Electrodeless Lorentz Force (ELF) Thruster for ISRU and Sample Return Mission

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Anthony P. Pancotti[1], Justin M. Little, Jordan S. Neuhoff, Barry M. Cornella, David E. Kirtley, and John T. Slough
MSNW LLC, Redmond, WA 98052

Operation of the Electrodeless Lorentz Force (ELF) Thruster on a variety of complex propellants is investigated to assess the application of this technology to in-situ resource utilization (ISRU) and sample return missions. Utilization of in-situ resources as spacecraft propellant is an enabling technology for a large class of missions. A case study is presented for a sample return mission from Ceres. The ELF-160 thruster is modified with an injection system for complex propellants to suit ISRU operation. Quasi-steady, pulsed operation is enabled by the development of a thermally viable power processing unit (PPU) and rapid puff valve array. Preliminary results using time-of-flight Langmuir probe measurements show FRC formation and acceleration on a number of molecular gases, and indicate a more elaborate dynamic compared to operation on noble gases.

I. Introduction

In Situ Resource Utilization is a concept that is key to the success of future manned missions. The “5:1 concept”, for Mars has shown that every 1 kg of mass saved on Mars through ISRU is the equivalent of 5 kg in LEO and 85 kg on Earth [1]. Using ISRU to provide propellant for propulsion systems is not a new concept. Several studies have been performed that show large gains if Martian atmosphere is processed to make Liquid Oxygen, Methane, or Hydrogen for traditional chemical systems [2]. Additionally, using asteroid, Lunar, or Martian water can also be immensely beneficial for solar and nuclear thermal propulsion systems [3]. In fact, the latest NASA Design Reference Mission Architecture 5.0 (2009) proposed a nuclear thermal rocket with ISRU [4]. However, ISRU propulsion studies to date have been limited to thermal and combustion systems and the low specific impulses that are associated with them. In order to increase payload fraction, electric propulsion having specific impulses of 2000-6000 seconds are required as will be shown with direct mission analyses. This is particularly important to Martian and asteroid cargo missions, where cost and payload fraction are driving factors, rather than trip time.

Traditional electric propulsion has been demonstrated to operate excellently on noble gases such as Xenon and Krypton. For propellant gasses that contain oxygen or volatiles, the primary failure mode for traditional EP is the rapid poisoning of the high-temperature neutralization cathodes. In addition, electrostatic thrusters have direct plasma-wall and plasma-electrode contact. This contact inevitably leads to erosion and high-temperature wall chemistry. If Oxygen, Hydrogen, or Carbon containing gases are passed through these thrusters there are significant detrimental effects including increased wall erosion, carbon deposition and, perhaps worst of all, the creation of conductive or partially-conductive layers on both insulating components and electrodes.

The ELF thruster [5] is a fully electromagnetic and electrodeless pulsed propulsion system that is of particular interest to ISRU missions. It creates a high-density, magnetized plasmoid known as a Field Reversed Configuration (FRC) using external RF antennas that produce a Rotating Magnetic Field (RMF) transverse to the thruster axis of symmetry (Figure 1). The synchronous motion of the electrons magnetized in this field produce a large azimuthal current that, when driven in a direction opposite to that flowing in the external solenoid, reduces and eventually

[1] Propulsion Research Scientist, pancotti@msnwllc.com

reverses the magnetic (bias) field inside the plasma, thus forming a closed magnetic field configuration separate from the external thruster fields (the FRC). These large FRC plasma currents, together with the greatly increased radial magnetic field created by the presence of the plasmoid, result in a substantial $J_\theta xB_r$ force that rapidly accelerates the FRC propellant out of the thruster.

The ELF thruster demonstrated operation on Nitrogen and Xenon at 50 Joule per pulse [5], and steady operation with a CO_2 and simulated Martian Air propellant [6]. The EMP thruster – a variant of the ELF thruster - demonstrated operation on Xenon and a Hydrazine simulant at 1 Joule per pulse and 1-5 kW average power [7]. The Xenon Plasma Liner Experiment [8] demonstrated RMF formation on both Xenon and Deuterium at greater than 400 Joules per pulse (400+ kW). This paper seeks to leverage these extensive Xenon thruster results, and the limited operating points on complex gases, to demonstrate a steady state, fully ISRU compatible electric propulsion thruster. Test results and thruster performance will be presented here along with developmental work on plasma pre-ionizers for FRCs, including a liquid water injector, and thermal tests of a steady operation 30 kW pulse power units for the RMF.

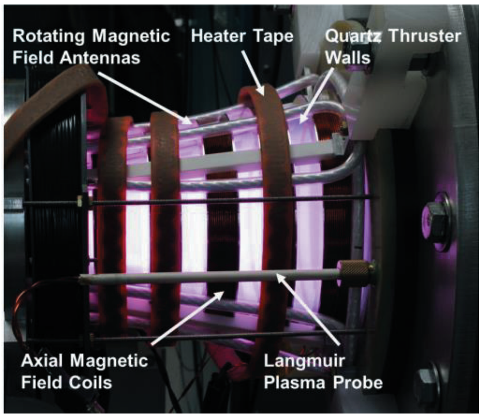


Figure 1. Cross Section of the ELF-160 thruster operating on water vapor.

II. Mission Application

NASA has recommitted its desire for both science and exploration missions to Mars and Near Earth Objects (NEO) [9]. Those destinations have a particularly interesting characteristic over the lunar environment, namely the availability of large amounts of non-silica materials. The long-term Martian missions are particularly exciting, as Mars has vast amounts of easily extractable Argon (1.6% atmospheric), carbon dioxide (95% atmospheric), and likely subsurface H_2O. Additionally, there is evidence that large amounts of other volatiles, such as ammonia and methane, may be locked in subsurface deposits [10]. The majority of Near Earth Asteroids (NEA) and Comets also contain many millions of tons of propulsively interesting volatiles. C-type asteroids, known as carbonaceous chondrite asteroids, contain up to 20% water by volume, as well as large amounts of hydrocarbons, organic material and other volatiles. M-type asteroids, known as metallic or iron asteroids, contain vast amounts of iron, nickel, cobalt, titanium, and platinum metals. While M-type asteroids do not have major deposits of volatile resources, it is believed that the majority do contain some trapped frozen water, nitrogen, and organic materials, equating to millions of tons. Besides purely scientific endeavors, asteroids have the potential for astronomical monetary value. The smallest know M-type NEA is only 2 km in diameter, but is estimated to contain as much as $32 Trillion worth of iron, nickel, cobalt, and platinum [11].

An important focus of NASA's 2014 Strategic Plan (Objective 1.5 [12]) is the question: How did our solar system form and evolve? Central to this objective are the asteroids Vesta and Ceres. Both objects, whose orbits lie within the asteroid belt, have survived largely intact since the formation of the solar system. As a result, a detailed understanding of their mineral properties could provide a wealth of insight into the physical and chemical conditions prevalent in the nebulous solar system.

NASA's DAWN spacecraft [13] launched in September 2007 with the goal of remotely characterizing the bulk density, gravitational field, topography, and mineral properties of both Vesta and Ceres. This ambitious eight-year mission, which requires a total Δv of around 11 km/s, is enabled by electric propulsion technology. Specifically, its three Xenon ion thrusters (based on NSTAR technology) provide a majority of the post-launch Δv.

Considered here is the application of ISRU to a CERES sample return mission. Compared to the remote sensing of the DAWN instrumentation, characterization of the returned surface material of Ceres in terrestrial laboratories will yield chronological information, dramatic improvements in measurement scale and precision, and the ability to manipulate the sample and revisit it with continually improving technology. Furthermore, contrasting the laboratory measurements with the remote data from DAWN will provide a unique perspective that will greatly improve the understanding of remote observation methods – an important benefit for future science missions.

Noting that a mission of this type is essentially impossible without the use of electric thrusters, we focus our attention on comparing two EP systems (Table 1): a Xenon Hall thruster and the ELF-160A. The spacecraft equipped with the Xenon Hall thruster carries all of its propellant to and from Ceres (non-ISRU), while the ELF-

160A uses Xenon for the outbound phase and in-situ derived water for the return phase (ISRU). Indeed, a recent study [12] found a strong presence of water vapor surrounding Ceres that is thought to result from either volcanic activity or surface ice sublimation.

The total onboard solar power is taken to be 30 kW with a mass of 5 kg/kW. The Hall thruster is assumed to operate at a specific impulse and thrust efficiency of 3,000 s and 60%, respectively. The alpha of the Hall thruster and PPU is estimated as 5 kg/kW. In the nominal outbound configuration, the ELF-160A will operate on Xenon at 3,000 s Isp and 60% thrust. The inbound phase will operate on water acquired from Ceres at 5,000 s Isp and 55% thrust efficiency. The ELF-160A allows the unique ability to operate at variable Isp on different propellants. The large Isp in the return phase is used to offset the decreased thrust efficiency due to increased ionization losses associated with the use of water as an EP propellant.

The compact nature of the ELF-160A and state-of-the-art PPU promise to deliver an EP system with around 1.5 kg/s. Finally, high thrust maneuvers required at the surface of Ceres are accomplished with a 300 s Isp chemical propulsion (CP) system. A simplified mission analysis consists of six phases beginning with the spacecraft in LEO: (1) EP transfer from LEO to Earth escape, (2) EP transfer from Earth escape to Ceres that leverages a Mars gravity assist [11], (3) CP powered descent to the surface of Ceres, and (4-6) the inverse of these phases on the return journey. The Δv-requirement of each phase is given in Table 1.

Using this mission model, the mass available for the payload (spacecraft structure, instrumentation, etc.) as a function of the LEO insertion mass was solved for. The benefit of ISRU is most clearly demonstrated in a plot of the payload mass fraction (payload mass divided by LEO insertion mass) versus the mass in LEO (Figure 2). ISRU enables significantly higher payload mass fractions. Furthermore, there exists a range of LEO insertion masses for which the mission becomes impossible without the use of ISRU. These benefits, shown in Figure 2 for the Ceres sample return mission, generally results from comparing ISRU and non-ISRU EP missions, and become even more apparent as the return Δv increases. Therefore, one can imagine a wide range of missions for which ISRU is crucial.

The LEO insertion mass and total mission duration requirements as a function of the payload mass are examined in Figure 3. Again, the benefit of ISRU for a given payload mass is clearly demonstrated in the form of significantly decreased insertion masses and mission times. These benefits can be quantified by choosing a payload mass of 1,000 kg (Table 1 and Figure 3). For comparison, the payload mass of the DAWN spacecraft is around 350 kg (28% payload mass fraction). The ELF-160A operating on Ceres water during the return phase allows the mission to be completed with nearly half the launch mass when compared to the same mission operating with a cluster of Xenon Hall thrusters. Assuming a launch cost of 10,000 $/kg, the mass savings translates to a $35 million decrease in cost. The vast majority of mass savings for the ISRU mission is reflected in the decreased Xenon propellant mass, which results from decreasing the spacecraft mass and eliminating the need for return propellant. In addition, this

Table 1. Simplified model of a Ceres sample return mission. Model assumes optimal transfer between Earth escape and Ceres using a Mars gravity assist [2]. Trip times are calculated assuming a 50% thrust duration for this phase. The payload mass consists of the spacecraft structure, attitude, reaction, and thermal control, CP (without propellant), and data subsystems.

	Non-ISRU	ISRU
Spacecraft Parameters		
Electric Propulsion (EP) System	Hall Thruster	ELF-160A
EP Propellant	Xenon	Xenon (outbound) Water (inbound)
EP Specific Impulse (s)	3,000	3,000 (out) 5,000 (in)
EP Thrust Efficiency (%)	60	60 (out) 55 (in)
Onboard Solar Power (kW)	30	30
Propulsion System Alpha (kg/kW)	5	1.5
Solar Power Alpha (kg/kW)	5	5
Chemical Propulsion (CP) System	Bipropellant	Bipropellant
CP Specific Impulse (s)	300	300
Mission ΔV's		
LEO to Earth Escape (m/s)	6,900	6,900
Earth Escape to Ceres (m/s)	13,000	13,000
Ceres Descent (m/s)	510	510
Ceres Ascent (m/s)	510	510
Ceres to Earth Capture (m/s)	13,000	13,000
Earth Capture to LEO (m/s)	6,900	6,900
Results for 1,000 kg Payload		
Payload Mass (kg)	1,000	1,000
Prop. and Power Mass (kg)	300	195
Neutral Propellant Mass (kg)	1,059	630
EP Propellant Mass (kg)	4,751	1,763
ISRU EP Propellant Mass (kg)	0	600
Total Mass to LEO (kg)	7,110	3,588
Total Mission Time (years)	7.1	5.2

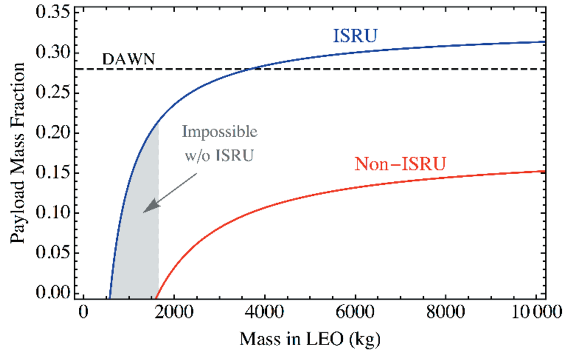


Figure 2. LEO insertion mass and mission time requirements as a function of the payload mass. Labels are shown for a payload mass requirement of 1,000 kg.

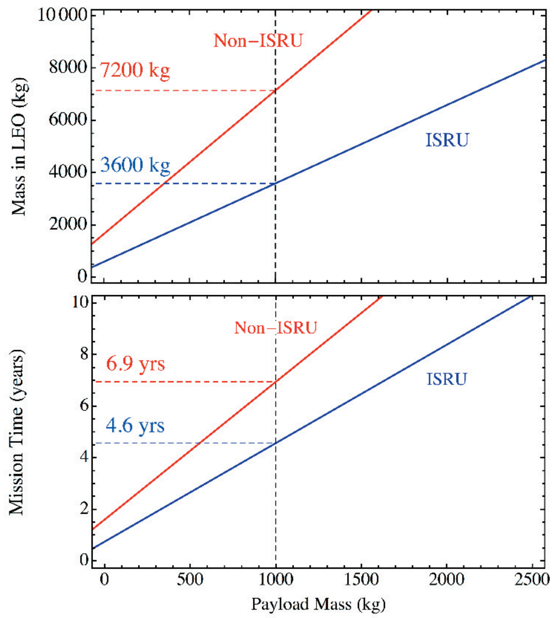


Figure 3. LEO insertion mass and mission time requirements as a function of the payload mass. Labels are shown for a payload mass requirement of 1,000 kg.

decreases both the neutral propellant mass and required thrust (to overcome Ceres gravity) of the CP system.

The large return Isp of the water-powered ELF-160 increases the duration of this phase of the mission with respect to the non-ISRU spacecraft. However, this downside is offset by the time-savings due to the significantly higher outbound acceleration for the lighter ISRU-enabled spacecraft. The confluence of these two effects is an overall trip time that is two years lower for the ISRU mission than the non-ISRU mission.

Decreased spending for planetary science missions brought about by budget cuts has created a strong impetus to drive down launch costs and mission times. Just as electric propulsion enables the DAWN mission, an ISRU operated ELF-160 may ultimately enable a CERES sample return mission.

III. Experimental Research

Experiments were carried out on the ELF-160, a ~160 mm diameter, 30kW ELF thruster, and its subsystems. The propellant injection system was modified to allow operation on complex gaseous and liquid propellants. Thermal tests were performed on a 15 kW PPU (the thruster uses one PPU for each of two RMF antennae). Finally, thruster performance testing at low background pressures was achieved using a fast-acting puff valve array.

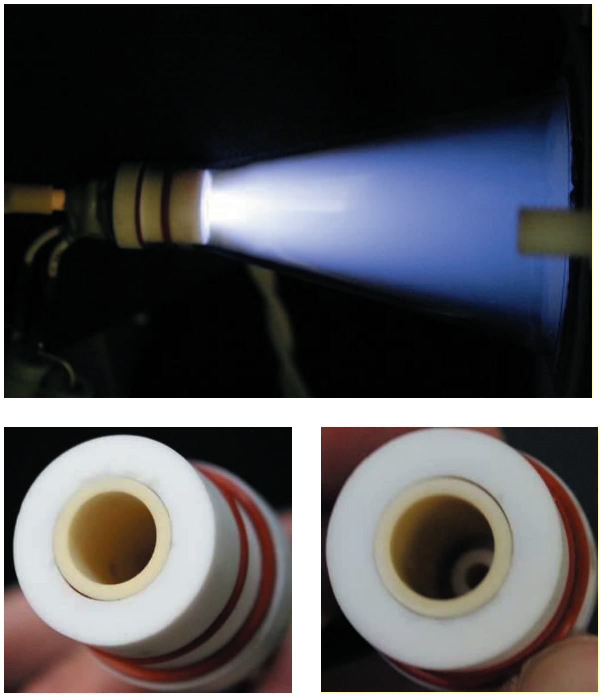
A. Injector Design

1. Gaseous Propellant

An inductive RF discharge is needed to operate on ISRU propellants. Although many forms of pre-ionization can provide the seed electron population of the RMF, an inductive system allows the use of harsh atomic and molecular propellants that would otherwise corrode or char an electroded system.

For gaseous propellants, a 13.56 MHz inductive RF pre-ionization (PI) stage was constructed of a 6 to 10 turn coil or antenna. The coil was wrapped around a 12 mm diameter alumina tube. The tube and coil were then potted in a machined ceramic (Macor) house designed to fit in the back of the thruster assembly. The PI is designed for high temperatures and could run above 300 °C. The number of coil turns was chosen, along with a low-loss tuning cap, to impedance match the RF power supply - a commercially available 13.56 MHz supply with an automatic tuning network. The tuning network allows for matching to a large range of loads and cable length; however, a higher inductance coil (around 1 μH) and the use of a tuning capacitor (68 nF) near the inductive load allowed for maximum coupling to the RF plasma, with the line length associated with the MSNW vacuum test facility. Baseline tests were performed over a variety of flow rates and pre-ionization power on the 30 kW ELF-160 thruster to determine the PI requirements for successful thruster operation.

Two long-duration PI tests were conducted, the first being done with argon. This test took six hours with a flow rate of 2.9 mg/s and a RF power of 20 W. The PI housing reached an equilibrium temperature of 94 °C after about 10 minutes. No noticeable changes in plasma brightness or color were observed during the test. There was no clear indication of erosion, degradation, or material deposits, which was expected for an inductive system running on a noble gas. To make these tests more relevant to an ISRU propellant, a second test was conducted with a more appropriate molecular propellant. CO_2 was used because it contains both carbon and oxygen. If there was to be an issue with certain atomic species, it was believed that carbon would pose the most likely threat of coking and forming deposits while atomic or ionic oxygen would be the most reactive. The CO_2 long-duration test was run for eight hours at a flow rate of 1.6 mg/s and a power of 10 W. Similar to the Ar tests, no damage was observed to the discharge channel or housing (see Figure 4).



Before *After*

Figure 4. An RF inductive pre-ionizer operating on CO_2 gas and before and after photos of the long-duration operation.

2. Liquid Propellant

Liquid propellants provide many interesting challenges for an electric propulsion device; however, they have significant advantages, which have been discussed previously. Although a single gas injector can typically be used on many different gases, liquid injectors usually need to be designed for a specific liquid. This is due to the large variation in viscosity, vapor pressure, and phase transition temperature. Described here is a water injection system for continual operation with an FRC thruster (Figure 5).

Figure 6 shows an exploded picture of the water injection system and all its major components. A 32 mm^2 disc of porous stainless steel was soldered onto a 3/8 stainless steel tube that coupled to the PI housing with a glass tube and O-ring seal. The other end of tube connected via Swagelok tee to a heater and a liquid water injection port. A 40 W heater was used for stable control of the temperature. The injector was assembled and mounted on the back of an ELF thruster cone. Figure 5 is a photo of the injector operating on a steady flow of 2.5 mg/s of H_20 and 50 W of RF power.

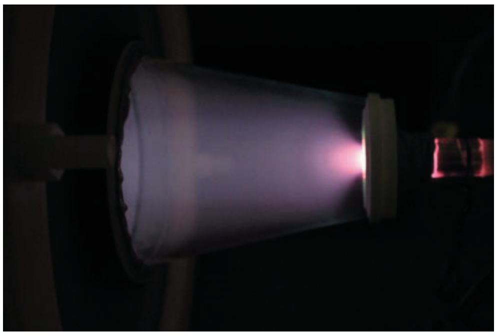


Figure 5. Photograph of the H2O plasma injector operating in a thruster geometry at 3 mg/s and 50 W of forward power.

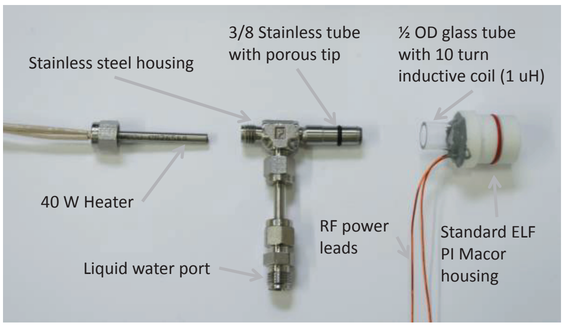


Figure 6. Picture of an RF inductive pre-ionizer assembly designed to fit into a standard FRC gas injection port.

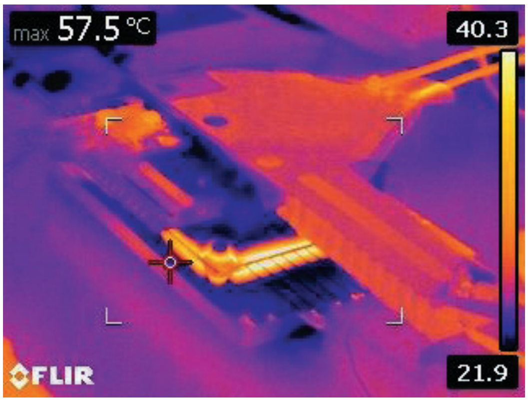


Figure 7. Thermal image of the IGBT switch, ceramic caps, and cooling plate.

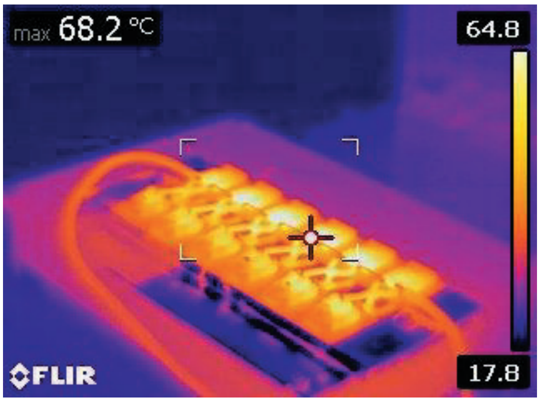


Figure 8. Thermal image of pulse charging diodes.

Figure 9. Thermal image of the antenna and water cooled dummy load.

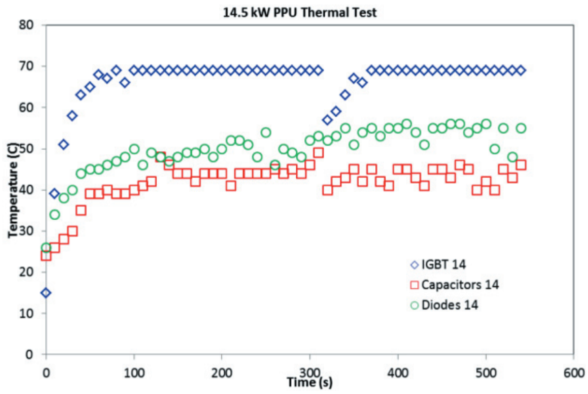


Figure 10. Thermocouple data for IGBT, ceramic capacitors, and charge diodes during the 14.5 kW PPU thermal test.

B. 30 kW Pulsed Power Units

A 30 kW thruster poses a number of interesting challenges for the power processing unit (PPU). These challenges, which stem from requirements on the repetition rate, energy per pulse, efficiency, and lifetime of the PPU, include high voltages that need to be electrically isolated from critical thruster components and large thermal loads on the PPU switches during steady operation. These thermal loads are particularly important at high powers because the IGBT switches used by MSNW to provide high-speed, high-efficiency switching, are inherently low-temperature devices. Therefore, it is crucial for the thermal management system to draw sufficient heat away from the switches to prevent failure.

To test the thermal behavior of the PPU, a rep-rated continuously operating PPU assembly was built and tested using highly efficient solid state switching and state-of-the-art high Q-factor ceramic capacitors. To test the PPU in a controlled bench top environment, a dummy load was created to simulate the plasma load. The antenna used to couple to the load was a 3 turn copper coil that was insulated with silicone tape. The geometry of the load and the antenna was designed to achieve an inductance of 1.3 μH, which is similar to an operating thruster. The energy storage consisted of an array of 22 ceramic capacitors, each with a capacitance of 22 nF. The bench top setup, with the capacitance of the energy storage and the inductance the antennas were designed for, resulted in a RMF frequency of 200 kHz. This frequency is standard for most FRC thrusters, which typically run from 150 to 300 kHz.

A set of four thermocouples was used to measure the temperature of the critical component of the bench top setup. The capacitors and switch as well as the pulse charging diode and inductor were all measured using floating DMM thermocouple probes. In addition to the thermocouples a near infrared (FLIR) camera was used to monitor the setup and individual components. This tool was very valuable, as it helped visualize the temperature gradients. Figure 7 through Figure 9 show images from the thermal camera. As one can see, the base plate gets much hotter than the rest of the switch; however, with proper thermal conduction, the base plate can be kept within appropriate operating conditions. The pulse charging diodes create a loss within the system. Originally, a single diode stack was used; however, this was changed to a 2 parallel stacks to share the charge current and, as a result, the heating decreased. The antenna is the hottest part of the bench top setup. Most of the energy was dumped into the stainless steel load, and without water cooling, that would certainly get the hottest. However, the load was easy to cool, but the antenna was not. A primary challenge that will need to be addressed in a stead-operating thruster is the electrical insulation of the antenna, which will need to survive high operating temperatures.

The thermal tests were run again starting at 2 kW for system checkout, and were repeated for 4 and 8 kW, before running at the full 15 kW. The 15 kW test ran at a capacitor or RMF voltage of 3.2 kV at a repetition rate of 130 us. This collated to a DC power supply voltage of 500 V and 29 A. As a note, the system was tested at slightly faster repetition rates compared to thruster operation (~500-1000 us) to achieve the full 30 A. Ultimately, transients associated with the DC power supply switching between current and voltage limiting modes motivated the thermal test to be performed at 14.5 kW. The 14.5 kW was run for nine minutes to insure that all the components rose to thermal equilibrium (as seen in Figure 10). The IGBT switch got the hottest at 70 °C, which is still lower than its maximum operation temperature of 125 °C. The ceramic capacitors only heated to about 45 °C. It is worth noting that the capacitor array was actively cooled with a fan during these tests; however, their temperature was well below their maximum value of 175 °C.

C. Quasi-Steady, Pulsed Operation

The final and largest challenge for running a 30 kW thruster at MSNW had to do with facility limitations. Previous thrust stand measurements for low power operation used a steady flow of gas. This is acceptable for 1 to 5 kW thrusters because the flow rate is tens of sccm of Xe, for which the 9,000 l/s of pump can keep the background pressure in the 1E-5 to 1E-4 Torr range. Higher pressures will cause erroneous performance measurements and, at worst, will result in arcing and destruction of thruster components. For example, a 30 kW thruster operating at 2,000s would require a flow rate of almost 200 sccm of Xe. At these flow rates, the background pressure would be in the mid to high 1E-3 Torr range.

The optimal solution was to run the thruster with a short pulse of gas and use the vacuum chamber volume as ballast. The solution to this was to create a pulse of gas with a fast-acting micro pulsed (puff) valve. These valves have been used extensively in FRC research at MSNW for both fusion and early spacecraft propulsion development. Custom 600 V pulsed valve drives have been developed in order to achieve puff times as short at several hundred microseconds. To create a square pulse pressure profile, four valves were fired in sequence. The valves were arranged on an aluminum cube plenum so that each valve output flowed a finite amount of working gas transversely into the plenum chamber, which has a variable volume by adjusting a stop located on the back end of the cube.

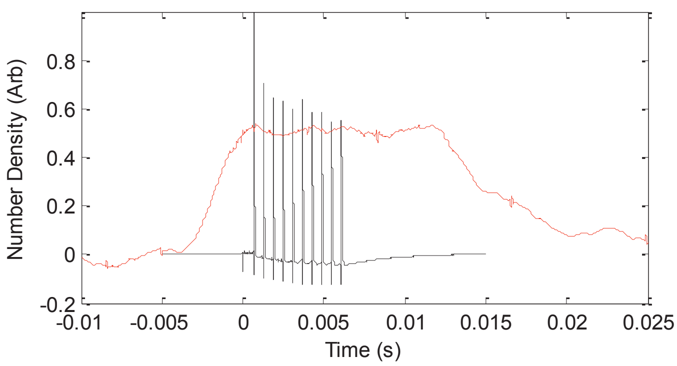


Figure 11. Plot of pressure as determine by the FIG overlaid with double probe 1's measurement of ion number density.

An overlay of the gas density and the FRCs formed are shown in Figure 11. As is shown, the gas pressure was allowed to reach the top of its profile before FRCs were formed. At this point, a steady pulse train is formed, with the total number of pulses determined by the rep rate of the thruster.

D. Operation on Molecular Gases

Preliminary performance measurements were conducted with the thruster mounted external to the vacuum chamber using downstream Time-of-Flight (TOF) asymmetric Langmuir probes as the primary diagnostic. The first probe was placed approximately one thruster diameter (14 cm) downstream from the thruster exit. The second probe was place 24 cm from the exit, resulting in a separation distance of 10 cm. The ELF-160 thruster body used parallel antennas with an inductance of 325 nH. The PPUs each used four 330 nF WIMA capacitors for a total capacitance of 1.32 μF, which resulted in an RMF frequency of 242 kHz. The PPUs were run at 3,400 V, supplying a total of 15.3 J of RMF energy. For most of the data presented here the repetition rate, or the time between FRC discharges, was 600 μs. If run steady, this would be equivalent to a 25 kW thruster.

Figure 12 shows both double probe 1 (DP1) and 2 (DP2) data for the formation and ejection of 10 FRCs on N_2O gas. Figure 13 illustrates how this data is analyzed to determine the average ion velocity of the FRC. Shown are both DP1 and DP2 signals and the integration bounds that are used to determine the center of mass of the FRC. The bounds are set to trim noise from pre-ionization, RF, and RMF pickup. The center of mass of each FRC is shown as a blue dot on the double probe trace. The time between these dots is used to give the most accurate measurement of average ion velocity. For this shot on N_2O the velocity was measured to be 12 km/s. With the exception of the first FRC, the measured velocity of each FRC is typically within a few percent of their mean velocity.

It is worth noting the double-peaked shape of the double probe single in Figure 13. This is accredited to the dissociation of N_2O molecule into atomic nitrogen and oxygen. Because of their different molecular weights, they are accelerated differently out of the thruster. This phenomenon can be seen with almost all molecular gasses, as shown in Figure 14. Here, it can be seen that Methane produces the largest double peaked distribution. It also shows that the leading edge of the first peak accelerates out of the thruster faster than any other gas tested. This would coincide with the proposed explanation, as methane would have a large number of hydrogen atoms, which would achieve the highest velocity due to their small mass. One more note of interest here is that the separation effect of atomic species was not witnessed as significantly in steady flowing gases. In these tests, the same conditions were run as in Figure 14, with the exception that the valve was held open and flow was controlled with a mass flow meter. The best explanation for this difference is that the higher background pressure from the steady flow causes more collisions and ion-neutral interaction, resulting in a more evenly distributed average ion velocity.

As can be seen from the double probe traces, the use of molecular propellants in FRC thrusters is a complicated and complex process. This study found that these propellants can be successfully ionized and utilized in an FRC thruster at the 30 kW level. Ion velocities ranged from 5,000 to 23,000 m/s, which is close to the maximum resolution of the TOF system based on current probe spacing. These results were far from optimized for each individual molecular gas. A more detailed characterization of the behavior of different multi-species plasmas and the influence of this behavior on thruster operation will be a topic of future study.

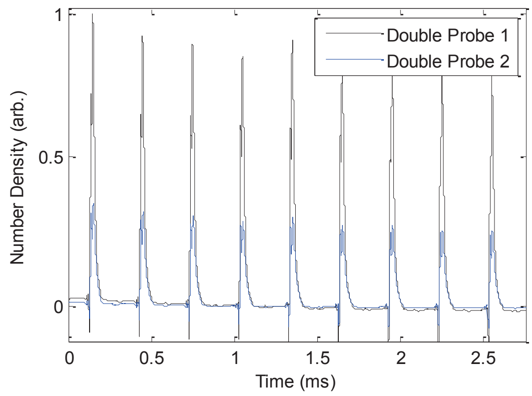


Figure 12. Double probe data for Shot # 26970. Puffed operation on N2O.

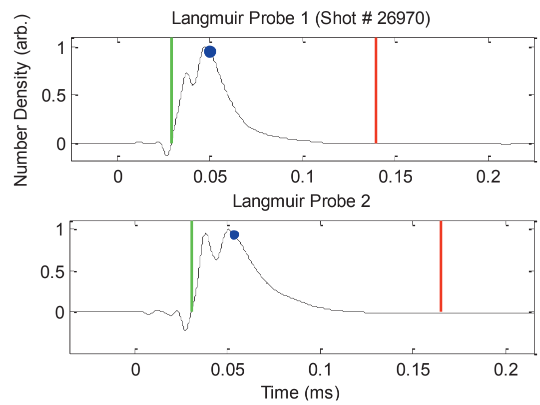


Figure 13. Close up of a single FRC formed during test number # 26970. Puffed operation on N2O.

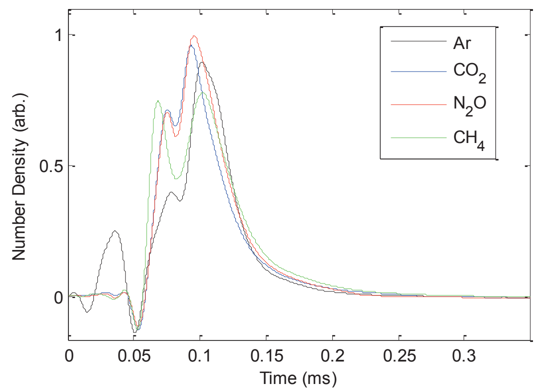


Figure 14. Double Probe 1 data of a single FRC for pulsed operation on Ar, CO2, N2O, and CH4.

IV. Conclusion

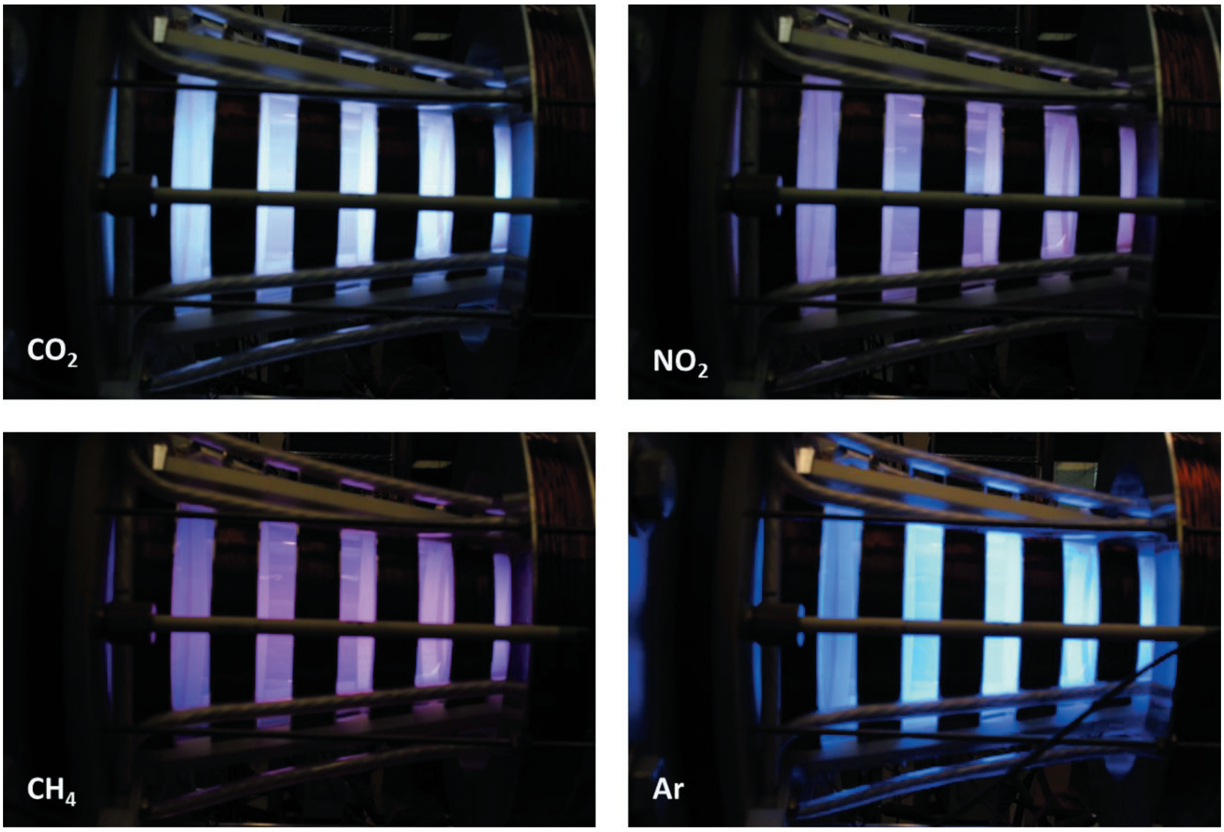


Figure 15. ELF-160A Operating on a variety of propellants

The ELF thruster represents an enabling technology for a number of ISRU missions based on its ability operate on a variety of complex propellants without the worry of electrode poisoning and wall erosion (Figure 15). Using the example of a sample return mission from the asteroid Ceres, it was shown that an ISRU-enabled spacecraft results in insertion masses nearly half as large as required for conventional xenon electric propulsion, with a trip time savings of around two years.

Previous research at MSNW, which focused on the operation of the ELF thruster on nitrogen [5], air [14], Martian atmosphere simulant [6], and hydrazine simulant [7], was expanded upon to assess the performance of the thruster on complex propellants. Two new propellant injection and pre-ionization systems were developed to operate on different gaseous and liquid propellants. Quasi-steady, pulsed operation was enabled by the development of a thermally viable power processing unit (PPU) and rapid puff valve array. Preliminary results showed exhaust velocities that ranged from 5 to 23 km/s for air, carbon dioxide, nitrous oxide, and methane.

With all the major subsystems validated in either a benchtop setup or pulsed mode of operation, a final ELF-160 for ISRU propellants can be designed, built, and tested. To do so, this work will require addition research into the thermal aspects of the thruster assembly as a whole. Future designs will be based primarily off the thermal analysis described here. Special attention is needed in the development of methods to cool both the low-temperature PPU as well as the high-temperature thruster body; in particular the RMF antennas. This will require a materials study to determine a coating that can withstand high-temperature operation, with enough flexibility to accommodate thermal cycling without cracking in order to prevent arcing of these high voltage components.

Once a suitable antenna insulation is determined, a complete thruster assemble can be built for steady state operation. However, in order to fully test and characterize an EP thruster of this power level (30 kW) a high vacuum facility with a pumping speed of greater than 200,000 l/s is required. This will ensure a background pressure low enough to minimize ingestion and neutral entrainment effects. This facility will also need to be equipped with the proper cooling mechanism as well as a diagnostic suite to adequately operate up to full temperature while properly assessing thruster performance. Plans are ongoing to test FRC propulsion devices over a range of power levels using direct thrust stand measurements. This diagnostic tool still remains the most reliable and accurate way to validate performance metrics such at thrust, specific impulse, and thruster efficiency.

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