A Pulsed Vacuum Arc Ion Thruster for SmallSat Applications

IEPC 2019-A-657

Presented at the 36th International Electric Propulsion Conference
University of Vienna • Vienna, Austria
September 15-20, 2019

Jonathan Kolbeck[1] and Michael Keidar[2]
The George Washington University, Washington, DC,20052, USA

Vacuum arc thrusters (VATs) are relatively simple electric propulsion devices based on the much-researched vacuum arc. Currently, this technology is primarily used for basic attitude control of small spacecraft, primarily CubeSats. This technology has already flown on BRICSat-P, BRICSat-2, and Canyval-X. We report on the development of a vacuum arc ion thruster (VAIT) currently in development at The George Washington University, called the "Micro-Cathode Arc Thruster Main Propulsion System", or µCAT-MPS for short. The thruster is designed to be used in 3 U or larger CubeSats as main propulsion system. The thruster uses a VAT as a plasma source, and a pair of grids are used to accelerate the ions electrostatically out of the thruster's chamber. Laboratory measurements show an increase in ion velocity when a 1 kV acceleration voltage is applied. For these measurements, titanium was used as the propellant. The use of other cathode materials will also be investigated, such as Mg, W, Mo, Al, and Nickel. The total power input of the thruster is expected to be in the range of 30-60 W, with most of the power going into ion acceleration. Thrust measurements will be performed on this thruster, and it is expected that it will reach approximately 100 µN. Ongoing research is focused on increasing the efficiency of the ion acceleration process with the grids by increasing their ion transparency.

I. Introduction

Vacuum arc thrusters, or VAT for short, have been suggested as viable technologies to provide thrust since the early 1960s [1-3]. VATs offer a simple way of producing thrust by means of an electric arc, caused by the electrical breakdown between two electrodes thanks to a thin carbon layer on the insulator between both electrodes. The physics and the background of this type of propulsion system have been extensively discussed[4]. Vacuum arc thrusters have been previously flown on various missions, including BRICSat-P[5], BRICSat-2, and CANYVAL-X[6]. This technology offers a low-cost option for attitude control for small spacecraft. BRICSat-P was equipped with 4 thrusters provided by The George Washington University. Telemetry showed that the thrusters were able to detumble the spacecraft from approximately 12 degrees/s per axis to below 1 degree/s per axis within the first 96 hours. The thrusters were not operated continuously, and the CubeSat had no other means of detumbling. BRICSat-2 was launched on STP-2 and is currently sending telemetry. It is expected to receive thruster performance data within the next few months.

II. Basics

The Power Processing Unit (PPU) in VATs is usually an inductive-type energy storage device capable of producing a large initial voltage spike to overcome the resistance of the thin carbon coating. Once breakdown occurs, usually at around 500-600 V depending on the sheet resistance of the carbon coating, the capacitors provide the current necessary to keep the discharge alive. The inductance and capacitance of the PPU are closely related to the maximum discharge current and discharge length. Therefore, these two parameters allow to tune the discharge to the requirements of the

[1] PhD candidate, Mechanical and Aerospace engineering, jkolbeck@gwu.edu
[2] Professor, Mechanical and Aerospace, keidar@gwu.edu

mission, be it a short, high-current discharge or a lower current, longer duration pulse. A simple schematic of a VAT can be seen in Figure 1.

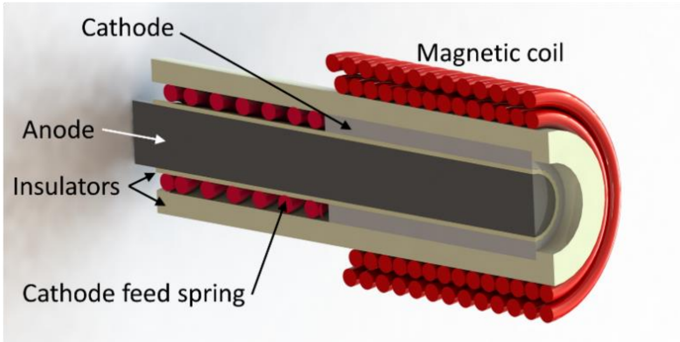

Figure 1: Schematic of a Vacuum Arc thruster.

One concern of using both VAT and VAIT is the triggering of arcs. Therefore, the triggering issue deserves special attention. VATs and VAITs are usually triggered by a "triggerless" or "self-triggering" method [7], and recent efforts focus on developing a better understanding of the arc ignition process [8]. It was shown that the insulator material plays an important role in "triggerless" or "self-trigger" arc ignition. To ignite vacuum arc thrusters, it is common to use booster circuits (Figure 2). The first segment of the circuit depicted by circuit loop 1 consists of a power supply, generally with a voltage between 15 to 25 volts powered either directly by the spacecraft's power supply unit, or stepped up from any of the satellite's bus voltages using DC-DC converters, which charges the capacitor C. When a 5 V square-wave is applied to the insulated-gate bipolar transistor (IGBT), the device "closes" and completes circuit loop 1, which leads to a buildup of the magnetic field of the inductor L. After a charging period defined by the inductance and capacitance of the components and controlled by the length of the pulse-width modulated square pulse, this signal is turned off and the IGBT opens again. The energy stored in the inductor is then released onto the thruster as a high-voltage pulse in the order of LdI/dt, causing breakdown between the two electrodes through a conductive film, and therefore, closing circuit loop 2. This process leads to the production of plasma which expands from the cathode and thereby produces thrust. Theoretically, this process can be repeated at high repetition rates, of hundreds to thousands of pulses per second. Typical repetition rates are below 50 Hz due to the charging time of the capacitor and energy stored in the inductor and the thermal properties of the thruster. For higher repetition rates (higher duty cycle), measures would need to be engineered to remove the dissipated energy. It is worth mentioning that the circuit, due to its transient magnetic fields, is prone to producing a large quantity of electromagnetic interference (EMI). Measures should be taken to shield this PPU from other critical components in a satellite, such as the onboard computer or other sensitive sensors. Moreover, the spacecraft operator should plan accordingly and avoid sensitive

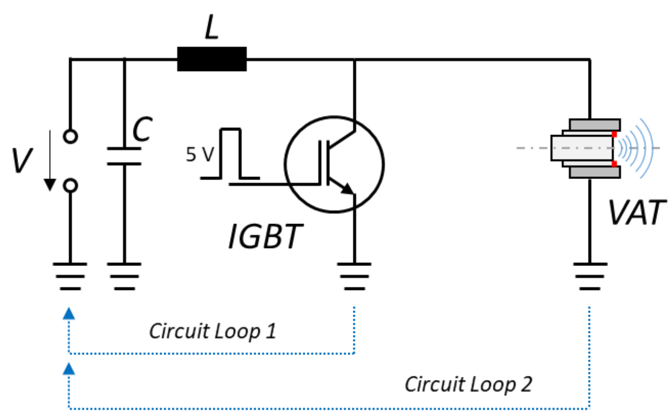

Figure 2. Schematic of a typical booster circuit for a pulsed vacuum arc thruster, where the interruption of circuit loop 1 produces the voltage to ignite the discharge in circuit loop 2.

measurements or radio communications while the thrusters are firing to avoid any issues. The CubeSat BRICSat-P operated in such a way that during propulsive maneuvers, only the vital subsystems were operational. The CubeSat operated without anomalies during these maneuvers.

III. Motivation

Generally, electric propulsion systems are operated in what is called "single-stage" operation, which means that a thruster will have specific capabilities regarding thrust, specific impulse, and efficiency. Some propulsion systems can be throttled and they do indeed change the different parameters mentioned above, but these are not part of the scope of this research. Usually, the propulsion system will be operated at an optimum point, although occasions for variation exist. For specific applications such as orbit raising maneuvers where high thrust is required, a specific thruster will be needed. For maneuvers such as station-keeping, one would prefer a thruster with high specific impulse (I_{sp}). Therefore, two separate single-stage thrusters will be required if both maneuvers are to be performed efficiently. This issue can be addressed by so-called "two-stage" propulsion systems. Research has been performed using a variety of devices to operate thrusters in such a configuration. In a two-stage mode, the ionization stage is separate from the acceleration stage. In single stage operation (i.e. 2nd stage turned off), the thruster provides a high thrust-to-power ratio (TPR) and is ideal for orbital change maneuvers, whereas the two-stage operation will provide a higher specific impulse and is ideal for station-keeping and interplanetary missions, where efficient use of the propellant is more important than thrust. The use of two-stage thrusters would benefit a broad amount of missions, such as all-electric satellites for geostationary applications (high-thrust LEO to GEO transfer, high I_{sp} station-keeping and drift corrections once in orbit). Interplanetary probes and sample return missions to asteroids would also benefit from this development. Such devices are being studied as part of NASA's Electric Propulsion Program[9]. Some examples of two-stage propulsion systems are the P5-2 (5 kW) thruster which was designed and tested at the University of Michigan[10], the Helicon Hall Effect Thruster (600 W – 1200 W)[11], the Linear Gridless Ion Thruster (2 kW)[12], the VASIMR engine (200 kW)[13, 14], the D-80 Thruster with Anode Layer (TAL, 3 kW)[15], and the Very High Specific Impulse Thruster with Anode Layer (VHITAL, 25-36 kW)[16], amongst many others. As an example, the VHITAL thruster yields a TPR of 26 mN/kW and an I_{sp} of 6000 s in single-stage mode, whereas in the two-stage mode, the TPR drops to 19.7 mN/kW but the I_{sp} increases to 8000 s. Single-stage operation in Hall-effect thrusters (HET) and TAL systems is limited to voltages of around 1 kV and after this point, they experience a significant drop in efficiency due to severe anode heating[16]. This can be overcome by separating the ionization and the acceleration regions, i.e. with a two-stage system[16]. These systems, however, have their drawbacks.

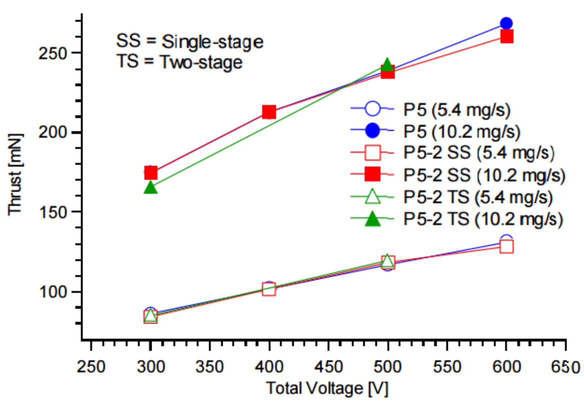

Figure 3. Thruster performance for the P5, P5-2 (single-stage), and P5-2 (two-stage) thruster from the University of Michigan. Figure taken from Hofer et al. Edited to fit this page format.

Two-stage systems tend to be more inefficient than their counterparts at lower voltages. This can be seen in Figure 3. The P5 thruster was developed by Air Force Research Labs (AFRL) and is a 5 kW HET thruster. The P5-2 is the two-stage development by the University of Michigan, also for 5 kW operation. At voltages below 450 V, the P5-2 (single-stage, SS) is equivalent to the P5, since they are practically the same thruster. With these parameters, the P5-2 (two-stage, TS) is slightly more inefficient than the P5 and P5-2 SS. At voltages above this threshold, the P5-2 SS loses performance compared to the P5 due to small changes that were made in the discharge channel but the P5-2 TS has a better performance than both single-stage thrusters. No data is available for values above 500 V due to damage to the thruster[10]. One of the most concerning issues is the higher erosion of the main thruster components such as the electrodes and dielectric components. In two-stage operation, the thruster is subject to higher powers, which are derived from the higher voltages that are applied. Consequently, the kinetic energy of the particles is significantly higher and cause a larger amount of surface sputtering on both metallic and non-metallic surfaces. Prolonged occurrence may lead to failure in the ceramic components, and the plasma could reach other electrical components, resulting in short circuits. Additionally, the higher electric power causes the thruster's temperature to increase, leading to damage or deformation of specific components. Furthermore, incorporating another stage to the system increases the complexity of the system, as more complicated circuitry is required for optimum operation. Another main issue that can be seen by the power at which the thrusters operate (number in parentheses after each thruster) is that none of these thrusters cater to small satellites.

All thrusters mentioned above operate at powers of at least 0.5 kW, a power level that is unfeasible for most small satellites. The literature review currently shows no available two-stage devices for CubeSat applications in the market. To summarize, the foremost technical challenges to date for two-stage systems are the higher component erosion, higher thermal loads, system complexity, and the current unavailability for smaller-sized spacecraft. The proposed solutions for these problems will be described in detail in the remainder of the document. The result will be a propulsion system on which different parameters can be studied to potentially mitigate some of these issues. Additionally, the system will be designed to cater to small satellites.

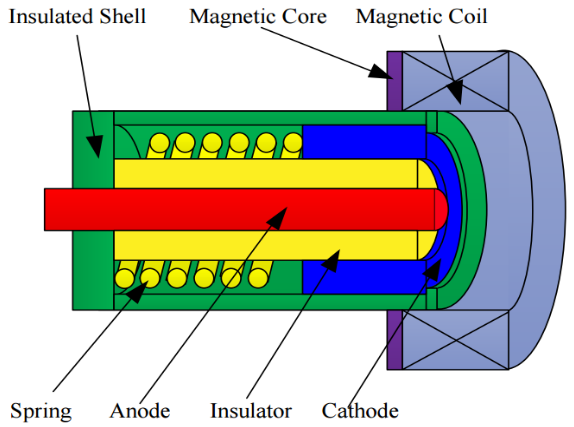

Figure 4. Schematic of a µCAT used for RCS. The size of the device shown is approximately. 2.5 cm in length.

IV. The First Stage

The first stage of the device proposed herein is a modified Micro-Cathode Arc Thruster (µCAT). This produces the plasma that will then be accelerated using an electric field. The µCAT is an electric propulsion system that is based on the well-researched ablative vacuum arc or 'cathodic arc' process[17, 18]. This physical phenomenon is known to erode the negative electrode (cathode) with every discharge. In this case, this is highly desirable as the cathode is the thruster's propellant. During each discharge, a small amount of metallic propellant is eroded, ionized, and accelerated. The efficiency is enhanced by a magnetic field[19]. The magnetic field is caused by the arc current as it travels through a magnetic coil prior to arcing between the electrodes. The device and the main components can be seen in Figure 4. A spring is used as a feed system to ensure that there is always propellant available for the next discharge. The system can be pulsed and therefore, throttled by simply changing the discharge frequency. At a discharge frequency of 10 Hz, the power consumption is approximately 1 W for the current version of the µCAT. The thruster can be pulsed at frequencies between 1 and 50 Hz, depending on the requirements and mission needs. The frequency is limited by the thermal properties of the components used for the casing and most importantly, the feed system's spring, which runs the risk of getting annealed at high temperatures.

The physical nature of the arc discharge allows any conductive (solid) material to be used as a propellant. The eroded material is, to a large degree, fully ionized. Additionally, it is common that the particles are multiply ionized. In the case of tungsten, the mean ion charge state $\bar{Q}$ is 4.6+ and approximately 94% of the ions have a charge state equal or higher than 3+[20].

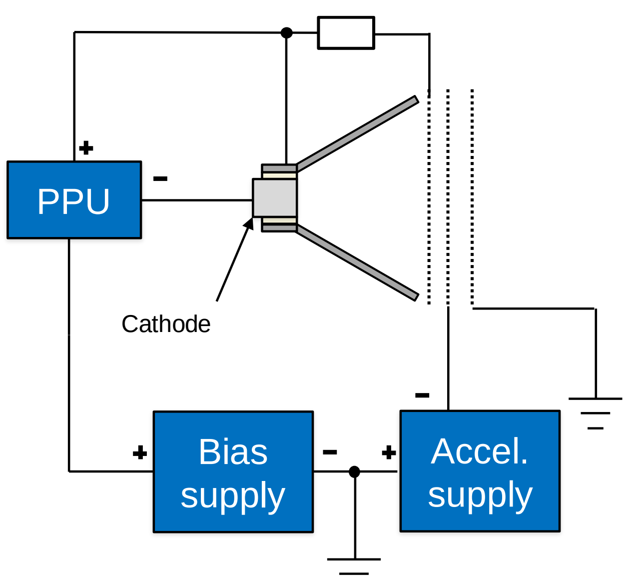

Figure 5: Proposed electrical schematic for thruster operation.

The fact that the discharge can ablate any conductive material allows the thruster to operate with different metals, each with different physical properties, giving the mission designer flexibility when it comes to the mission's design, e.g. nickel will produce a higher thrust compared to titanium, but the latter would offer a higher specific impulse under unchanged discharge conditions. The system does not require any pressurized tanks or other components that may be required when dealing with propellants such as xenon. This is advantageous because it greatly reduces the system's complexity and risk involved. Additionally, only an electrical connection is required to operate the thruster, since the propellant and all necessary components are integrated within the thruster's structure. Therefore, the µCAT technology offers opportunities that are unmatched by most gas-fed propulsion systems, such as the ability and flexibility of attaching the thrusters to deployable booms to increase the torque for RCS maneuvers. To operate the thrusters, a voltage between 15 to 25 volts is required to

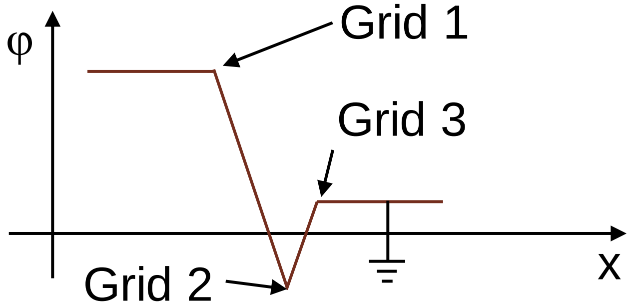

Figure 6: Proposed electrical schematic for thruster operation.

energize the system. The booster circuits convert the energy, producing an instantaneous peak arc discharge of approximately 50 A. This instantaneous current ablates a small fraction of the cathode and ionizes it, producing a quasi-neutral plasma that does not require a neutralizer. The plasma plume is almost fully ionized hence eliminating potential self-contamination due to the charge exchange process in the case of a weakly ionized plasma. Nickel and Titanium cathodes (propellant) have been characterized for the use in this technology and have resulted in specific impulses of 2200 s and 2800 s. The energy consumption is approximately 0.1W/Hz for 2 micro-N-s impulse bits.

V. Thruster and Experimental Setup

In order to accelerate the plasma, it is required to have a large potential difference between the plasma and the grids. It is also desired that the plasma potential is high compared to the local ground. Figure *5* shows the proposed circuit schematic for the thruster system. The PPU is raised/biased to a high positive potential (around 500V to 1kV) with respect to ground by means of a biasing power supply. First grid is connected to the anode of the thruster with a resistor in series, to reduce electron loss on the grid and to reduce Joule heating on the gird. The second grid is biased negatively with respect to the first grid and ground, to create a strong electric field to accelerate the ions and to suppress electrons. The third grid is referenced to ground. Figure *6* shows the potential distribution from the different grid, starting with the plasma potential on the right side. The plasma will assume a potential similar to the potential of the anode, which is also shared by the first grid. Once the plasma goes past the first grid, the electrons are repelled by the strong negative voltage and the ions are accelerated. Once outside the thruster, the ions return to a potential similar to the ground potential. Referencing the PPU to a higher potential is a challenging endeavor because the laboratory power supplies that are normally used are not designed to be floated or biased to high voltages. Furthermore, some of the integrated circuits are not designed to be at high potentials. Therefore, the experiments were split into Phase 1 and Phase 2. Phase 1 consisted of performing basic tests without the use of a bias supply. Due to time constraints, we will only report on Phase 1 results. Phase 2 is scheduled to start within the next week and will include a complete revamp of the discharge circuit to allow it to be biased positively to ground. During Phase 1, the original voltages were used (i.e. the first grid was biased positively relative to ground) as it would be in the actual system. Other grid configurations and voltages were also tested to see if a configuration would

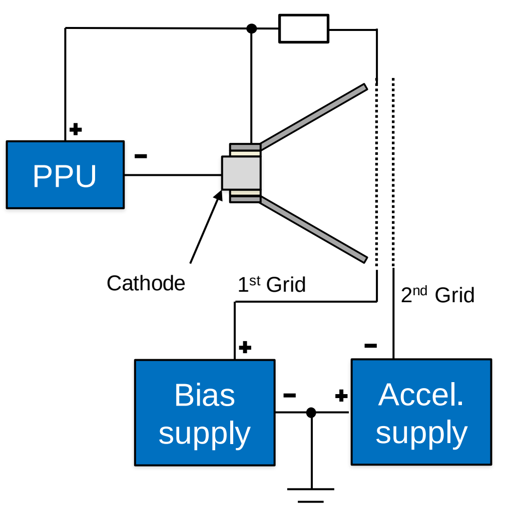

Figure 7. Image of the laboratory model of the gridded µCAT-MPS system inside the chamber.

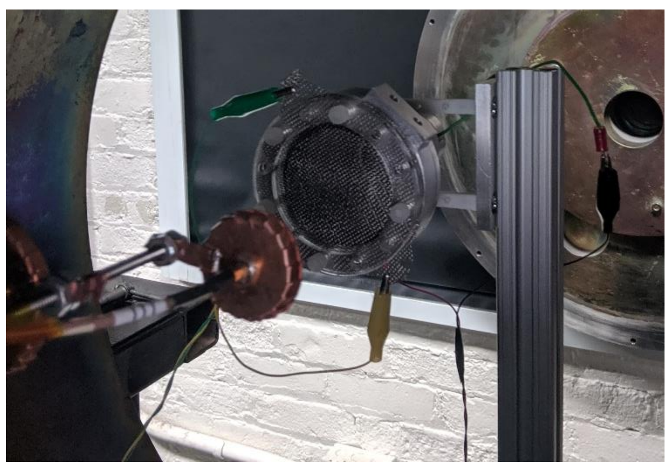

Figure 8. Image of the laboratory model of the gridded µCAT-MPS system inside the chamber.

work without having to bias the PPU. Furthermore, Phase 1 experiments were conducted using only two grids, instead of three. Figure *7* shows the schematic of the Phase 1 testing device.

VI. Phase 1 Test Campaign and Results

Phase 1 experiments are being carried out at Stanford University's Plasma and Propulsion Laboratory. The experiments were carried in a small vacuum chamber with a 60cm diameter and a length of approximately 1 meter with a 60 cm diameter cryogenic pump that is able to bring the pressure down to the low 10^{-6} Torr range to the high 10^{-7} Torr range. The turnaround time after the cryogenic pump has reached nominal temperature is approximately 30 minutes. This allows for experiments to be carried out quickly if something needs to be changed, such as a loose connection or a missing magnet. To measure the total ion current of the thruster, a large graphite plate was placed in front of the thruster's exit. The graphite plate is large so that it catches all the ions coming from the thruster. Time of flight measurements were used with small copper Faraday cups that were separated by 105 mm from each other in front of the thruster. The first detector was at 100 mm from the outermost thruster grid. The cups were previously used for other experiments performed at Stanford[21]. The voltage to the first grid was varied between 0 to 750 V, while the second grid was left at -76V. Figure *13* shows an image of the thruster firint.

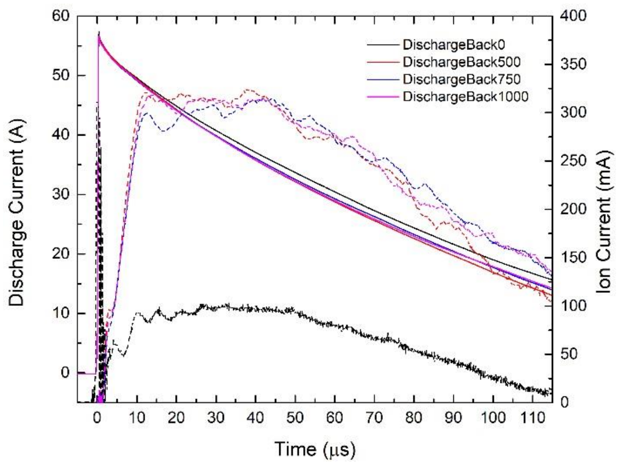

Figure 9. Total ion current measurements with different first grid voltages (100 shot average).

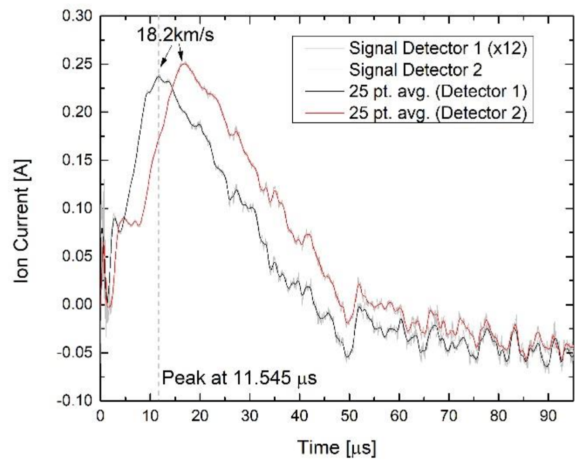

Figure 10. TOF measurement with the first grid biased to 0 V and the second grid to -76V (10 shot average).

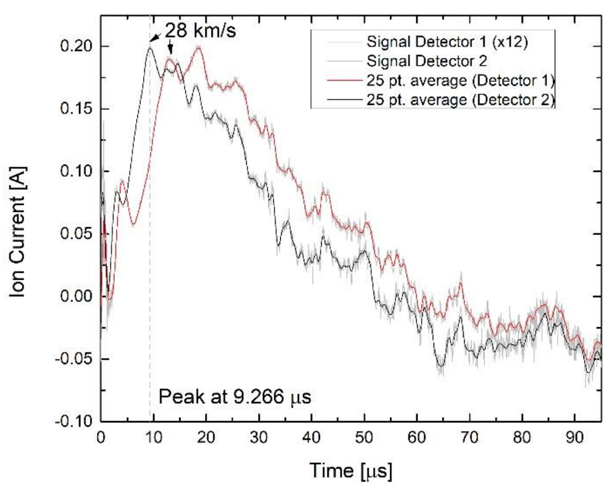

Figure 11. TOF measurement with the first grid biased to 500V and the second grid to -76V (10 shot average).

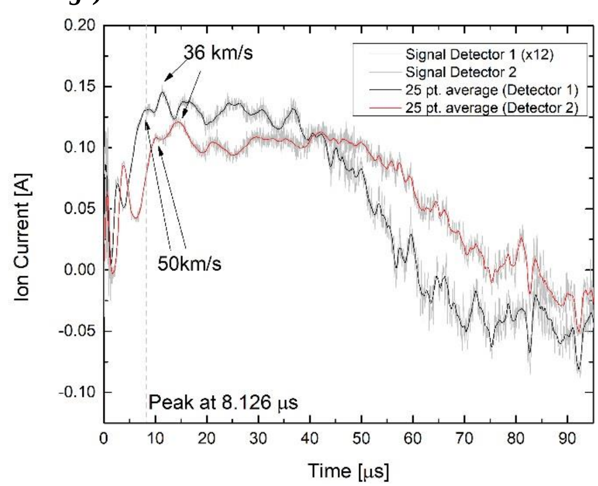

Figure 12. TOF measurement with the first grid biased to 750V and the second grid to -76V (10 shot average).

VII. Discussion and further steps: Phase 2

The time shift between distinct peaks in both signals was used to measure the velocity of the plasma. Since the distance between the detectors is known, it was only necessary to make sure that the peaks corresponded to each other. The data above shows that there seems to be acceleration of ions during the very first few microseconds, as seen on the detectors. Here, the peaks are closest to each other (i.e. the fastest). As time progresses, the distance between matching peaks (between the two signals) the ion velocity drifts back to approximately 18 km/s, which resembles the velocity of ions without any acceleration, i.e. the ions produced by the source. This is explained by the fact that, at the beginning, the ions see the potential difference between the grids and are accelerated, but then, since the first grid is so positively biased compared to the anode, it is a trap for electrons, which saturate the power supply and cause the grid's voltage do drop to virtually 0 V. Therefore, the plasma sees only the second grid, biased to -76V, which acts as an electrons screen but not as an accelerator. To continue with Phase 2, the PPU will have to be biased to the same potential as the first grid to increase the plasma potential.

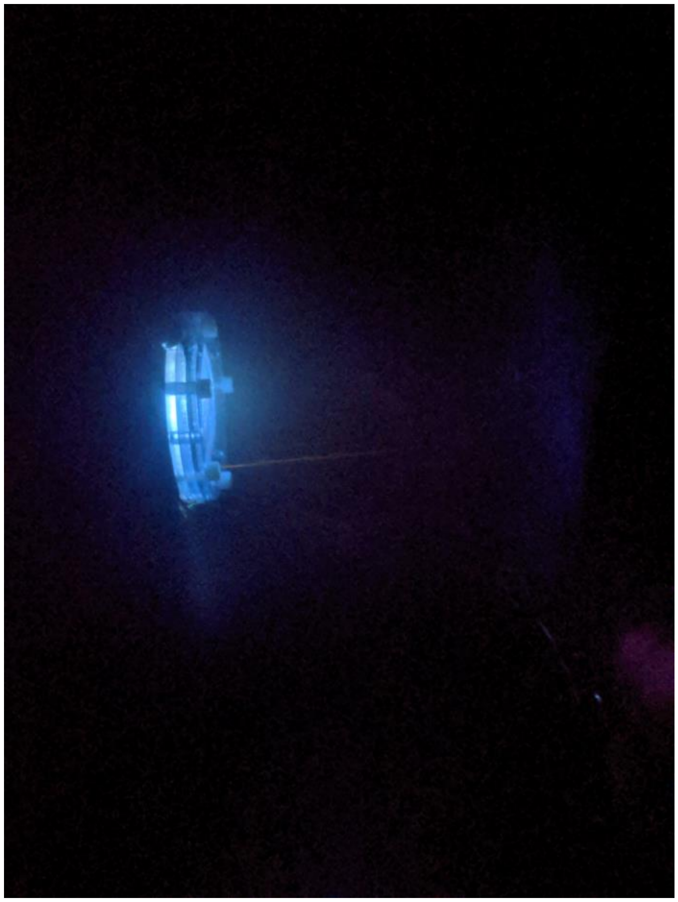

Figure 13. Long exposure image of the thruster firing.

VIII. Conclusion

The development of the Micro-Cathode Arc Thruster at The George Washington University has shifted towards higher-power applications within the small satellite community. The current development focuses on increasing the overall system efficiency of the µCAT system by adding a secondary acceleration stage to the system. This will occur in two phases, with Phase 1 already been completed. This device will go through extensive testing within the next weeks as part of Phase 2 and will be improved with the gained knowledge. This thruster will enable mission designers to push the boundaries of what is currently possible with conventional CubeSat propulsion systems.

Acknowledgments

The authors acknowledge support from NASA DC Space Consortium. Special thanks go to Prof. Mark Cappelli at Stanford University for allowing me to use his facilities for this endeavor.

References

1. Gilmour, A.S. *Concerning the Feasibility of a Vacuum-Arc Thrustor*. in *5th Electric Propulsion Conference*. 1966. San Diego, California.
2. Gilmour, A.S. and D. Lockwood, *Pulsed metallic-plasma generators.* Proceedings of the IEEE, 1972. **60**(8): p. 977-991.
3. Gilmour, A.S., J.R. R. Clark, and V. H, *Pulsed vacuum-arc microthrustors*, in *6th Electric Propulsion and Plasmadynamics Conference*. 1967, American Institute of Aeronautics and Astronautics.
4. Kolbeck, J., et al., *Micro-propulsion based on vacuum arcs.* Journal of Applied Physics, 2019. **125**(22): p. 220902.
5. Hurley, S., et al. *Thruster Subsystem for the United States Naval Academy's (USNA) Ballistically Reinforced Communication Satellite (BRICSat-P).* in *34th International Electric Propulsion Conference and 6th Nano-satellite Symposium, IEPC-2015-37*. 2015. Kobe-Hyogo, Japan.
6. Earth Observation Portal. https://directory.eoportal.org/web/eoportal/satellite-missions/c-missions/canyval-x, *Accessed on June 18, 2016*.

7. Anders, A., J. Schein, and N. Qi, *Pulsed vacuum-arc ion source operated with a "triggerless" arc initiation method.* Review of Scientific Instruments, 2000. **71**(2): p. 827-829.
8. Teel, G., et al., *Discharge ignition in the micro-cathode arc thruster.* Journal of Applied Physics, 2017. **121**(2): p. 023303.
9. Dunning, J.W., S. Benson, and S. Oleson. *NASA's Electric Propulsion Program*. in *27th International Electric Propulsion Conference, Pasadena, CA, US*. 2001.
10. Hofer, R.R., et al. *A high specific impulse two-stage Hall thruster with plasma lens focusing*. in *IEPC-01-036, 27th International Electric Propulsion Conference, Pasadena, CA*. 2001.
11. Shabshelowitz, A., A.D. Gallimore, and P.Y. Peterson, *Performance of a Helicon Hall Thruster Operating with Xenon, Argon, and Nitrogen.* Journal of Propulsion and Power, 2014. **30**(3): p. 664-671.
12. Beal, B.E. and A.D. Gallimore. *Development of the linear gridless ion thruster*. in *37th Joint Propulsion Conference and Exhibit, AIAA Paper*. 2001.
13. Chang-Diaz, F.R., *The VASIMR.* Scientific American, 2000. **283**(5): p. 90-97.
14. Longmier, B.W., et al. *VASIMR® VX-200 Improved Throttling Range*. in *Proc. 48th AIAA/ASME/SAE/ASEE Joint Propuls. Conf*. 2012.
15. Solodukhin, A., et al. *Parameters of D-80 anode layer thruster in one-and two-stage operation modes*. in *International Electric Propulsion Conference, IEPC-01-032, Pasadena, Ca*. 2001.
16. Sengupta, A., et al., *An overview of the VHITAL program: a two-stage bismuth fed very high specific impulse thruster with anode layer*. 2005: Pasadena, CA, US: Jet Propulsion Laboratory, National Aeronautics and Space Administration.
17. Boxman, R.L., D.M. Sanders, and P.J. Martin, *Handbook of Vacuum Arc Science & Technology: Fundamentals and Applications*. 1996: William Andrew.
18. Anders, A., *Cathodic Arcs: From Fractal Spots to Energetic Condensation*. Springer Series on Atomic, Optical, and Plasma Physics. 2009: Springer.
19. Keidar, M., et al., *Magnetically enhanced vacuum arc thruster.* Plasma Sources Science and Technology, 2005. **14**(4): p. 661.
20. Oks, E.M., et al., *Ion charge state distributions in high current vacuum arc plasmas in a magnetic field.* IEEE Transactions on Plasma Science, 1996. **24**(3): p. 1174-1183.
21. Lee, N., et al., *Theory and experiments characterizing hypervelocity impact plasmas on biased spacecraft materials.* Physics of Plasmas, 2013. **20**(3): p. 032901.