gallium (40W/m-K) are higher than for the

other eight propellants and could result in differences in how the energy from the discharge is transferred to the solid propellant. The model underestimates the performance of chalcopyrite, while overestimating the performance of lead and gallium.



**Figure 5.14. Impulse bit measured data overlaid with the empirical equation approximation (Eq. 5.4).**

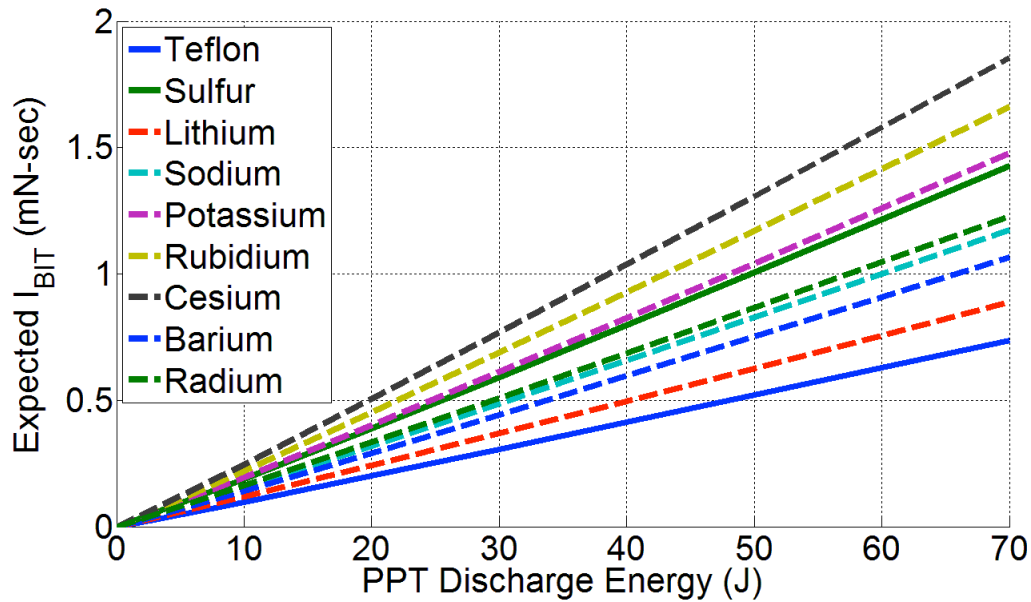


**Figure 5.15. Specific thrust measured data (Fig. 5.13) overlaid with the empirical equation approximation (Eq. 5.5).**

**Table 5.4. Percent error and  $R^2$  coefficient of determination between the empirically derived model and the measured results.**

| <b>Propellant</b>       | <b>Percent Error (%)</b> | <b>Coefficient of determination (<math>R^2</math>)</b> |
|-------------------------|--------------------------|--------------------------------------------------------|
| <b>Teflon</b>           | 4.9                      | 0.99                                                   |
| <b>Epoxy</b>            | 6.5                      | 0.98                                                   |
| <b>Sulfur</b>           | 6.6                      | 0.97                                                   |
| <b>Olivine</b>          | 11.2                     | 0.96                                                   |
| <b>Chalcopyrite</b>     | 21.2                     | 0.78                                                   |
| <b>Bismuth Sulfide</b>  | 10.4                     | 0.95                                                   |
| <b>Volcanic Ash</b>     | 4.0                      | 0.99                                                   |
| <b>Lead</b>             | 16.3                     | 0.76                                                   |
| <b>Gallium</b>          | 46.5                     | 0.66                                                   |
| <b>Bismuth</b>          | 4.3                      | 0.99                                                   |
| <b>Plastic Average</b>  | 5.7                      | 0.98                                                   |
| <b>Mineral Average</b>  | 10.7                     | 0.93                                                   |
| <b>Metallic Average</b> | 22.4                     | 0.80                                                   |
| <b>Total Average</b>    | 13.2                     | 0.90                                                   |

Additional testing with a higher number of propellants is required to determine a more precise form of Eq. 5.5, however, it's believed that the model is a means to give qualitative estimates to what solid fuel pulsed plasma thruster propellants can produce the highest performance. Figure 5.16 shows the results of the model applied to seven propellants that all have relatively low ionization energies and/or low sublimation enthalpies (listed in Table 5.1). The model predicts that 3 elements will outperform sulfur (30%), cesium, rubidium (16%), and potassium (3%). While the other four, sodium, barium, radium, and lithium, were all predicted to outperform Teflon. Cesium has previously been tested as an electric propulsion propellant with Hall thrusters at the NASA Glenn Research Center, however concerns over safety precluded wide scale use [73]. Along with rubidium, cesium has also seen use with FEEP thrusters [74, 75]. Potassium Nitrate is often used as an oxidizer with solid propellant chemical rockets.

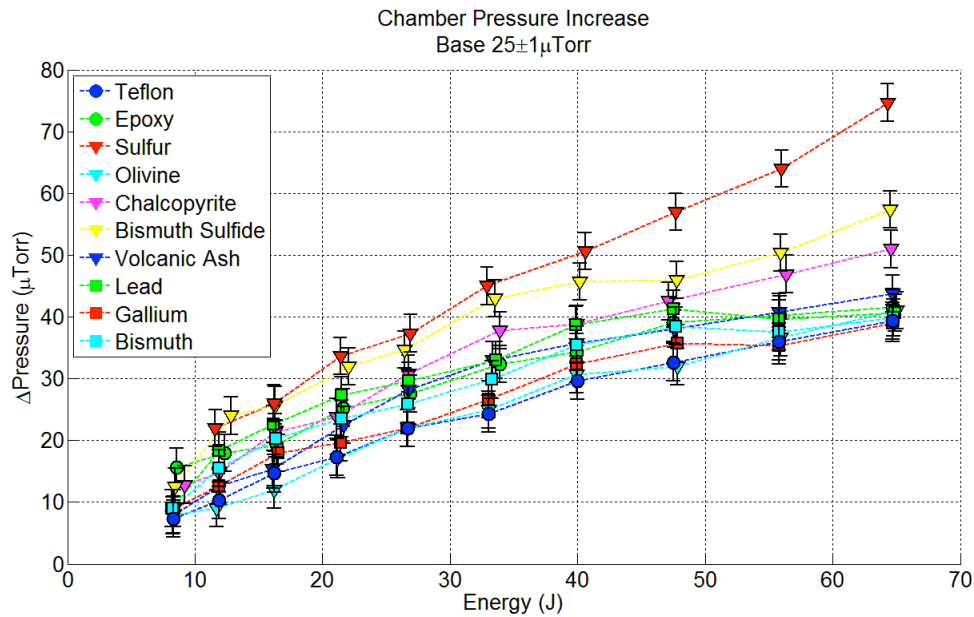


**Figure 5.16. The expected impulse bit (Eq. 5.5) for selected propellants compared against Sulfur and Teflon.**

### 5.3.2 Pressure change and ablated mass estimates

The change in pressure of the belljar vacuum chamber, from a base of  $25\mu\text{Torr}$ , was measured for each of the propellants. The chambers pressure was measured with a cold cathode sensor that updated at a 3Hz frequency. It typically took 0.6-1.3sec for the sensor to reach its maximum value after the discharge. The peak of this pressure increase for each propellant at varying discharge energies is plotted in Fig. 5.17. On a scale of 100's of milliseconds, the pressure is expected to be due entirely from the expansion of a cold and neutral gas, the plasma having recombined on a time scale 3-4 orders of magnitude faster. Therefore the pressure recorded should be an indication of the total mass ablated. Given that Teflon has a mass ablation rate of  $\sim 2\mu\text{g/J}$ , a chamber pressure increase twice that of Teflon at a given discharge energy would imply a mass ablation rate of  $\sim 4\mu\text{g/J}$ . The pressure increase with Teflon propellant ranged from  $7\mu\text{Torr}$  at 8J to  $40\mu\text{Torr}$  at 65J. Sulfur showed the highest pressure increase, ranging from  $22\mu\text{Torr}$  at 8J up to  $74\mu\text{Torr}$  at 65J. The next two largest pressure increases came from the propellants with the highest concentrations of sulfur within them, bismuth sulfide and chalcopyrite. Both plastics and all the mineral propellants showed fairly linearly increases in pressure as the energy was

increased. The three metallic propellants however, all showed a 2<sup>nd</sup> order effect at the higher discharge energies, appearing to level off once the energy reached above ~40J.



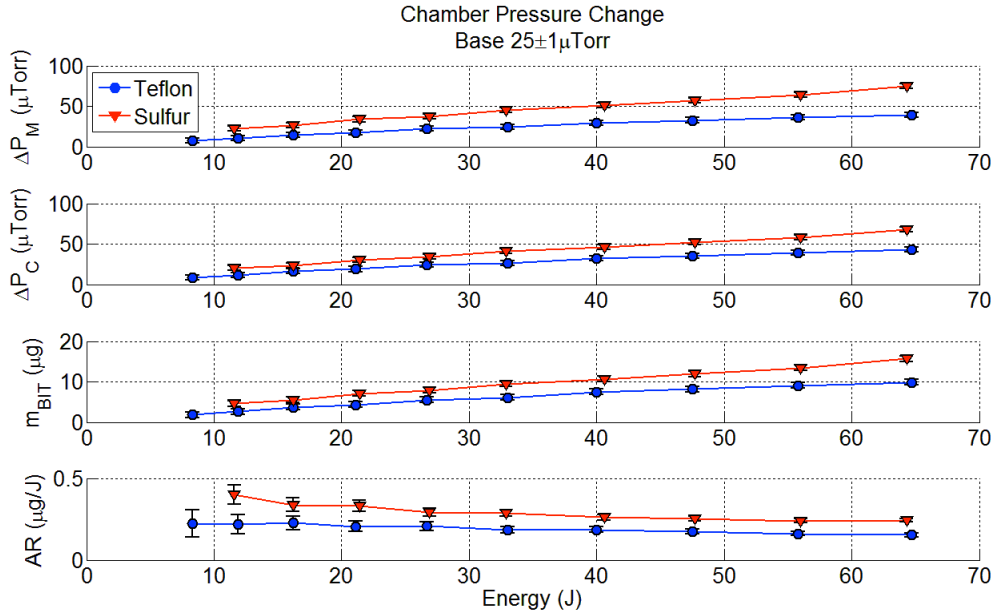
**Figure 5.17. The measured pressure increase from PPT discharges at varying energy levels for the 10 propellants tested.**

These pressure measurements are related to the ablated mass by the ideal gas law (Eq. 4.7), when it's assumed that all ablated material exists in gas form ~100's milliseconds after the discharge. As was shown in Fig. 5.13, sulfur propellant produced the highest specific thrust, while Teflon is the common baseline for the thruster. The ablated mass bit and mass ablation rate for Teflon and sulfur are shown in Fig. 5.18. Other than the belljar chamber vacuum chamber, volume of 130L, being smaller than the main chamber for the solid fuel experiment, the mass ablation estimate process here is identical to what was described in Chapter 4. It was found that both Teflon and sulfur showed a decreasing ablation rate at higher discharge energies; Teflon varied from  $0.2 \mu\text{g/J}$  at 10J to  $0.13 \mu\text{g/J}$  at 65J, while sulfur decreased from  $0.46$  to  $0.26 \mu\text{g/J}$  over the same energy range. Results from Chapter 4 show that the main chamber Teflon PPT had an ablation rate of  $0.2 \mu\text{g/J}$  at 100J, which matches the values found here.

These mass ablation rate are lower than expected and what has been measured previously with other experiment. This may be due to solid macroparticle ablation, which would not be measured by the pressure sensor. Macroparticles have been



measured previously to reach up to 30% of the total ablated mass. More accurate methods for measuring the ablated material have been completed with mass balances and should be considered for future alternative propellant testing [7, 8, 76].



**Figure 5.18.** The measured and corrected chamber pressure increase, calculated mass bit and mass ablation rate for Teflon and sulfur propellants (top to bottom).

## Chapter 6 Pulsed Plasma Thruster Atmospheric Operation

A propulsion system capable of geostationary positioning for high-altitude (20-40km), long-duration airships would allow for communication, weather observation, and surveillance operations over large surface areas at a fraction of cost for a satellite mission. The design of such a system at these mid-level atmospheric pressures offers a unique challenge, as the background density is too low for standard propeller devices, while being too high for most in-space thrusters. The low but constant wind speed at these altitudes necessitates the need for continuous propulsive ability, suggesting a thruster capable of using the ambient air as an in-situ propellant, eliminating the use of chemical propulsion rockets.

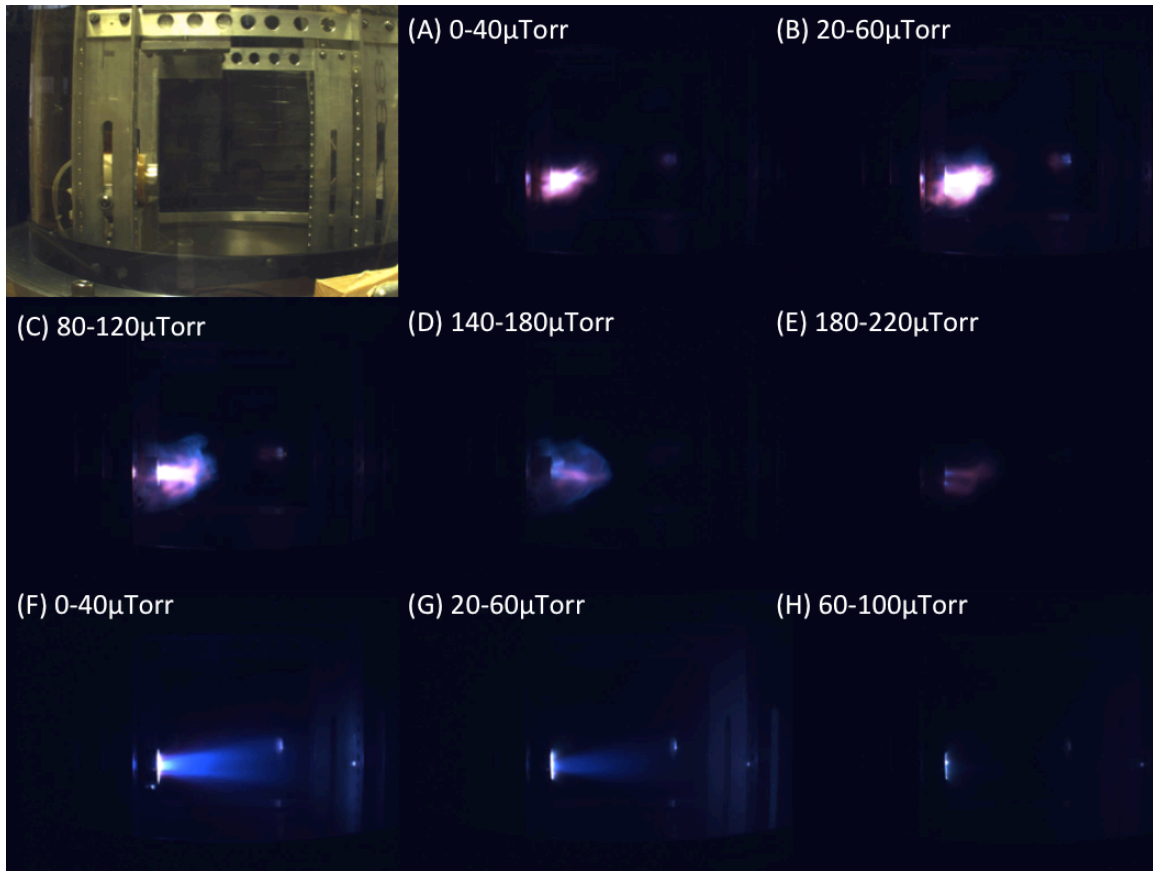
Use of electric propulsion for an air-breathing operation is not unheard of. Historically, research and funding has been allocated to electric propulsion devices in high-pressure environments in three primary areas: using the ambient gas in Earth's upper atmosphere as propellant, terrestrial drag reduction on supersonic flight, and extraterrestrial atmospheric flight. The first studies analyzing the potential for air-breathing Earth-orbital vehicles were published in the 1960s [77, 78]. More recently, Ion, Hall, and RF-heated thrusters have been introduced as feasible concepts for flight in Earth's atmosphere [79, 80, 81]. However, to date, none have been tested. Numerous studies have looked at using plasma injection to significantly alter the aerodynamic behavior of an aircraft in supersonic flight by reducing drag [82, 83, 84]. Mars, with its lower gravity and thinner atmosphere (4.5Torr surface pressure) [1], and Saturn's moon Titan, which is much less massive than Earth but harbors a thicker (1.1kTorr surface pressure) atmosphere [85], have both been identified as targets for atmospheric flight. Busek Co. Inc. was recently funded to research the Martian atmosphere breathing Hall Effect thruster (MABHET), in which the ambient gas in the atmosphere would be ionized and accelerated via the Hall Effect. MABHET was found to produce a maximum specific thrust of 33mN/kW when operating with in-situ fuel [86].

The inherent simplicity and robustness of the PPT, along with its ability to use the background atmosphere as an in-situ propellant makes it uniquely suited as a lightweight propulsion device for these airships. The thruster variants studied here operated at

background pressures of 5-50Torr. Such a pressure environment allows operation in both Earth's stratosphere (10-50km), which is above the average height of the powerful jetstream, and at low altitudes on Mars, ideal for an aircraft taking detailed images of the surface, while still able to cover large distances.

### 6.1 High Speed Camera Analysis With 1" Diameter Short Barrel Thruster

Based on the mean free path calculations described in Section 2.1 ( $\sim\mu\text{m}$  at 40Torr), any Lorentz acceleration would be canceled by a high ion-neutral collision frequency. Assuming a neutral particle radius of 150pm and ion velocity of 25km/s, the ion-neutral collision frequency varies between 0.5-2.5GHz at pressures of 10-40Torr. This suggests the atmospheric PPT will almost entirely be an electrothermal thruster, while at vacuum pressures, a hybrid electrothermal and electromagnetic device. This is supported by high-speed camera imagery (Fig. 6.1), which shows the thruster plasma plume for 33J discharges at background pressures of 40Torr and 50 $\mu$ Torr.



**Figure 6.1. Teflon propellant PPT high-speed camera images for 62J discharges at background pressures of 40Torr (top two rows) and 50 $\mu$ Torr (bottom).**

As shown in Fig. 6.1, at atmospheric pressures the plume appears to curl back towards the thruster as the expelled material collides with neutral particles inside and outside the discharge chamber. The intense light emission lasts for  $\sim 200\mu\text{s}$  when firing at atmospheric pressures;  $\sim 100\mu\text{s}$  longer than at microTorr pressures. This is due to the increased neutral collision frequency and ionization of the ambient air, creating what's commonly referred to as the snowplow effect. At microTorr pressures the plume expands outward in a diverging funnel shape with the light emission dying off after  $\sim 100\mu\text{s}$ . As described in Chapters 2 and 3, the vast majority of the plasma in space-like conditions travels from the thruster and interacts with the wall without any neutral collisions.



**Figure 6.2. High speed imagery taken with the camera set to a high exposure at discharge times of 0-100 $\mu\text{s}$ , 100 $\mu\text{s}$  – 5ms, 500 $\mu\text{s}$  – 5.5ms (left to right) relative to the igniter discharge for a 32J main discharge.**

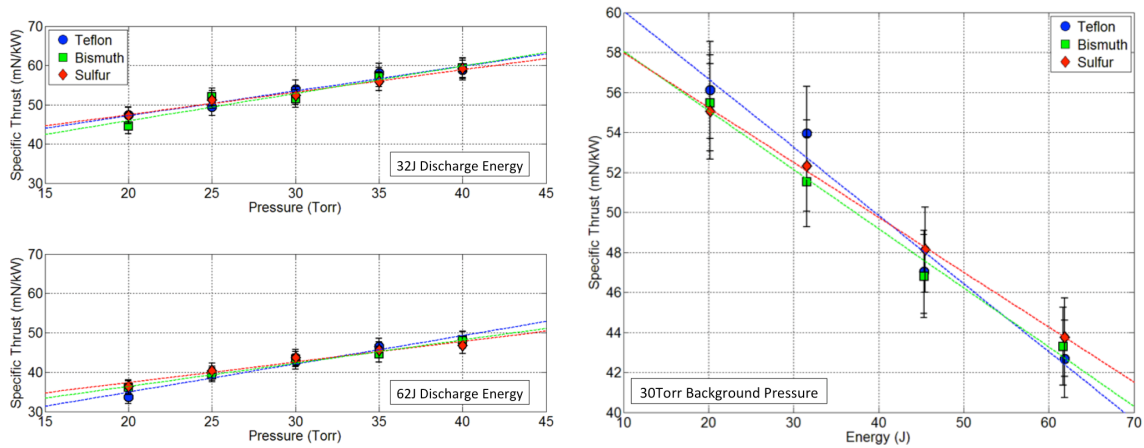
High-speed imagery with the camera set to its highest exposure was taken to better examine the exhaust plume late in time. The images in Fig. 6.2 were taken with the thrust stand's pendulum placed in front of the plume and with olivine propellant. Once the bulk of the plasma has recombined, large macroparticles from the discharge can be seen being expelled from the discharge chamber. These are captured as glowing orange streaks, and are the late-time ablation macroparticles discussed in previous studies. These particles have heavy mass ( $\sim \mu\text{g}$ ), but move relatively slowly ( $\sim 500\text{m/s}$ ). They account for the bulk of the thrusters ejected mass and their low velocity is the primary reason for the PPTs low ionization fraction and thrust efficiency.

## **6.2 Atmospheric Specific Thrust**

### **6.2.1 Thruster Geometry: 1" Diameter / 0.25" Height**

For atmospheric PPT operation, the pressure regime which the thruster can operate in should be set by the Paschen curve breakdown at the lower end (maximum capacitor

voltage set by electrode distance and background pressure) and the ability for the igniter to complete the main discharge circuit at higher pressures (large neutral collision frequency). The 1" diameter thrusters tested here could operate at background pressures between 5-45Torr, corresponding to altitudes of 20 – 33km in Earth's atmosphere. The atmospheric specific thrust results given here can be compared to the vacuum results in Section 5.2. All data shown in this section were taken with thrusters consisting of two 33 $\mu$ F capacitors rated to 1.4kV in parallel for the main discharge energy storage. It was found that by increasing the background pressure of the discharge chamber the mass bit (and subsequently the specific thrust) could be increased.

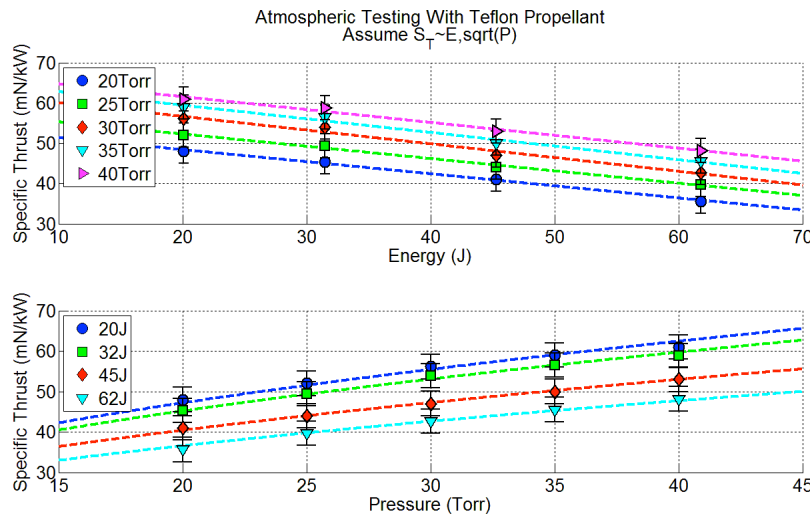


**Figure 6.3. The specific thrust of Teflon, bismuth sulfide, and sulfur PPTs with discharge energies of 32J and 62J fired at varying atmospheric pressures (left) and at a background pressure of 30Torr over varying discharge energies (right).**

It was found that the impulse bit and specific thrust increase with increasing pressure. For main discharge energies of 32 and 62J, this effect is shown for the Teflon, sulfur, and bismuth sulfide propellants in Fig. 6.3 for the 1in diameter thrusters shown in Figs. 6.1 and 6.2. At 32J, the specific thrust increases from 45 to 60mN/kW as the background pressure is increased from 20 to 40Torr. While at 62J, the specific thrust increases from 35 to 45mN/kW over the same pressure range. All three propellants, within the margin of error, result in the same specific thrust, leading to the conclusion that the majority of the thrust is produced by the acceleration of the ambient air and not the ablated fuel. This is further supported by the increase in specific thrust with increasing background pressure, as there is more ambient air mass in the discharge chamber at 40Torr than at 20. Assuming that the air density in the chamber decreases linearly with pressure from

$1.225\text{kg/m}^3$  at 760Torr, and with a discharge chamber volume of  $10\text{cm}^3$ , the mass of the ambient air inside the discharge chamber is estimated to be between 300 and 600 $\mu\text{g}$ , at 20 and 40Torr, respectively. These estimates are an order of magnitude higher than the ablated mass estimates of  $2\mu\text{g/J}$  [8], implying that the contribution to impulse from the heated ambient air is much greater than the contribution from the ablated propellant.

The right panel of Fig. 6.3 shows how the thrusters specific thrust varies with increasing capacitor energy at 30Torr. The impulse bit continues to increase with increasing capacitor energy, however when normalized by the energy, the specific thrust decreases linearly over this energy range. This result is in opposition to the vacuum data shown in Section 5.2, where the specific thrust increases and then levels off with increasing energy over the same energy range, again suggesting that the PPT has different acceleration mechanisms between the atmospheric and vacuum operating regimes.



**Figure 6.4. Measured specific thrust results for a Teflon thruster at pressures between 20-40Torr and energies of 30-62J. The best-fit curves (Eq. 6.2) have a linear relationship with energy and square root with pressure.**

Knowing that the performance of the atmospheric PPT is irrelevant to the type of propellant used an expanded data set was completed with only Teflon. The data set contained discharge energies of 20, 32, 45, and 62J at each background pressures between 20-40Torr in 5Torr increments and is shown in Fig. 6.4. This data allowed for an empirically derived formula of the thrusters performance, based entirely on the background pressure and discharge energy to be calculated. Over the range of energies

and pressures tested, the performance could be accurately modeled ( $R^2 > 0.95$ ) as being linearly dependent on both pressure and energy with

$$S_T = 0.64P_e - 0.32E_c + 43 \quad 6.1$$

Eq. 6.1 is a linear expression with two dependent variables that matches the measured results at pressures of 20-40Torr to within 3mN/kW. Expanding to lower pressures, it's known that the specific thrust of an identical Teflon thruster at vacuum has been measured to be  $10 \pm 2$  mN/kW for discharge energies of 10-70J (Chapter 5). This implies that at low pressures, the specific thrust should be 10mN/kW, irrespective of the discharge energy. A number of different power and exponential curves which required  $S_T = 10$  mN/kW at 0Torr were fit to the data, the most accurate ( $R^2 > 0.97$  between 20-40Torr) was

$$S_T = (-0.06E_c + 9.42)P_e^{0.5} - 10 \quad 6.2$$

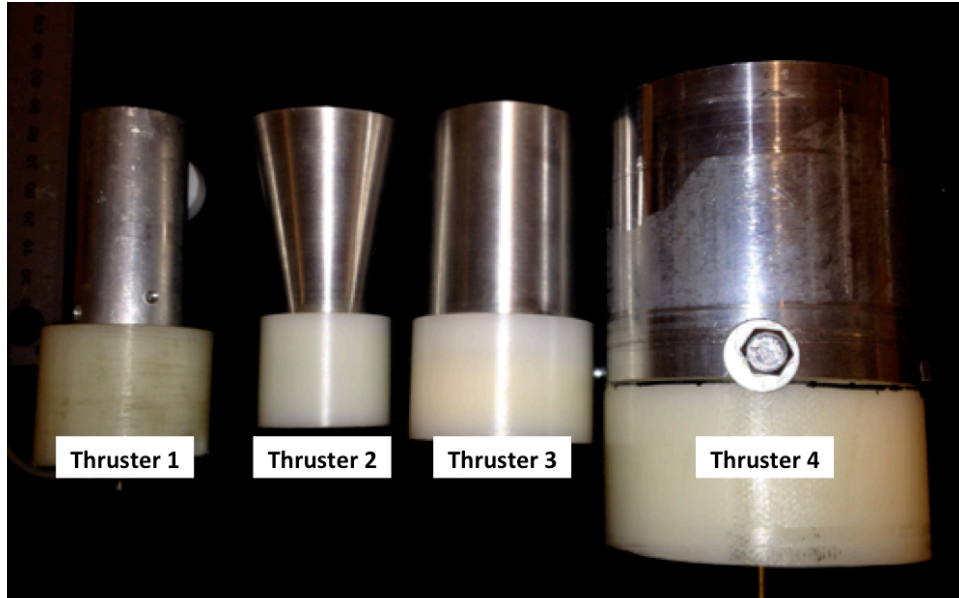
Between 20-40Torr and 20-62J, Eq. 6.1 and 6.2 are identical to within 2mN/kW ( $R^2 > 0.95$ ). A data set over larger pressure and energy ranges could not be completed due to the high neutral collision frequency (limiting higher pressures and lower capacitor voltages) and uncontrollable discharges due to Paschen breakdowns (limiting lower pressures and higher capacitor voltages).

Although considerably different operating regimes, the performance of the PPT (widely considered the simplest electric propulsion device) in the atmosphere yields higher specific thrusts than any of the well optimized and highly efficient Hall and Ion thrusters currently in use on satellites.

### **6.2.2 Thruster Geometry: 1-3" Diameter / 3" Height**

Based on results from the previous section which said that a higher number density of particles within the discharge chamber resulted in more thrust, the volume of the discharge chamber was varied to determine how the thrust of an atmospheric PPT varies with size. In addition to the short barrel PPT (Sections 5.1, 5.2, 6.1, and 6.2), four longer barreled thrusters were tested. All thrust results with the short barrel PPT had the thrust stand pendulum at 1cm from the cathode (2cm from propellant surface). The longer PPTs

tested here had a cathode that extended 8cm from the propellant surface. The pendulum was again placed 1cm from the cathode and therefore was 9cm from the propellant face. The four PPTs tested are shown in Fig. 6.5 and their dimensions given in Table 6.1.



**Figure 6.5. The 8cm electrode length PPTs tested for atmospheric operation.**

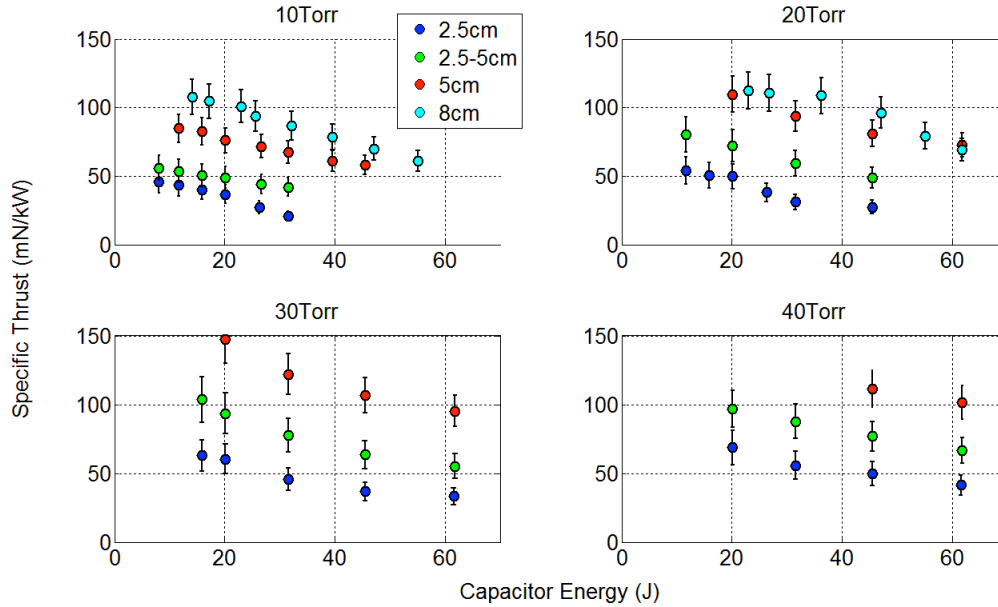
**Table 6.1. The physical dimensions of the four 8cm cathode length PPTs tested.**

|                                            | Thruster 1 | Thruster 2 | Thruster 3 | Thruster 4 |
|--------------------------------------------|------------|------------|------------|------------|
| Electrode Length (cm)                      | 8          | 8          | 8          | 8          |
| Minimum Cathode Diameter (cm)              | 2.5        | 2.5        | 4.8        | 7.6        |
| Cathode Exit Diameter (cm)                 | 2.5        | 4.8        | 4.8        | 7.6        |
| Minimum Electrode Distance (cm)            | 1.1        | 1.1        | 2.2        | 3.6        |
| Chamber Volume (cm <sup>3</sup> )          | 35         | 120        | 135        | 400        |
| Propellant Surface Area (cm <sup>2</sup> ) | 4.9        | 4.9        | 18.1       | 45.4       |

Similar trends to the smaller barrel thruster design were found. As expected, the general trend was that more ambient air in the discharge chamber resulted in a higher impulse. At a background pressure of 10Torr the smallest of the four thrusters (Thruster-1) had a decreasing specific thrust from 48–19mN/kW as the main discharge capacitor energy was varied from 8-32J, while the largest of the thrusters (Thruster-4) had a specific thrust of 111-59mN/kW as the energy increased between 14-55J. At 40Torr Thruster-1 varied its specific thrust between 72-41mN/kW at energies from 20-62J, while thruster-4 thruster couldn't fire. The operational pressure range will be discussed further in Section 6.3.



The fact that the specific thrust between the smallest and largest thruster continues to increase for a given discharge energy at 10 and 20Torr implies that the highest efficiency thruster cross-section is larger than what's been tested to date. Larger diameters however restrict operation to lower pressures (higher altitudes). Assuming a 1Hz firing frequency, the 8cm diameter thruster had 3.3mN of thrust at 10Torr and 4.3mN at 20Torr, while the 5cm thruster had 5.8mN at 30Torr and 6.5mN at 40Torr.



**Figure 6.6.** The specific thrust of the long (blue), flared (red), wide (green), and wider (cyan) PPTs tested at background pressures of 10-40Torr.

The general trend of Fig. 6.6 is for the data points to move up in specific thrust and to the right in energy as the background pressure and electrode distance are increased. This will be explained further in Section 6.3.

### 6.3 Operating Pressure Range

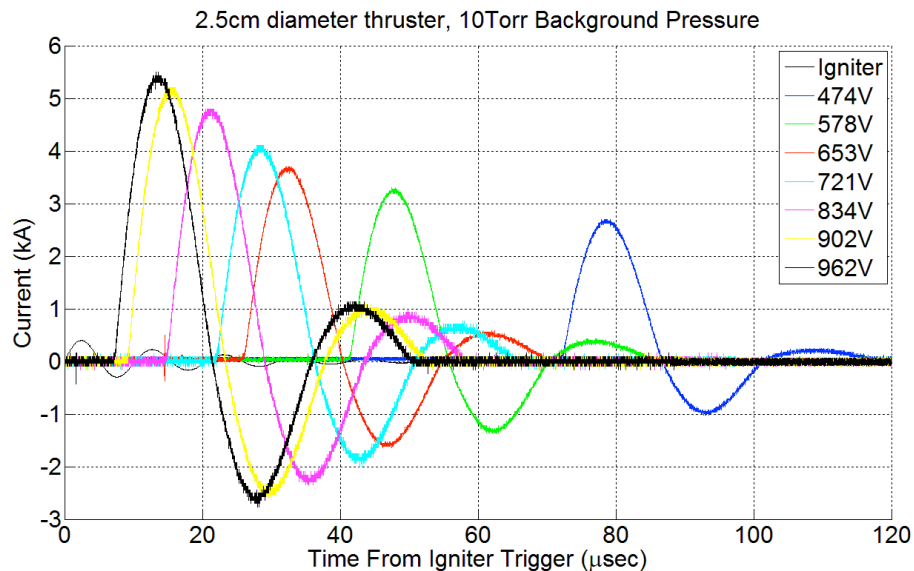
Using the four varying diameter thrusters in Section 6.2.2, it was found that the operating regime of the atmospheric PPT is a function of the background pressure, main capacitor voltage, and the minimum anode-cathode electrode distance.

$$PPT_{Range} = f(P, V, d_{electrode}) \quad 6.3$$

At higher background pressures and larger electrode spacing, a larger voltage potential was required to allow for the igniter arc to complete the main discharge circuit. The

voltage standoff between the thruster electrodes was lower at shorter distances as well as lower background pressures due to operating on the right-hand side of the Paschen curve. The lowest voltage required to fire is set by the electron-neutral collision frequency. Higher background pressures required higher discharge voltages as an increase in the electron-neutral collision frequency required more energy to complete the main discharge circuit. Uncontrollable discharges due to Paschen's law prevented higher voltages from being placed on the electrodes at low background pressures. These two limits set the operational range for these thrusters in the atmosphere.

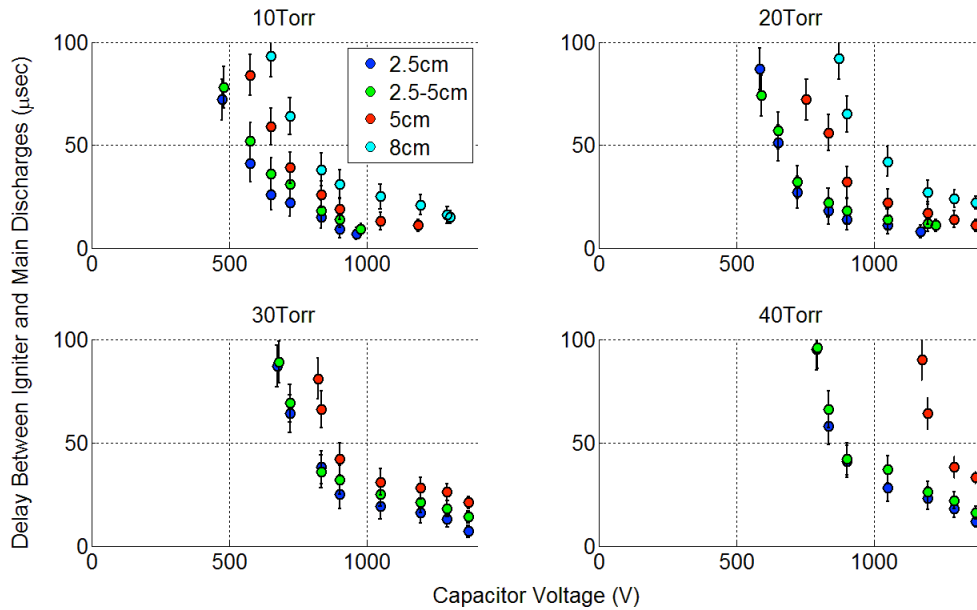
All four thrusters shown in Fig. 6.5 were tested at pressures between 5-40Torr, in 5Torr increments and at voltages between 400-1400V, in 100V increments. It was found that, at a given pressure, the time delay between the igniter triggering and the main discharge current flowing would increase from  $\sim 10\mu\text{sec}$  at high voltages to  $\sim 50\mu\text{sec}$  at low voltages, before the main discharge voltage was lowered to a point where the igniter was unable to complete the main discharge circuit. Figure 6.7 shows the igniter and main discharge current waveforms at a background pressure of 10Torr for main voltage potentials between 470-960V with the 2.5cm diameter thruster.



**Figure 6.7. Igniter and main discharge current traces for the 2.5cm diameter thruster at 10Torr pressure for main discharge voltages between 474-962V.**

It was found that no thruster was able to fire when the delay between the igniter and main was greater than  $100\mu\text{sec}$ . The variance to this time delay for low voltages was  $\sim 10\mu\text{sec}$

and reduced to  $\sim 3\mu\text{sec}$  at higher voltages. Fig 6.8 shows the time delay for the four thrusters at pressures of 10-40Torr in 10Torr increments.



**Figure 6.8. Time delay between the start of the igniter current and main discharge currents for thrusters with cathode diameters of 2.5cm, 2.5-5cm flared, 5cm, and 8cm at pressures of 10-40Torr.**

The maximum and minimum voltages found for each thruster are given in Table 6.2 and Fig. 6.9. Both the Paschen and electron-neutral collision frequency voltage boundaries result in linear relationships when compared to product of the ambient pressure and minimum electrode spacing. The maximum error between the lines of best and measured data is 7% for the upper voltage limit and 10% for the lower.

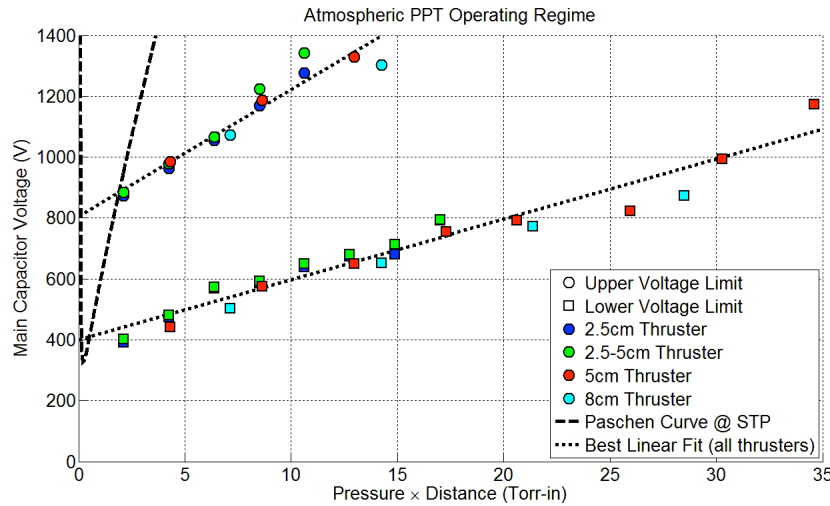
**Table 6.2. The minimum and maximum voltages for controlled PPT discharges 5-40Torr pressures. The voltages are given to the nearest 10V between 350-1400V. Thrusters numbered per Fig. 6.5.**

| Thruster        | 1                                          | 2   | 3    | 4     | 1                                                  | 2     | 3     | 4     |
|-----------------|--------------------------------------------|-----|------|-------|----------------------------------------------------|-------|-------|-------|
| Pressure (Torr) | Minimum Voltage Required For Discharge (V) |     |      |       | Maximum Voltage Without Uncontrolled Breakdown (V) |       |       |       |
| 5               | 390                                        | 400 | 440  | 500   | 870                                                | 880   | 990   | 1070  |
| 10              | 470                                        | 480 | 580  | 650   | 960                                                | 980   | 1190  | 1300  |
| 15              | 570                                        | 570 | 650  | 770   | 1060                                               | 1070  | 1330  | >1400 |
| 20              | 590                                        | 590 | 760  | 870   | 1170                                               | 1220  | >1400 | >1400 |
| 25              | 640                                        | 650 | 800  | >1400 | 1280                                               | 1340  | >1400 | >1400 |
| 30              | 670                                        | 680 | 820  | >1400 | >1400                                              | >1400 | >1400 | >1400 |
| 35              | 680                                        | 710 | 990  | >1400 | >1400                                              | >1400 | >1400 | >1400 |
| 40              | 690                                        | 800 | 1180 | >1400 | >1400                                              | >1400 | >1400 | >1400 |

The expression for the Paschen curve at standard temperature and pressure (STP) and with units of pressure (P) in atmosphere, electrode spacing (d) in meters, and voltage (V) in volts is given as

$$V = \frac{(4.366 \times 10^7) Pd}{\ln(Pd) + 12.8} \quad 6.4$$

As shown in Fig. 6.9, this expression overestimated what maximum voltage could be placed on the electrodes. It's assumed that Paschen's law at STP is approximately valid at low pressures, but may not be exact, partially explaining the upper voltage limit variation. Also possible is imperfections in the machining of the electrodes, possibly causing micro-sized sharp edges on the anode and cathode, resulting in corona regions [87] and reducing the effective distance (d) in Eq. 6.4.



**Figure 6.9. The operational regime of the four varying diameter atmospheric thrusters. The thrusters were tested at pressures between 5-40Torr and voltages from 400-1400V.**

For operation on the right-hand side of the Paschen curve, if the atmospheric PPT is to be operated at high altitudes (lower pressures) then larger electrode spacing is required to reach the higher discharge voltages. No testing has been completed to the left of the Paschen curve due to pumping capabilities of the vacuum chamber, however operation in this low-pressure regime should allow for higher voltages. This could be important for operation in Earth's atmosphere at altitudes above 100,000ft where the pressure falls below 10Torr or on Mars with a 5Torr ground pressure. To operate at higher altitudes in

Earth's atmosphere or at all on Mars will require a regime to the left of the Paschen curve.

The above results show the feasibility of using the PPT as a long duration station-keeping thruster onboard high-altitude airships in Earth's upper atmosphere. The acceleration mechanism of the atmospheric thruster was found to be highly electrothermal with the majority of the thrust from the heated and accelerated ambient air, in essence creating an in-situ thruster. Multiple electrode geometry thrusters were tested, with the larger barrels showing increased performance, while the smaller diameter barrels could be operated at higher background pressures. Higher discharge energies showed a higher impulse bit, while lower energies showed higher efficiency. To test the operation of the thruster outside the laboratory, two atmospheric flights were flown, the first in 2013 (Section 7.2) and the second in 2015 (Section 7.4).

## Chapter 7 Atmospheric Pulsed Plasma Thruster Flights

The ability to provide station-keeping capabilities for high-altitude airships would allow for communication and surveillance operations over a larger surface area than aircraft can provide and at a drastically reduced cost from in-space satellite. Propulsion at stratospheric altitudes is a difficult engineering challenge as the pressure is too low for propeller devices and too high for most traditional in-space thrusters. The physical testing of electric propulsion systems in the atmosphere provide for three learning opportunities, first to allow for testing in a environment not repeatedly in the laboratory, secondly to detach the experiment from laboratory computers and power, and the building and testing of an airship structure in its operational environment

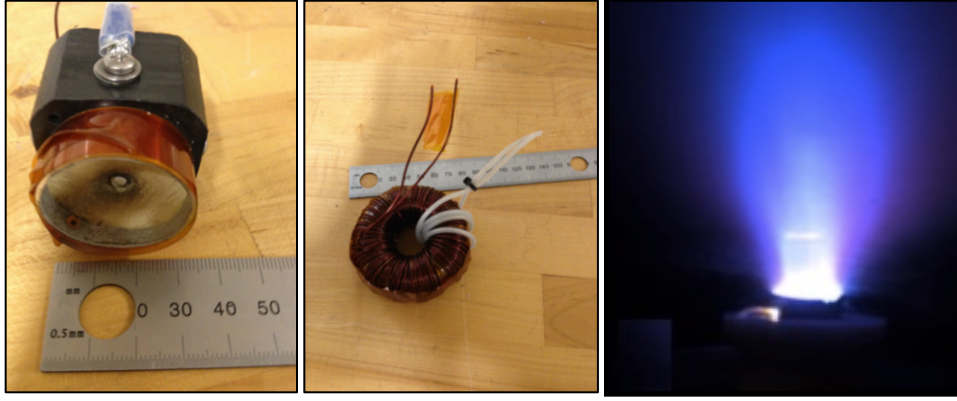
Two PPTs were launched from high-altitude balloons to test autonomous operation in the atmosphere. The first in 2013 attached to a latex burst balloon; the second in 2015 onboard a semirigid airship. These flights were both in compliance with the FAA regulations on small ballooning covered in Part 101 [88]. In particular, our launches were:

- (1) Individual payload mass under 4lbs if weight/size ratio  $>3\text{oz/in}^2$
- (2) Individual payload mass under 6lbs if weight/size ratio  $<3\text{oz/in}^2$
- (3) Total payload mass under 12lbs
- (4) Cloud coverage  $<50\%$
- (5) Visibility  $>5\text{miles}$
- (6) Impact of payload with ground could not damage person/property

### 7.1 Construction and Laboratory Calibration: 2013 Launch

A PPT (Fig. 7.1) was launched on board a high-altitude balloon on June 1<sup>st</sup>, 2013 from the Grant County International Airport in Moses Lake, Washington. The objective of the test was to gain the knowledge that accompanies flight heritage in a non-laboratory setting. In particular we were testing the ability of the thruster to fire autonomously in-situ in the stratosphere. The flight PPT discharge chamber was similar in design to the short barrel laboratory model. The inner diameter of the cathode was 1.2” and the height

of electrodes was  $\frac{1}{4}$ ". Sulfur was chosen as the propellant to further demonstrate the feasibility of alternative propellants in PPTs.



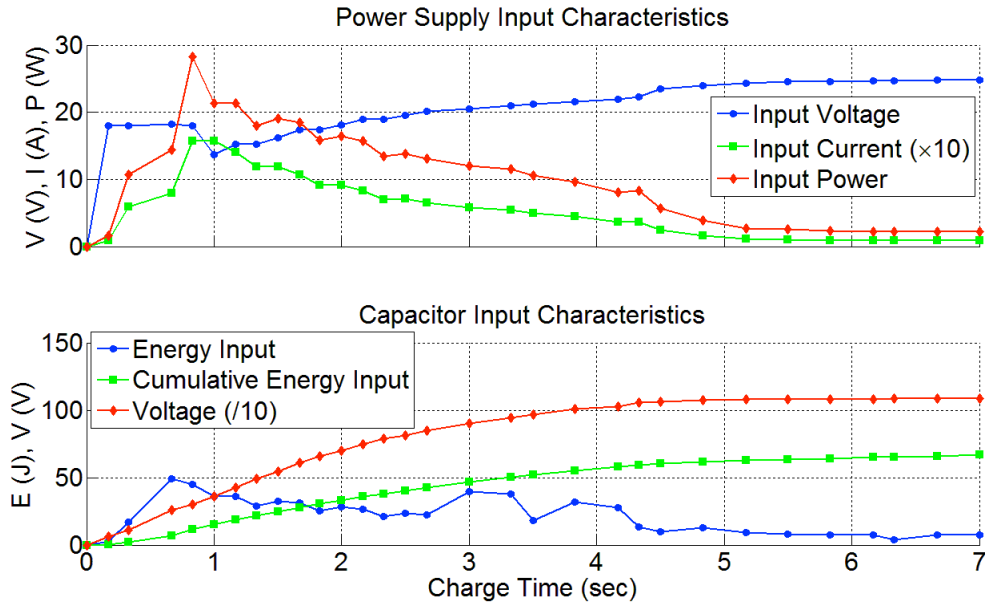
**Figure 7.1. The 2013 flight PPT (left), 90mm diameter transformer (center, and plume (right) of the thruster firing at an energy of 42J in the laboratory.**

The main discharge energy storage was two 33 $\mu$ F capacitors in parallel, charged with a 30W, 1kV DC-DC converter. The entire thruster was powered by six 9V procell batteries. The power draw from the procell battery pack by the DC-DC converter to charge the 66 $\mu$ F capacitor bank to 1000V (33J) was measured and is shown in Fig. 7.2. A constant voltage power source such as used here, has a maximum electrical efficiency ( $\eta_{eff}$ ) of 50%, as resistive losses in the wires ( $IR^2$ ) will equal the capacitor energy ( $0.5CV^2$ ).

$$E_{RES} = \int I^2 R dt = \int \left[ \frac{V}{R} \exp\left(-\frac{t}{RC}\right) \right]^2 R dt = 0.5CV^2 \quad 7.1$$

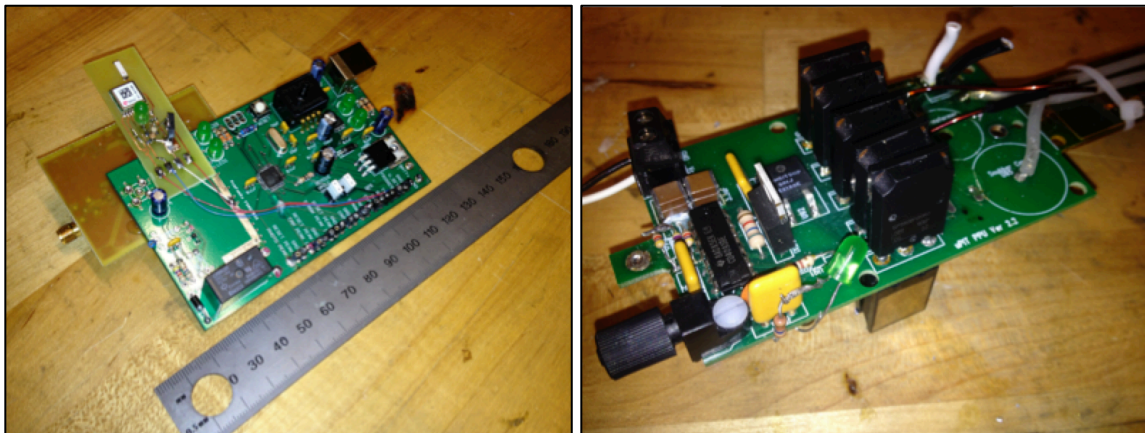
$$\eta_{ele} = \frac{E_{CAP}}{E_{TOTAL}} = \frac{E_{CAP}}{E_{CAP} + E_{RES}} = 50\% \quad 7.2$$

Fig. 7.2 shows that the power system draws 66J from the procell battery pack to charge the PPT main discharge capacitor to 33J, resulting in an electrical efficiency of 50%.



**Figure 7.2. Power, voltage, and current draw from a laboratory power supply (top), and energy, cumulative energy, and charge voltage to the capacitors (lower).**

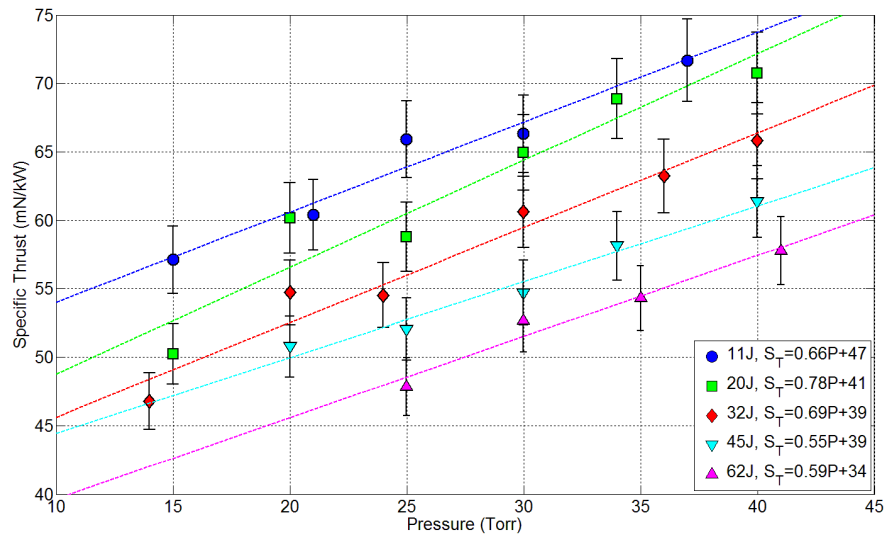
The igniter circuit consisted of one Cornell Dubilier 1.5uF capacitor (typically charged to 400V), an IGBT switch (Fig. 7.3), and a 4:200 iron core step-up toroidal transformer (Fig. 7.1). The secondary side of the transformer was wound with 14AWG silicon coated wire rated to 40kV. The toroidal ferrite core was undersized (OD=62mm, 35mm, and h=25mm) for typical operations at these power levels based on the minimum transformer cross-sectional equation  $A_{c,min} = VT/NB_{SAT}$ . However as the system only required an initial arc from the igniter and not sustained power, saturating the core was acceptable. The primary side had an inductance of 3.2μH, resulting in a current trace roughly shaped as a decaying sinusoidal, with an initial peak current of 285A and 13.8μs period.



**Figure 7.3. The telemetry board (left) and IGBT trigger board (right).**



To control the experiment during flight, an Arduino code running on an AMTEL ATmega328P 20MHz microcontroller (Fig. 7.3) was developed to autonomously charge and discharge the PPT capacitors based on firing results. The code partitioned the flight into three distinct periods of operation. While rising at pressures above 40Torr, the system transmitted the flight time and GPS coordinates, leaving the rest of the system unpowered. When the pressure decreased below 40Torr, the computer charged the capacitors to a set value of 500V and triggered the igniter IGBTs to close, firing the PPT. If the PPT fired successfully, another shot at the same voltage was attempted. If the computer detected a misfire, the capacitor voltage was varied in two ways. If a breakdown occurred without the igniter firing, the capacitor voltage was lowered by 10%. If the igniter fired and the capacitors failed to discharge, the capacitor voltage was raised 10%. All discharges that were measured less than 1ms after the igniter was triggered to fire were transmitted as confirmed PPT firings. Once the pressure increased above 50Torr during the descent, the system shut down the PPT energy storage system but continued to transmit the GPS coordinates of the balloon.



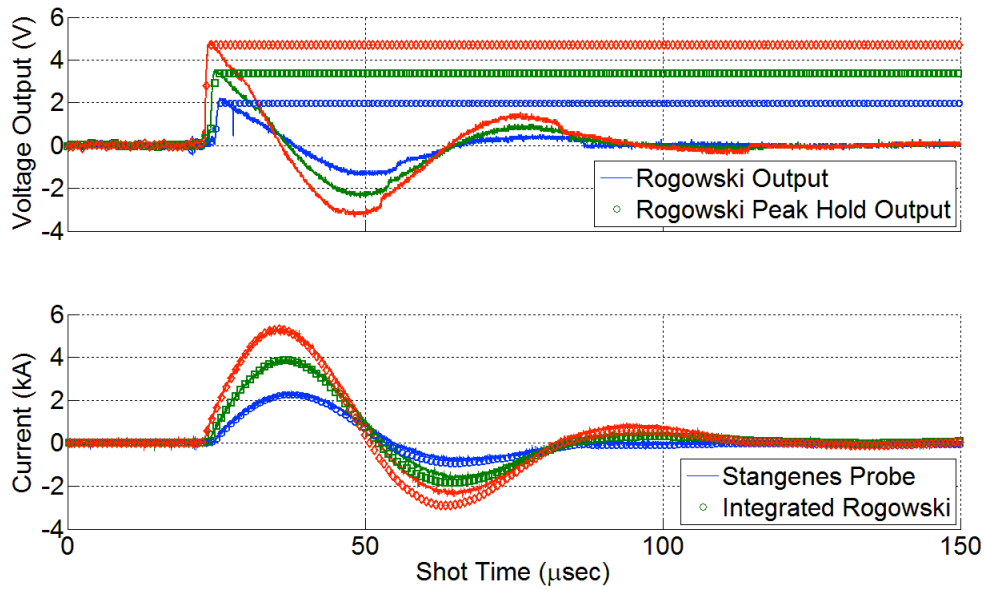
**Figure 7.4. Laboratory specific thrust measurements of the flight PPT at main discharge energies between 11 and 64J over a pressure range of 14-41Torr.**

As shown in Fig. 7.4, laboratory thrust measurements were taken of the flight PPT at varying discharge energies for varying background pressures. Similar to the smaller

thruster detailed in Sections 6.1 and 6.2, the specific thrust increases with increasing pressures and decreases with increasing energy and can be described by

$$S_T = 0.66P_e - 0.23E + 47 \quad 7.3$$

an empirically derived linear equation of two dependent variables. For identical background pressures and capacitor energies, the specific thrust of the flight PPT was measured to be 14-22% higher than for the smaller laboratory version. It is assumed that this increase was primarily due to the discharge chamber volume being increased 29%.

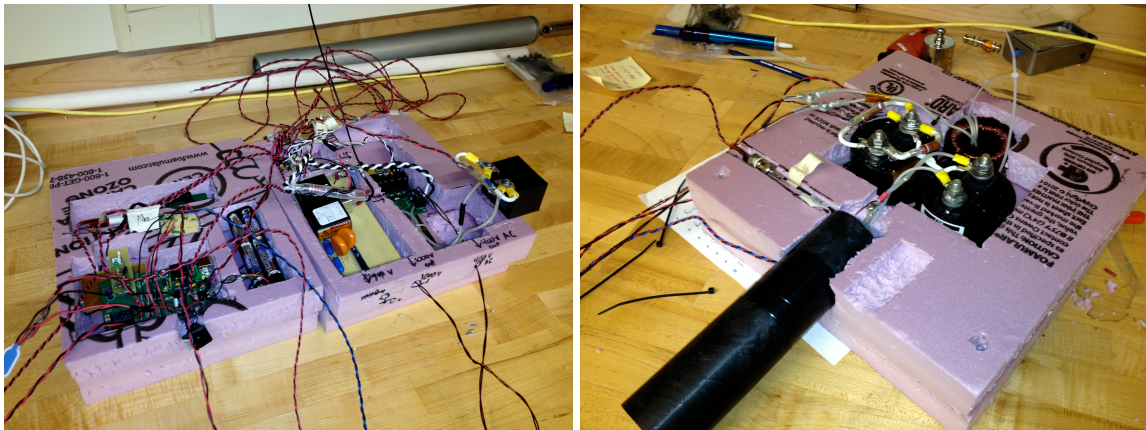


**Figure 7.5. The Rogowski coil signal before and after the peak hold circuit (top) and the integrated Rogowski output plotted against the Stangenes transformer output (bottom) at 20Torr for discharge energies of 8 (blue), 21 (green), and 33J (red).**

A Rogowski coil was used to measure the PPT main discharge current on the flight, the output of which, measured from a laboratory oscilloscope, is shown in Fig. 7.5a. The voltage output from the Rogowski coil is proportional to the time derivative of the current passing through the loop, given by Eq. 3.15. As the flight computer had a 5V upper limit, the Rogowski coil was designed so that  $V_{\text{rog, max}} = 5\text{V}$  for a 33J discharge. Due to power and bandwidth constraints on the PPT flight, only maximum voltage values could be recorded and transmitted. To measure this voltage, a peak-hold circuit was used to maintain the maximum Rogowski coil output voltage over 100ms (Fig. 7.5a), which allowed the telemetry system to sample and transmit the value. The integrated Rogowski

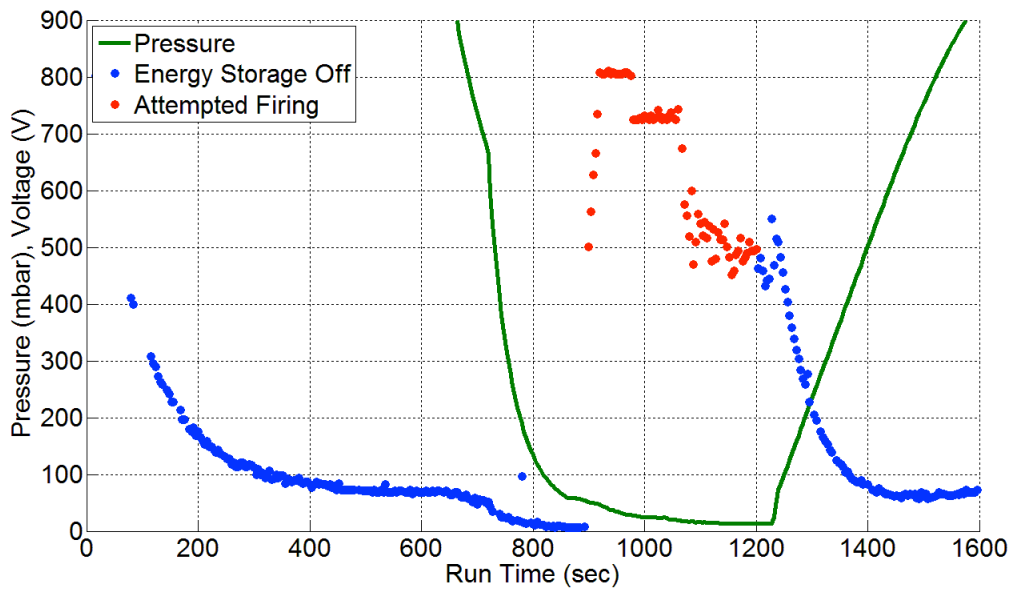
output is plotted against the Stangenes output in Fig. 7.5b. The Rogowski coil accurately models the first half-period of the current trace and then begins to differ after the current first changes direction. This was acceptable as only the peak Rogowski value could be transmitted with the limited-bandwidth telemetry system on the flight.

The entire PPT flight electronics and hardware fit into four foam-insulated packages (Fig. 7.6) weighing 3.75kg and occupying 972 in<sup>3</sup>. Package 1, nearest to the balloon, housed a vertically mounted high-definition camera looking down towards the PPT. Package 2 included the flight computer, high-voltage power supplies, igniter capacitor, IGBT board, and batteries. Package 3 housed the PPT, main discharge capacitors, igniter transformer, and horizontal high-definition camera. The GPS unit was stored in package 4, furthest from the balloon. An 8244-meteorological 1200g balloon and an 8ft diameter parachute were used to lift the payload and return it safely to Earth.



**Figure 7.6. PPT flight packages 2 (left) and 3 (right) during construction.**

The complete system, with all data transmitted wirelessly, was tested in the belljar vacuum chamber before the flight. The measured background pressure and discharge voltage from one such test is shown in Fig. 7.7. Once the background pressure reached below 40Torr, the thruster charged and began firing at voltages between 500-800V until the chamber was brought up to air.



**Figure 7.7. The thrusters main discharge voltage and chamber background pressure for a laboratory test completed before the 2013 flight.**

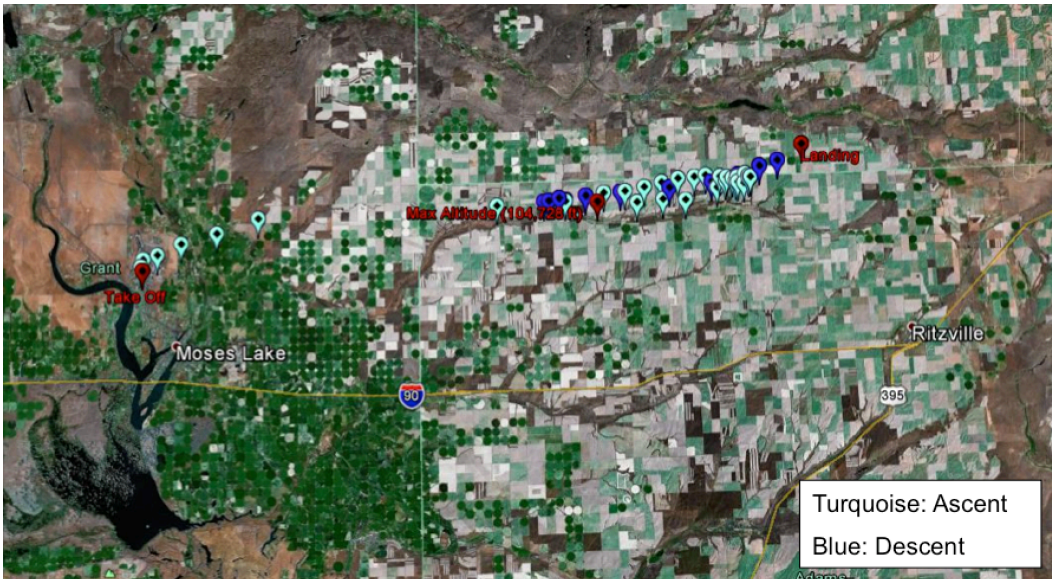
## 7.2 Atmospheric Burst Balloon: 2013 LEAF Mission

The high-altitude balloon was launched at 7:29am on June 1<sup>st</sup>, 2013 from the Grant County International Airport in Moses Lake, Washington. The entire flight took 130min with the balloon popping 100min into the flight. The maximum altitude was 31.9km with a minimum pressure of 12Torr at apogee and a minimum temperature of -46.2C occurring at an altitude of 17.2km. The balloon traveled a total of 56km east, landing in a wheat field north of the city of Ritzville.

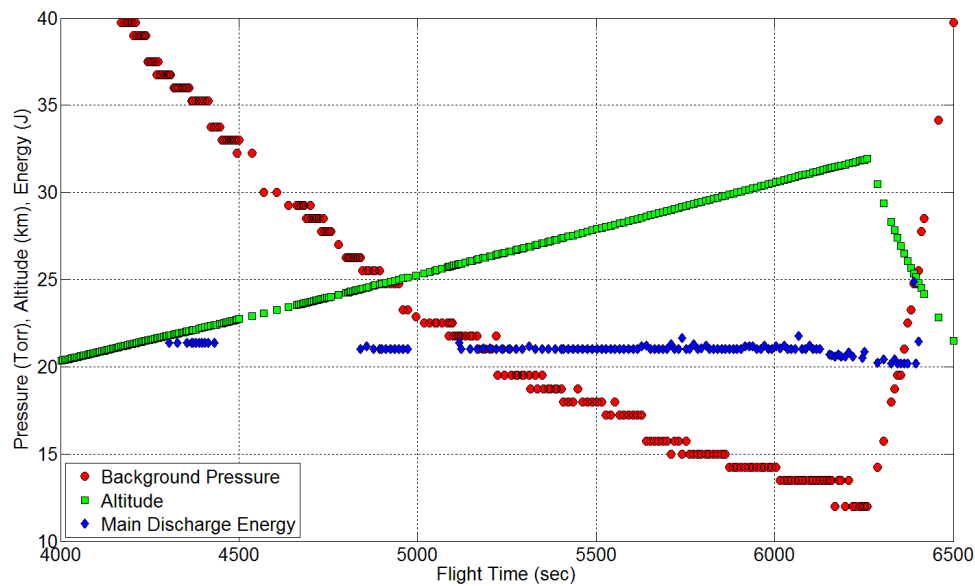




**Figure 7.8.** Onboard images of (top left to bottom right) launch, 10min after launch, 60min after launch while firing at 70,000ft, and 90min after launch while firing at apogee (2sec after balloon burst).

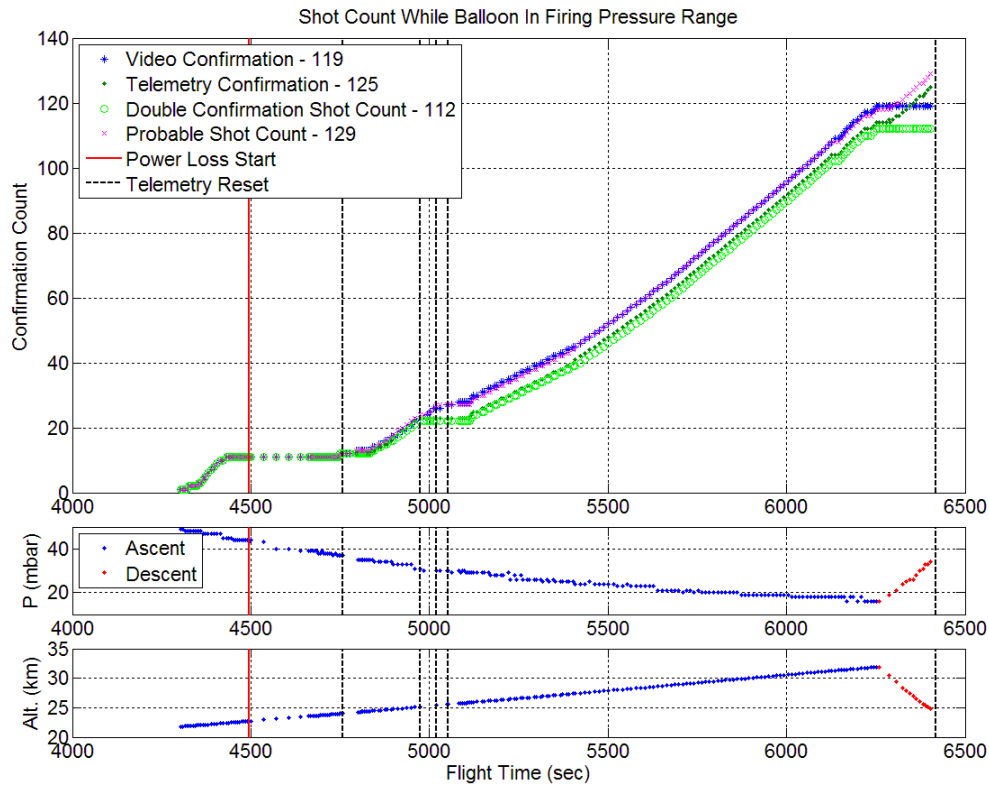


**Figure 7.9.** The GPS tracking data from the balloon payload over the duration of the flight. The turquoise points are from the ascent, the blue the descent.



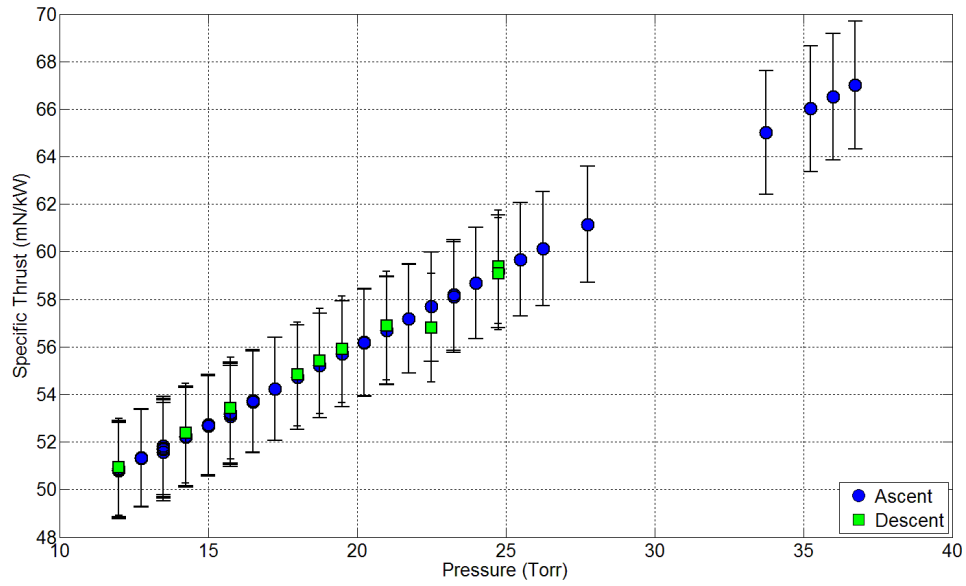
**Figure 7.10. The background pressure, balloon altitude, and main discharge capacitor energy for the 129 successfully recorded PPT firings.**

The PPT was able to charge and fire nominally for 25min, as shown in Fig. 7.10. The computer telemetry system confirmed 125 successful discharges (114 ascent, 11 descent), while the audio from the HV video camera picked up 119 (ascent only). There was extensive background noise on the camera after apogee resulting in no PPT audio on the descent. There were 112 discharges confirmed by both the telemetry and camera systems, and we estimate 129 probable controlled discharges (112 dual confirmations, 13 muffled audio but confirmed telemetry, and 4 confirmed audio without telemetry). The computer initially charged and attempted to fire the PPT once the pressure reached 64mbar (64,800ft), with the first confirmed firing at 49mbar (71,500ft). The system reset multiple times at altitudes between 23-26km, as show in Fig. 7.11. It's believed that operation at these altitudes falls near the minimum of the Paschen curve for typical circuit board components with pin spacing of 0.1".



**Figure 7.11. The shot count of the PPT as confirmed via the audio from the HV video camera (blue) and from the telemetry (green dots).**

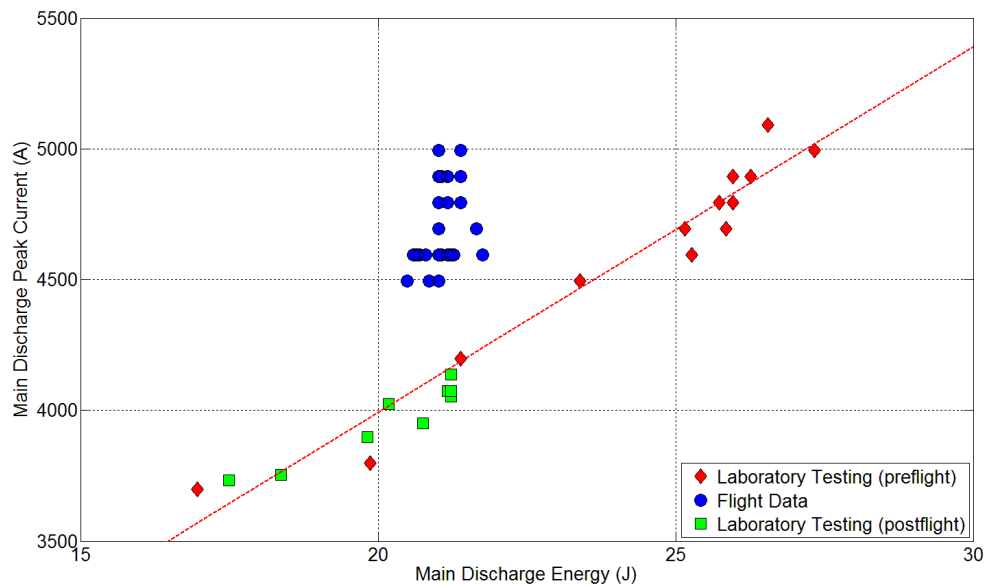
Relating the background pressure and capacitor energy measurements taken during the flight to the laboratory calibration results with Eq. (7.3), the specific thrusts of the 129 discharges were calculated and are shown in Fig. 7.12. These inferred specific thrust results, which are highly dependent on the exterior pressure, were found to vary between 50.8 and 67.0mN/kW, corresponding to impulse bits of 1.0 – 1.4mN-s throughout the flight. This is potentially a doubling in specific thrust of the 33mN/kW Hall thruster currently in development for flight in the Martian atmosphere [86]. The slight larger spread in the decent results can be attributed to the violent shaking and high velocity of the payload as it fell, compared to the relatively smooth ascending flight. These vibrations could have caused inaccuracies in the telemetry system and the high speed could result in errors in the recorded position.



**Figure 7.12. The calculated specific thrusts for the 129 successful discharges of the flight PPT for the ascent (blue) and descent (red).**

Inferring the specific thrust of the flight PPT based on laboratory results requires knowledge that operation was similar between the two environments. This was done by measuring the main discharge current peak with a Rogowski coil, as shown in Fig. 7.13. Higher peaks were recorded on the flight (4.5-5kA) than during laboratory testing (3.9-4.2kA) at discharge energies of 21J. It is known that low temperature reduces wire resistance and increase capacitor capacitance. Eq. (1.7) shows that both those effects would increase the peak current produced. Using the temperature coefficient data provided for the Cornell Dubilier polyethylene capacitors [89] a decrease of 60°C would result in a capacitance increase of 0.8%. Using the temperature coefficient for copper of  $\alpha = 3.93\text{mOhm}/^\circ\text{C}$  and an initial resistance of 110mΩ at 20°C, it can be assumed that the stray wire resistance would decrease to 85mΩ at flight temperatures. Modeling these modifications with Eq. (1.10) shows that the temperatures experienced throughout the flight would result in a peak current increase from 4.2kA to 4.6kA, explaining the current increase measured on the flight. This increase in peak current could result in a thrust increase; however the vacuum chamber is not presently equipped with temperature controls to verify.





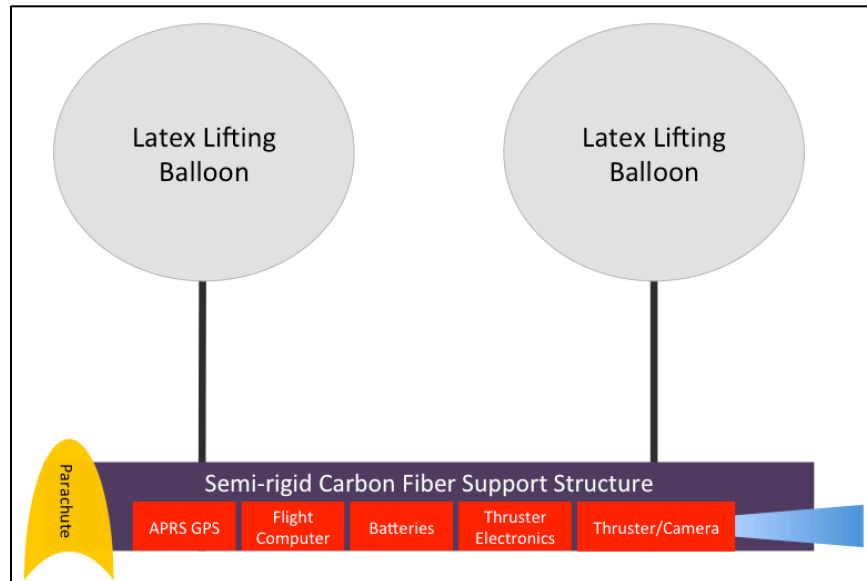
**Figure 7.13. The maximum currents measured with the peak hold Rogowski coil are shown for preflight laboratory testing (red), recorded flight data (blue), and postflight laboratory testing (green).**

### 7.3 Construction and Laboratory Calibration: 2015 Launch

The 2013 flight showed that a PPT can operate in the pressure and temperature regime of Earth's stratosphere at an altitude of ~30km. This operating regime allows for testing of in-space hardware outside the laboratory in addition to being the ideal location for atmospheric airships as this altitude is easily achievable by current day balloon technology at low-costs and is above the high-winds of the jetstream. Work completed on the atmospheric propulsion system since 2013 has focused on the creation of an airship with less drag and a high impulse thruster for propulsion.

The laboratories second atmospheric launch, held in 2015, was called the High Atmospheric Zeppelin Experiment (HAZE) and its objectives were two fold:

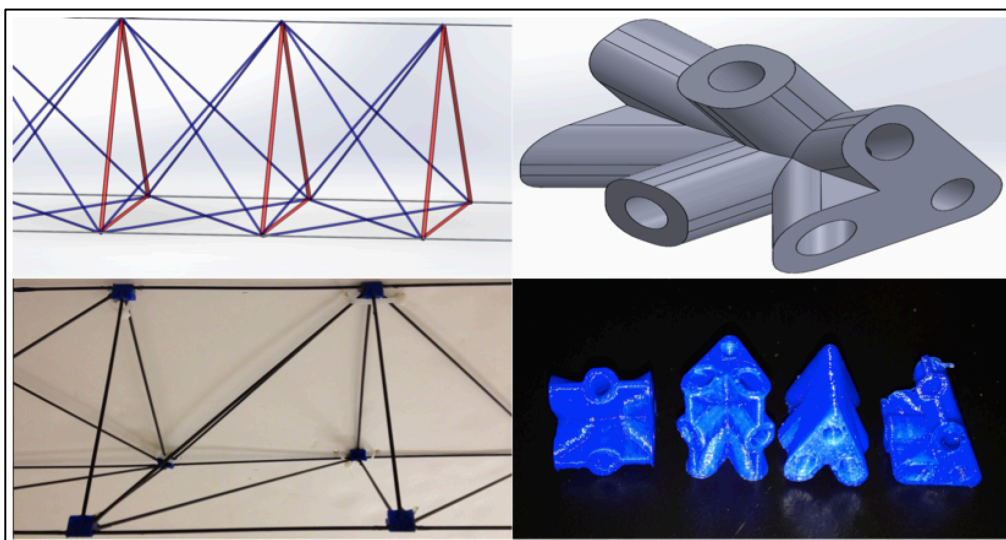
- (1) increase the thrusters impulse by optimizing the electrode geometry
- (2) show that a semi-rigid airship structure, which falls under the FAA criteria for small ballooning, can support the necessary electronics for PPT operation



**Figure 7.14. Overview of the HAZE airship structural design.**

### 7.3.1 Airship Design

The 2013 flight simply used string to attach the various thruster payloads together. To create a payload capable of station-keeping or controlled movement, a rigid or semi-rigid structure is required. As a first step towards a high-altitude airship (Fig. 7.28), 0.1” diameter carbon fiber, chosen due to its high strength/weight ratio, and 3D-printed interconnects were used to create the triangular truss structure shown in Fig. 7.15. The carbon fiber was press fit, and then epoxied into the interconnects.



**Figure 7.15. The triangular truss airship structure created from carbon fiber rods and 3D-printed interconnects.**

The structure was created in four 8ft sections in order to transport in vans and each section was connected together onsite with metal compression tubes and epoxy. The airship was completed with two 1200g latex burst balloons, where each balloon was connected to the semi-rigid structure at three locations. It was feasible that one balloon could rupture while the 2<sup>nd</sup> remained inflated, resulting in the payload slowly returning to Earth and potentially traveling hundreds of miles while in the jet stream. This necessitated a cutdown relay on each balloon, consisting of a Nickel-chrome wire wrapped around the load lines, that would be engaged once the altitude begins to decrease.

### 7.3.2 Thruster Design

Based on the results from Section 6.2, a 5cm diameter, 8cm length electrode coaxial PPT (Fig. 7.16) was chosen due to its high specific thrust and ability to operate over a wide pressure range. Based on Fig. 6.6 the thruster's specific thrust, over a range of 20-40Torr and 10-70J, varies linearly with both pressure and energy and can be described, to within 10%, by the linear equation with two dependent variables

$$S_T = -0.95E + 2P + 85 \quad 7.4$$

The electronics for the main and igniter circuit discharges was based on similar designs to the 2013 launch. A single 33 $\mu$ F, 1400V capacitor was chosen as the main discharge energy storage and was charged with a 2kV, 30W power supply. The igniter circuitry consisted of a 400V, 500mW power supply, one IGBT to switch a 400V, 3 $\mu$ F capacitor into a 3:150 turn ratio step-up, iron core transformer. Both the secondary leads of the igniter transformer were allowed to electrically float relative to the rest of the electronics. This configuration was found to allow the thruster to discharge at the highest pressures in the atmosphere. This is in opposition to vacuum operation of the thruster where the anode of the igniter wire was tied to the cathode of the main discharge to produce the most repetitive discharges.

All thruster circuitry was powered with six 12V Lithium Poly (LiPoly) batteries (Fig. 7.16), capable of providing 6000mA-hrs of current. Identical rogowski coils of 200 wraps each were placed around the main discharge anode and cathode wires, while a smaller

50-wrap rogowski coil was placed around the igniter wires on the primary side of the transformer. The output of all three of these current sensors were connected to separate peak-hold circuits, allowing the computer to record the highest voltage signal produced. Both the main and igniter energy storage capacitors had a voltage divider placed across the leads which served dual purposes, (1) allowing the computer to sample and record the capacitor voltage and (2) acting as a bleed resistor to drain the capacitor as the payload descended from apogee back down to ground where. A photoresistor peak-hold circuit powered off a 9V battery was placed inside the electrodes of the thruster to sample the light emission of the plume. The components within the three rogowski coils, both voltage dividers, and the photoresistor were sized to give a output voltage of  $<4.5V$  when the thruster was fired at its maximum discharge energy of 33J. 5V was the maximum voltage the microprocessor could sample.



**Figure 7.16. The PPT electrodes and main discharge energy storage (top left), igniter and main discharge circuitry (top center), ATMEGA1280 microprocessor (bottom left), LiPoly battery pack (bottom center), and thruster housed within a single 8ft carbon fiber truss structure (right).**

The thruster was controlled by an ATMEGA1280 microprocessor on a custom built PCB (Fig. 7.16). The computer transmitted and stored thruster and ambient atmospheric data via a 144MHz radio for real-time telemetry and onboard flash memory capable of storing 131kbytes of data. A 40in, half-wave dipole antenna was attached to the radio for flight. The computer PCB housed three balloon cutdown relay switches, a pressure sensor accurate at  $>5\text{Torr}$ , and four op-amps to isolate the thruster from the microprocessor, leaving 14 analog and 5 digital channels for thruster operation.

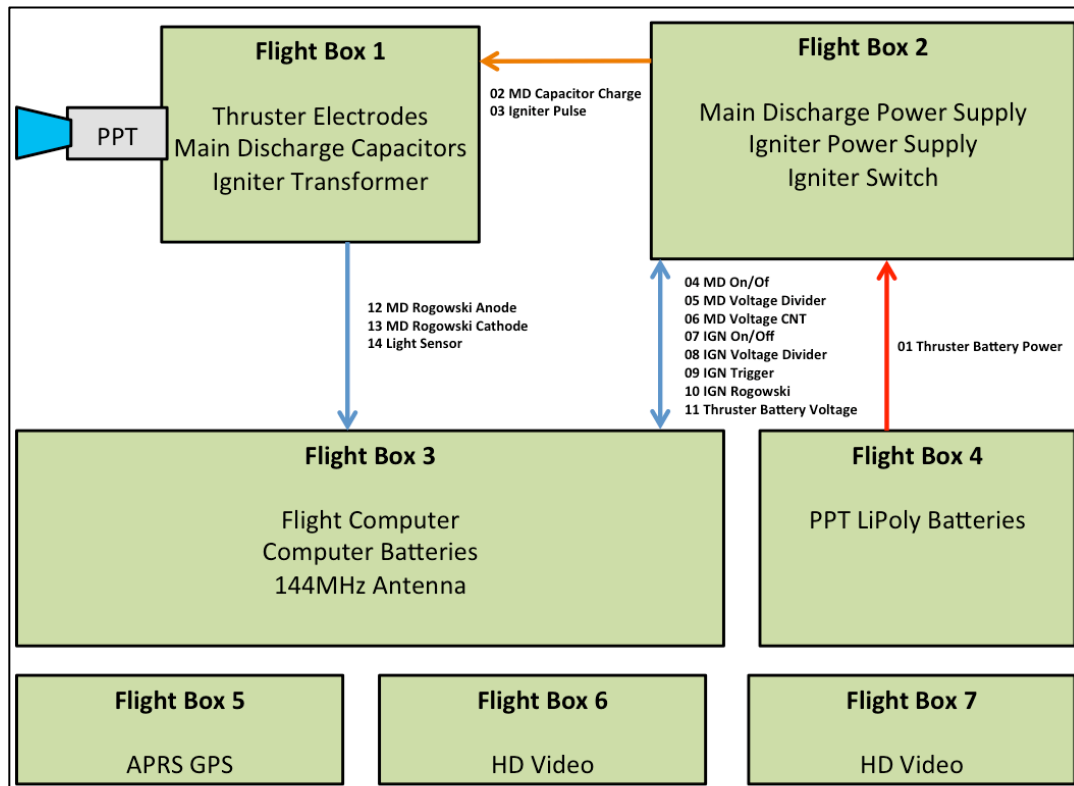
The code for the HAZE flight was a modification of the 2013 atmospheric test while having the same overall structure. The PPT was set to start attempting to fire when the pressure fell below 50Torr. While not firing, the computer samples each channel (#1-9 in Table 7.1) once and transmits the value along with the GPS position. When the pressure falls below 49Torr the igniter and main discharge power supplies turned on and the code follows the below steps:

- (1) Igniter capacitor charged to 400V
- (2) Main capacitor charged to preset voltage (initially 630V)
- (3) Sample main discharge capacitor voltage until charged
- (4) If capacitor discharges before preset value reached, reduce preset voltage 10%, and return to step (1)
- (5) Trigger igniter IGBT and sample channels 6-9 in Table 7.1 for 2ms at maximum frequency of microprocessor and store maximum value
- (6) Read igniter and main discharge capacitor voltages
- (7) If both main and igniter capacitors are <50V, count as successful thruster discharger and return to step (1)
- (8) If main capacitor did not discharge, increase preset voltage 10% (maximum preset voltage of 1400V) and return to step (1)

**Table 7.1. The 13 analog and digital inputs/outputs for PPT operation.**

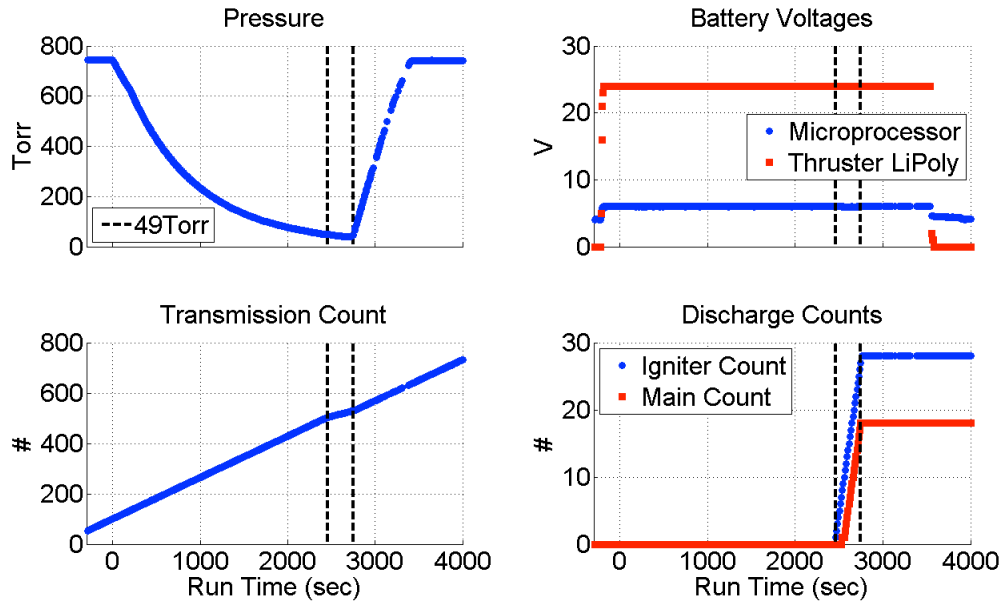
| <b>Analog Inputs (read/transmitted prior to igniter trigger)</b>                                     |                                       |
|------------------------------------------------------------------------------------------------------|---------------------------------------|
| <b>1</b>                                                                                             | Microprocessor temperature            |
| <b>2</b>                                                                                             | Ambient air temperature               |
| <b>3</b>                                                                                             | Main discharge capacitor voltage      |
| <b>4</b>                                                                                             | Igniter discharge capacitor voltage   |
| <b>5</b>                                                                                             | Battery voltage                       |
| <b>Analog Inputs (continuously sampled for 2ms after igniter trigger, highest value transmitted)</b> |                                       |
| <b>6</b>                                                                                             | Plume photoresistor                   |
| <b>7</b>                                                                                             | Main discharge anode rogowski coil    |
| <b>8</b>                                                                                             | Main discharge cathode rogowski coil  |
| <b>9</b>                                                                                             | Igniter discharge anode rogowski coil |
| <b>Digital Outputs</b>                                                                               |                                       |
| <b>10</b>                                                                                            | Igniter discharge power supply on/off |
| <b>11</b>                                                                                            | Igniter discharge IGBT trigger        |
| <b>12</b>                                                                                            | Main discharge power supply on/off    |
| <b>13</b>                                                                                            | Main discharge preset voltage         |

Two high-definition video cameras capable of storing 3hr of 720p video footage each were included on the flight. One was positioned directly in front of the thruster in an attempt to capture the light emission of the plasma plume. In addition to the 144MHz telemetry system which included GPS position, an APRS (automatic packet reporting system) transmitter was included on the flight. The APRS was a secondary GPS system and transmitted via the HAM radio network to the internet in real-time.



**Figure 7.17. Overview of the HAZE airship PPT electrical system.**

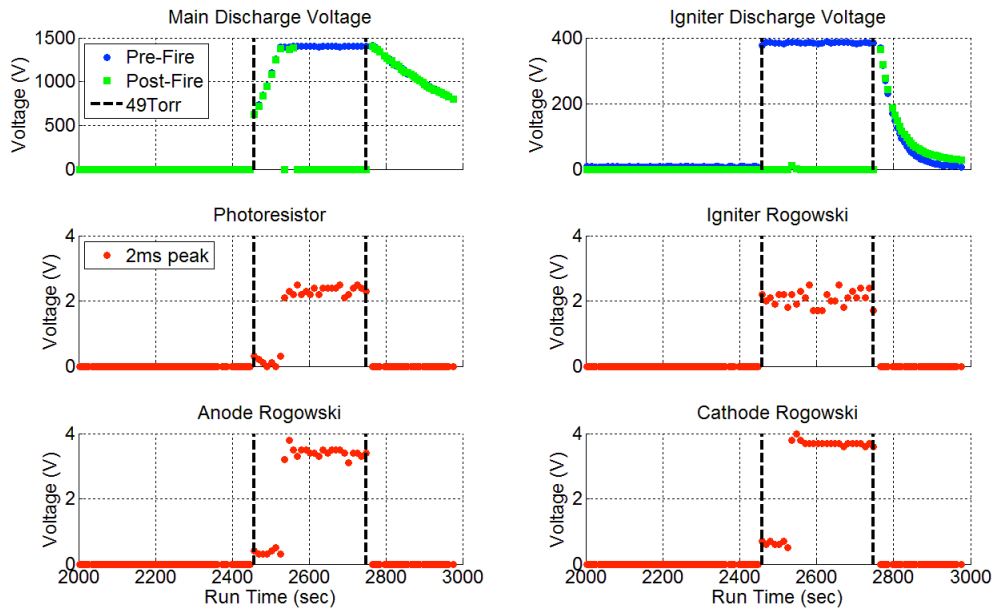
A number of tests were run with the thruster in the laboratories large vacuum chamber where the PPT was housed within a single carbon fiber truss section and electrically isolated from the chamber and building power. The computer relayed data via its 144MHz radio through the windows on the chamber in real-time. The results shown below are for the last laboratory test held two days before the launch and lasted for an hour from pump-down to up-to-air with 650 telemetry transmissions over the duration of the test.



**Figure 7.18. Vacuum chamber pressure (top left), microprocessor board and PPT battery voltages (top right), telemetry transmission count (bottom left), and thruster igniter and main discharge counts (bottom right).**

The chamber pressure was below 49Torr, the maximum pressure where the thruster attempted to fire, for 5min, reaching a minimum of 39Torr. The main discharge power supply initially charges the capacitor to 630V, however is unable to fire until a pressure of 43Torr and voltage of 1.4kV is reached. During this test, the igniter discharged all 28 times it was attempted, while the main discharged 18, all at 1.4kV. The igniter and main capacitor voltages pre and post fire can be seen in Fig. 7.19, while the discharge counts are shown in Fig. 7.18.

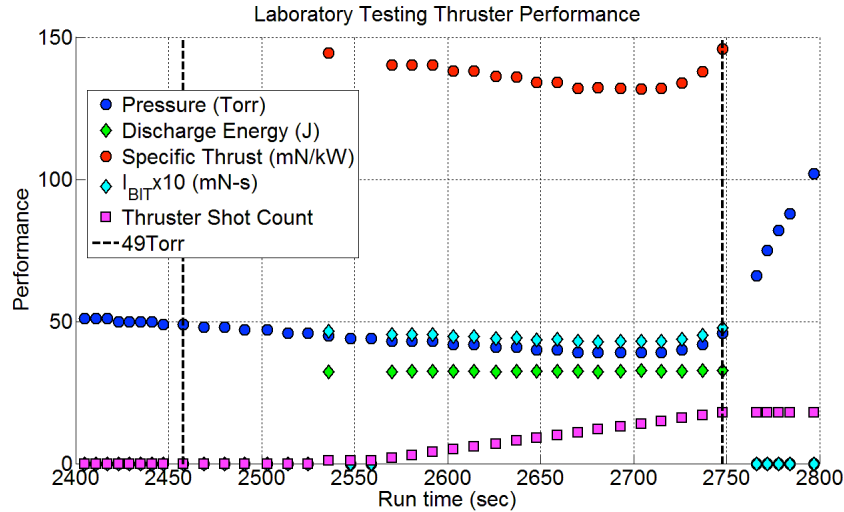
Inferring the specific thrust of the flight PPT based on laboratory results requires knowledge that operation was similar between the two environments. This was done by measuring the main discharge current peak, igniter discharge current peak, and light emission of the thruster (shown in Fig. 7.19 for the laboratory and 7.25 for the flight).



**Figure 7.19. Laboratory testing thruster sensors: main discharge pre/post voltages (top left), igniter discharge pre/post voltages (top right), photoresistor peak output (center left), igniter anode rogowski coil peak signal (center right), main discharge anode (bottom left) and cathode (bottom right) rogowski coil peak signals.**

The igniter circuitry caused 0.1-0.3V of noise on the photoresistor signal, while the 1.4kV PPT discharge produced a signal of 2.0-2.5V, with an average of 2.3V as shown in Fig. 7.19. There was no signal while the thruster wasn't firing as the room lights were turned down so that very little ambient light entered the vacuum chamber. The main discharge anode rogowski coil had 0.3-0.5V of noise from the igniter discharge, and a signal of 3.1-3.8V, while the cathode rogowski coil had 0.5-0.7V of igniter noise and a signal of 3.6-4.0V. The current through the anode and cathode wires should be same and although the rogowski coils were made identical, small differences in the size of the rogowski loop could cause the 10% variation. The igniter discharge rogowski coil varied between 1.7-2.2V when firing without the main, and 1.7-2.5V with the main discharge. Assuming that the specific thrust relationship with pressure and energy are the same as Eq. 7.4, the thrusters performance was calculated and is shown in Fig. 7.20. The laboratory test set a baseline for the results gathered during the flight in Section 7.4.

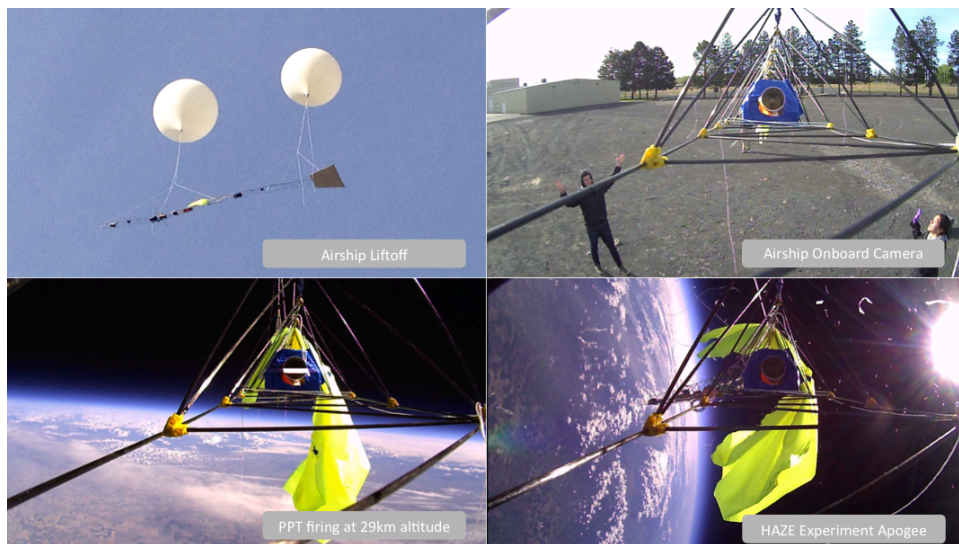




**Figure 7.20. Overview of the thrusters pressure, discharge energy, specific thrust, impulse bit, and shot count from the laboratory test.**

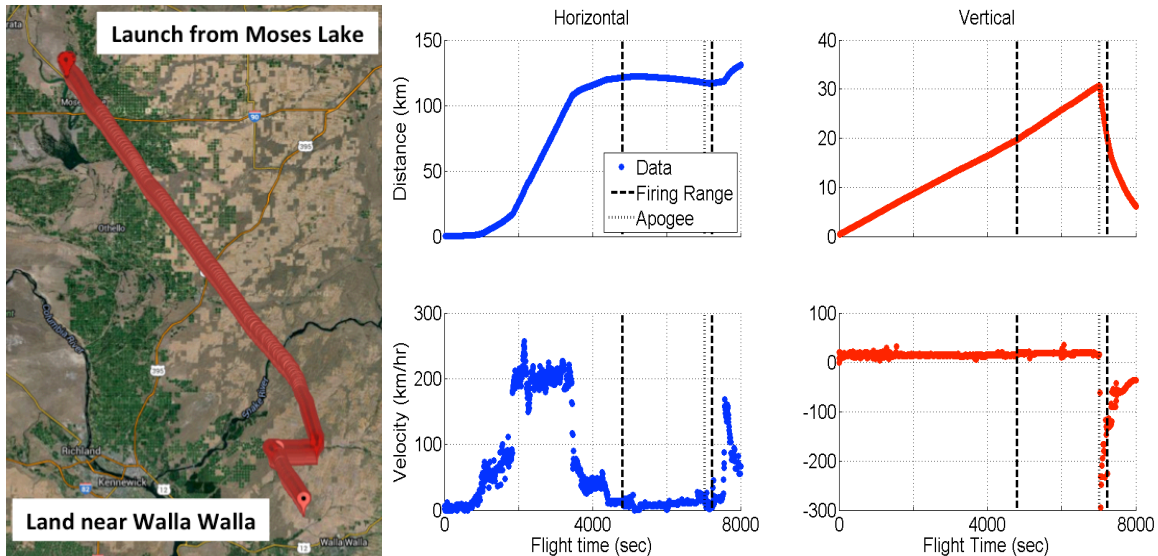
#### 7.4 Atmospheric Zeppelin Airship: 2015 HAZE Mission

The HAZE airship was launched from the Grant County International Airport in Moses Lake, WA on April 26, 2015. The airship rose to an altitude of 30.7km in 90min and took another 30min to return to the ground, landing ~10km NW of Walla Walla, WA. Onboard video shows the airship undergoing considerable stress while rising through the jetstream and yet remained intact until after balloon burst at apogee. It's believed that the shock wave from the first balloon bursting caused the 2<sup>nd</sup> to rupture at the same time, negating the need for the balloon cutdown mechanism.



**Figure 7.21. Images of the HAZE flight of liftoff (top), the PPT firing at 29km altitude (bottom left), and immediately after balloon burst (bottom right).**

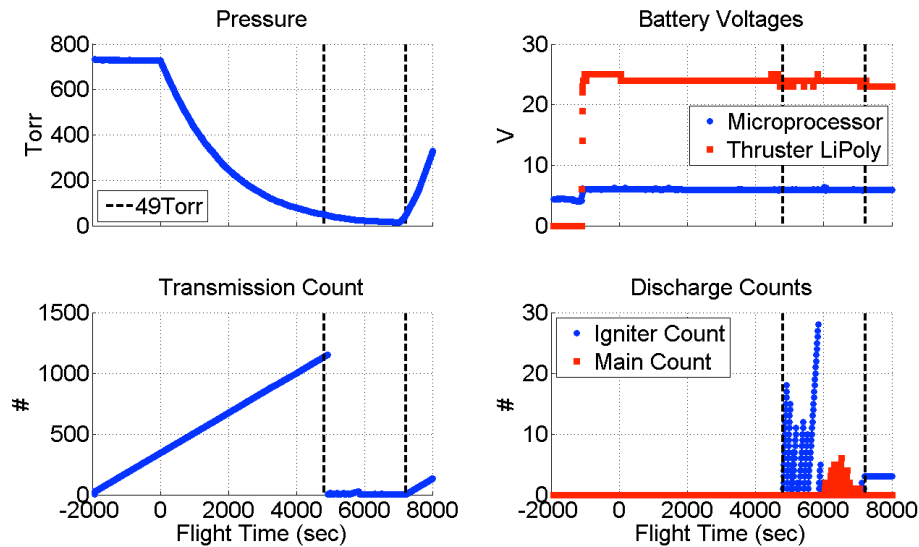
The airship traveled a total of 133km from Moses Lake, reaching a ground speed of 200km/hr while rising through the jet stream. It had a constant vertical velocity of 10km/hr while rising, and initially fell at  $\sim 300$ km/hr before impacting the ground at  $\sim 10$ km/hr. The horizontal distance and velocity were much higher than the 2013 flight as the jet stream was stronger, while the vertical results were similar to what was seen in 2013 with a single balloon.



**Figure 7.22. GPS track of the airship flight (left), ground distance from Moses Lake (top center), ground velocity (bottom center), vertical altitude (top right), vertical velocity (bottom right).**

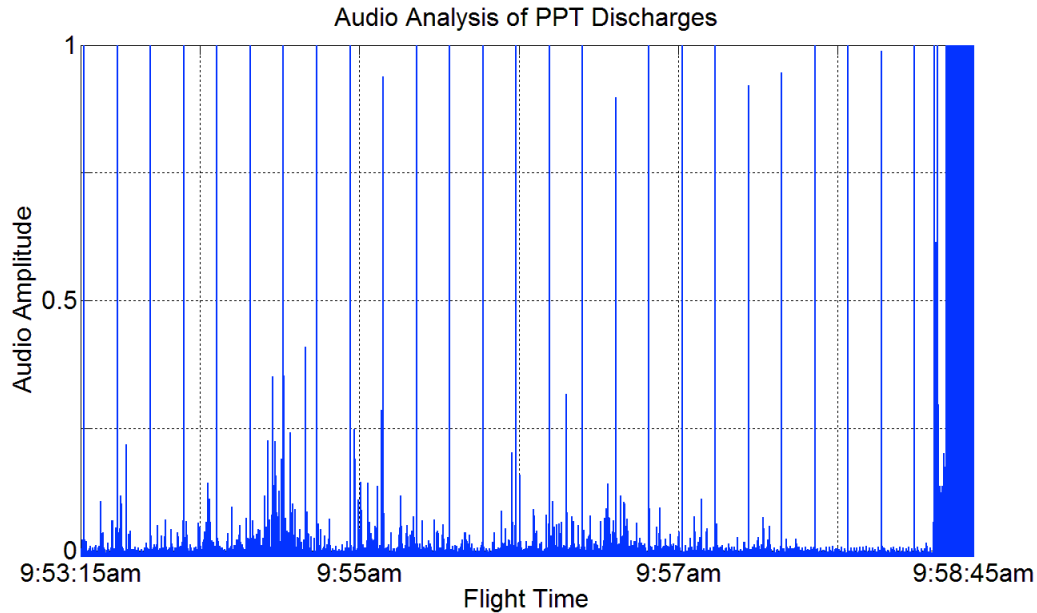
The payload was located after landing and the onboard memory storage allowed for all data to be recovered. The payload reached 49Torr in 80min and spent 40min at lower pressures. As can be seen in Figs. 7.23 & 7.25 there were two major differences between the laboratory test in Section 7.3 and the flight. First that the thruster didn't begin firing until a pressure of 23Torr, unlike the 43Torr in the laboratory test, and secondly that the ATMEGA1280 microprocessor reset multiple times during the test. There were 66 resets in total. The reasons for both of these are unknown, however it's believed that the cold temperature (typically  $-30$ - $50^{\circ}\text{C}$  at 25-30km altitude) could be the cause as it's the only variable that could not be replicated in the lab. The cold lowers wire resistance, allowing for higher currents to flow into the power supplies and microprocessor. Better insulation around the electronics and more temperature sensors could help exactly identify and fix these issues for future flights. Another possibility is an unintended electrical connection

between the thruster and the microprocessor. While charging, the thrusters DC-DC converters pull 30W (24V, 1.2A) from the battery pack for 3-4 seconds. Shown in Fig. 7.23, the microprocessor bus voltage increases from 4.7 to 6.2V when the PPT LiPoly batteries are turned on. The cause of this increase is unknown, however it was found during both the laboratory testing and during the flight and may be a result of a shifting common electrical plane between the two separate battery packs.



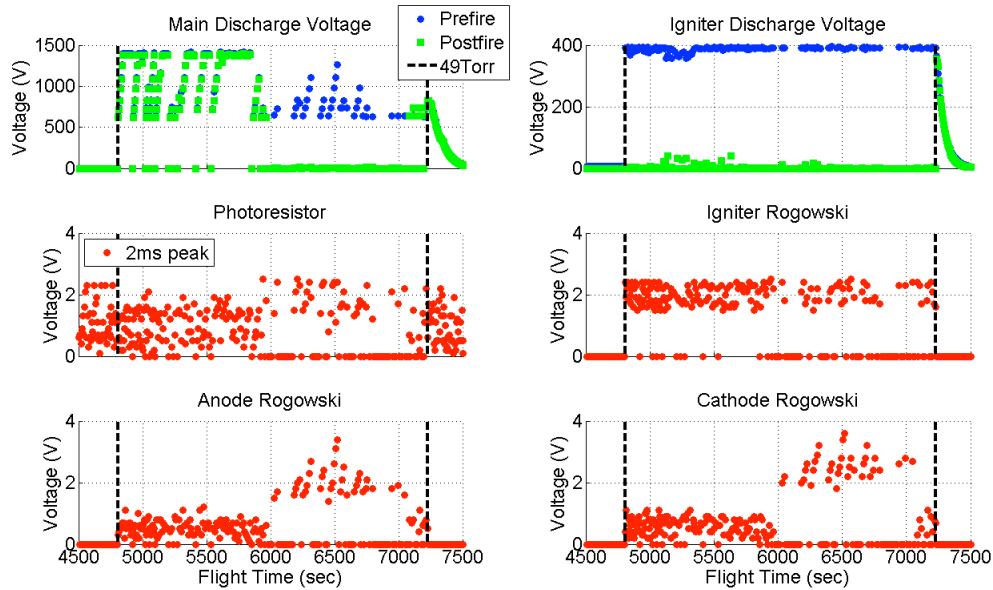
**Figure 7.23. Pressure (top left), ATMEGA1280/PPT supply voltages (top right), transmission count (bottom left), and igniter/main discharges (bottom right).**

The igniter successfully discharged all 228 times it was attempted over the 40min duration at pressures lower than 50Torr. Once the thrusters main discharge began firing at 23Torr, 80 discharges were completed without a thruster misfire, even with microprocessor resets. Each microprocessor reset caused the thrusters main discharge voltage to be reduced to 630V, and consecutive successful discharges without a board reset resulted in a 10% increase in voltage. The shot count was confirmed by visual light emission on the HD video as well as through the audio of the camera, which is plotted in Fig. 7.24 for the 5min leading up to the balloon burst at 9:58:30am. A firing frequency of 10-14 seconds was confirmed by both the telemetry and camera.



**Figure 7.24. Audio amplitude from the onboard video cameras showing the PPT discharges leading up to the balloon burst for the 5min prior to apogee.**

To compare the operation of the thruster between the flight and laboratory, four sensors, three rogowski coils and a photoresistor for light emission, were used. The photoresistor signal varied between 0.1-2.3V with an average of 0.9V when the thruster wasn't firing due to being pointed the sun or picking up the reflected light emission of the Earth. This same range was found with the igniter discharges as well as full thruster firing. However, with the full thruster, the photoresistors output signal had a minimum value of 0.8V and an average of 1.8V, double that of the non-firing signal. The photoresistors signal while firing is a combination of the background ambient light and the thrusters light emission. The lack of low output signals from the photoresistor is a result of the thrusters light emission. The igniter rogowski signal had an average of 2.1V with a maximum of 2.5V, similar to the laboratory test, suggesting that the igniter discharge performed similarly between the flight the laboratory. The main discharge anode rogowski coil had a noise value of 0.6V while the igniter fired without the main, and varied between 1.4-3.4V with the full thruster. The cathode rogowski coil had a noise average of 0.6V with the igniter and varied between 1.8-3.6V with the full thruster.



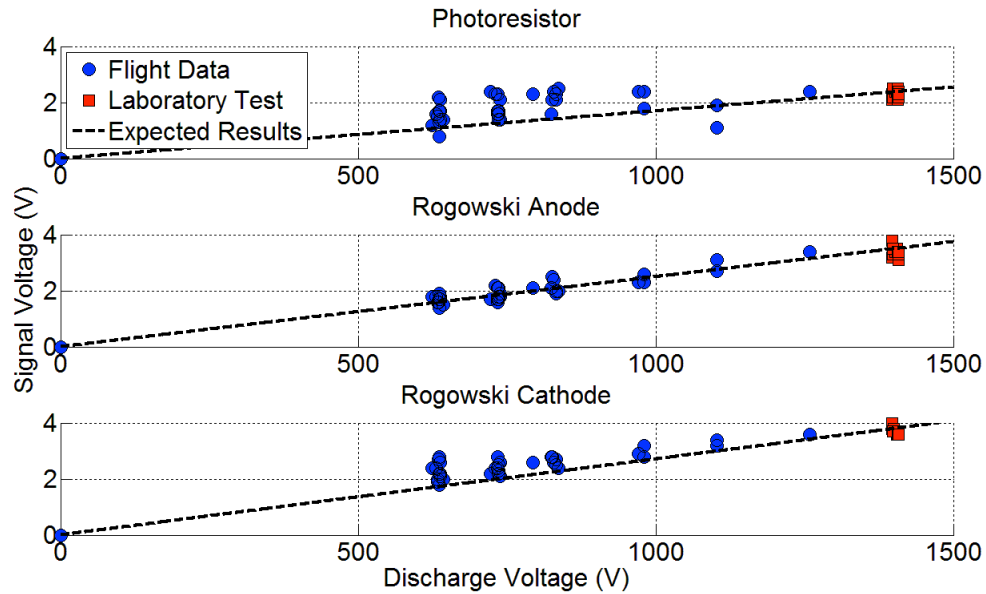
**Figure 7.25. Thruster sensor data: main discharge pre/post voltages (top left), igniter discharge pre/post voltages (top right), photoresistor peak output (center left), igniter rogowski coil peak signal (center right), main discharge anode (bottom left) and cathode (bottom right) rogowski coil peak signals.**

**Table 7.2. Average values of the four thruster sensors for laboratory and flight during periods of non-firing, igniter discharge, and full thruster operation.**

|                                 |        | Ambient | Igniter Discharge | Full Thruster Discharge |
|---------------------------------|--------|---------|-------------------|-------------------------|
| Photoresistor Light Sensor      | Lab    | 0       | 0.2               | 2.3                     |
|                                 | Flight | 0.9     | 1.0               | 1.8                     |
| Main Discharge Anode Rogowski   | Lab    | 0       | 0.4               | 3.4                     |
|                                 | Flight | 0       | 0.6               | 2                       |
| Main Discharge Cathode Rogowski | Lab    | 0       | 0.6               | 3.7                     |
|                                 | Flight | 0       | 0.6               | 2.5                     |
| Igniter Discharge Rogowski      | Lab    | 0       | 2.1               | 2.1                     |
|                                 | Flight | 0       | 2.0               | 2.1                     |

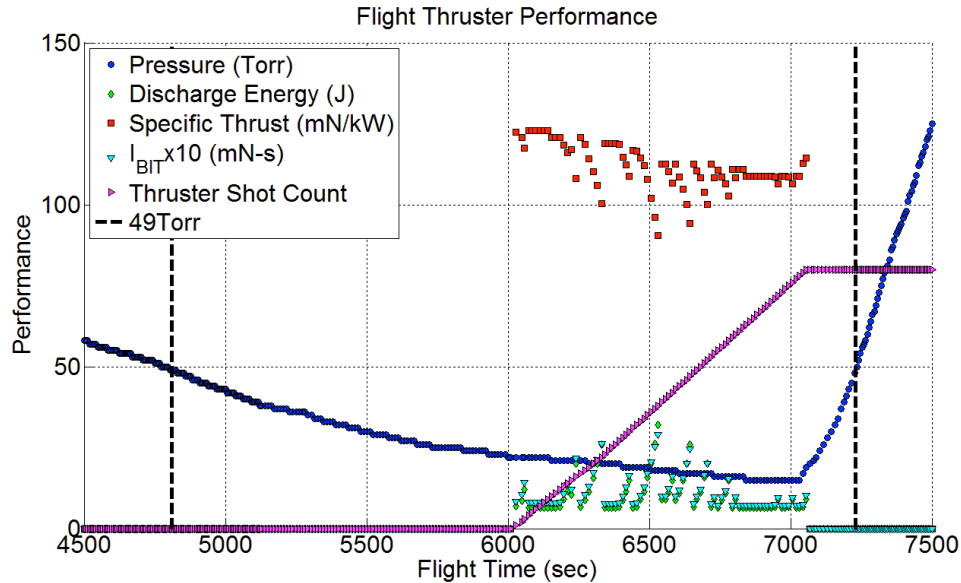
This increase in the signal to noise ratio on the photoresistor, main discharge anode rogowski, and main discharge cathode rogowski coil with the full thruster are all indications that the main discharge fired. Laboratory testing was completed entirely at 1.4kV discharges, while the flight had discharge voltages ranging between 600-1300V due to the microprocessor resets. Figure 7.26 plots the photoresistor, main discharge anode rogowski, and main discharge cathode rogowski coils as a function of discharge voltage for both the flight and laboratory testing. Eq. 1.6 suggests that the main discharge current peak should increase linearly with voltage, while results in Chapter 5 suggest the

photoresistor peak signal should increase linearly. The rogowski coil signals show a linear relationship between the flight and laboratory suggesting that the current flowing through the main discharge was similar between the flight and the laboratory. The photoresistor flight data is higher than what would be expected based on the 2.4V signal during laboratory testing. This is attributed to the ambient light contribution from the sun and Earth for some shots.



**Figure 7.26. Photoresistor, anode rogowski, and cathode rogowski coil peak output signals for the flight and laboratory testing. The expected results based on laboratory testing are shown in black.**

The photoresistor, anode rogowski, and cathode rogowski coil agreement to the recorded values in the laboratory test implies that the specific thrust relationship in Eq. 7.4 is a valid approximation for the performance of the thruster onboard the airship. The performance of the thruster is plotted in Fig. 7.27, along with the ambient pressure and discharge energy for all successful firings of the thruster over the course of the flight.

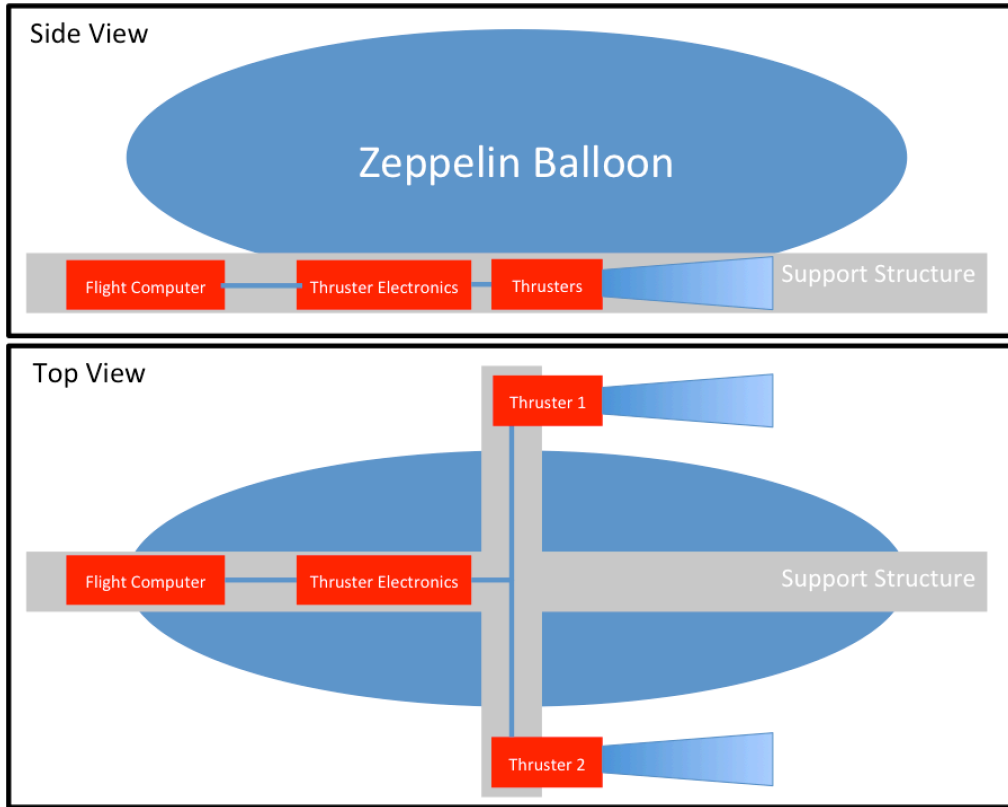


**Figure 7.27. Overview of the thrusters pressure, discharge energy, specific thrust, impulse bit and shot count for the atmospheric flight.**

The 2015 airship flight showed that a high efficiency (through larger diameter electrodes) PPT can operate in the atmosphere at altitudes above the jetstream. A light-weight, semirigid, carbon fiber and 3D printer truss structure was tested and shown to withstand the high forces of the lower atmosphere. Keeping the mass of this structure below FAA requirements for permitting allows for inexpensive testing of the concept. Future work should look to increase the power of the main discharge electronics system to allow for a higher firing frequency of the thruster, better protect the microprocessor from thruster EM noise, and work towards a full zeppelin airship capable of long duration testing at stratospheric altitudes.

## 7.5 Future Airship Direction

The 2013 and 2015 flights showed that a pulsed plasma thruster can operate in the pressure and temperature regime of Earth's stratosphere. A semirigid airship with mass under the FAA required 12lb limit was built for the 2015 flight and proved successful at staying intact while rising through the jetstream and firing at altitude. The creation of a zeppelin airship capable of long duration operation at high-altitude is the next logical step for this research. This can be accomplished either through the use of a valve/ballast system on a latex balloon or through an elongated mylar balloon. Steering capability for the airship could be given via a thruster mounted off-axis of the airship.



**Figure 7.28. A semirigid zeppelin airship design with a mylar balloon, carbon fiber triangular truss structure, and two pulsed plasma thrusters mounted off-axis for steering capability.**

The necessary helium required for such an airship can be calculated by balancing the buoyancy and gravitational forces. The buoyancy force

$$F_b = \rho g V_{dis} \quad 7.5$$

is a function of the fluid density the object is moving through ( $\rho$ ) and the volume of displaced air ( $V_{dis}$ ), while the gravitational force

$$F_g = gm \quad 7.6$$

is simply a function of the objects mass ( $m$ ). Setting them equal results in

$$\rho V_{dis} = m \quad 7.7$$

The average density, pressure, temperatures at Moses Lake in the months of April and May are  $1.225\text{kg/m}^3$ ,  $101.2\text{kPa}$ , and  $287.1\text{K}$ , respectively.  $0.017\text{ kg/m}^3$ ,  $1.2\text{kPa}$ , and  $226\text{K}$

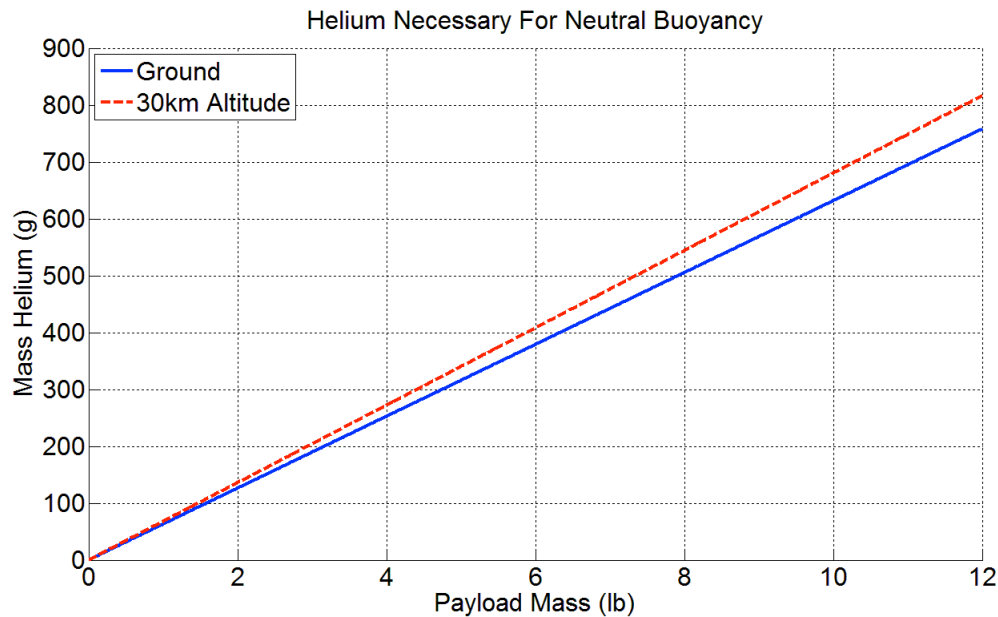


were assumed for an altitude of 30km. For a 12lb payload, these values lead to a balloon volume of 4.5m<sup>3</sup> on the ground and 320m<sup>3</sup> at altitude.

The volume and temperature of helium changes from the ground to 30km altitude, and as such the ideal gas law (Eq. 4.7) can determine the necessary helium mass to hover the airship. The number of moles of a gas is related to the mass (m) through the molar mass (M).

$$m = \frac{PVM}{RT} \quad 7.8$$

The molar mass of helium is 4.003g/mol, resulting in a necessary helium mass of 815g to balance a 12lb payload mass at a 30km altitude. A typically K-type bottle of helium has a ground volume of 250m<sup>3</sup> and helium mass of 1160g, implying that 70% of one bottle is necessary to balance a 12lb payload.



**Figure 7.29. The necessary helium mass to for neutral buoyancy of payload masses up to 12lbs at the ground and an altitude of 30km based on launching from Moses Lake, WA in April or May.**

## **Chapter 8 CubeSat Pulsed Plasma Thruster Design**

Our lab began actively researching CubeSat components in 2013. This was partially to give students hands-on experience with spacecraft systems, as well as to develop a pipeline to move research experiments from the laboratory to flight. The particular focus of my work has been to modify our existing Pulsed Plasma Thruster technology to function autonomously onboard a CubeSat satellite. The lab is actively researching, and funded for, CubeSat missions around Jupiter as part of the Europa Clipper mission [90] and CubeSat operation in lunar orbit as part of NASAs Lunar CubeQuest Centennial Challenge [91].

### **8.1 CubeSat Introduction**

Miniature satellites offer universities the ability to bring experimental engineering out of the laboratory and into a testing environment at relatively low cost. These missions allow for practical, hands-on experience for the teaching of spacecraft design and operation. CubeSat missions have typically been defined by single-instrument science, testing new technologies, low-rate communications, and low power, data, and lifetime needs.

The standardized CubeSat design can be traced back only 15 years to 1999 when Professors Bob Twiggs and Jordi Puig-Suari developed the concept [92, 93]. A CubeSat is a 10x10x10cm satellite, or multiples thereof, generally weighing 1kg per 1U cube. The standardization of the satellite dimensions allowed for a standardized launcher interface, called the Poly-Picosatellite Orbital Deployer (P-POD). The P-POD launcher allowed for decoupling the development of the satellite from the launch vehicle, greatly decreasing the cost and timeline of the mission. The first CubeSats were launched in 2003, the 100<sup>th</sup> in 2012, and currently over 270 missions have been flown.



**Figure 8.1. The University of Aalborg 1U CubeSat, Planet Labs Flock-1, and a proposed 6U CubeSat for interplanetary travel by University of Michigan (left to right).**

The first university built CubeSat launched from the standard P-POD launch was from the University of Aalborg in 2003 [94]. Their AAU CubeSat-1 had a 1U form factor and survived 2.5 months until an antenna failure [95]. Since then, the industry has flourished. Predominantly, Planet Labs, a private company based out of San Francisco, has launched over 100 3U satellites for Earth imaging [96]. Future plans include concepts such as propulsion for interplanetary travel, currently being developed in a 6U structure by the University of Michigan [39]. These three CubeSats are shown in Fig. 8.1.

The most accurate history of CubeSat missions comes from Michael Swartwout of Saint Louis University [97]. He claims that the CubeSat industry appears to be entering its third phase of operation. Phase one saw an average of 5 launches per year between 2000-2005, phase two had 13 launches, while the past three years have had an average of 80 launches per year. This third phase in particular has been defined by a large increase in the number of government and private satellites as well a shift from 1U to 3U form factors. Swartwout also shows that Universities in particular have had a challenging time building successful first satellites, however their success rate on second launches is on par with industry. The two large conclusions that Swartwout draws are that as long as new programs build new CubeSats, failure rates will be high and that the general trend is to go larger and more complex, the exact opposite of what initially started the CubeSat interest.

Due to a current day lack of propulsive capabilities, CubeSat missions are confined to their dispersal orbits. In LEO this not only limits the science capability of these satellites, but the lack of station keeping ability sets a hard timeline on mission duration. A 1U CubeSat in an initial 400km altitude orbit (assumed to be launched from the ISS) will have a nominal life of ~6 months. Including a small propulsion unit (total impulse of 40Ns) can increase the lifetime to 15 months [98]. The standard CubeSat deployment method is from a P-POD launcher. These launchers use a coiled spring to release the satellite at a relative velocity of 1.5m/s. Over an arbitrary three-month mission duration, this constant velocity will result in a separation distance of  $10^4$ km. As long as CubeSat missions are limited to LEO, this won't be a concern for communication with ground stations, however this distance will create drastic data transmission limitations for missions outside of Earth orbit as CubeSats are highly power limited and monopole antennas have an  $r^2$  dependency.

CubeSat propulsion is less advanced than other nanosatellite technologies due to, in part, the difficulties in miniaturizing conventional thruster designs. Common CubeSat subsystems, such as microprocessors, communication, and energy storage have seen drastic advancements in miniaturization in recent years due to their use in common, commercially available products as part of the gaming, computing, and mobile device industries. In-space propulsion systems do not enjoy wide spread commercial use with the general public, nor do they scale effectively due to their inherent requirements of high voltage, high current, and/or electrode spacing.

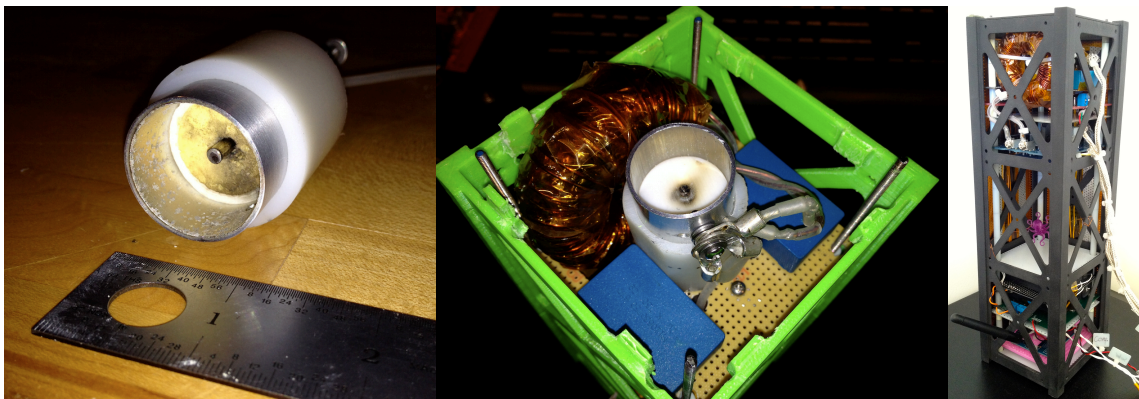
Chemical propulsion systems, using mono- or bipropellant rockets, exist or are in development with thrusts of <1N and small  $\Delta V$  capabilities of <100m/s [99, 100, 101]. Solid propellant won't be appropriate for these satellites due to their one-time use. Cold-gas, Butane, and Hydrazine have been the common propellants to date [102]. For moderate  $\Delta V$ s (0.1-1km/s) electric propulsion is needed. A wide variety of electric thrusters are in development, however, to date, none have flown [39, 103, 104]. Work has been completed to scale down Ion and Hall thrusters, while arc jets and Pulsed Plasma Thrusters can be easily operated at low power levels.

The overall mission objective will determine the specific propulsion system required. We believe that the sulfur PPT can find a niche in this market for missions requiring the longevity of electric propulsion in addition to higher acceleration rates, albeit at higher mass consumption rates.

## 8.2 Thruster Design and Satellite Integration

The Advanced Propulsion Laboratory began working solid fuel plasma sources in 2011 as a means to provide the propellant for the High Power Helicon Thruster (Chapter 4). Increased performance was found for the higher energy (40-100J) discharges. In 2012 effort was placed into using these plasma sources as thrusters in their own right to test varying propellants types, creating the traditional Pulsed Plasma Thruster with nontraditional propellants. This involved a slight reduction to the size of the main discharge capacitors to energy levels of 10-60 J. 2013 saw the first attempt to move a thruster out of the laboratory with the start of our atmospheric testing (Chapter 7), which was completed at energy levels of 5-30J, involved an onboard autonomous computer, a primary battery for power, and a large reduction and simplification to the thrusters electronics. Modifying the technology of the atmospheric Pulsed Plasma Thruster for autonomous CubeSat operation required further reductions to the size, mass, and power of the device, as well as integration with other satellite subsystems. The section below details the work that has been completed in that regard.

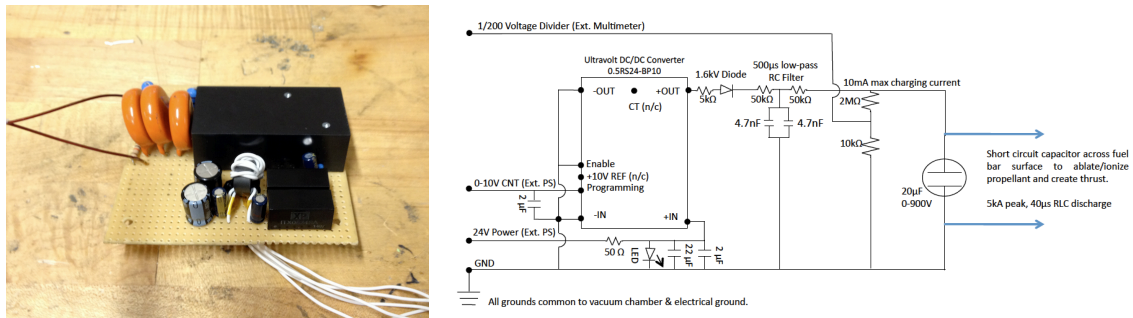
### 8.2.1 Pulsed Plasma Thruster



**Figure 8.2.** The sulfur PPT (left) integrated into PLA (center) and windform (right) CubeSat frames.

The CubeSat PPT detailed here was created from aluminum electrodes and uses a solid Sulfur propellant rod. Sulfur has been shown in the laboratory to have a specific thrust, mass ablation rate, and ionization fraction, twice that of Teflon (Chapter 5). The discharge chamber, formed by the outer electrode (cathode) is 2.5cm in diameter and 1cm in height. Nylon was machined to house the thruster and interface with the satellite. The thruster was designed to be operated autonomously from an off-the-shelf Arduino Uno microprocessor.

The main discharge circuit consists of four primary components, a power supply, low-pass filter, voltage divider, and energy storage capacitor. A 10W, 1kV power supply from Ultravolt was chosen. The supply takes an input voltage of 24V and has three output pins (-500V, 0V, and +500V). The -500V pin was grounded, resulting in a 500-1kV output. Low pass RC filters as well as ferrite beads in a common core configuration were included to protect against EM noise and voltage spikes. A voltage divider converts the 1kV capacitor voltage to a 5V signal to be read by the microprocessor. Two 2 $\mu$ F, 1kV, capacitors were chosen for the main discharge energy storage. The leads of the capacitor were directly attached to the PPT electrodes.

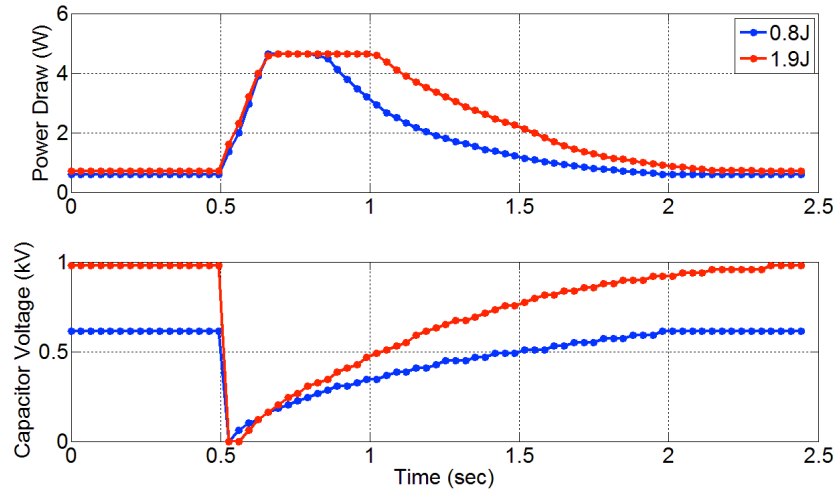


**Figure 8.3. PPT main discharge electronics and circuit diagram.**

The power consumed by the main discharge circuit over a 2.5s span with a capacitor discharge at 0.5s is shown in the top panel of Fig. 8.4 for initial capacitor energies of 0.8 and 1.9J, while the measured capacitor voltages for the same discharges are shown in the lower panel. The power draw when the capacitors are fully charged is  $0.7 \pm 0.1$ W, the variance depending on the set capacitor energy, while the peak power draw is 4.2W when initially charging. The capacitors take  $\sim 2$ sec to fully charge, limiting the firing frequency with this electrical setup to 0.5Hz. For any firing frequency slower than 0.5Hz, the power

draw of the main discharge circuit is dependent on the firing frequency and capacitor energy:

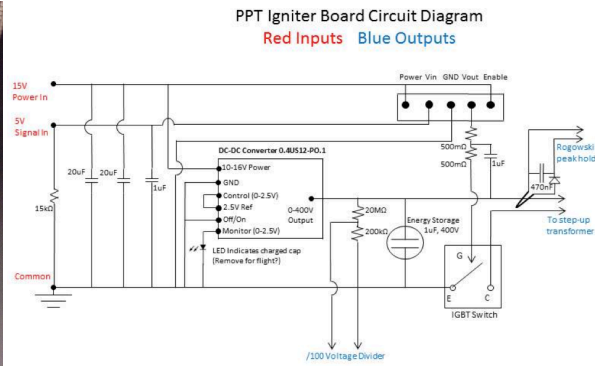
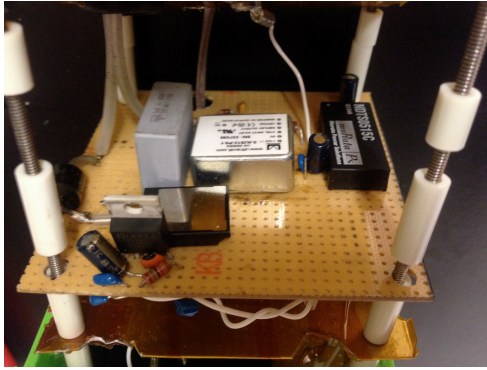
$$P_{MD}[W] = (1.15E_c + 1.1)f + 0.7 \quad 8.1a$$



**Figure 8.4. The power draw (top) and voltage (lower) for the main discharge electronics for energies of 0.8J (blue) and 1.9J (red) for a single discharge.**

The PPT requires an igniter circuit to initiate the discharge. The circuit diagram and current status of the board are shown below. The igniter board is powered with 15V, which is the maximum voltage of the DC-DC converter and the minimum voltage for the IGBT switch. The DC-DC converter steps up the voltage to 400V and is placed in parallel with a 1 $\mu$ F energy storage capacitor, taking ~400ms to full charge. The IGBT switch requires a driver, which receives a 5V triggering voltage. When triggered, the IGBT connects the energy storage capacitor with the primary side of a 3:150 transformer, stepping up the voltage to a 20kV, <8J arc, which occurs within the anode and cathode electrodes of the PPT. This arc creates the initial seed plasma to complete the circuit between the anode and cathode of the main discharge, allowing the PPT to fire.

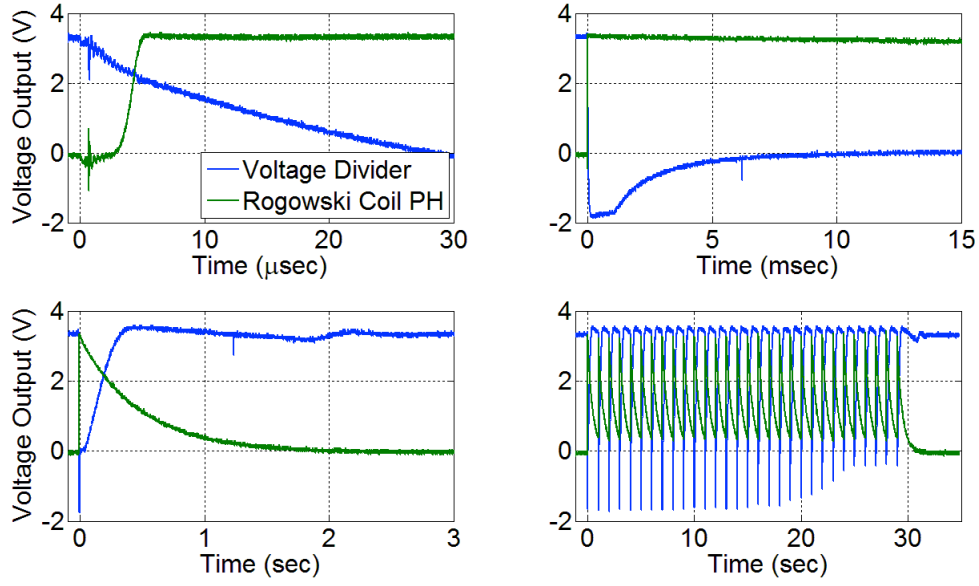




**Figure 8.5. The igniter discharge electronics mounted onto the CubeSat rail system (left) and corresponding circuit diagram (right).**

A voltage divider reduces the 400V capacitor voltage to 5V that can be read by the microprocessor. A rogowski coil was included to measure the current on the transformer primary (400A peak), the output of which as fed into a peak hold circuit to be sampled by the microprocessor. Figure 8.6 shows the voltage divider and rogowski peak-hold signals for a single igniter pulse over three different time frames, measured by a laboratory oscilloscope. The igniter was triggered at 0sec and oscilloscope traces were captured for 30 $\mu$ s, 15ms, and 3s. The capacitor voltage is recorded as 0V between 10-50msec, while the rogowski peak-hold maintains a voltage within 10% of the maximum value for 20ms, allowing for sampling by the microprocessor. Firing the igniter circuit for 30 pulses at a frequency of 1Hz shows that the voltage divider and rogowski output signals remain relatively constant and uniform from shot to shot in terms of amplitude, charge time, and discharge time.



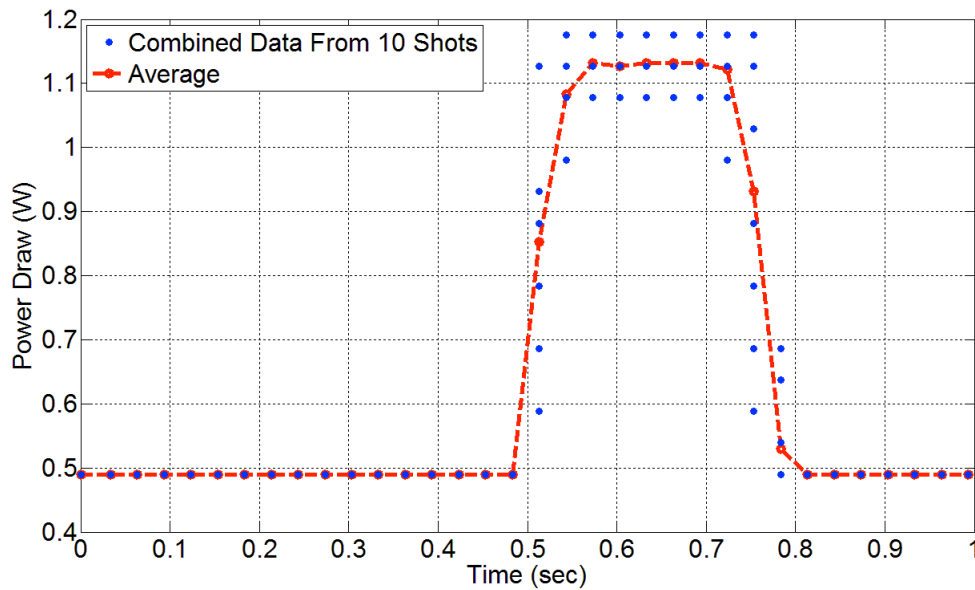


**Figure 8.6.** The voltage divider (blue) and rogowski peak-hold (green) signals for a single igniter pulse over time spans of 30 $\mu$ s (top left), 15ms (top right), and 3s (lower left), as well as 1Hz discharges for 30 pulses (lower right).

The igniter circuit was connected to a laboratory, wall-plug power supply, in order to the measure the power consumed. The data from 10 shots, taken at a frequency of 1Hz, was collected and is shown in Fig. 8.7. In this data run, the igniter was triggered at 0.5s into each pulse. The igniter board pulls 0.5W when not firing and a peak of 1.2W when firing and recharging the capacitor, for an average power draw of 0.7W at 1Hz. For any firing frequency slower than 1Hz, the power that the igniter circuit will pull from the batteries while functioning is shown in Eq. 8.1b. Combining Eq. 8.1a and 8.1b, results in an expression for the total power input to the PPT as a function of main capacitor discharge energy and firing frequency, which will be the same for both the igniter and main discharges.

$$P_{IGN}[W] = 0.2f + 0.5 \quad 8.1b$$

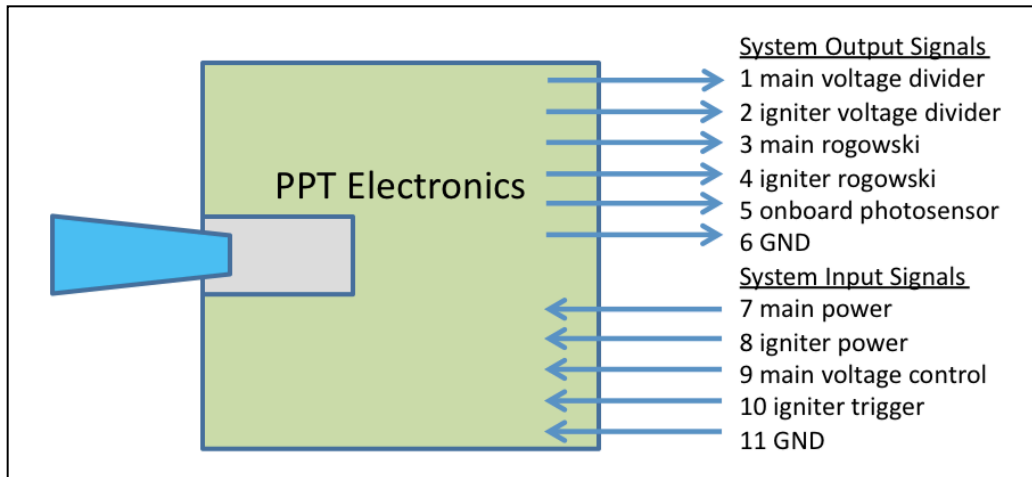
$$P_{PPT}[W] = (1.1E_c + 1.3)f + 1.2 \quad 8.2$$



**Figure 8.7. The power consumed by the igniter circuit over 1sec. The results from 10 data runs are shown (blue dots), as well as the average (red line).**

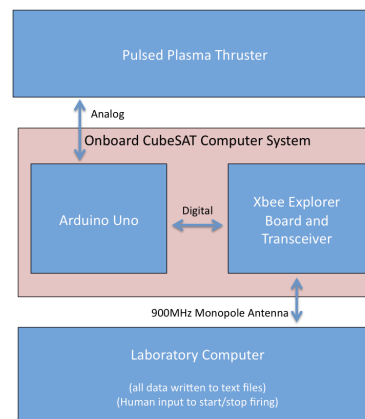
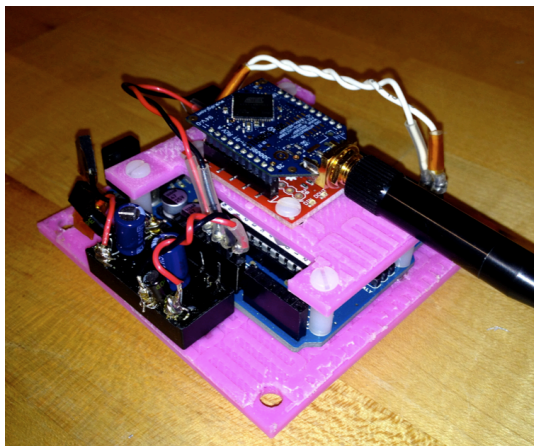
### 8.2.2 Microprocessor and Communication

The PPT requires inputs from a computer as well as power from the spacecraft. By running all necessary wires through a 15-pin d-sub connector, it was easily interchangeable to run the PPT from the computer and batteries onboard the satellite to external laboratory sources for testing and troubleshooting. The PPT requires power for the main and igniter circuits, means to vary the main capacitor voltage, and a trigger for the igniter circuit. The onboard Arduino computer was capable of reading and storing the main and igniter capacitor voltages before and after discharge through a voltage divider, rogowski coil maximum output to record the discharge current, as well as an onboard photoresistor to measure the light intensity of the created plasma plume.



**Figure 8.8. The PPT electrical inputs and outputs to and from the microprocessor and satellite PPU.**

A standard Arduino Uno microprocessor (ATmega328) with a 900MHz XBee-Pro RF module, XBee explorer board, and 900Mhz monopole antenna were used to control the PPT and transmit its status while firing in the laboratory vacuum chamber. The device was mounted together onto two 3D printed boards for integration with the CubeSat rails. It was placed within a grounded aluminum box in addition to having 2.2 $\mu$ F filter capacitors placed on the power and reset pins to the microprocessor to protect against EMI from the PPT discharge. The computer and transmitter were powered directly from the outputs of a 12V DC-DC converter, the inputs of which were tied to the bus voltage.



**Figure 8.9. The satellites onboard computer and transceiver system mounted together (left) and overall system data transmission schematic (right).**

An Arduino code (32KB in size) was written to run the experiment in the laboratory. The code consists of six distinct steps (Table 8.1). The 900MHz XBee transceiver could not

only transmit satellite and thruster status, but received a true/false signal from the human-controlled computer outside the chamber to determine if the PPT should be charged and fired. An overview of six primary portions of the software is given below. It was written in Arduino 1.5.6 (2014).

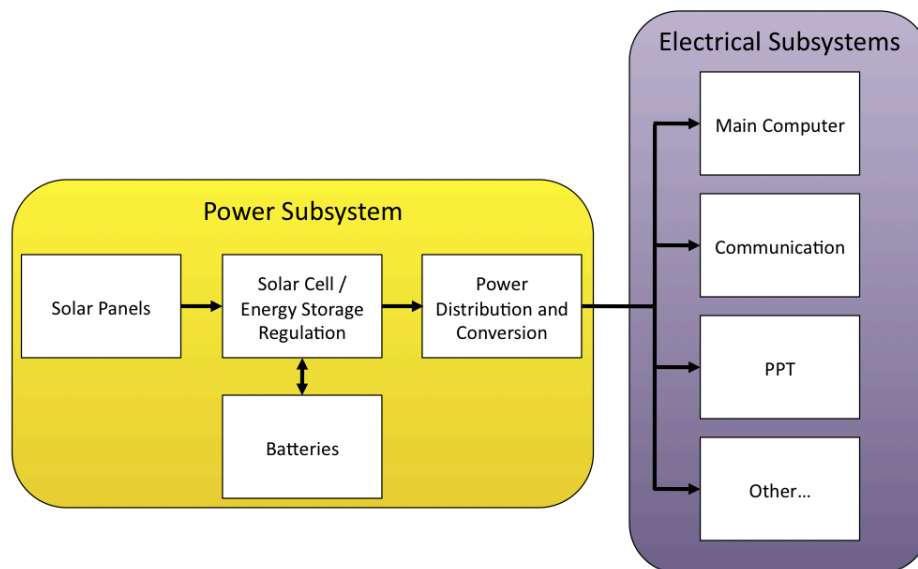
```
// DAWGSAT 2015 – CubeSat PPT Testing Arduino Code Overview
#define A_REF 5.00           // Define global constants
// Setup Loop
void setup(){
  Serial.begin(9600);        // Baud rate
  pinMode(A1,OUTPUT);        // Initialize analog output pins
  digitalWrite(D1,LOW);      // Initialize digital pins
  fire_PPT = false; }        // Set PPT to not fire on first shot
// System loop
void loop(){
  // (1) Read satellite status signals
  read_status(){
    v1 = analogRead(V1)*(A_REF/1024);} // Read satellite sensors
  // Run PPT code if human input is TRUE
  if(fire_PPT){
    float ctr_volt = CTR_VOLT;        // 0-5V signal → 0-1kV main discharge
    digitalWrite(IGN_CTR, HIGH);      // Charge igniter capacitor with digital
    5V
    // (2) Charge main capacitor, wait 2 seconds to complete, record voltages
    analogWrite(MAIN_CTR, ctr_volt / (A_REF * 255));
    delay(2000);
    VDpre = analogRead(VD) * (A_REF / 1024);
    // (3) Fire Igniter Circuit of PPT
    digitalWrite(IGN_TRIG, HIGH);     // Trigger the igniter circuit
    igniter_count++;                  // Update igniter count
    digitalWrite(IGN_TRIG, LOW);      // Turn off igniter circuit
    // (4) Record maximum value of sensors for 2ms after PPT discharge
    read_peakvalues(){
      // Create 2ms timer and set peak voltages to 0
      unsigned long timer2ms = millis() + 2000;
      PEAK_v = 0;
      // For 2ms read analog input and record maximum value
      while(millis() < timer2ms){
        PEAK_rog_main1 = max(PEAK_v, (float)(analogRead(V1))); } }
      VDpost = analogRead(VD) * (A_REF / 1024); } // Read voltages post-fire
  // (5) Buffer and transmit data
  buffer_data();
  transmit_buffer();
  // (6) Human input: PPT start/stop firing
  read_fire_YN(); }
```

**Table 8.1. The six primary portions of the Arduino code to control the PPT while firing within the belljar.**

| Step # | Function                   |
|--------|----------------------------|
| 1      | Read satellite sensors     |
| 2      | Charge PPT capacitors      |
| 3      | Fire PPT                   |
| 4      | Read PPT discharge sensors |
| 5      | Transmit data              |
| 6      | Human input                |

### 8.2.3 Power Processing Unit

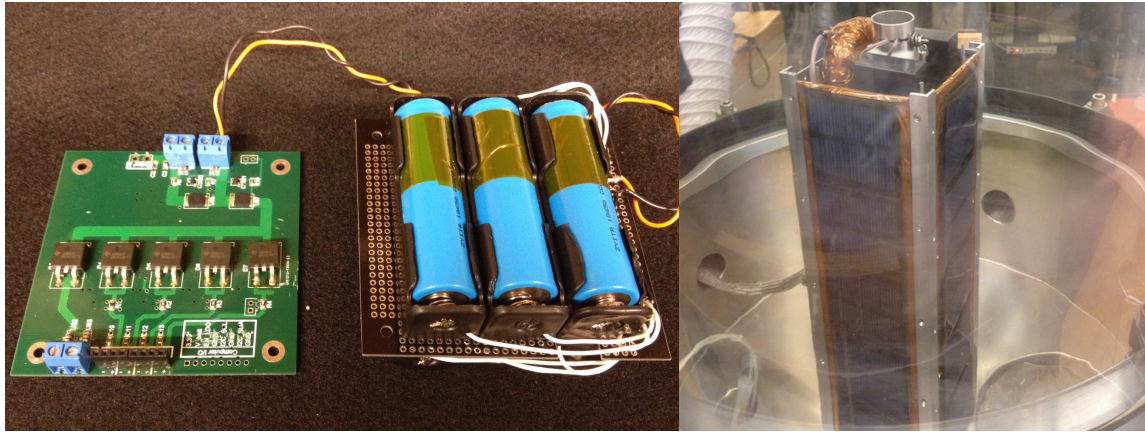
To create a fully autonomous thruster, off the shelf rechargeable LiIon batteries and solar panels were implemented for an energy storage system. A top-level system diagram of this power processing unit (PPU) is shown in Fig. 8.10. Regulation of the panels is required to maximize the power output of the panels and to provide the correct power input to the batteries, necessitated by the varying battery charge state. Battery regulation also entails resettable fuses for short-circuit protection. A DC-DC converter isolates the batteries from the spacecraft subsystems and distributes a constant bus voltage for all systems to draw off of.



**Figure 8.10. Top-level systems diagram for the electrical subsystems built.**

The components used for the PPU are shown below in Fig. 8.11. It's important to note that this initial in-house PPU, created from off the shelf components, is not space rated,

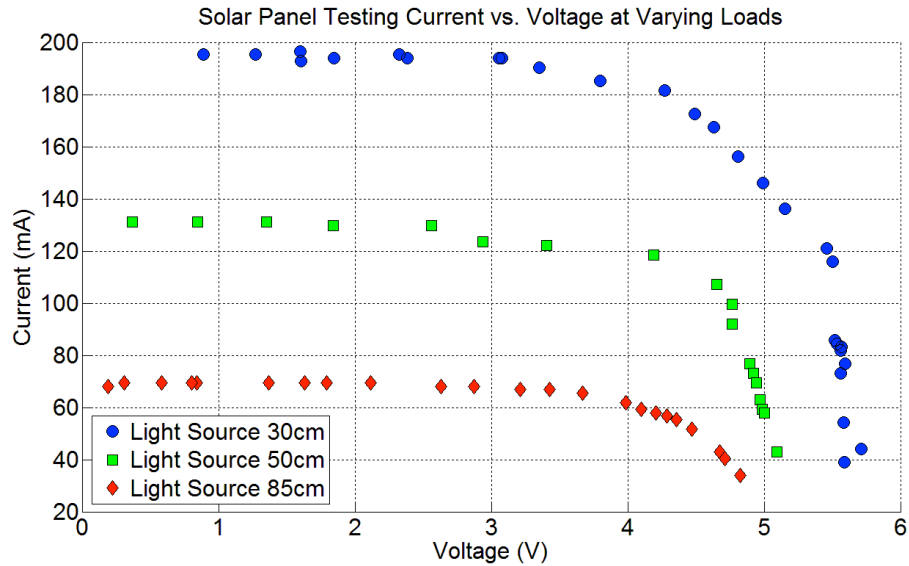
nor nearly efficient enough to be used on a spacecraft. The sole purpose was to gain experimental knowledge of a generic PPU layout, before advancing to more complex and expensive components.



**Figure 8.11. The Li-Ion battery pack and the power regulation board (left), and the solar panel arrays mounted onto a 3U CubeSat (right).**

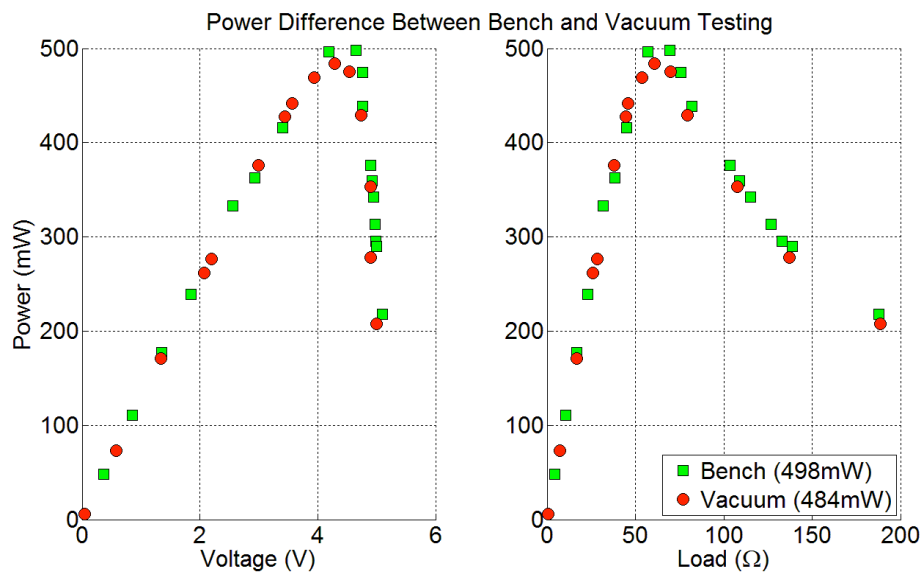
The solar panels used are comprised of commercially available silicon polycrystalline photovoltaic (PV) wafers. They were chosen for initial laboratory testing and do not possess the high efficiency or temperature requirements required for space applications. Four solar arrays, each 6.5x32.5cm, were used for laboratory testing. Each of the four arrays was placed in series with a separate regulator, the outputs of which were tied together in parallel.

Bench testing of a single solar array was completed with a wall-powered light source at varying distances between 30 and 85cm. Testing between the four arrays showed a variation of <2% in the power output between them. As can be seen in Fig. 8.12, the output current of the solar array varies nonlinearly with the array voltage. When short-circuited, the output power is zero as the output voltage is zero. When open-circuited, the output power is zero as the output current is zero. As expected, decreasing the light source distance increases the light intensity and correspondingly the output power. The maximum power point (MPP) falls between the open and short-circuited extremes and was found to occur at 4.2V, 75% of the open circuit voltage at a light distance of 50cm and at 90% of the short-circuited current (118mA at a light distance of 50cm). This resulted in a maximum power of 498mW at a light distance of 50cm.



**Figure 8.12. Power output from bench testing of a single solar array at varying resistive loads.**

When the satellite was placed within the vacuum chamber, the closest distance the light source could be placed was 50cm due to the plexiglass safety shield. It was expected to find a decrease in solar panel power output between bench and vacuum chamber testing due to dispersion and reflection from plastic safety shield and glass vacuum/atmospheric interface. Given in Fig. 8.13, testing showed an MPP decrease of 498mW on the bench to 484mW in vacuum, a 3% power decrease.



**Figure 8.13. Current and voltage output from bench (green) and vacuum (red) testing of a single solar array.**

Ideally, any system using solar power would operate the panel at its maximum power output. This is particular true of a solar powered battery charger, such as our use, where the purpose is to capture and store as much solar energy as is possible in as little time as possible. There are numerous ways to operate a solar panel at its maximum power point. One such method is an approach that implements a maximum power point tracking (MPPT) algorithm to optimize the match between the solar array and the battery bank. An SPV1040 chip was chosen to serve this purpose. It is a low-power, monolithic step-up converter with an input voltage range from 0.3-5.5V. The SPV1040 has been flown previously and can operated under wide irradiance and temperature ranges. A power regulation board was designed and printed to house these regulators and easily interface between the solar arrays, energy storage batteries, and the subsystem electrical loads.

Using four light sources in the laboratory at a distance of 50cm from the panels, and the SPV1040 regulator, a maximum power of 1.9W (~480mW from each of the four arrays) could be continuously supplied to the satellite.

Six 18650 Lithium Ion (Li-Ion) batteries were used for energy storage. When fully charged each could output 3.7V for 2.2Ahrs, resulting in 8.1Whrs. Each was 1.8cm in diameter and 6.5cm in height, weighing 47grams, making the batteries the heaviest part of the satellite. They were connected in series with a 7A resettable fuse to the power regulation board. The power regulation and battery boards are shown in Fig. 8.11.

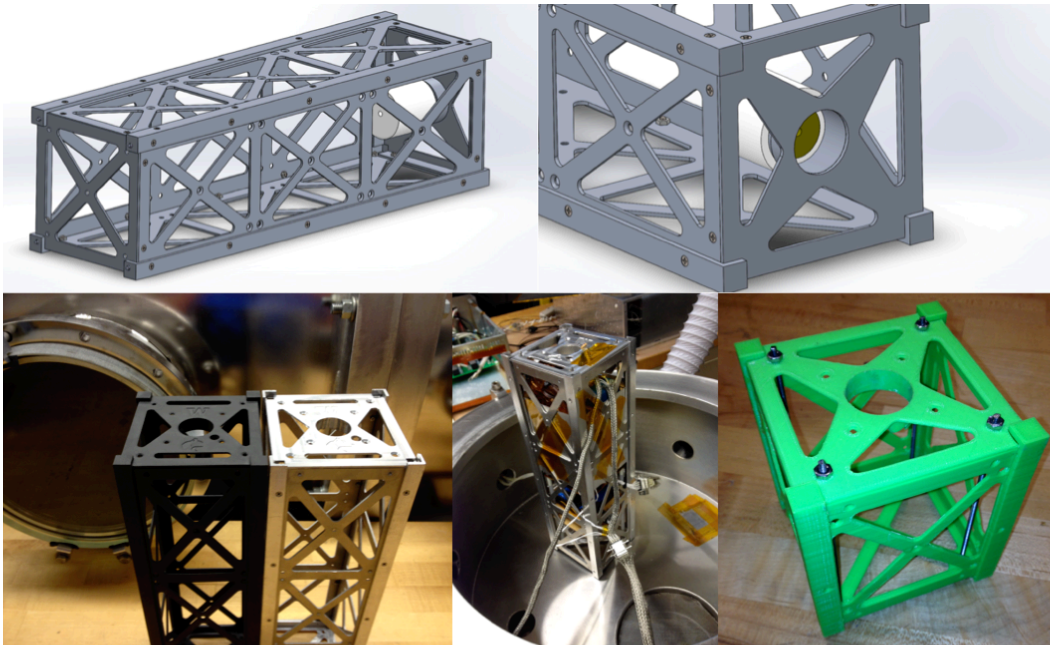
#### **8.2.4 Satellite Structure**

In addition to building the subsystem hardware and software, a satellite body to house the thruster was constructed. New advances in 3D printing allow for complex structural designs to be manufactured in a matter of hours at extremely low cost. We are currently testing the feasibility of 3D printed frames against Al6061. All 3U designs must be  $340.5 \pm 0.3$  mm in length, and  $100 \pm 0.1$  mm in width and breadth to be compatible with standard CubeSat deployment mechanisms [92].

Manufacturing of CubeSat bodies is relatively straightforward for university research labs with access to state of the art machine shops. The side panels are 1/16" thick and machined with a water-jet into an isogrid design for a high strength/mass ratio. The



panels were bolted together with four L-beam rails machined from solid extruded square rod stock. Custom milled front and back plates complete the satellite. The side panels were recessed below the planes defined by the rail sides to allow for mounting of solid panels.



**Figure 8.14. Solidworks structure models (top), windform and Al6061 3U frames (left), Al6601 CubeSat in testing (center), and a PLA 3D printed 1U frame (right).**

The total mass of the Al6061 structure is 550g, and although small compared to the 3U limit of 4kg, it isn't extraordinary. Small satellites can often be starved for mass and future efforts can be made in mass reduction once stress and strain points are known. For Earth based testing this design was deemed to be sufficient as it offered a solid, nonflexible structure of the right dimensions for initial prototyping. Future designs could use commercially available skeletons or reduce wall thickness, both of which would create a more flexible structure before component installation, but are far lower overall mass. The other major design factor to the skeleton left out of this version was removable side panels for access and maintenance.

For rapid prototyping of the CubeSat structure to test interfacing, placement, and layout of components, 3D printing was utilized. A modular design was created which allows for 1-3U structures to be assembled from the same components. The freely accessible 3D

printers on campus allow for a <24-hour turn around on components created from PLA plastic.

A relatively new material, windform XT has been introduced within the 3D printing community due to R&D financing from the high-performance racecar industry to lower the weight of their vehicles while maintaining structural integrity. CRP Technologies is the primary manufacture of this material, which is a polyamide-based carbon fiber reinforced composite that is characterized by high stiffness, high strength, and low weight [105].

Comparing average values for density and tensile strength shows that 3D printed PLA is half as heavy for a given volume than Aluminum, while windform has a 30% higher strength/weight ratio.

**Table 8.2. Density, strength, and the resulting specific strength of tested CubeSat materials.**

|          | Density<br>(g/cc) | Tensile Strength<br>(MPa) | Specific Strength<br>(kN-m/kg) | Machined Wall<br>Thickness (in) |
|----------|-------------------|---------------------------|--------------------------------|---------------------------------|
| Aluminum | 2.7               | 125                       | 46                             | 1/16                            |
| PLA      | 1.3               | 50                        | 38                             | 1/5                             |
| Windform | 1.1               | 80                        | 72                             | 1/16                            |

### 8.3 Thruster Performance

A simple conservation of momentum analysis was completed to predict the thrusters performance. This was followed by density measurements in the plume of the thruster to more accurately quantify the thrusters impulse bit at these low discharge energies. Our laboratory does not currently have a thrust stand for the belljar vacuum chamber that can measure these small impulses.

Using experimental data in addition to previous published work, estimates for the performance of the thrusters was completed. Laboratory data (Section 5.2) shows the plasma exhaust velocity to be constant over a range of 2-10J for both Teflon (25km/s) and sulfur (16.8km/s). Previous experiments [8] have shown an average neutral velocity of 500m/s for Teflon. Due to the fairly negligible influence of the neutral particles to the total thrust output, this velocity was assumed for both Teflon and sulfur experiments.

The mass ablation rate of Teflon has previously been found to be 1-3 $\mu\text{g}/\text{J}$  [8], and is assumed to be 2 $\mu\text{g}/\text{J}$  here. Based on chamber pressure measurements (Section 5.3.1) the sulfur propellant ablates 90% more material as the Teflon variant, implying that sulfurs ablation rate is 3.8 $\mu\text{g}/\text{J}$ . The percentage of ablated material that is ionized for Teflon has been shown in previous experiments [7, 8] to be 8-16% and is assumed to be 12% here. Based on the ionization energies of Sulfur (10.36eV) and Teflon (average of Carbon at 11.26eV and Fluorine at 17.4eV) the plasma fraction ( $f_p$ ) for sulfur is expected to be 16%.

The values from the above assumptions were used in the following equations to predict the PPT performance. The thrusters mass bit ( $m_{bit}$ ), or fuel ablated per shot, is simply the mass ablation rate ( $m_{AR}$ ) multiplied by the discharge energy ( $E_C$ ), while plasma and neutral mass comes from the ionization fractions.

$$m_{bit} = m_{AR} E_C \quad 8.3$$

$$m_{p,n} = m_{bit} f_{p,n} \quad 8.4$$

The maximum number of discharges from the thruster, based strictly on the amount of propellant available, is found dividing the total mass of the fuel by the mass bit.

$$\#pulses = \frac{m_{propellant}}{m_{bit}} \quad 8.5$$

The impulse bit ( $I_{bit}$ ) of the thruster is the total momentum transfer from both the plasma and neutral particles ablated per discharge. While the total impulse ( $I_{total}$ ) of the thruster is the impulse bit multiplied by the number of discharges.

$$I_{bit} = m_p v_p + m_n v_n \quad 8.6$$

$$I_{total} = \sum_{\#pulses} I_{bit} \quad 8.7$$

Once the specific impulse ( $I_{sp}$ ) of the ablated material is calculated (Eq. 8.8), the rocket equation (Eq. 1.1) can be used to find the velocity changed imparted to the spacecraft.

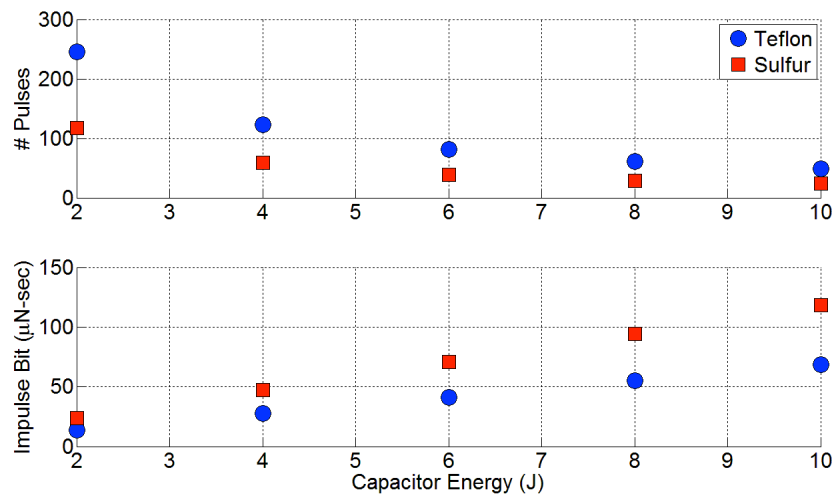
$$I_{SP} = \frac{f_p v_p + f_n v_n}{g_0} \quad 8.8$$

Based on the above assumption and equations a comparison between Teflon and sulfur was completed for the CubeSat thruster geometry and energy levels (Table 8.3 and Fig.

8.15). Based on a propellant geometry of 5cm<sup>2</sup> surface area and 7.5cm length, the satellite can be expected to hold up to 75g of solid sulfur and 85g of Teflon. A Teflon thruster will have enough propellant for 4-21 million pulses, while the sulfur variant will have 2-10 million. Sulfur's main advantage is its 70% increase in  $I_{bit}$ , from 68.8 to 118 $\mu$ N-s at 10J, allowing for more rapid spacecraft velocity changes. Mission with higher total impulse and  $\Delta V$  requirements would be more suited for a Teflon variant as these values are 18% larger. This is due to Teflon's higher  $I_{SP}$  of 350s, compared to 317s for sulfur.

**Table 8.3. The predicted performance for Teflon and Sulfur propellants at 2 and 10J.**

|                                 | Teflon |      | Sulfur |       |
|---------------------------------|--------|------|--------|-------|
| Discharge Energy (J)            | 2      | 10   | 2      | 10    |
| Total Ablated Mass ( $\mu$ g)   | 4      | 20   | 7.6    | 38    |
| Plasma Mass ( $\mu$ g)          | 0.5    | 2.4  | 1.2    | 6.1   |
| # Pulses (millions)             | 21     | 4    | 10.2   | 2     |
| Impulse Bit ( $\mu$ N-s)        | 13.8   | 68.8 | 23.6   | 118.1 |
| Plasma Velocity (km/s)          | 25     |      | 16.8   |       |
| Exhaust Isp (sec)               | 350    |      | 317    |       |
| Total Impulse (N-s)             | 292    |      | 240    |       |
| $\Delta V$ with 4kg $m_0$ (m/s) | 74     |      | 60     |       |

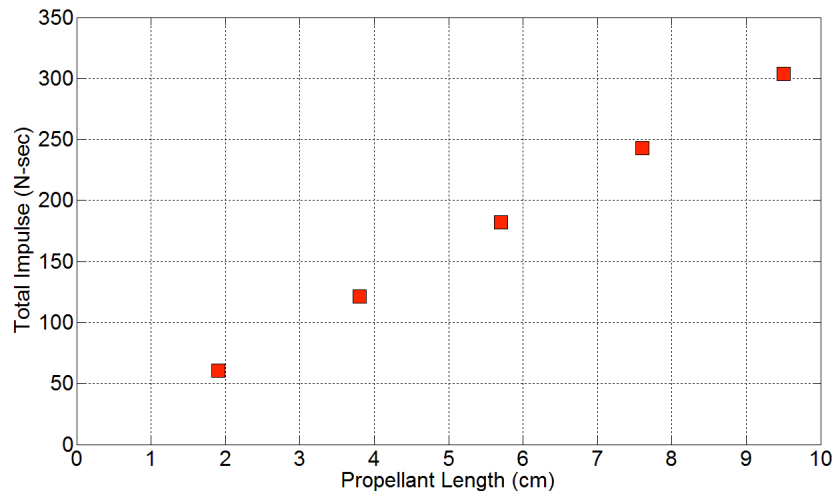


**Figure 8.15. Thruster  $I_{bit}$  (top) and lifetime (bottom) comparison between Teflon (blue) and sulfur (red) propellants over an energy range of 2-10J.**

The above calculations were for a set propellant rod length of 7.5cm. Allowing this parameter to vary between 1.9 and 9.5cm for a sulfur propellant thruster leads to the results in Table 8.4 and Fig. 8.16. Increasing the propellant rod length, linearly increases the mass of propellant, and the resulting total impulse and  $\Delta V$ .

**Table 8.4. 10J Sulfur PPT performance estimates for propellant lengths of 1.9-9.5cm.**

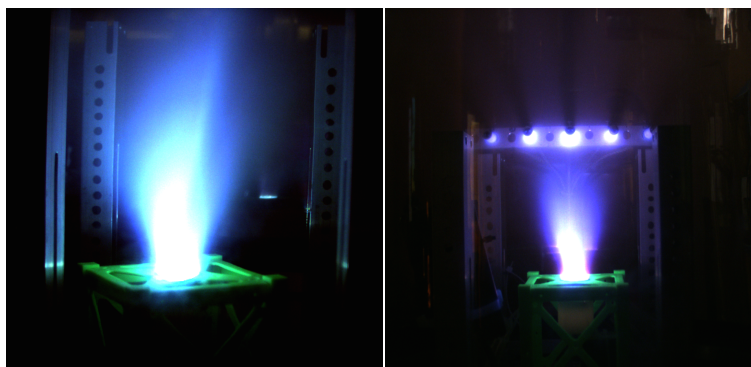
| <b>Propellant Length (cm)</b>   | <b>1.9</b> | <b>3.8</b> | <b>5.7</b> | <b>7.6</b> | <b>9.5</b> |
|---------------------------------|------------|------------|------------|------------|------------|
| Propellant mass (g)             | 19.6       | 39.1       | 58.7       | 78.2       | 97.8       |
| # Pulses (millions)             | 0.5        | 1.0        | 1.5        | 2.1        | 2.6        |
| Total Impulse (N-s)             | 61         | 122        | 182        | 243        | 304        |
| $\Delta V$ with 4kg $m_0$ (m/s) | 15         | 31         | 46         | 61         | 77         |



**Figure 8.16. Total impulse vs. propellant length for a sulfur propellant PPT.**

Propellant selection, length, firing frequency, and discharge energy would all be mission specific. The possibility of sulfur propellant enables missions requiring lower propellant mass and faster orbit changing than what Teflon is capable of.

The thrusters performance was experimentally measured for each discharge energy level (0.5, 0.9, and 1.6J) and at each probe sampling height (13, 18, and 23cm) using a 5-Langmuir probe array (discussed in Section 3.1). The Teflon and sulfur PPT firing from within the CubeSat can be seen in Fig. 8.17.



**Figure 8.17. Teflon (left) and sulfur (right) propellant PPTs firing within the belljar vacuum chamber. The sulfur is shown firing into the Langmuir probe array.**

The mass of the ions was assumed to be half carbon (12amu) / half fluorine (18amu) for Teflon propellant, and entirely sulfur (32amu) for the sulfur propellant. The plasma bulk velocity was calculated with the time of flight method (detailed in Section 3.1). The traces were integrated in time to determine a mass flux (kg-s/m<sup>3</sup>), starting from 5μs, which is after the noise from the igniter circuit subsides, and extending to when the Langmuir trace reduced to below the noise level (typically between 35-50μs).

$$\Phi_m = \int_{5\mu s}^{t_f} nmdt \quad 8.9$$

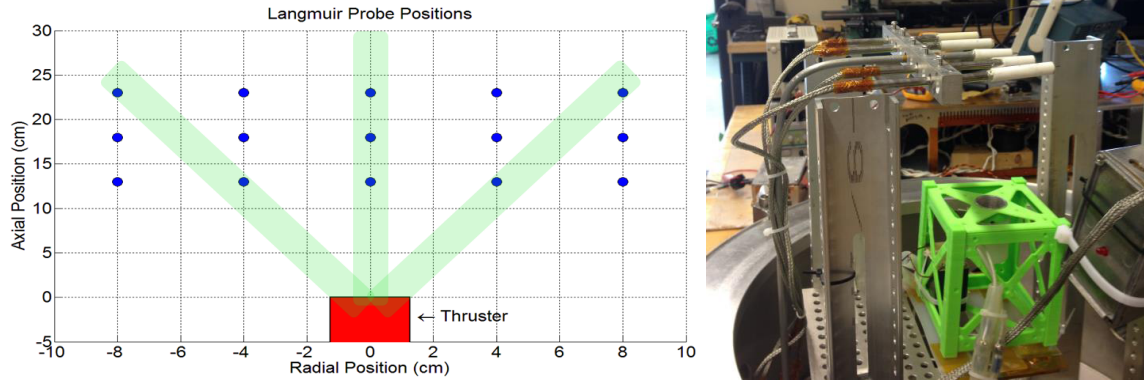
These integrated density-time values were plotted spatially, after which a Gaussian distribution (G) was fitted to the five data points. The plumes cross-sectional area at each height, although not necessary for the impulse calculation, was calculated by assuming circular symmetry and using the Gaussians standard deviation as the cross-sectional radius. Multiplying the Gaussian by the velocity squared and an angle correction factor, and then integrating the result radially and azimuthally determined the impulse bit. Only the axial component of the particles momentum will contribute to useful thrust; this is accounted for with the angle correction factor,  $c_\theta = \cos(\tan^{-1}(r/h))$ .

$$I_{bit} = nmv^2At = v^2 \int_0^{2\pi} \int_0^{40cm} G_{r,n,m} C_\theta r dr d\theta \quad 8.10$$

The specific thrust, formally defined as the thrust divided by the input power (N/W), is unit equivalent to the impulse divided by the capacitor energy (N-s/J).

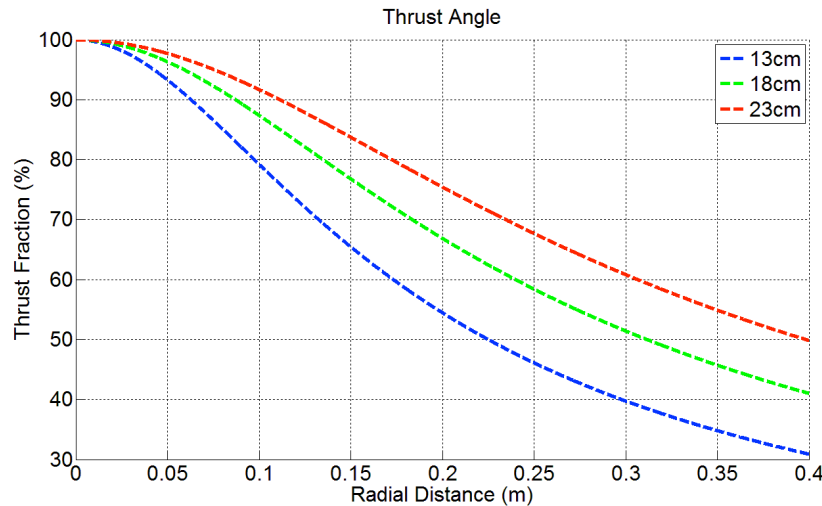
$$T_s = \frac{T}{P} = \frac{I_{BIT}}{E_C} = \frac{2I_{BIT}}{CV^2} \quad 8.11$$

The Langmuir probe array was placed in the plume of the thrust to sample a radial cut through the cross-section from -8 to +8cm and was varied axially from 13-23cm above the propellant surface. The igniter arc was on the left hand side of the thruster from this viewpoint, as was the screw that made the cathode electrical connection.



**Figure 8.18.** The Langmuir probe sampling region (left) with the probe locations (blue), thruster (red), and straight line paths between thruster and probes (green). The thruster is housed within a 1U frame under the Langmuir probe array (right).

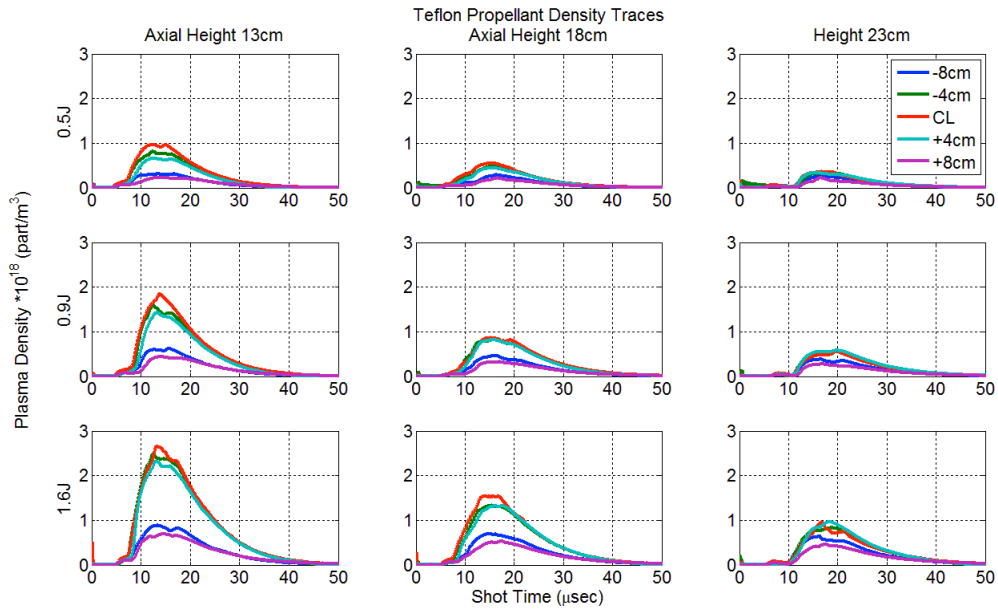
Only the axial component of the impulse will contribute to useful thrust, hence the cosine term in Eq. 8.11. This thrust angle was found through simple geometry at the three downstream sampling heights and is plotted below. The loss due to the thrust angle is 0 on the centerline and increases to 70% at a height of 13cm and radial position of 40cm.



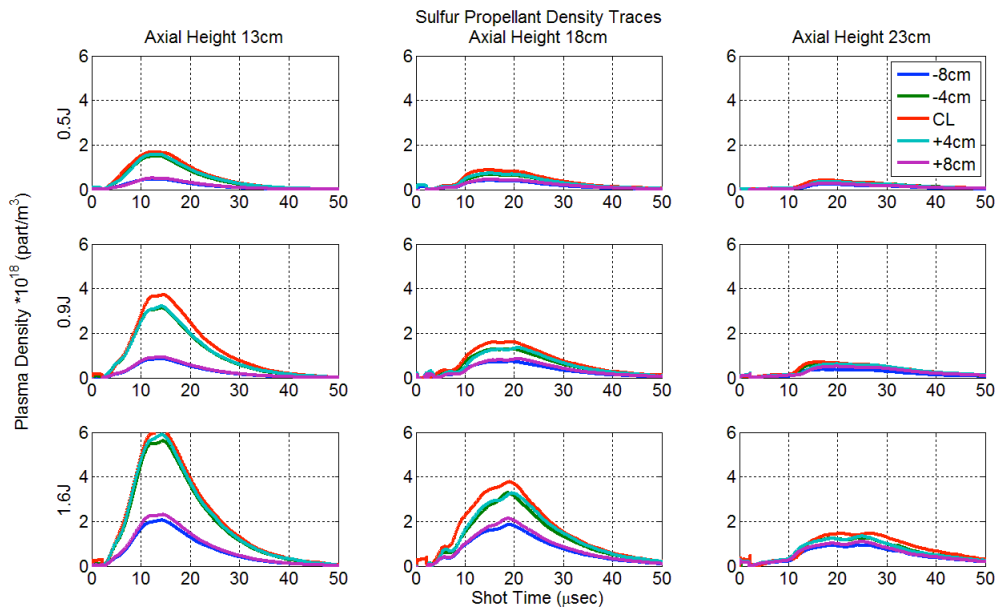
**Figure 8.19.** The useful component of thrust is only that in the axial direction. The percentage of useful thrust at any given point in the experiment is shown plotted at radial distances of 0-40cm and axial distances of 13, 18, and 23cm.

The density at each energy level (0.5, 0.9, and 1.6J) was measured by each of the five probes at heights of 13, 18, and 23cm. For each energy level and height, five separate shots were fired, the average of which are shown below in Figs. 8.20 and 8.21 for Teflon sulfur, respectively. For any given discharge, there was up to a  $\pm 2\mu\text{s}$  variation between when the igniter would arc. This variation propagated to the main discharge current and

Langmuir probe outputs. This was corrected by comparing and normalizing each of the main discharge current traces, and then adding the same offset to the corresponding Langmuir outputs. After these offsets were accounted for, the variation between Langmuir probe outputs for each of the five repetitive shots was <2% in maximum density and <1% in peak density time



**Figure 8.20. Teflon propellant density traces at 0.5 (top), 0.9 (center), and 1.6J (bottom) discharges for probe heights of 13 (left), 18 (center), and 23cm (right).**



**Figure 8.21. Sulfur propellant density traces at 0.5 (top), 0.9 (center), and 1.6J (bottom) discharges for probe heights of 13 (left), 18 (center), and 23cm (right).**

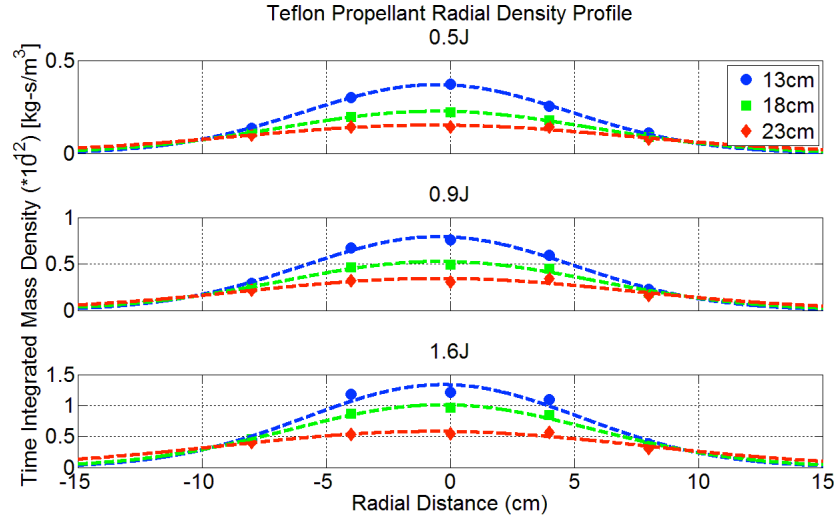


Using the probes shown in the green highlighted sections of Fig. 8.18, the bulk velocity of the plasma exhaust was estimated using the time of flight (TOF) method. The experimental setup allowed for five separate TOF velocities to be calculated at each energy level; the average of the five was taken as the plasma exhaust velocity. Three of the five locations were on the centerline, between probes at 13 & 18cm, 18 & 23cm, and 13 & 23cm. The two remaining locations are on an angle of  $\pm 70^\circ$ . The two probes that make up the  $70^\circ$  TOF velocity are only within  $2^\circ$  of a straight-line path to each other, this is considered close enough to be sampling the same plasma at each probe.

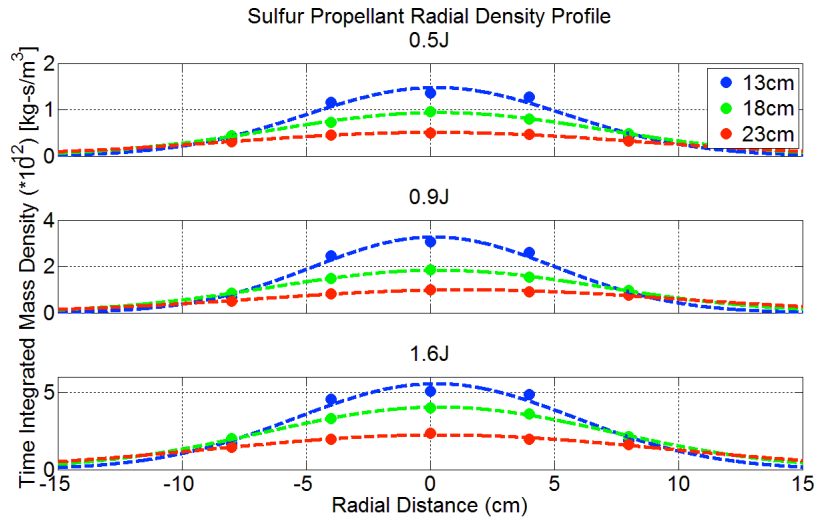
**Table 8.5. Time of flight velocity results for 0.5, 0.9, and 1.6J between probes on the centerline at heights of 13 & 23cm, 13 & 18cm, and 18 & 23cm, as well as at angles of  $-70^\circ$  and  $+70^\circ$ , and the average of the five velocity measurements.**

| Energy (J)    | V <sub>13-23</sub> (km/s) | V <sub>13-18</sub> (km/s) | V <sub>18-23</sub> (km/s) | V <sub>-70°</sub> (km/s) | V <sub>+70°</sub> (km/s) | V <sub>avg</sub> (km/s) |
|---------------|---------------------------|---------------------------|---------------------------|--------------------------|--------------------------|-------------------------|
| <b>Teflon</b> |                           |                           |                           |                          |                          |                         |
| <b>0.5</b>    | 22.7                      | 19.2                      | 27.8                      | 26.1                     | 27.4                     | 24.7                    |
| <b>0.9</b>    | 25.0                      | 22.7                      | 27.8                      | 25.5                     | 27.4                     | 25.7                    |
| <b>1.6</b>    | 27.0                      | 26.3                      | 27.8                      | 26.8                     | 24.9                     | 26.6                    |
| <b>Sulfur</b> |                           |                           |                           |                          |                          |                         |
| <b>0.5</b>    | 17.1                      | 18.2                      | 19.5                      | 18.9                     | 18.6                     | 18.5                    |
| <b>0.9</b>    | 16.1                      | 17.2                      | 18.5                      | 21.1                     | 21.5                     | 18.9                    |
| <b>1.6</b>    | 16.4                      | 17.2                      | 18.3                      | 16.2                     | 16.2                     | 16.9                    |

The density traces in Figs. 8.20 and 8.21 were integrated and multiplied by the ion mass (Eq. 8.10) to calculate a time integrated mass density for each Langmuir measurement taken. The results are shown plotted against the radial position of the measurement and a Gaussian curve was fit to the data points. The FWHM for each of the Gaussians was found to increase linearly with energy and decrease as the square root of the height. The Gaussian fits applied show the center of the plume a constant  $-0.3 \pm 0.2$ cm off the centerline (towards the left in Fig. 8.18) for both Teflon and sulfur propellants, implying that the plume is not perfectly centered on the axis of the thruster. This is believed to be due to the location of the igniter arc and/or the cathode connection screw (both to the left). However, this off-axis tilt is minor ( $<1\%$  at a height of 23cm) and isn't considered a major drawback of the thruster.



**Figure 8.22.** Time integrated mass density radial profiles of the Teflon plasma plume at 0.5 (top), 0.9 (center), and 1.6J (lower) discharges, at 13 (blue), 18 (green) and 23cm (red) heights overlaid with best-fit Gaussian distributions.

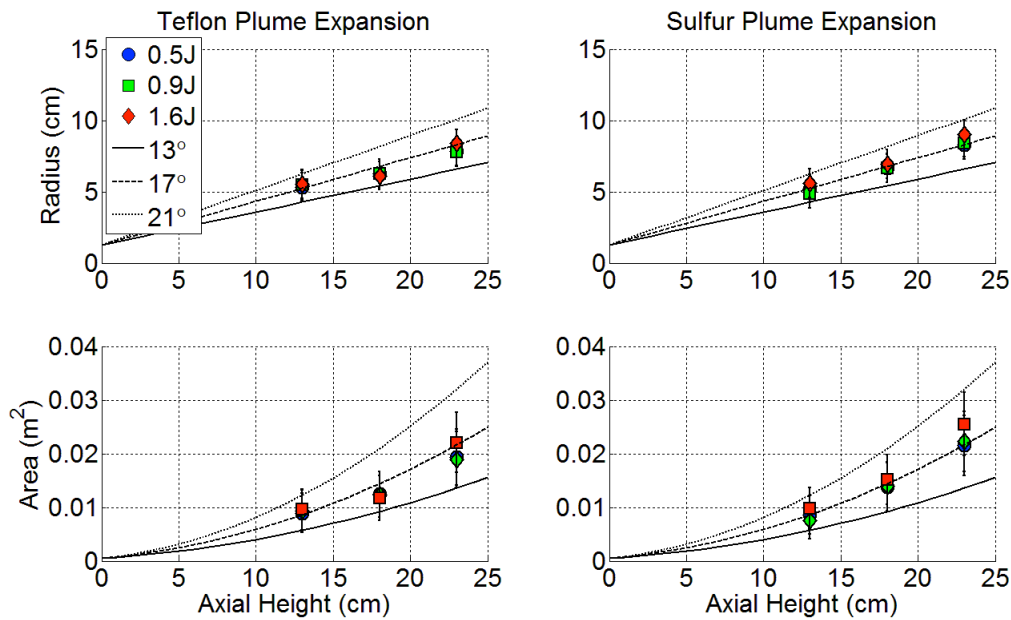


**Figure 8.23.** Time integrated mass density radial profiles of the sulfur plasma plume at 0.5 (top), 0.9 (center), and 1.6J (lower) discharges, at 13 (blue), 18 (green) and 23cm (red) heights overlaid with best-fit Gaussian distributions.

Although not required for the impulse calculation, as the area is accounted within the integration in Eq. 8.11, the radius and cross-sectional area of the plume as a function of axial height is plotted below in Fig. 8.24. The plume cross-sectional area was determined by defining the standard deviation of the best-fit Gaussian curve as the circular cross-sectional radius. The radius of the plume was found to increase linearly with height, suggesting a cone shaped, thermally expanding plasma plume. A linearly increasing plume radius also implies that the cross-sectional area of the plume will increase as  $h^2$ .

Using the standard deviation to determine the plume area results in an expansion angle of  $17 \pm 4^\circ$ , which is similar to results from other PPTs [8]. Based on shot to shot variation, a maximum error of 1cm in the radius of the plume was found. There was no statistically difference between the Teflon and sulfur plume areas.

If the plasma is fairly collisionless over the downstream region being sampled, the only effect on the density should be the thermal expansion of the plume. If the collisionless assumption is true, then the density should fall off as  $h^2$ , inversely proportionally to the expanding area, and the impulse measured at each of the three heights should be similar.

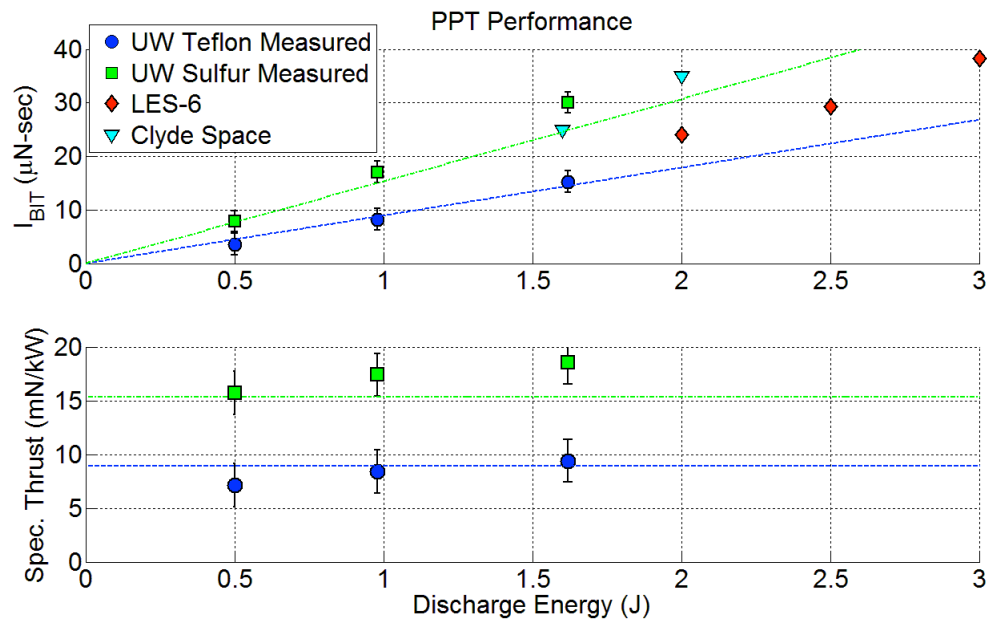


**Figure 8.24.** The expanding plume radius (top) and cross-sectional area (lower) from the best fit Gaussian distributions in Figs. 8.23-8.24. 0.5J is shown in blue, 0.9J in green, and 1.6J in red. Expected axisymmetric plume areas starting at a diameter of 2.54cm at a height of 0cm are shown for angles of  $13^\circ$  (solid),  $17^\circ$  (dashed), and  $21^\circ$  (dotted).

Using the Teflon and sulfur results above, Eqs. 8.10-8.12 were used to estimate the thrusters performance for each propellant at each energy level and axial height (Fig. 8.25). It was seen that the impulse bit calculated at the three different axial heights agreed to within 8%, and that the impulse increased linearly with energy for both propellants. Similar trends were found with the LES-6 [8] and Clyde Space PPTs [106]. Similar to the higher discharge energies (Chapter 5), the sulfur PPT shows a doubling in impulse over the Teflon propellant. The linear increase in impulse, matches the expected impulse

(Section 8.4.1) to within 15%. The linear increase in impulse, results in a constant specific thrust, again matching the expected values to within 15%.

If the same performance increase from sulfur to Teflon translates to the Clyde-Space PPT geometry and electronics, specific thrusts nearing 30mN/kW should be expected, far outperforming any PPT that's been flown or currently on the market for sale.



**Figure 8.25. The calculated impulse bit (top) and resulting specific thrust (bottom) for the Teflon (blue) and Sulfur (green) propellants. Impulse bit for the LES-6 PPT (red) and the current state of the art PPT with Teflon propellant designed for CubeSat operation by Clyde-Space (cyan) are shown [8, 104].**

These results suggest that for short mission durations with small satellites where higher impulse is required, but the small propellant mass and volume that electric propulsion can provide, a sulfur propellant PPT may be a viable thruster. Future laboratory testing should look to optimize the shape and size of the PPT electrodes, as well as the capacitor charging electronics to allow for higher frequency discharges. One of Teflon's attributes has been its ability to be used for millions of pulses with little change to the thrusters performance. The sulfur propellant tested here was fired for hundreds of pulses without drastic variation to the thrusters performance, however future testing should examine longer lifetime characteristics with sulfur propellant.

An initial CubeSat frame and top-level satellite subsystems were designed and built in the laboratory for the purpose of testing a sulfur PPT specifically designed for small satellite

operation. The thruster and associated subsystems performed as required for autonomous operation of the thruster, which had a specific thrust of 18.7mN/kW at a discharge energy of 1.6J and plasma densities of  $10^{18}$ part/m<sup>3</sup> at distances of 10-20cm. Future work should transition to space-rated components as well as better characterize the long duration performance of the sulfur propellant.

**Table 8.6. The plume geometry, impulse bit, and specific thrust measurements.**

|        | Energy (J) | Height (cm) | Radius (cm) | Area (m <sup>2</sup> ) | Impulse Bit (μN-s) | Specific Thrust (mN/kW) |
|--------|------------|-------------|-------------|------------------------|--------------------|-------------------------|
| Teflon | 0.5        | 13          | 5.33        | 0.009                  | 3.8                | 7.6                     |
|        |            | 18          | 6.29        | 0.012                  | 3.4                | 6.7                     |
|        |            | 23          | 7.84        | 0.019                  | 3.5                | 7.1                     |
|        | 0.9        | 13          | 5.47        | 0.009                  | 8.6                | 9.6                     |
|        |            | 18          | 6.28        | 0.012                  | 7.9                | 8.9                     |
|        |            | 23          | 7.76        | 0.018                  | 8.0                | 9.0                     |
|        | 1.6        | 13          | 5.53        | 0.010                  | 15.5               | 9.7                     |
|        |            | 18          | 6.11        | 0.012                  | 14.9               | 9.7                     |
|        |            | 23          | 8.38        | 0.022                  | 15.4               | 9.6                     |
| Sulfur | 0.5        | 13          | 5.22        | 0.009                  | 8.1                | 16.2                    |
|        |            | 18          | 6.64        | 0.014                  | 8.5                | 17.0                    |
|        |            | 23          | 8.28        | 0.022                  | 7.0                | 13.9                    |
|        | 0.9        | 13          | 4.87        | 0.008                  | 16.3               | 18.1                    |
|        |            | 18          | 6.66        | 0.014                  | 17.4               | 19.3                    |
|        |            | 23          | 8.42        | 0.022                  | 17.4               | 19.4                    |
|        | 1.6        | 13          | 5.59        | 0.010                  | 28.5               | 17.8                    |
|        |            | 18          | 6.93        | 0.015                  | 31.1               | 19.5                    |
|        |            | 23          | 9.00        | 0.026                  | 30.4               | 19.0                    |

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