An investigation of alternative propellants for pulsed plasma thrusters

IEPC-2019-798

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

William Yeong Liang Ling[1], Xiangyang Liu[2], Mengcheng Huang[3], and Justin Quansah[4]
Beijing Institute of Technology, School of Aerospace Engineering, Beijing, 100081, China

Abstract: Pulsed plasma thrusters are structurally simple forms of electric propulsion and were the first form of electric propulsion to be tested in space. They generally consist of only an energy storage capacitor, propellant, electrodes, and a spark plug. However, while they are structurally simple, the physics behind their operation involves many complicated interrelated phenomena. Understandably, these thrusters have therefore been under investigation for many decades. Nevertheless, their structural simplicity lends itself well towards scalability and makes them ideally suitable for use in micropropulsion for small satellites that can weigh only several kg. On such small satellites, there is currently a critical need for propulsion systems to serve as either main propulsion systems or orbital control systems. At present, the propellant used in pulsed plasma thrusters is typically solid polytetrafluoroethylene (PTFE). Unfortunately, at lower discharge energy levels such those necessitated by the power limitations of micro/nano-satellites, PTFE has a tendency to exhibit carbon deposition, which can ultimately lead to thruster failure. In this new era of small satellites, it is important to consider alternative propellants in the miniaturization of pulsed plasma thrusters. While the use of PTFE has become almost standard in the last several decades, there is also some promise in re-investigating previously considered propellants in the context of application in micropropulsion. Some of these propellants were discarded in favor of PTFE due to performance factors such as a lower impulse bit, but for certain applications in micropropulsion, this may result in a higher specific impulse that may actually be preferable in certain missions.

I. Introduction

The miniaturization of technology has enabled the development of devices such as smartphones that are much smaller than their predecessors but with significantly more computing capabilities. In recent years, this advantage has also been extended to satellite systems. While conventional satellites such as weather satellites and those that make up the Global Positioning System (GPS) can weigh several thousand kilograms, the miniaturization of technology has enabled the development of small satellites that can weigh as little as 1 kg. Of these, the CubeSat architecture is a common miniaturized satellite architecture where 1 unit (1U) corresponds to dimensions of 10 × 10 × 10 cm.[1] The typical weight of a 1U CubeSat is around 1 to 2 kg and multiple units can be combined together to form progressively larger systems such as 2U, 3U, 6U, etc.

It has been estimated in the SpaceWorks 2019 Nano/Microsatellite Market Forecast that around 2000 to 2800 micro/nano-satellites (1 to 50 kg) will require launch over 5 years. There has also been a significant increase in commercial adoption, with the most popular weight range being 4 to 6 kg. Such satellites are used in applications such

[1] Associate Professor, School of Aerospace Engineering, wling@bit.edu.cn (corresponding author).
[2] Associate Professor, School of Aerospace Engineering, liuxy@bit.edu.cn.
[3] Graduate student, School of Aerospace Engineering.
[4] Graduate student, School of Aerospace Engineering.

as downstream data analytics and communications. They are also being used in satellite constellations, with many satellites acting together to deliver a service.

Currently, most of these satellites are delivered to orbit with no propulsion systems. Within Earth orbit, a micropropulsion system can be used to enable spacecraft to maneuver, extend their orbital lifetimes, perform orbital changes, and to deorbit at the end of their lifetime. However, while it has been possible to miniaturize electronic systems to create these small satellites, it has proven much more challenging to develop suitable efficient micropropulsion systems. Typical electric thrusters such as Hall thrusters and ion thrusters that are used for larger satellites either do not function or exhibit extremely low efficiencies when scaled down in size.[2]

One promising micropropulsion system for these small satellites is the pulsed plasma thruster (PPT).[3,4] PPTs are a simple form of space propulsion that have been studied since the dawn of the space age and have also seen actual use in space. They were the first form of electric propulsion to fly in space (on the Zond-2) and have significant flight her-itage due to their structural simplicity.[5] This structural simplicity also makes them ideally suited for miniaturization. More recently, several PPTs have been developed around the world specifically for use in CubeSats.[6–8] Companies such as Busek Company, Inc., Clyde Space Ltd., and Mars Space Ltd.[9,10] have also developed PPT CubeSat modules. However, all of these thrusters still use a typical solid fluoropolymer called polytetrafluoroethylene (PTFE) as propellant. While the vast majority of PPTs in research and development currently use PTFE, gaseous, liquid, and solid propellants have all been studied in the past, and further research is still ongoing. Here, we will briefly discuss the propulsion needs in micropropulsion applications such as on CubeSats. We will also cover a brief history of propellant types that are suitable for micropropulsion applications.

II. The Pulsed Plasma Thruster

A. History

The first PPT spaceflight was on the Soviet Zond-2, where it was used as the actuators of the altitude control system.[5] While the mission itself was intended to perform a flyby of Mars, communication was eventually lost with the spacecraft. Nevertheless, the PPTs were successfully tested and operated in space before the loss of the spacecraft. In the US, the first PPT flight was achieved on the LES-6 satellite, where solid PTFE propellant was used instead of gas.[3,11] This was because the initially conceived gas-fed PPT design failed due to the mechanical metering valve. Most modern PPT systems today have many things in common with the thrusters used on the Zond-2 and LES-6 spacecraft. However, due to the need for further miniaturization and to improve the thruster efficiency, research into other propellants is currently still ongoing.

B. Operation

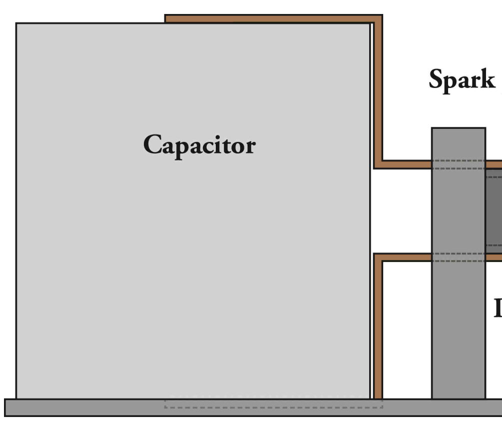

Figure 1. Schematic of a typical PPT with parallel-plate electrodes. A circuit diagram for the spark plug is omitted for simplicity. Figure adapted from ref. 12.

A typical PPT is structurally simple, and typically consists of electrodes, a main capacitor, propellant, and a spark plug to initiate the main discharge process. A schematic of a typical PPT with parallel-plate electrodes is shown in Fig. 1. Energy is stored within the main capacitor, with typical charging times being in the order of 1 s and the corresponding pulse cycle in the order of 1 Hz. The discharge time of the capacitor is typically in the order of microseconds, resulting in an instantaneous power in the order of MW, much higher than the average power consumption level in the order of W.

The PPT discharge is initiated by a spark plug, with the spark completing the circuit between the electrodes. A discharge arc then forms between the electrodes and ablates and ionizes the propellant to produce plasma. A self-induced magnetic field results from the discharge current passing through the plasma, producing a Lorentz force acting in the downstream direction that accelerates the plasma to produce thrust. Figure 2(a) shows a side-view time-integrated broadband image of a single PPT discharge, with the discharge arc and plasma plume clearly visible. A front-view of the plasma is shown in Fig. 2(b).

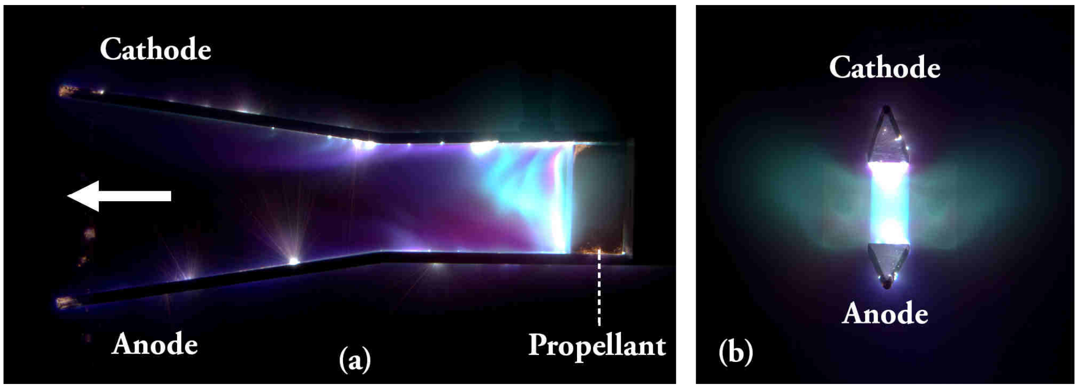

Figure 2. (a) Side-view and (b) front-view time-integrated broadband emission from a single PPT discharge. The plasma plume is accelerated in the direction of the arrow in (a) and out of the page in (b). Figure adapted from ref. 12.

PPTs can generally be divided into two configurations based on their electrode geometry: coaxial and parallel-plate electrodes.[3] In general, the parallel-plate configuration shown in Fig. 1 and Fig. 2 will have a greater electromagnetic thrust contribution while the coaxial configuration will be dominated by electrothermal thrust as it is better able to confine and expand heated neutral gas. As the discharge energy of the thruster decreases, the proportion of electrothermal thrust from a PPT will increase, suggesting that coaxial configurations may be more suitable in low-power low-energy systems such as in micropropulsion applications.

III. Requirements in Micropropulsion

Micropropulsion systems can be used for spacecraft orientation as well as main propulsion. Thrusters that are used only for orientation or station-keeping can have a lower thrust and a lower total Δv capability. Conversely, a main propulsion system may be required to perform significant orbital changes, requiring a larger total Δv, which can be up to several 100 m/s. In order to escape Earth orbit, a capability in excess of 1 km/s is then required, and interplanetary transfers would require several km/s. Compared with station-keeping, these applications will require a significantly larger propellant mass, which will then increase the mass that has to be reserved for the propulsion system.

Small satellites such as CubeSats are severely limited in mass and power, with typical power budgets being in the order of W. This can severely limit the propulsion options available. While temporary increases in peak power can be achieved using onboard batteries, this will further strain the mass limitations of CubeSats. If we are to consider the performance of a micropropulsion system, we need to consider the balance between the specific impulse and the thrust or impulse bit of the system. Generally, an increase in thrust will decrease the time required to perform orbital maneuvers, but will result in decreased performance and a lower specific impulse. In order to deliver a given total thrust or Δv, more propellant will then be needed due to higher propellant mass inefficiencies. Conversely, if the time required for a maneuver can be increased, a system with lower thrust and higher specific impulse can then be used, reducing the total propellant mass required.

If we consider the severe mass limitations of nanosatellites, it may be more beneficial to have a system with a higher specific impulse at the expense of thrust. With PPTs, this can be achieved by either optimizing the system towards a certain configuration, or by using propellants other than PTFE, the latter of which will be discussed here. Different propellants will produce different specific impulse vs. impulse bit behaviors, and the ideal propellant can then be selected based on the specific mission requirements.

IV. Pulsed Plasma Thruster Propellants

A. Polytetrafluoroethylene

As mentioned previously, the most common propellant in use today is solid PTFE. As such, it can be expected that there has been significant research done regarding it. PTFE is attractive due to its performance in PPTs, ease of storage, and the lack of a need for valves, injectors, heaters, etc. Conversely, the primary drawbacks of PTFE are the need for a mechanical propellant feeding system (which may become significantly complex for larger propellant masses), carbon deposition on the propellant surface leading to thruster failure, and late-time ablation resulting in particulate emission and a decrease in the propellant utilization efficiency.

For the PPT on the LES-6 satellite, qualification testing of the thruster included accelerated life cycle testing at an operation rate of up to 4 Hz for the flight model prototype.[11] The number of pulses demonstrated was 10^7, which was the order of magnitude required by the satellite's mission.

At the University of Stuttgart in Germany, the ADD SIMP-LEX electromagnetic PPT was stress-tested with PTFE propellant.[13] The carbon in PTFE propellant was observed to have a tendency to build up in structured surface layers that deposits on both the propellant surface and the electrodes. On the electrodes, they will reduce the effective electrode surface area available for discharge. The mass bit was also found to increase with thruster operation, with the highest measured mass bit surpassing predictions by almost 50%. Unusual carbon deposition was observed, and was concluded to be due to insufficient PTFE propellant bar alignment, leading to thermal deformation at the propellant tip. This demonstrates a mechanical limitation to the use of solid propellants.

For lower discharge energy levels, tests on the μ-Lab SAT II (50 kg) at the Tokyo Metropolitan Institute of Technology showed that there was a sensitivity of the carbon deposition process with the pulse energy.[14] Operation below 3.6 J resulted in the PTFE propellant feeding system being susceptible to instability, causing non-uniform ablation of the propellant. Carbon deposition was found to affect sustained thruster operation at certain conditions. The tendency of PTFE propellant to result in thruster failure due to carbon deposition was also demonstrated with testing on the Air Force Research Laboratory coaxial MicroPPT by Keidar et al.[15] Here, carbon deposition was observed to develop over time, especially during operation at lower discharge energies. This is especially crucial for applications in micro/nano-satellites where the available power is severely limited. The cause of the carbon deposition was suspected to be carbon flux returned from the plasma rather than the incomplete decomposition of the PTFE propellant. It is believed that at higher discharge energies, there is a higher PTFE surface temperature and higher ablation rate, resulting in the prevention of carbon build-up. However, as micro/nano-satellites are typically limited by available power, this suggests that carbon deposition with PTFE will become a more serious problem as PPTs are scaled down in size.

It is important to note that the developmental history of PTFE as a PPT propellant is largely based on applications in larger satellites with more available power. For example, the LES satellite series had a mass of around 150 to 200 kg, and the measured weight of the PPT system was around 7 kg. The discharge energy of the LES 8/9 flight thrusters was around 20 J, and the thrusters were fired at 0.5 to 1 Hz, suggesting an operating power requirement of 20 to 40 W. Another example would be the PPT on the 600 kg EO-1 satellite, which had a discharge energy around twice that of the LES PPT and weighed around 5 kg.[16] These values are around an order of magnitude higher than what is needed for micro/nano-satellites and CubeSats. Nevertheless, almost all modern PPTs for CubeSats still use PTFE as a propellant by default due to its research maturity. In the current era of miniaturized satellites, it is important to reassess its viability and to consider alternative propellants.

B. Ethylene tetrafluoroethylene

Palumbo and Guman examined other thermoplastics as possible alternatives to PTFE very early during the development of PPTs.[17] However, most of these were found to exhibit significant carbon formation during thruster operation, resulting in a noticeable decrease in the impulse bit as a function of the operation time. For some thermoplastics, the exposed surface was completely black after approximately 1000 discharges. The most promising alternative was ethylene tetrafluoroethylene (ETFE), which was found to have a specific impulse around 60% higher than that of PTFE. The reason for this is possibly due to the replacement of two fluorine atoms by two hydrogen atoms when compared with PTFE. However, none of the tested alternatives produced a thrust-to-power ratio that was comparable with that of PTFE. Nevertheless, depending on the specific mission requirements of micro/nano-satellites, a higher specific impulse may be preferable to a higher thrust to power ratio. As ETFE is a solid propellant similar to

PTFE, it is simple to implement in existing PPT designs. However, it is likely to exhibit different discharge characteristics and plasma parameters, thus requiring further investigation in the future.

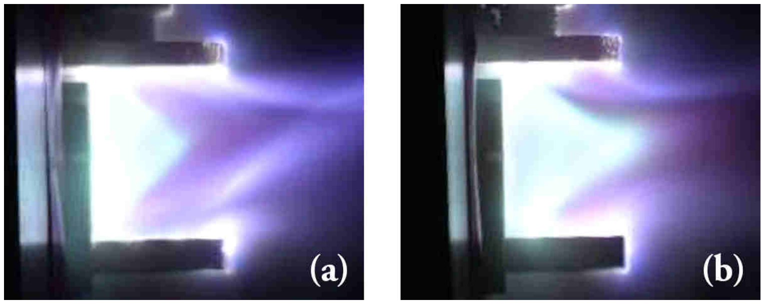

Figure 3. Time-integrated discharge images of a PPT with (a) PTFE and (b) ETFE propellant. The discharge energy is 15 J for both propellants.

Side-views of the discharge plasma profiles for PTFE and ETFE can be seen in Fig. 3. These propellants were tested on an identical PPT, with the images showing discharge energies of 15 J. While the profiles are qualitatively similar, the discharge arc for ETFE extends notably further than that of PTFE, even reaching past the exit of the thruster at the greatest extent.

C. Non-volatile liquid perfluoropolyether

Liquid propellants have long been studied for use in PPTs. These can be divided into liquid metals (e.g., mercury, gallium, lithium, etc.) and other liquids (e.g., water, methanol, etc.) Unfortunately, these generally require complex propellant storage and feeding systems that run contrary to micropropulsion requirements. More recently however, non-volatile liquids have received some attention as a PPT propellant. The volatility of a liquid refers to its tendency to vaporize from a liquid to a gaseous state at a given background pressure. A useful quantification of the volatility is the vapor pressure of a liquid. This is the pressure produced by a vapor in thermal equilibrium with its solid or liquid state at a given temperature inside a closed system. It is related to the tendency of particles to escape from the liquid or solid, i.e., its evaporation rate. All liquids will boil when the background pressure is equal to their vapor pressure.

The pressure levels in experimental vacuum chambers are typically in the order of 10^{-3} to 10^{-5} Pa. Furthermore, the background pressure in deep space is even lower, with the local environment around the Rosetta spacecraft having a pressure in the order of 10^{-9} Pa during cruise and 10^{-8} Pa during thruster operation.[18] Non-volatile liquid propellants are propellants that can be considered to be non-volatile even under vacuum pressure conditions. These can have extremely low pressures ranging from 10^{-4} Pa to negligible values.

One type of non-volatile liquid propellant for PPTs are liquids from the perfluoropolyether (PFPE) family. They were first studied as part of the Liquid Micro Pulsed Plasma Thruster FP7 project in Europe.[19] Figure 4 shows an image of liquid PFPE in a beaker in comparison with solid PTFE. PFPEs are non-hazardous, have excellent chemical stability, are stable over a wide temperature range, and have almost negligible evaporation in vacuum at temperatures of up to 100 to 150 °C. They are available in various grades, with some having a vapor pressure that is even lower than that of the environment around a spacecraft in deep space. Due to this, there is no need for pressurization and propellant storage and delivery are dramatically simplified when compared with other liquid propellants.

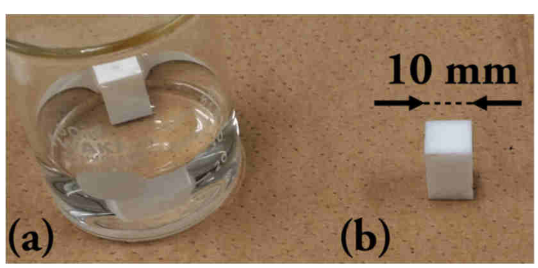

Figure 4. (a) Liquid PFPE propellant in a beaker with a solid porous ceramic block, and (b) solid PTFE propellant. (From ref. 12).

To simplify the propellant delivery system, a passive flow method using a porous ceramic has been investigated for PFPEs.[12,20] This method further simplifies propellant delivery as the propellant is automatically replenished by capillary forces. An experimental thruster with PFPE was operated at The University of Tokyo with energy levels of 5 to 45 J. The carbon deposition resistance of PFPE was studied over several thousand discharges, with liquid PFPE exhibiting no carbon deposition at all (Figure 5). This confirms that one of the major failure mechanisms of PPTs is completely absent when using liquid PFPE as a propellant.

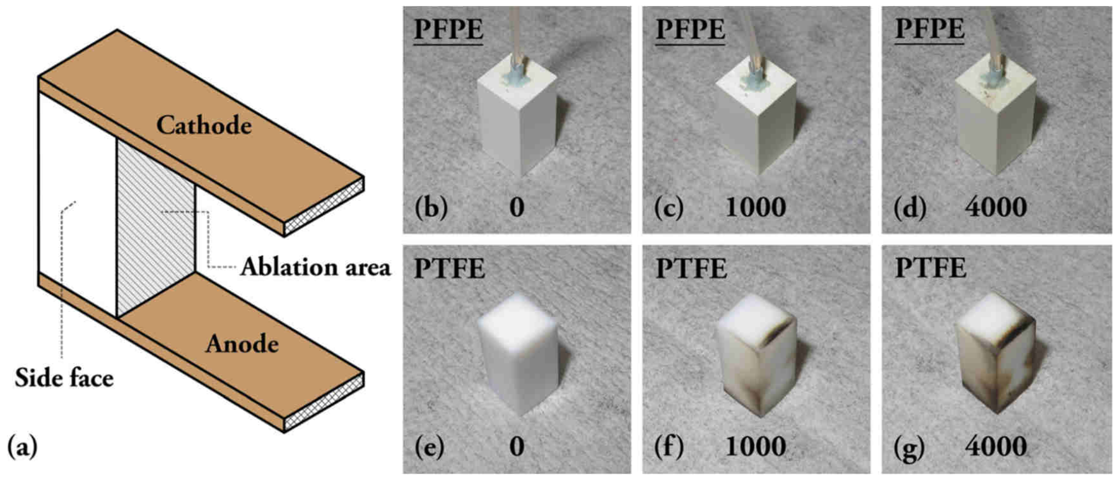

Figure 5. (a) Orientation of the propellant block with respect to the electrodes. PFPE-impregnated ceramic after (b) 0, (c) 1000, and (d) 4000 shots, and PTFE after (e) 0, (f), 1000, and (g), 4000 shots at a discharge energy of 7.5 J. (From ref. 12).

D. Electric solid propellant

Electric solid propellants (ESPs) are an emerging area of research with potentially major implications in propulsion. The application of electric power to an ESP results in the solid propellant being ignited and exothermically decomposed. This continues until the power is removed. The continued development of ESPs for rocket propulsion applications resulted in the production of a formula that achieves a higher specific impulse and conductivity compared with earlier formulations. This is referred to as HIPEP, or higher-performance electric propellant. It is a hydroxyl-ammonium-nitrate (HAN)-based energetic material. The plasma plume of a PPT using HIPEP as the propellant was recently studied.[21,22] The impulse bit was found to vary significantly from pulse to pulse and thruster to thruster. Variations of up to 40% over the thruster lifetime, and 60% from thruster to thruster were observed, with the main suspected cause being inconsistencies in the propellant production process. The plume was found to be weakly ionized gas consisting of high-temperature electrons in low relative temperature ions and neutrals, indicating that the acceleration mechanism is dominated by electrothermal energy. Thermal inefficiency was identified relative to PTFE PPT plasma plumes, despite a stored energy of 40 J that is in the upper range of PPT operation. This suggests that most of the thermal energy from the discharge is deposited into the thruster components instead of the propellant and plasma. In summary, HIPEP behaves similar to PTFE, but with a larger mass bit, with the mass bit per ablation area being more than an order of magnitude above typical PPTs (~790 $\mu g/cm^2$ vs. 1 to 50 $\mu g/cm^2$). However, the electron temperature, electron density, and ionization fraction are lower when compared to PTFE.

V. Discussion

The last several decades have seen the use of propellants in PPTs ranging across the entire physical spectrum of gases, solids, and liquids. While initial experiments were performed using gas-fed PPTs, these were eventually limited technologically by the lack of suitable fast-acting valves with reaction times suitable for the extremely short PPT discharge times. In the context of micropropulsion, the requirement for these additional components increases the challenges in miniaturization. Eventually, focused shifted to the use of solid propellant, of which PTFE was found to be the most promising due to its higher thrust-to-power ratio. Since then, PTFE was and is still considered to be the "gold standard" propellant for PPTs, and almost all designs and flight models have used some form of PTFE as the propellant.

However, while PTFE better resists carbon deposition and charring compared to other solid propellants, it is still prone to the problem, especially at the lower discharge energies at which micropropulsion systems must operate. This is one of the major limitations in the ultimate lifetime of a PPT. Larger propellant masses also require an increasingly complex feeding system due to the solid nature of the propellant. Due to these drawbacks, research was also conducted using liquid propellants. However, the issue of their stability in vacuum and leakage issues had not been resolved until the very recent introduction of non-volatile liquid propellants. These potentially have the capability to replace solid PTFE in certain PPT applications in the near future.

With respect to micropropulsion, the fundamental requirement for any system is simplicity and the ability to be miniaturized. When considering this, solid propellants are the most suitable due to their lack of a need for secondary propellant systems. Non-volatile liquid propellants with passive feeding systems also exhibit similar advantages. However, the use of gaseous or volatile liquid propellants will necessitate secondary systems such as pressurized tanks and feeding valves, increasing system complexity and the number of challenges associated with miniaturization. The most suitable propellant for a micropropulsion PPT system will depend on the mission requirements. While non-volatile liquid PFPE shows promise, compared with solid propellant designs, it is still earlier in the development phase, with further study required to identify the ideal use-case scenarios and implementation process. With solid propellants, several alternatives to PTFE such as ETFE and electric solid propellants are available. These have different advantages and disadvantages, and are also not as well studied as PTFE.

ETFE exhibits a higher specific impulse but a lower thrust-to-power ratio when compared with PTFE. As mentioned in the introduction, this trade-off may be advantageous in certain applications since mass is a premium on micro/nano-satellites. For missions with higher Δv requirements, the propellant mass fraction may become a significant part of the propulsion system. In these situations, a higher specific impulse may then become more desirable as it will reduce the amount of propellant required to deliver a given Δv. For applications such as spacecraft orientation or station-keeping, the propellant mass fraction will be significantly smaller, meaning that specific impulse may not then be a primary concern.

Chemical propellants such as electric solid propellants are also attractive in that they combine the chemical release of thermal energy with a coaxial configuration to supplement electrothermal thrust. However, as with the alternatives to PTFE, they are also earlier in the development phase and have exhibited significant variation between propellant samples and higher thermal inefficiencies compared with PTFE. Further studies are required to properly characterize their performance and ideal use-cases.

As PTFE has been the “gold standard” PPT propellant for several decades, it has been studied in much more detail than other alternatives. However, with the advent of micro/nano-satellites, it has become necessary to consider alternatives in the miniaturization of the PPT. In the future, it will be necessary to understand the ablation process and plasma parameters of these new propellants in order to best utilize them.

VI. Concluding Remarks

The miniaturization of electronics has enabled the development of micro/nano-satellites weighing as little as several kg, and the cost of development and access to space has decreased by several orders of magnitude. With the development of suitable micropropulsion systems, the capabilities of these micro/nano-satellites can be significantly extended, possibly even putting interplanetary missions within reach. The long history of PPT development and their inherent simplicity make them ideally suitable as micropropulsion systems. Current research in alternative propellants further open the door to PPTs that can be tailor-designed for specific mission requirements. With such propellants available, it will then be possible to develop a wide range of micropropulsion systems based on PPTs to meet particular mission requirements, ranging from a high specific impulse to a high thrust-to-power ratio.

It is apparent that micro/nano-satellites have different propulsion requirements compared with conventional satellites. However, the PPT has been developed over several decades with the primary focus being implementation in larger satellites. With the advent of small satellites in the kg weight level, it has become necessary to revisit the adoption of PTFE as the propellant of choice in miniaturized PPT systems.

Acknowledgments

This work was partially supported by the National Natural Science Foundation of China [No. 11802022] and the Beijing Institute of Technology Research Fund Program for Young Scholars.

References

[1]Crusan, J. and Galica, C., “NASA’s CubeSat Launch Initiative: Enabling broad access to space,” *Acta Astronautica*, Vol. 157, 2019, pp. 51-60.

[2]Schönherr, T., "Plasma Acceleration in Ablative Pulsed Plasma Thrusters," *Encyclopedia of Plasma Technology*, edited by J. Shohet, Taylor & Francis, New York, 2016, Vol. 2, pp. 1452-1461.

[3]Burton, R. L., and Turchi, P. J., "Pulsed Plasma Thruster," *Journal of Propulsion and Power*, Vol. 14, No. 5, 1998, pp. 716-735.

[4]Zhang, Z., Ling, W. Y. L., Tang, H., Cao, J., Liu, X., Wang, N., "A review of the characterization and optimization of ablative pulsed plasma thrusters," *Reviews of Modern Plasma Physics*, Vol. 3, No. 1, 5.

[5]Kazeev, M. N., Kozubskiy K. N., Popov G. A., "Victor Khrabrov – Pioneer of the First Space Electric Propulsion System Development and Space Tests," *Proc. 31st International Electric Propulsion Conference*, IEPC-2009-235, The Electric Rocket Propulsion Society, Michigan, USA, 2009.

[6]Pottinger, S., Krejci, D., Scharlemann, C., "Development of a µPPT for CubeSat Applications," *44th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit*, AIAA-2008-4532, American Institute of Aeronautics and Astronautics, Reston, Virginia, 2008.

[7]Coletti, M., Guarducci, F., Gabriel, S. B., "A micro PPT for Cubesat application: Design and preliminary experimental results," *Acta Astronautica*, Vol. 69, No. 3-4, 2011, pp. 200-208.

[8]Krejci, D., Seifert, B., Scharlemann, C., "Endurance testing of a pulsed plasma thruster for nanosatellites," *Acta Astronautica*, Vol. 91, 2013, pp. 187-193.

[9]Ciaralli, S., Coletti, M., Gabriel, S. B., "Performance and lifetime testing of a pulsed plasma thruster for Cubesat applications," *Aerospace Science and Technology*, Vol. 47, 2015, pp. 291-298.

[10]Ciaralli, S., Coletti, M., Gabriel, S. B., "Results of the qualification test campaign of a Pulsed Plasma Thruster for Cubesat Propulsion (PPTCUP)," *Acta Astronautica*, Vol. 121, 2016, pp. 314-322.

[11]Guman, W. J. and Nathanson, D. M., "Pulsed plasma microthruster propulsion system for synchronous orbit satellite," *Journal of Spacecraft and Rockets*, Vol. 7, No. 4, 1970, pp. 409-415.

[12]Ling, W. Y. L., Schönherr, T., and Koizumi, H., "Characteristics of a non-volatile liquid propellant in liquid-fed ablative pulsed plasma thrusters," *Journal of Applied Physics*, Vol. 121, 2017, 073301.

[13]Lau, M. and Herdrich, G., "Pulsed Plasma Thruster Endurance Operation Stress Testing at IRS," *Proc. 33rd International Electric Propulsion Conference*, IEPC-2013-255, The Electric Rocket Propulsion Society, Washington, D. C., 2013.

[14]Igarashi, M., Kumagai, N., Sato, K., Tamura, K., Takegahara, H., Okamoto, H., et al., "Performance Improvement of Pulsed Plasma Thruster for Micro Satellite," *Proc. 27th International Electric Propulsion Conference*, IEPC-01-152, The Electric Rocket Propulsion Society, Pasadena, CA, USA, 2001.

[15]Keidar, M., Boyd, I., Antonsen, E. L., Gulczinski III, F. S., Spanjers, G. G., "Propellant Charring in Pulsed Plasma Thrusters," *Journal of Propulsion and Power*, Vol. 20, No. 6, 2004, pp. 978-984.

[16]Zakrzwski, C., Benson, S., Sanneman, P., Hoskins, A., "On-Orbit Testing of the EO-1 Pulsed Plasma Thruster," *38th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit*, AIAA-2002-3973, American Institute of Aeronautics and Astronautics, Reston, Virginia, 2002.

[17]Palumbo, D. J. and Guman, W. J., "Effects of Propellant and Electrode Geometry on Pulsed Ablative Plasma Thruster Performance," *Journal of Spacecraft and Rockets*, Vol. 13, No. 3, 1976, pp. 163-167.

[18]Schläppi, B., Altwegg, K., Balsiger, H., Hässig, M., Jäckel, A., Wurz, P., et al., "Influence of spacecraft outgassing on the exploration of tenuous atmospheres with in situ mass spectrometry," *Journal of Geophysical Research: Space Physics*, Vol. 115, 2010, A12313.

[19]Szelecka, A., Kurzyna, J., Daniłko, D., Barral, S., "Liquid micro pulsed plasma thruster," *Nukleonika*, Vol. 60, No. 2, 2015, pp. 257-261.

[20]Ling, W. Y. L., Schönherr, T., and Koizumi, H., "Discharge characteristics of an ablative pulsed plasma thruster with non-volatile liquid propellant," *Applied Physics Letters*, Vol. 111, No. 1, 2017, 014101.

[21]Glascock, M. S., Rovey, J. L., Williams, S., Thrasher, J., "Plume Characterization of Electric Solid Propellant Pulsed Microthrusters," *Journal of Propulsion and Power*, Vol. 33, No. 4, 2017, pp. 870-880.

[22]Glascock, M. S., Rovey, J. L., Polzin, K. A., "Electric Solid Propellant Ablation in an Arc Discharge," *Journal of Propulsion and Power*, Vol. 35, No. 5, 2019, pp. 984-993.