Micro Pulsed Plasma Thruster Development

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Abstract: The pulsed plasma thruster (PPT) is a space rated technology that has performed station keeping tasks for a variety of former missions. There has recently been an interest in diversifying the range of propulsion tasks carried out by PPTs. The missions appropriate for the use of PPTs vary from lunar missions where the PPT would provide primary propulsion to on orbit maneuvers for satellites with a mass of much less than 100 kg (e.g. CubeSats). The latter application requires PPTs to be miniaturized and adapted to operate at power levels in the range of 1-3 W. The reduction of the propellant surface area of miniaturized PPTs (μPPTs) and a reduction in the dimensions of the thruster electrodes gives rise to issues associated with geometric scaling laws. Although the issues relating to μPPT scaling have been investigated to a certain degree in the past it is felt that for an application on CubeSats this topic has to be investigated in greater detail for even smaller dimensions. Presently the performance of a μPPT with a propellant area of 1 cm^2 has been analyzed over a discharge energy range of 2 J to 8 J. The present paper discusses the performance evaluation and provides an outlook of upcoming research.

I. Introduction

THERE has been a trend within the space sector to develop smaller satellites with a faster turn around time and reduced costs. This trend has resulted in an increasing number of scheduled future missions which will utilize satellites in the microsatellite class or smaller. Academic institutions are also actively participating in the development of nanosatellites and picosatellites, over 40 Universities worldwide are currently working on CubeSat projects. CubeSats are satellites, with, as the name indicates, a cubic shape with a side length of 10 cm and a mass smaller than 1 kg. While in the beginning CubeSats were only used for Sputnik type missions now their mission tasks include sophisticated scientific experiments. Increasing mission complexity however requires active attitude and orbit control. Therefore, the need for propulsion systems which meet the performance requirements of these missions whilst conforming to the demanding mass and power constraints imposed by satellites with a mass of less than 100 kg is paramount. Mission analysis studies show that the use of active on board propulsion opposed to reaction wheels or passive magnetic attitude control can dramatically increase mission capabilities for a CubeSat. An additional concern with regard to CubeSats is the fact that while their mission time is only on average 1 year, due to their orbits with an altitude of up to 800 km the actual in orbit time can extend 25 years. With increasing amounts of CubeSats being launched, concerns are mounting that these satellites constitute a hazard for telecommunication satellites during their orbit transfer. Propulsion systems can provide means to deorbit and hence to reduce at least the in-orbit residence time. Whilst the advantages of having active propulsion on-board CubeSat type satellites are clear, the challenge is to develop a propulsion system capable of performing within the mission requirements whilst utilising minimal power and spacecraft volume. CubeSats pose the most stringent constraints for propulsion systems in terms of mass and power. Table 1 shows a representative mass and power budget for such a picosatellite. However, the values given in Table 1 can significantly shift, additional mass could be allocated to the propulsion system with the use body mounted solar panels. Thus, allowing smaller batteries to be used whilst

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providing 1 W of continuous power to the spacecraft. As a result of the incorporation of solar cells an additional 200 to 250 g would become available to accommodate additional payload or extra mass for the propulsion and attitude control subsystems.

Table 1 Typical mass and power budget for Cubesats modified from Puig - Suari et. al.[1]

Component	Mass, g	Power
Batteries	470	~400 Whr storage
Microcontroller	10	0.1 W
Propulsion	200	4 W
Receiver	5	0.0012W
Structure	300	-
Thermister	≤ 1	-
Transmitter	16	0.12 W standby 1.11 W transmitting
Voltage Sensor	≤ 1	-
Total	~ 1000	4.25 W standby 5.25 W transmitting

Active on board propulsion is a mission enabling technology that allows satellites to perform more advanced operations. Satellites with a mass of less than 100 kg may perform propulsive maneuvers including formation flying, satellite inspection, drag compensation, station keeping and attitude control in future missions. The maximum delta V requirement for these missions assuming a duration of 6 to 12 months is 300 ms^{-1}, within the expected performance range of µPPTs.[2]

Pulsed Plasma Thrusters (PPTs) have been successfully used on spacecraft since the 1960's performing station keeping and attitude control tasks.[3] PPTs have operated in a variety of geometries such as coaxial, side fed and breech fed. Figure 1 shows the conventional breech fed PPT also known as the rectangular geometry. Although these thrusters have been operated in laboratory environments both using solid and fluid propellants, Teflon® (PTFE) is the propellant of choice for space missions. The use of a solid propellant avoids the implementation of complex propellant feed systems and hence maintains the simplicity and robustness of the thruster design. The pulsed nature of the thrust produced makes PPTs ideal for propulsive maneuvers that require small impulse bits such as drag compensation, station keeping and attitude control. The low power requirements of PPTs compared to other electric thrusters and the simplicity of its mechanical design with relatively few moving parts presents the ideal solution to providing active on board propulsion for nano and picosatellites. They also allow easy integration into existing satellite architectures, see Fig. 2 for an example of possible µPPT integration within a CubeSat. In order for the advantages of PPTs to be exploited as a mission enabling technology they require miniaturization and enhanced performance exceeding the present level.

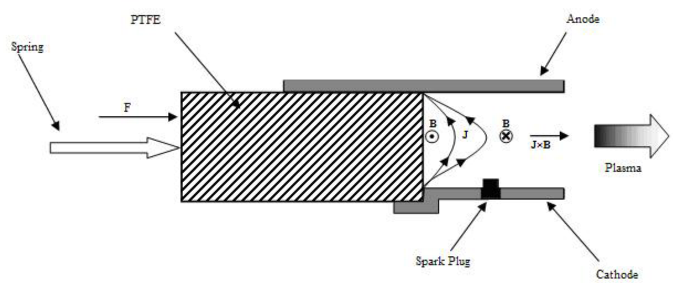


Fig. 1 PPT system in the conventional rectangular (breech) spring fed geometry operating with Teflon® (PTFE) propellant.

II. The Miniaturisation of Pulsed Plasma Thrusters

Three major challenges exist in the effort to miniaturize a PPT. (i) scaling laws as evaluated in the past for standard sized PPTs might not be valid and require renewed investigation, (ii) spark plugs which are used for discharge initiation are too large, heavy and power consuming for µPPTs, (iii) identification of suitable capacitors. In the following sections these issues are discussed in detail.

A. Scaling Factors and Implications for Thruster Performance

The μPPT design will be such that thrusters may be housed within a CubeSat structure whilst leaving sufficient volume to allow the incorporation of scientific/technological payload and attitude control systems. Fig. 2 shows one result of a system analysis. A single PPU (green) provides power and control to 6 μPPTs (red).

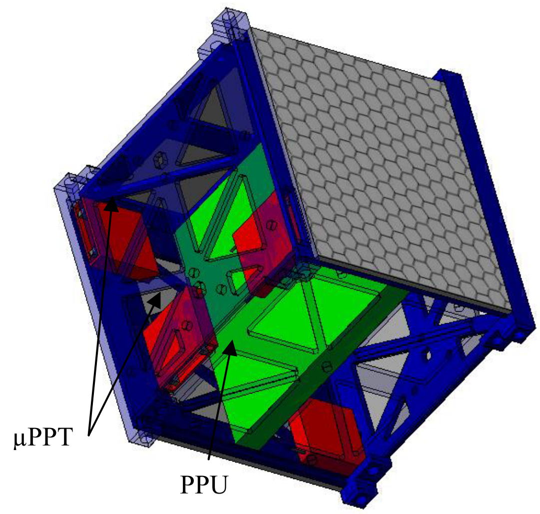


Fig. 2 Possible μPPT integration within a CubeSat.

Electrode geometry plays a key role not only in defining thruster dimensions but also in terms of thruster performance. Past investigations with regard to electrode geometries were conducted with rather large electrode widths and electrode gaps compared with the anticipated size of a μPPT, with dimensions ranging from 10 to 40 mm.[4,5,6] Besides the open question of whether these results can be simply scaled down to a μPPT level, the different studies sometimes offer contradictory results. For example, Arrington states that increasing the length of electrodes results in an increase in efficiency[7]. For the case of increasing the length from 2.54 cm to 3.81 cm an increase in efficiency of half a percent from 9% to 9.5% was observed at an energy of 43 J but the specific thrust was unaffected. In contrast, the research of Guman and Peko demonstrated that decreasing electrode length from 7.6 cm to 0.66 cm increased specific thrust by 100% and specific impulse by 160% for 5 J operation.[8] These contradictory results also persist partially for the variation in electrode gap and electrode width. Although some of those differences might be due to the different thruster geometries, utilized capacitors, thermal effects etc. they are puzzling and require clarification.

The critical thruster design criteria appears to be the aspect ratio (the ratio of the electrode separation h to the width w). It seems the common understanding in the PPT community that by increasing the aspect ratio performance improvements can be achieved as shown by Eq.1 which relates the thrust T and power P to the aspect ratio.

$$\frac{T}{P} \propto \frac{h}{w} \tag{1}$$

However, while Eq. 1 is helpful for a general design it only partially represents the situation in a PPT. It for example does not allow the identification of a limit in increasing the aspect ratio. In reality it was found that an excessive increase in electrode gap leads to charring of the Teflon surface. A good historical indicator of this limit is the energy per cm^2 of propellant surface. This ratio ranges for breech-fed PPTs between 0.69 and 3.8 with the smallest ratio from the LES-6 thruster and the largest from the TIP-II (NOVA) thruster[3]. Taking these data as the baseline can provide an envelope for the aspect ratio variation. However, the situation is even more complex than it appears. Increasing the aspect ratio can be achieved by decreasing the electrode width or increasing the electrode gap. The most detailed experimental data with regard to these variations was published by Kuan Yuan-Zhu.[9] Analyzing this experimental data in more detail shows a curious dependency of thruster performance with the method utilized to increase the aspect ratio (increasing h or decreasing w). All the data shown in Fig. 3 have been obtained for a discharge energy of 3.7 J. As can be seen in Fig. 3, the extent and fashion in which the performance changes depends not simply on the aspect ratio but on how the aspect ratio change is obtained. It can not simply be explained by the changing energy per unit area. Available semi-empirical equations between energy per unit mass and performance fail to predict such a correlation[10]. Points A and B for example correspond to equal propellant surface area, hence according to semi-empirical relationships for thruster performance the I_{Bit} should also be equal, clearly this is not the case.

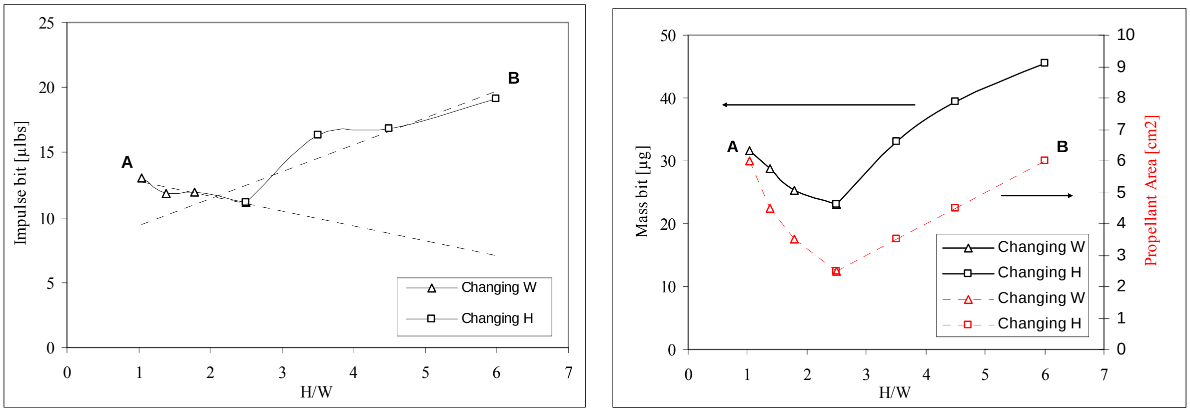


Fig. 3 Performance change correlated with changes in electrode width or electrode gap (Data extracted from reference [11])

To date the smallest propellant surface area of a laboratory model μPPT is 0.5 cm^2 and was operated at discharge energies of 2 J to 3.6 J.[11] It was demonstrated that a possible advantage of miniaturization is an increase in energy density i.e. the ratio of discharge energy to propellant surface area. This increase in energy density results in a decrease in propellant charring or carbonization of the Teflon® surface. However, studies have shown that decreasing propellant surface area to prevent propellant charring has a limit, after which no discernible difference is observed. For example a reduction in electrode width and therefore propellant surface area from 12 to 6 mm had little effect on charring at 2 J, but no charring was observed on increasing discharge energy from 2 to 2.3 J.[11] Also, increasing energy density results in the PPT operating at higher temperature thus increasing late time ablation which may be a factor that causes miniaturized PPTs to display reduced efficiencies compared to larger models.

Due to the ambiguity surrounding the impact of electrode geometry on performance particularly for miniature PPTs it is necessary to investigate in more depth the dependence of performance on geometry and understand the discrepancies presented in literature. This is of particular importance because the envisioned μPPT design will reduce the PPT structure by considerably more than attempted in previous efforts. It is expected that such an extreme miniaturization will even change the physics of the acceleration process, e.g. effects such as viscous boundary losses might become more important.

B. Discharge Initiation

Ablative PPTs use a spark plug to provide an initial discharge, thus increasing charge density and facilitating the initiation of the main discharge across the exposed face of the solid propellant. The method chosen for discharge initiation plays a significant role in the performance of the thruster as well as having an affect on mass and volume. The consistent emission of the pseudo discharge from the spark plug is critical for repeatable shot to shot performance. Alternative methods for discharge initiation include the use of a thermionic electron source or laser ignition.[12,13]

Both methods were investigated using gas fed PPTs operating on argon propellant. These unconventional methods for discharge initiation have yet to be tested on a breech fed PPT operating with solid propellant. Due to the increased complexity that these techniques introduce into the system without an enhancement of observed performance, they are unsuitable for small satellite missions where severe restrictions on mass and power apply. Additionally, these methods remain immature and only by an extensive development effort may become a viable alternative to the use of spark plugs in the future. At this point, spark plug type ignition initiation seems to be the only viable method. However, there are limitations to the degree to which a spark plug may be miniaturised dictated by the insulation required due to the high voltages achieved during sparking. The electrodes of the spark plug must also be sufficiently robust in order to prevent severe erosion, therefore the diameter of the central electrode is typically no smaller than approximately 1 mm. The erosion of spark plug electrodes is a life limiting factor for PPTs, although the selection of an appropriate electrode thickness and material such as iridium which has shown little erosion after one million sparks may prevent such problems[14].

C. Energy Storage Devices

The energy storage units used for space proven PPTs are typically film capacitors.[15] The capacitors chosen for space applications must display good reliability i.e. no aging phenomena or hysterisis over prolonged operation, high specific energy, low equivalent series resistance and inductance, and high peak current pulse capability. Mica capacitors display the required properties for PPT applications, they are able to operate at high frequencies and are insensitive to current reversals, but with current technology cannot be manufactured to conform to the stringent mass and volume requirements of picosatellites. An alternative is the ceramic capacitor which is able to conform to the performance requirements whilst adhering to mass constraints[16]. However, this is still a relatively new application for ceramic capacitors and further testing and characterization is required before they may be accepted as an option for µPPT flight models.

III. Experimental Results

As outlined in the preceding paragraphs, the present experimental effort has in its initial phase the goal to investigate the PPT performance as a function of the electrode geometries. To accomplish this, a simple structure was designed which, due to its modularity, allows a convenient exchange of electrodes and variation of separation. According to the test plan, the electrode geometry and discharge energies used in the initial tests are comparable to values found in literature. Only after verification (if possible) of literature values, it is intended to decrease incrementally the size of the thruster to a level which is considered necessary for a CubeSat application. The following paragraphs summarize these initial steps and presents first test results.

D. Test Facility

The testing and characterization of the µPPT takes place in a high vacuum facility capable of achieving a chamber pressure of 10^{-6} mbar without load, during thruster operation a pressure of the order of 10^{-5} is achieved. The vacuum chamber has dimensions of 0.5 m in length and 0.4 m diameter, ensuring that facility effects are kept to a minimum.

The µPPT test bench was designed to allow maximum flexibility in the thruster configuration. The components of the thruster are easily interchangeable and the electrode geometry may be adjusted in order to investigate a variety of test matrices (see Fig. 4). The thruster design conforms to the standard rectangular geometry with parallel electrodes utilizing solid Teflon® propellant. The test bench allows electrode separations of up to 3 cm to be investigated, which is more than sufficient for applications for the microsatellite class and smaller.

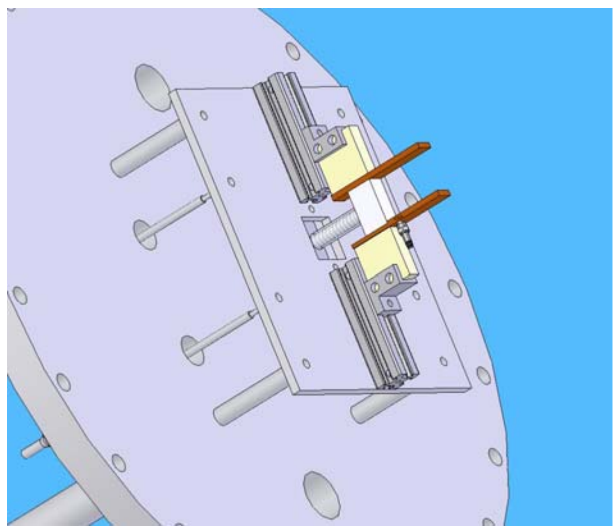


Fig. 4 The µPPT test bench shown mounted on the vacuum chamber flange excluding electrical connections

The spark plug initially utilized for these tests is a MicroViper from Rimfire, USA with a diameter of approximately 5 mm. However, initial tests showed that the discharge reliability of this spark plug decreases with decreasing vacuum pressures. At 10^{-5} mbar the spark plug ceased to function. Instead a Unison spark plug with a diameter of 10 mm was used. Although this spark plug functions well, due to its size it was not possible to assemble it such that its front face is flush with the cathode but instead the spark plug is offset and fires through the cathode aperture. Although this is not a satisfactory solution it is at this point in time the only viable solution.

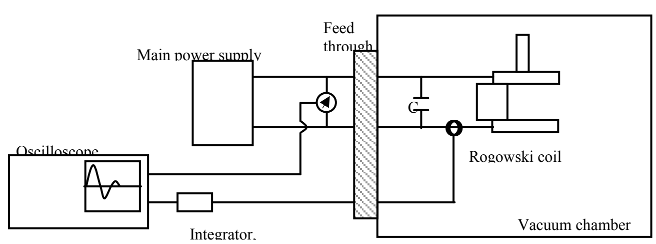


Fig. 5 Schematic of the current measurement system

A digital oscilloscope is used in conjunction with a high voltage probe and current measurement coil in order to evaluate discharge currents and voltages, a schematic of the system used to monitor the µPPT current and voltage is shown in Fig. 5. LabView software provides an interface between the oscilloscope and PC allowing experimental data to be recorded. The current measurement system consists of a twelve turn Rogowski coil and an integrator with an RC constant of 100 µs. The energy storage device used is an oil filled capacitor from Maxwell Laboratories Inc. and has a capacitance of 31.1µF.

The µPPT discharge current curves are analyzed to provide an estimate of impulse bit, I_{Bit}. The circuit parameters are also calculated from the experimental current curves. The I_{Bit} is related to the discharge current via Eq. 2 and is determined by integrating the discharge current curve using a trapezoidal method.

$$I_{Bit} = \frac{L'}{2}\int_0^t i^2 dt \tag{2}$$

Where the inductance gradient L' is approximated by Eq. 3 and expressed in terms of the permeability of free space also known as the magnetic constant $\mu_0 = 4\pi \times 10^{-6}$, the electrode separation h and electrode width w.

$$L' \approx \mu_0 \frac{h}{w} \tag{3}$$

PPT circuit parameters may also be determined from the discharge current curve, assuming that the µPPT electrical system responds as an underdamped LCR circuit results in a discharge current represented by Eq. 4.

$$i = -\frac{V_0}{L}\frac{1}{\sqrt{\frac{1}{LC}-\frac{R^2}{4L^2}}}\exp\left(-\frac{R}{2L}t\right)\sin\left\{\left(\sqrt{\frac{1}{LC}-\frac{R^2}{4L^2}}\right)t\right\} \tag{4}$$

Taking the ratio of the current at the peak current I_1 and the minimum current I_2 with an elapsed time between the maxima and minima of a half period $T = 2\pi(LC)^{0.5}$ the resistance R may be expressed as the following;

$$R = \frac{4L}{T}\ln\left(\frac{I_1}{I_2}\right) \tag{5}$$

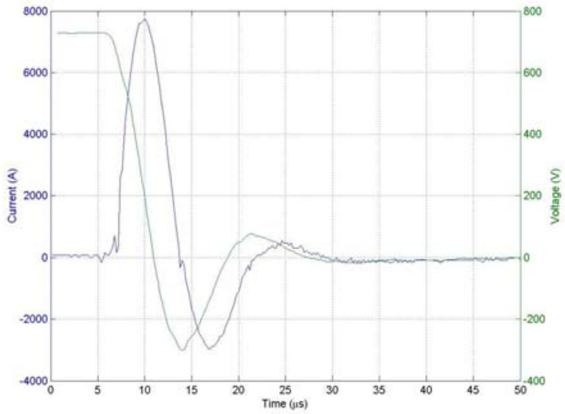
Where the inductance is calculated from the period of the current curve for a known capacitance (see Eq. 6).

$$L = \frac{T^2}{4\pi^2 C} \tag{6}$$

The equations described above were used to determine the µPPT performance characteristics and circuit parameters which are presented in the following section.

E. Preliminary Results

A µPPT with an aspect ratio of 1 was investigated, the electrode configuration of h=10 mm and w=10 mm was selected in order to conform to previously tested geometries and hence provide a basis for the comparison of the µPPT performance with earlier models. The µPPT was tested at discharge energies of 8 J, 4.7 J and 2 J. At higher energies tests were run for a duration of 5000 shots and were repeated twice. The test at 2 J was aborted after approximately 2000 shots due to propellant charring. The fact that charring is observed at 2 J (energy per cm^2 = 2 Jcm^2) indicates that the energy coupling into the plasma is not efficient. It is assumed that this is due to the possible flawed discharge initiation, which may be due to the assembly of the spark plug into the thruster (see discussion in chapter B). Another explanation might be the relative slow current increase in the initial µs of the discharge. This again is due to the excessively high capacitance of the utilized Maxwell capacitor (for the same discharge energy, a capacitor with smaller capacitance results in a faster current increase). For future tests, it is planned to exchange this particular capacitor with a capacitor more appropriate for the small discharge energies used in these tests.



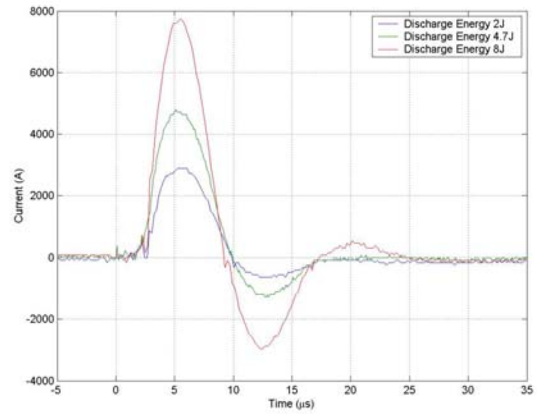


Fig. 6 Representative current and voltage characteristics for the µPPT operating at 8J

Fig. 7 Characteristic current curves for various discharge energies

Figure 6 shows the typical current and voltage characteristics for an 8 J discharge. A peak current is achieved after approximately 5 µs following the discharge initiation at which point the capacitor is fully discharged. The characteristic current curves obtained at different discharge energies are compared in Fig. 7, as expected the peak current observed increases with increasing energy. Figure 8 shows the linear relationship between impulse bit and energy. The higher impulse bits achieved during 8 J and 4.7 J operation correspond to mass bits of m_{Bit} = 4.185 and m_{Bit} = 2.569 µg shot^{-1} respectively. Mass bit calculations were prohibited for 2 J operation due to propellant charring which would result in erroneous values.

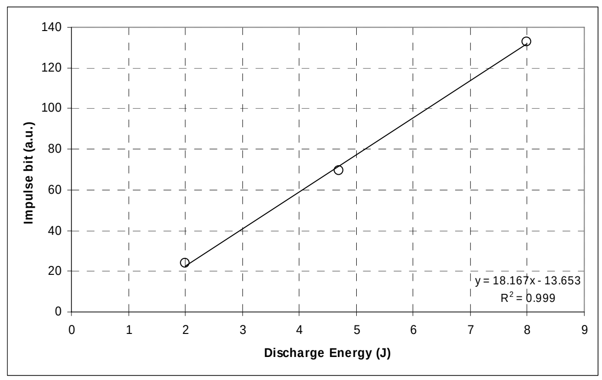


Fig. 8 Impulse bit as a function of discharge energy

The values determined for the µPPT inductance and resistance indicate that performance is improved with increasing discharge energy. But the values shown in Table 2 are excessively high. This is due to the non-optimized fashion of connecting the capacitor with the electrodes, which introduces additional resistance and inductance into the system. However, this is not a concern at this point as the focus of the investigation is the relative change of performance parameters with respect to changing electrode geometries. In order to optimize µPPT performance characteristics the circuit inductance and resistance should be minimized which reduces parasitic losses from the energy storage system, in this way a maximum transfer of energy from the capacitor to the thruster discharge may be achieved.

Energy, J	Resistance, mΩ	Inductance, nH
2	71.8 ± 6.5	181.2 ± 20.2
4.7	54.3 ± 6.7	180.1 ± 20.0
8	50.3 ± 6.4	178.4 ± 25.1

Table 2 Calculated inductance and resistance values

IV. Conclusions and Future Work

The development of a µPPT for application on a CubeSat was recently initiated. Due to the small available mass and volume this development indeed pushes the envelope in the PPT development. Several major challenges such as the discharge initiation, type of energy storage and the PPU design are at this point under investigation but still not yet resolved.

In an initial step a renewed investigation of the electrode geometry was initiated. A µPPT test bench thruster with a propellant surface area of 1 cm^2 was designed and integrated into a new high vacuum facility and operated. Results for the initial electrode geometry have shown that operation at low energy levels results in propellant charring. This result is probably be at least partially due to the fashion of discharge initiation used in these low energy tests. Increasing the discharge energy eradicates charring. It is planned to utilize an in-house made discharge initiation for future tests.

These initial tests provided however a first verification of the test facility and test software. For the envisioned use of a µPPT on-board a CubeSat a significant miniaturization of existing systems has to be performed. The critical components which require further development in order to provide adequate performance for picosatellite applications and adherence to mass restrictions are the energy storage system and, as mentioned, the discharge initiation system.

Further test series will be performed for reduced electrode lengths and varied aspect ratios in order to establish the influence of electrode geometry on thruster performance for miniaturized dimensions. An additional objective will be to gain an insight into the physical processes occurring on such a scale with the aim of applying this knowledge to further enhance thruster performance.

Acknowledgments

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