Review of RF Plasma Thruster Development

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Abstract: The RF plasma thruster has considerable potential to ease the impact of severe constraints on power, mass, volume and lifetime of microsatellite propulsion systems. This concept is classified as an electrothermal propulsio

n system and exploits RF capacitively coupled discharge (RFCCD) for heating of a propellant. Based on a proof-of-concept study, two new test articles were designed, built and tested. Thruster behavior and operating envelopes were investigated as a function of frequency, power, mass flow rated, and electrode separation.

I. Introduction

In the last few years, government agencies, military and industry have recognized the need to deploy small satellites and spacecraft. Key attributes of such spacecraft, which dictate design, are modularity, maneuverability, maintainability, lifetime, autonomous operation and launch/hardware cost. Feasibility studies have been initiated to investigate the needs of microspacecraft pertaining to development of key components used in propulsion and power. As the size of a spacecraft is reduced, the propulsion wet mass tends to become a more significant portion of the overall system mass. Chemical and electric propulsion alike have to address and deliver propulsive capability within constrains of a mission. In particular, electric propulsion systems have to take into account severe power limitation, which is an inherit characteristic of these proposed spacecraft. Integration of electric micro-thrusters exceedingly relies on reducing the power conditioning requirements to an absolute minimum, while ensuring reliable, long-term performance.

NASA’s New Millennium Program, launched more than a decade ago, initiated research and development of advanced technologies for spacecraft. One of its major aspects was miniaturization to support high launch frequency of small, low-cost spacecraft. To date, this program spun off many micropropulsion system developments of both chemical and electric propulsion technologies. Some concepts, which promise to provide some relief to severe constrains, are found in the area of electrothermal propulsion. Since the 1950’s, extensive research has been conducted in the field of electrothermal systems, particularly with high power arcjets and resistojets. To date, electrothermal systems represent the largest number of thrusters propelling satellites in space. Concepts such as vaporizing liquid micro-thrusters or micro-resistojets are emerging new technologies considered for microspacecraft. Another interesting electrothermal propulsion concept is the AC gaseous discharge. Currently there are two distinct approaches. The first one is the electrodeless microwave arcjet. The second is the inductive RF discharge thrusters.

Another prospective micropropulsion concept relies on a capacitive RF discharge between co-axial electrodes. This RF plasma concept is classified as an electrothermal system. A glow discharge between co-axial electrodes provides heating of a propellant, and thermodynamic expansion of the hot gas generates thrust. Preliminary proof-of-

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concept experiments explored electrode wear by comparing DC and RF operation at comparable levels of power, mass flow rate, and electrode separation. The RF mode showed favorable results with respect to surface condition and discharge stability. Propulsive capabilities of this concept rely on macroscopic parameters such as geometry (electrode separation), mass flow rate (propellant kind and pressure resulting in heating chamber), frequency (1 MHz to 1GHz), and power. These parameters will be strong determining factors on plasma formation and plasma characteristics, which ultimately will affect heating, erosion, and acceleration.

This paper summarizes the state-of-the-art (SOA) micropropulsion technologies and their performance characteristics, and outlines the motivation for using RF plasma discharges for micropropulsion devices. Furthermore, it reviews important results from a proof-of-concept study, which is the basis for the second-generation hardware development. Feasibility experiments and preliminary test results are discussed and presented.

II. SOA Micropropulsion Technologies and Performance

In the last few years, government agencies, military and industry have recognized the need to deploy small satellites and spacecraft. Key attributes of such spacecraft, which dictate design, are modularity, maneuverability, maintainability, lifetime, autonomous operation and launch/hardware cost. Feasibility studies have been initiated to investigate the needs of microspacecraft pertaining to development of key components used in propulsion and power. As the size of a spacecraft is reduced, the propulsion wet mass tends to become a more significant portion of the overall system mass. Chemical and electric propulsion alike have to address and deliver propulsive capability within constraints of a mission. In particular, electric propulsion systems have to take into account severe power limitation, which is a inherit characteristic of these proposed spacecraft. Integration of electric micro-thrusters exceedingly relies on reducing the power conditioning requirements to an absolute minimum, while ensuring reliable, long-term performance.

Today, small satellites are commonly classified in three categories, namely microspacecraft (10–100 kg), nanospacecraft (≤ 10 kg) and picospacecraft (≤ 1 kg). Advanced mission scenarios considered by the scientific and military community push propulsion and power needs for these spacecraft. Primary propulsion and attitude control are the most basic propulsion requirements, but propulsive capability for precise positioning control is increasingly becoming a mission-enabling factor. Other significant handicaps of small satellite operation are severe constraints on power, mass, volume and lifetime.

Since the 1990’s, electric propulsion (EP) has been a vital part of spacecraft propulsion for a wide spectrum of space missions and applications for both industry and government agencies. EP provides significant performance benefits compared to cold gas and conventional chemical systems. Commercial satellite manufactures have embraced EP due to significant economic advantages. To date, electrothermal (arc- and resistojet), electrostatic (Hall and ion thruster) or electromagnetic (pulsed plasma thruster) systems propel close to 200 spacecraft in various mission scenarios spanning lower earth orbit (LEO), geo stationary/synchronous orbit (GST/GSO), and trajectories within the inner planets. However, these applications employ electric thrusters which function best at power levels greater than 1 kW.

In general, microsatellites draw power from solar arrays or batteries to cover their energy need. The main power bus typically operates well below 100 V and, depending on the spacecraft, might deliver power levels up to 100 W. State-of-the-art micropropulsion systems are based on electrostatic acceleration mechanisms, namely FEEP, colloid, ion and Hall thrusters. Other successful systems encompass micro pulsed plasma thrusters (µPPT) and vacuum arc thrusters (VAT). These micro propulsion systems require high operating (hundreds to thousands of volts) voltages, all of which exceed typical voltage ceilings for microspacecraft. Power processing units (PPU) must provide significant voltage conversion between the main bus and the microthruster to enable its operating requirements. Extremely high potential differences are present on the microspacecraft at all times increasing the risk of undesirable discharges, which can lead to destruction of sensitive science/mission instrumentation or compromise the propulsion subsystem. Further, these high voltage requirements of most microthrusters preclude a direct-drive scheme, which could reduce power processing to an absolute minimum. Undoubtedly, electric micropropulsion systems afford significantly higher specific impulse over chemical propulsion, thus yielding a substantial reduction in the propulsion system’s wet mass. It is important that the thruster’s power subsystem doesn’t outweigh this benefit. The power supply for an electric thruster is the largest contributor to the propulsion subsystem with regard to mass and volume. On average, the power
Plasma discharge couples energy into the propellant and raises its enthalpy. Thermodynamic expansion of the hot applied potential and RF frequency dictate the oscillation of these particles between the coaxial electrodes. Energy exchange occurs through collisions between electrons and neutral propellant particles. The mechanism of heating the working fluid, while stochastic heating is an addition stochastic heating is confined to the locale of the sheaths. Typically at high pressures, ohmic heating is the prevalent momentum transfer due to high voltage moving sheaths. Plasma resistivity resulting from electron electrode wear could be suppressed and by doing so achieve the opposite outcome to material processing. Similarly, propellant is injected into the cavity formed by the electrodes. Ohmic and stochastic heating are eliminated with geometry and electrical characteristics. It was speculated that under certain operating characteristics it lent itself to the supposition that electrode wear (sputtering, erosion) could be significantly reduced or even eliminated with geometry and electrical characteristics. The basic idea of this concept lent itself to the supposition that electrode wear could be suppressed and by doing so achieve the opposite outcome to material processing.

The RF plasma thruster concept has considerable potential to ease the impact of severe constraints on power, mass, volume and lifetime on satellite propulsion systems. It addresses all the pertinent issues and alleviates most drawbacks of SOA micropropulsion systems. The RF plasma thruster is based on RF capacitively couple discharge (RFCCD) between coaxial electrodes. Figure 1 illustrates the basic geometry and configuration underlying this proposed propulsion system. Preliminary proof-of-concept experiments were conducted at Auburn University a few years ago. The main goal of that study was to explore feasibility of the concept regarding possible propulsive capability for small satellite applications. Furthermore, it evaluated fundamental operating characteristics as a function of frequency, power, and mass flow rate. Capacitively coupled RF discharges are commonly used in material processing and etching. Here the quality of the surface will depend on electrical characteristics. The basic idea of this concept was to use the supposition that electrode wear (sputtering, erosion) could be significantly reduced or even eliminated with geometry and electrical characteristics. It was speculated that under certain operating characteristics electrode wear could be suppressed and by doing so achieve the opposite outcome to material processing.

This RF plasma concept is classified as an electrothermal system. RF power is fed co-axially to the inner and outer electrodes. Similarly, propellant is injected into the cavity formed by the electrodes. Ohmic and stochastic heating are the main mechanisms for heating the propellant in a RF capacitive coupled glow discharge between co-axial electrodes. Plasma resistivity resulting from electron-neutral collisions leads to ohmic heating. Stochastic heating results from momentum transfer due to high voltage moving sheaths. Ohmic heating is considered a bulk phenomenon, while stochastic heating is confined to the locale of the sheaths. Typically at high pressures, ohmic heating is the prevalent mechanism of heating the working fluid, while stochastic heating is an additional mechanism affecting the heating at low pressures. Energy exchange occurs through collisions between electrons and neutral propellant particles. The applied potential and RF frequency dictate the oscillation of these particles between the coaxial electrodes. This RF plasma discharge couples energy into the propellant and raises its enthalpy. Thermodynamic expansion of the hot

<table>
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III. RF Plasma Thruster Proof-of-Concept

The RF plasma thruster concept has considerable potential to ease the impact of severe constraints on power, mass, volume and lifetime on satellite propulsion systems. It addresses all the pertinent issues and alleviates most drawbacks of SOA micropropulsion systems. The RF plasma thruster is based on RF capacitively couple discharge (RFCCD) between coaxial electrodes. Figure 1 illustrates the basic geometry and configuration underlying this proposed propulsion system. Preliminary proof-of-concept experiments were conducted at Auburn University a few years ago. The main goal of that study was to explore feasibility of the concept regarding possible propulsive capability for small satellite applications. Furthermore, it evaluated fundamental operating characteristics as a function of frequency, power, and mass flow rate. Capacitively coupled RF discharges are commonly used in material processing and etching. Here the quality of the surface will depend on electrical characteristics. The basic idea of this concept lent itself to the supposition that electrode wear (sputtering, erosion) could be significantly reduced or even eliminated with geometry and electrical characteristics. It was speculated that under certain operating characteristics electrode wear could be suppressed and by doing so achieve the opposite outcome to material processing.

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propellant through a nozzle generates thrust. The total heating depends on frequency, power input, pressure, electrode geometry and propellant properties.

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Figure 1. Schematic of the RF Plasma Thruster Concept.

The proof-of-concept device featured a coaxial arrangement of inner and outer electrodes. The power supply and transmission lines were based on 50-Ω impedance. The RF power supply has a frequency range between 10 and 175 MHz. Depending on frequency and gain of the signal, it delivers power up to 300 W. The contour between inner and outer electrodes (diameter ratio of inner/outer electrodes) maintains an 50-Ω impedance. Thereby it minimizes power losses through reflection that occur at impedance mismatches. The inner and outer electrodes taper of towards the throat area. In this region, the gap distance is varied between 0.5 and 1.5 cm by inserting 5-mm wide disks at the base of the inner electrode. Table 2 summarizes the test matrix conducted during the proof-of-concept study at Auburn University.

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Table 2. Testing Matrix for Proof-of-Concept

To explore electrode wear of the RF thruster concept, tests were performed supplying the thruster with DC and RF power. Prior to the experiments, the electrodes were cleaned and polished to a shiny finish. Both operating conditions maintained equal power level (20 W), gas mass flow (0.54 mg/s), and electrode separation (15 mm). The frequency for the RF power mode varied between 30 and 150 MHz. The DC mode operated of a DC power supply with arc suppression circuitry. Both the DC and RF modes were tested for 60 minutes, respectively. Figure 2 shows the inner electrode after RF and DC mode testing, respectively. The electrode after RF testing revealed no noticeable changes in appearance when compared with the surface condition before testing. However, inspection of the DC mode revealed severe discoloration of the surface and rounder edge of the inner electrode. Sputtering effects alter surface morphology, degrade surface quality and consequently affect operating conditions. Over the course of DC testing, the voltage had to be readjusted to maintain the test power level.

The temperature of the bulk propellant was measured using resistance temperature detectors (RTD). RTDs were distributed in key locations (throat and exhaust plume) to access temperature trends as a function of frequency, electrode separation, mass flow rate, and power. For the constant power testing series, Figure 3 illustrates a
representative diagram for the throat temperature as a function of time and frequency. For this particular case, the gap distance is 0.5 cm and the mass flow rate is 0.22 mg/s. The frequency varies between 30 and 150 MHz, while the power is held constant at 20 W. The temperature of the bulk propellant is higher at lower frequencies and approaches a distinct asymptotic value after about 100 seconds. The second test series incorporated varying the power between 25 and 55 W in 10-W increments. Figure 5 shows the stagnation temperature as a function of time and power. The diagram depicts temperature trends for a frequency of 120 MHz, an electrode separation of 0.5 cm, and a mass flow rate of 0.54 mg/s. Similar to the first test series the temperature approaches an asymptotic value after about 100 s reaching an equilibrium condition. Figure 5 summarizes the temperature end values as a function of power and gap distance. The temperature increases with increasing power and gap distance.

Figure 2. Inner Electrode after Testing in RF (left) and DC (right) Mode.

Figure 3. Temperature as a Function of Frequency (20 W, 0.5 cm, 0.22 mg/s).

Figure 4. Temperature as a Function of Power (120 MHz, 0.5 cm, 0.54 mg/s).

Figure 5. Temperature as a Function of Power and Gap Distance (120 MHz, 0.54 mg/s).
IV. Second-Generation Hardware Development

For design and operation of the second-generation test article, the challenges are addressing and alleviating some of the identified constraints with which micropropulsion systems are confronted. The proof-of-concept experiments were conducted with a large device, whose inner electrode measured 2.54 cm in diameter. We designed and built two test articles, designated RF25-1 and RF50-1, whose inner electrode diameter are 0.64 cm and 1.27 cm, respectively. Figure 7 shows a schematic of the basic design and indicates typical dimensions of the smaller test article. The RF powertrain is based on 50Ω transmission lines. Similar to the proof-of-concept device, the inner/outer electrode design (contour, diameter ratio) maintains as close as possible a 50Ω impedance to minimize impedance mismatch and power loss across the thruster. The dielectric spacer is made of machineable alumina bisque. It provides electrode separation and incorporates an elegant solution to introduce the propellant into the discharge annulus. It features radial propellant feed and axial propellant injection into the annulus between the electrodes. Both electrodes and the contoured nozzle are made of aluminum. However, the contoured nozzle is also machined from Lexan to afford visual excess into the discharge annulus. In addition, it features two different geometries, namely a flat or conical contour. Nozzles are manufactured with three different orifice diameters, namely 0.711 mm (nozzle A), 1.397 mm (nozzle B), and 2.794 mm (nozzle C). The two laboratory-scale test articles have a mass of 113 g and 229 g, respectively. Both are compact cubes, where the sides of the cube measure about 3.8 cm for RF25-1 (55 cm³) and 4.5 cm for RF50-1 (85 cm³). Figure 6 illustrates the smaller of the two prototypes. Neither of the two laboratory test articles represents a mass or volume optimized design.

As with the proof-of-concept experiments, preliminary experiments were conducted with both test articles under various test conditions to explore operating regimes. Helium, nitrogen, and argon served as propellants. DC and RF power supplies provided a range of different electrical conditions at power levels below 100 W. The main objective of these experiments was to understand the control and its resulting behavior of a significantly smaller RF plasma thruster. Figure 7 shows RF25-1 operating with argon at a frequency of 900 MHz and power level of about 40 W.

An IR camera was employed to reveal a likely temperature distribution in the discharge. The bandwidth of the camera was between 1.5 and 5 μm. Figure 8 shows the front and side views of RF50-1 operating in DC power mode with argon. These images indicate qualitatively infrared emissions caused by the discharge. Unfortunately, a quantitative evaluation of the temperature distribution was not possible due to the low intensity counts. Singly ionized argon (Ar I) exhibits a strong line at a wavelength of 1.694 μm, which lies in the range of the IR camera.

As with other electric propulsion devices, it is important to understand what relationships exist between mass flow rate, discharge current and discharge voltage. A preliminary sensitivity study was conducted to explore DC discharge characteristics. Operating in a DC power mode, Figure 9 shows a sequence of plumes as a function of mass flow rate. Both the RF25-1 and RF50-1 underwent extensive testing to determine discharge current as a function of DC discharge voltage and mass flow rate. The graph in Figure 10 shows a typical example of the results achieved with RF50-1 and argon as the propellant. As typical in a DC glow discharge, the discharge current increases with increasing...
discharge voltage for a given mass flow rate. The mass flow rate controls the discharge current at fixed discharge voltages. At a given voltage, the discharge current increases with increasing mass flow rate.

**Figure 7.** Operation of RF25-1 with Argon at 900 MHz.

**Figure 8.** IR Camera Images of RF50-1 Operating in DC Power Mode with Argon.

**Figure 9.** Plume Size as a Function of Mass Flow Rate running RF50-1 in DC Power Mode.

**Figure 10.** Current as a Function of DC Voltage and Mass Flow Rate.

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Another vital parameter controlling propulsive capability is the amount of power coupled into the plasma. To access this particular characteristic, experiments were conducted to determine the influence of heating on the pressure in the discharge annulus. A baseline was established by measuring the pressure in the discharge annulus as a function of mass flow rate and orifice diameter. This cold gas baseline is then compared to the pressure with RF heating at various frequencies. Figure 11 shows the pressure in the discharge annulus for the cold and hot gas cases. The frequency is held at 160 MHz. This data reflects the behavior of the RF25-1 with an orifice diameter of 0.711 mm and nitrogen as propellant. RF heating increases the enthalpy of the propellant. The effect of power is even more pronounced in Figure 12. Heat addition causes the pressure in the discharge annulus to rise at fixed mass flow rates. Pressure is a function of both power and mass flow rate. This particular set of data was obtained by operating RF25-1 at 160 MHz with nitrogen.

Operating the RF25-1 in RF mode for extensive periods of time with nitrogen produced a golden-yellow deposit on both electrodes and the dielectric spacer. The combination of materials in a nitrogen glow discharge environment is most likely the cause of an aluminum nitride deposition. Figure 13 shows the severe discoloration due to this deposition. Further investigation is necessary to confirm the compounds of this deposition.

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**Figure 11.** Pressure as a Function of Mass Flow Rate for Cold Gas and Power Input at 160 MHz.

**Figure 12.** Pressure as a Function of Power and Mass Flow Rate at 160 MHz.

**Figure 13.** Aluminum Nitride Deposition During RF Power Mode Operation.
Due to the relative large number of possible operating conditions (frequency, pressure, mass flow rate, power, geometry), a modeling effort was initiated. The goal is to identify operating regimes with favorable propulsive capabilities with respect to device geometry, electric power input and propellant. The numeric model encompasses Particle-in-Cell/Monte Carlo (PIC/MCC) and Direct Simulation Monte Carlo (DSMC) algorithms. PIC/MCC determines plasma characteristics (density, heating, temperature, etc.) in the discharge annulus, while DSMC models the expansion of neutral gas from the discharge annulus to the vacuum chamber. Considering the RF25-1 geometry and dimensions, RF heating (3 Torr, 200 MHz, 50 W) increased the specific impulse by 125% when compared to the cold gas case under the same conditions. So far the modeling indicates that the RF plasma thruster can generate milli-Newton thrust complying with power and size limitations dictated by microsatellite requirements.

V. Conclusion

The feasibility of a significantly smaller second-generation RF thruster has been demonstrated in this study. RF and DC glow discharges were initiated and sustained at a variety of conditions, which were controlled by frequency, power, mass flow rate, and electrode separation. Further testing and analysis are necessary to understand the impact of a wider frequency and power range, and other geometry effects (namely longer discharge annulus, contour of inner electrode, etc). The next test phase will incorporated thrust stand measurements to access propulsive capability within the established operating envelop. Careful attention needs to be attributed to integration issues of thruster hardware and propellant/power feed. Miniaturization and mass/volume optimization will further emphasize these issues. This low-power RF plasma thruster concept promises to be a viable alternative to other SOA micropropulsion systems.

References


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