Test Results of ExoTerra’s Halo Micro Electric Propulsion System

M. VanWoerkom¹, V. Gorokhovsky², G. Pulido³, R. Pettigrew⁴, A. Seidcheck⁵
ExoTerra Resource, LLC, Littleton, CO 80127 USA

Abstract: While CubeSats have begun to disrupt the entire satellite industry, a lack of adequate propulsion and radiation tolerant options continues to limit their adoption beyond experimental missions. Rideshare regulations require that secondary payloads “do no harm” to the primary mission resulting in bans on toxic or combustible materials, which in turn, exclude most conventional propellant options. These restrictions limit CubeSats to the non-optimal orbits into which the primary mission delivers them, and often leaves them unable to maintain their orbits against drag from Earth’s upper atmosphere. Additionally, lack of radiation tolerant parts necessitates orbits in LEO and prevents spiral through the Van Allen belts into GEO or interplanetary orbits. Together, these limitations restrict CubeSat utility for commercial, military, or interplanetary missions.

ExoTerra’s Modular Micro Electric Propulsion (μ-EP) system enables more ambitious NASA, DoD and commercial CubeSats applications with a high-impulse, radiation tolerant propulsion option. The μ-EP system packages into 6-9U depending on specific mission requirements. ExoTerra’s magnetically shielded Halo Micro Hall-Effect Thruster operates between 85-450 W and generates between 4-33 mN of thrust at an Isp ranging from 700-1500 s. The thruster itself occupies less than 1/4U of volume and weighs <.65 kg, providing a significantly smaller option for its power class. Depending on tank size and operating point, we deliver over 100 kNs of impulse. The accompanying propellant storage and distribution system uses a Composite Overwrap Pressure Vessel (COPV) to minimize distribution system volume and mass. Powering the system is ExoTerra’s high-efficiency power processing unit (PPU), with a demonstrated 96-98% efficiency, mass under 0.45 kg, and 100 krad radiation tolerance to survive transit through the Van Allen belts.

ExoTerra recently concluded several rounds of testing of its Micro Electric Propulsion system for CubeSats, including performance testing of the thruster operating envelope, initial life testing, integrated system testing, and is currently undergoing qualification testing. The performance testing demonstrated an operating range between 85-450 W and 150-500 V with up to 40% efficiency, providing operating capabilities for low power CubeSats up to microsat scale satellites. The initial life testing operated for 300 hrs. The resulting data provided evidence of magnetic shielding at the operating point and projected a lifetime of >2000 hrs, resulting in total impulses up to 100 kNs. The integrated testing successfully demonstrated the ability of the PPU, propellant distribution system and thruster to work in concert and provided data on overall system efficiency, with demonstrated PPU efficiencies of up to 98%. Qualification testing is currently in work. Results from the qualification unit’s functional testing show thrust measurements in line with prior testing and significant improvements to Isp and total efficiency.

Together, the system provides demonstrated capabilities to enable CubeSats to perform missions requiring longer life, higher impulse, improved radiation tolerance, higher altitudes and increased maneuverability in a CubeSat scale.

¹ President, ExoTerra Resource LLC.
² Sr. Staff Electric Propulsion Engineer, Propulsion Division.
³ Propulsion Engineer, Propulsion Division.
⁴ Mechanical Design Engineer, Engineering Department.
⁵ Mechanical Engineer – Stress Analysis, Engineering Department.
1 Nomenclature

\[ A_a = \text{orifice area for anode gas supply line [m}^2\text{]} \]
\[ A_{cs} = \text{orifice area for stationary cathode operation gas supply line [m}^2\text{]} \]
\[ A_{ci} = \text{orifice area for cathode ignition gas supply line [m}^2\text{]} \]
\[ a = \text{channel length [mm]} \]
\[ b = \text{channel width [mm]} \]
\[ c_{Xe} = \text{Xe speed of sound at } T=300^\circ\text{C [m/s]} \]
\[ c_{Ar} = \text{Ar speed of sound at } T=300^\circ\text{C [m/s]} \]
\[ m_{\dot{a}} = \text{Ar mass flow rate through thin orifice into vacuum [kg/s]} \]
\[ m_{\dot{c}} = \text{Xe mass flow rate through thin orifice into vacuum [kg/s]} \]
\[ e = \text{charge of an electron [C]} \]
\[ n_i = \text{ion density in a plume [m}^{-3}\text{]} \]
\[ n_o = \text{neutral density [m}^{-3}\text{]} \]
\[ p = \text{chamber pressure [Pa]} \]
\[ p_o = \text{standard pressure [Pa]} \]
\[ P_{in-a} = \text{inlet pressure of anode supply line [Pa]} \]
\[ P_{in-c} = \text{inlet pressure of cathode supply line for stationary operation [Pa]} \]
\[ P_{in-ci} = \text{inlet pressure of cathode supply line for ignition [Pa]} \]
\[ P_a = \text{power deposited to the anode [W]} \]
\[ P_b = \text{beam power [W]} \]
\[ P_{ad} = \text{anode discharge power [W]} \]
\[ P_{jet} = \text{jet power [W]} \]
\[ P_k = \text{keeper power [W]} \]
\[ P_{mag} = \text{magnet power [W]} \]
\[ P_t = \text{total power [W]} \]
\[ P_w = \text{power deposited to the channel wall [W]} \]
\[ E_i = \text{ion energy, [eV]} \]
\[ T = \text{thrust [mN]} \]
\[ T_e = \text{electron temperature [K, eV]} \]

\[ g = \text{acceleration due to gravity [m/s}^2\text{]} \]
\[ I_{anode} = \text{anode current [A]} \]
\[ J_b = \text{ion beam current [A]} \]
\[ I_p = \text{specific impulse [s]} \]
\[ J = \text{ion current density [A/m}^2\text{]} \]
\[ k = \text{Boltzmann constant [m}^2\text{kg/s}^2\text{K]} \]
\[ R = \text{gas constant [J/mole K]} \]
\[ M_{Xe} = \text{Xe atomic mass [kg/mole]} \]
\[ M_{Ar} = \text{argon atomic mass [kg/mole]} \]
\[ m_a = \text{anode propellant mass flow rate [kg/s]} \]
\[ m_b = \text{beam propellant mass flow rate [kg/s]} \]
\[ m_c = \text{cathode propellant mass flow rate [kg/s]} \]
\[ m_i = \text{ion mass flow rate [kg/s]} \]
\[ m_t = \text{total propellant mass flow rate [kg/s]} \]
\[ T_a = \text{neutral gas temperature [K]} \]
\[ V_k = \text{cathode keeper voltage [V]} \]
\[ V_{anode} = \text{anode discharge voltage [V]} \]
\[ v_i = \text{ion velocity [m/s]} \]
\[ \gamma_{g} = \text{heat capacity ratio for gases} \]
\[ \rho_{Xe} = \text{Xe density [kg/m}^3\text{]} \]
\[ \rho_{Ar} = \text{Xe density [kg/m}^3\text{]} \]
\[ \rho_{Xe_o} = \text{standard xenon density at } T_o, p_o [kg/m}^3\text{]} \]
\[ \rho_{Ar_o} = \text{standard argon density at } T_o, p_o [kg/m}^3\text{]} \]
\[ \eta_a = \text{anode efficiency} \]
\[ \eta_r = \text{total efficiency} \]
\[ \Upsilon = \text{form factor of the thruster’s plasma plume} \]
\[ \theta = \text{beam divergence angle [°]} \]
\[ L_b = \text{ballast inductance [H]} \]
\[ R_b = \text{ballast resistance [Ohm]} \]
\[ C = \text{capacitance [mF]} \]
2 Introduction

While CubeSats have begun to disrupt the entire satellite industry, a lack of adequate propulsion options continues to limit their adoption beyond experimental missions. Rideshare regulations require that secondary payloads “do no harm” to the primary mission resulting in bans on toxic or combustible materials, which in turn, exclude most conventional propellant options. These restrictions often limit CubeSats to non-optimal orbits based on the primary mission needs, and often leaves them unable to maintain their orbits against drag from Earth’s upper atmosphere. This restricts CubeSat utility for commercial, persistent Earth science, military, or interplanetary missions [1].

ExoTerra’s Modular Xenon Micro Electric Propulsion (μ-EP) system enables more ambitious NASA, DoD and commercial CubeSats applications with a high-impulse, high-density propulsion option. The Xe μ-EP system packages into as little as 4U. Fig.1 depicts a 9U configuration being developed under a NASA Phase II SBIR. ExoTerra’s Halo Micro Hall-Effect Thruster (μ-HET) operates between 85-450 W and generates between 4-33 mN of thrust at an Isp ranging from 700-1500 s. The accompanying propellant storage and distribution system uses a Composite Overwrap Pressure Vessel (COPV) and a custom designed feed system to minimize distribution system volume and mass. Powering the system is ExoTerra’s high-efficiency power processing unit (PPU), with a demonstrated 96-98% efficiency, mass under 0.45 kg, and 100 krad radiation tolerance to survive transit through the Van Allen belts. The total 9U system has a dry mass of 6.8 kg and holds 4 kg of propellant. It can provide up to 60 kNs of impulse.

ExoTerra recently conducted a series of 4 tests on the system over the last 24 months, using steadily improved thrusters: Functional Testing, Life Testing, and Integrated Testing and Mark II Functional Testing. Functional testing occurred in 2017 and focused on exploring its operating envelope using the Halo-C iteration. Initial Life testing with Halo-D was performed in June of 2018 and focused on understanding the erosion rates to project lifetime. Integrated testing using Halo-E concluded in January 2019 and focused on demonstrating operation with the full PPU and representative propellant distribution system. Mark II Functional testing is ongoing and focused on measuring the performance of a flight ready Halo Thruster. This paper documents the results of the test results to date.

3 Functional Testing

The first set of testing performed was a functional test of the Halo Thruster. The objective of the testing was to measure the performance envelope of the thruster at various power and voltage settings using Xenon propellant. Key performance metrics included thrust, Isp and efficiency.

3.1 Functional Test Setup.

For the first three sets of tests, testing of Halo was conducted inside the Orion vacuum chamber in the Colorado State University (CSU) Electric Propulsion and Plasma Engineering (CEPPE) lab at the CSU Engineering Research Center. The testing was performed in the Orion vacuum chamber shown in Figure 2. Orion is 1.7 m diameter by 4.6 m long. Cryo pumps provide 20,000 L/s pumping capability. The ultimate vacuum prior to testing was maintained at p<
3E-6 Torr, while during the testing the pressure in the vacuum chamber never exceeded 3E-5 Torr leaving background neutral gas density at $n_b<1.08E19 [m^{-3}]$, which allows us, in a first approximation, to neglect the influence of the residual gas atmosphere on the thruster performance [2].

The chamber’s milli-Newton thrust stand uses a linear LVDT transducer to output thrust readings to LabVIEW. A set of finely calibrated weights are attached to a fishing line and induce a known force on the thrust stand that is used to calibrate the LVDT transducer. The thrust stand is hooked up to a LabVIEW algorithm that controls the calibration and thrust output. Total accuracy of the thrust measurement is +/- 0.5 mN.

A set of lab bench power supplies were used to provide the outer electromagnet power, inner electromagnet power, keeper power, cathode heater power, and anode power. The hollow cathode-to-anode-keeper circuit was powered by a DC power supply (Sorensen XG 600-1.4) operating in current mode. The cathode-to-main anode circuit was powered by a programmable DC power supply (Chroma 62024P-600-8) which was operating both in current and in voltage control mode in the different testing cycles. The magnetic coils of the thruster were powered by two DC power supplies (TDK Lambda GENH60-12.5). An oscilloscope was set up to monitor the frequencies and breathing modes of the thruster. Thermocouples and their thermometers were also set up to monitor the temperature of the thruster at different operating points. Ballast resistors between the thruster and power supplies were set up to protect the facility equipment in the case of an electrical fault. To obtain more accurate current and voltage readings than displayed by the power supplies, four-digit multimeters and Flukes were set up and wired directly to the feedthrough lines. Multimeters were set up for the anode voltage and current, keeper voltage and current, float voltage, and PPU supply voltage and current. A float voltage reading was set up to measure the voltage drop between the thruster and the tank wall (ground).

Xenon was provided by the lab provided feed system. Flow rates were measured to an accuracy .1 sccm.

The schematic for a typical small sized Hall Effect Thruster (HET) [6] is presented illustratively in Fig.3. The figure shows two possible positions for the hollow cathode (HC)-anode-keeper pair: external, outside of the thruster and internal, positioned along the thruster axis within the inner pole of the thruster. The external cathode used a heater to raise the cathode temperature before the low work function insert reached the thermal emission temperature, while the internal cathode was heaterless. The external cathode was used only for the second half of the functional testing, while the first half of the functional test, life test, integrated tests and Mark II functional tests utilized a heaterless internal HC [8]. The outer portion of the body included the outer bobbin with outer screen and the outer magnetic coil positioned between the outer wall of the body and the outer screen. The inner bobbin with its inner magnetic coil was

![Diagram](image.png)

Figure 3. Halo thruster scheme. Positions of the thermocouples across the thruster are shown as red dots.
made between the inner pole and the inner screen. Three thermocouples, TC1, TC2 and TC3 were installed to the inner and outer coils and the back side of the thruster as shown in Fig.9.

Figure 4 shows the thruster on the thrust stand in its internal centerline cathode configuration.

### 3.2 Functional Test Results

Testing began with ignition of the cathode. The cathode flowrate was established within the range between 0.5 and 2 sccm, approximately 10% of the main anode. The anode-keeper current was first established within the range 1.0-1.5 and then reduced to ~0.3A during steady-state operation. Target keeper voltage during testing was 25 V.

Once the cathode was lit, the anode was turned on. An optimization of the magnet settings was performed along with varying the voltage & power to determine the performance envelope. Figure 5 provides an image of the Performance testing. The plume is concentrated in the center of the channel, providing visual evidence of magnetic shielding.

The anode power level, anode voltage level, and anode flow rate level ranges that were tested are shown in Table 2.

The testing began by finding the optimal inner coil and outer coil currents. The thruster was set to an anode voltage of 300 V and power of 100 W. To optimize the thruster performance parameters, the inner and outer electromagnets were dialed to the predicted optimized ratio of 1.91. To confirm this, we then swept the magnets across a range to determine the optimal thrust to power ratio (T/P). The sweep consisted of gradually adjusting both the inner and outer electromagnet current supply levels about estimated optimum levels – within 0.05 to 0.5 Amps about the estimated optimum setting, in increments of 0.05 or 0.1 Amps. Electromagnet sweeps were repeated at anode voltage levels of 150, 225, 300, and 400 volts, at an anode power level of 100W. After each electromagnet current sweep, the same anode voltage level was swept with respect to anode power, and the resulting power sweep data was correlated with previous data at similar anode voltage and power levels to determine the effectiveness of the

<table>
<thead>
<tr>
<th>Thruster parameter</th>
<th>Tested range</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>Anode Power</td>
<td>85 - 450</td>
<td>Watts</td>
</tr>
<tr>
<td>Anode Voltage</td>
<td>150 - 500</td>
<td>Volts</td>
</tr>
<tr>
<td>Anode Flow Rate</td>
<td>4 - 48</td>
<td>SCCM</td>
</tr>
</tbody>
</table>

![Figure 5. Initial Functional Test Operation with Internal Cathode](image)

![Figure 6. Derivation sample of data sheet curves from raw data spread across all anode powers.](image)
performance parameter increase, the optimization method and the 1.71 electromagnet ratio were determined to be effective.

With the magnets optimized, we proceeded to perform testing of the full operating envelope. Testing was conducted by setting the voltage level and then adjusting the mass flow rate to achieve the desired anode power. Magnet power levels were adjusted, keeping the ratio fixed, to optimize on thrust. Figure 6 shows a typical profile, with specific impulse increasing with increase of the anode power reaching near 1500s at the anode power of 450W.

The performance envelope tested ranged from 85-600 W (anode). We measured anode voltages of 150V, 250V, 400V and 500V. While the thruster operated above 450W, steady state temperatures exceeded the design limits and only limited testing was performed in the region. During day 3 of testing, the centerline cathode experienced a failure. In order to maximize testing within our vacuum chamber test window, we installed an external cathode as shown in Figure 7.

Fig. 8 provides the measured thrust versus the anode power. Good linear relationships were shown as the power level increased. Thrust ranged from 4-33 mN. The highest thrust point was measured at 450 W and 150 V. The lowest anode operating power was at 250 V and 85 W and resulted in 4 mN of thrust. We operated the thruster at 550 W and 500 V and 600 W/400 V and showed good agreement with thrust projections from lower power levels. However, at power levels above 450 W, we would exceed temperature limits, necessitating that we operate at those levels for < 5 minutes.

Figure 9 shows the total Isp, including the cathode contribution, versus the total power (including magnet & keeper). Total Isp ranged from 636 s to 1540 s. Operating points above 450 W supported performance trends.

Total efficiency was then calculated. This included the anode, cathode and magnet power. These are plotted against total power in Figure 10. Total efficiency ranges from 10-35%. Excluding the magnet and cathode power, the anode efficiency exceeds 40% at 400 W and higher levels. At 85 W anode power levels, the anode efficiency was 16%.

4 Life Testing

The second set of tests was life testing. Small Hall Effect thrusters have traditionally struggled with lifetime. To counteract this, the Halo thruster is designed to provide magnetic shielding to the channel walls. This improves efficiency and lifetime. Testing was conducted to determine the efficacy of the shielding.
4.1 Life Test Setup

The life testing was conducted with a heaterless hollow cathode positioned along the axes of the inner pole. The Halo-D thruster was powered by facility power sources and fed via a chamber mounted propellant distribution system as outlined in Section 3.1. The goal of the life test was to aggregate sufficient hours of operation time of Halo to determine the erosion, if any, of the channel walls due to electron and ion bombardment. Testing was performed using Krypton to reduce testing costs. Additionally, magnetic field characterization was to be performed to correlate Halo to MagNet analysis. The thruster was run at ~160 W for 300 hrs. A sample from the 300 hrs life time testing records is shown in Figure 9. A stable average anode voltage and anode current can be seen. The low frequency oscillations have a small amplitude compared to the average $I_{\text{anode}}$ and $V_{\text{anode}}$ values.

4.2 Life Test Results

Testing was stopped every 100 hrs to measure the erosion rate. The thruster plume was directed to impinge on a graphite sheet in order to create particles in the chamber that would coat the channel and provide visual evidence of erosion. If erosion was occurring, it would scour away the graphite film accumulating on the channel. As seen in Figure 12, the outer wall remained dirty, while the inner wall showed a slight erosion along the outermost .25” of the inner channel wall. This indicates that the outer wall is at least scouring away at a slower rate than the carbon buildup.

Measurements were taken at 0, 45, 90 and 135 degrees around the channel using calipers. After 300 hr of testing, we measured .10 mm (.004”) of erosion along the inner channel at the worst case point. Erosion along the outer wall was below our ability to measure. This results in an erosion rate of .0003 mm/hr. Erosion of the channel is the primary failure mechanism of the thruster, as once it erodes, the plasma may begin to degrade the magnetic circuit. Based on the erosion rate measured and dimensions of the channel, we project an effective life of 2000+ hours. Improvements in the Mark II thruster are anticipated to improve the

**Figure 9.** Measured Total Isp vs. Total Power at various voltage settings.

**Figure 10:** Measured Total Efficiency vs. Total Power

**Figure 11.** Anode voltage and anode current during 300 hrs life performance test.
magnetic field by centering the field better, and provides a thicker channel which is anticipated to increase lifetime to 4000 hr.

5 Integrated Testing

5.1 Integrated Test Setup

Fig. 13 depicts the integrated test setup which was mounted inside the Orion vacuum chamber to the thrust stand. The Halo-E thruster was installed to a radiator plate (not shown in Fig. 13). In addition, nano-graphite strips were inserted between the outer body of the thruster and the radiator to further advance heat transfer between the thruster and the radiator.

The gas feed subsystem of the recent integrated test was installed on a skid panel together with the PPU and the thruster inside of the chamber as shown illustratively in Fig. 13. The gas feed subsystem consisted of three propellant gas feed lines, which split off from a regulator attached to the gas tank and led to the cathode and anode of the thruster. Orifices restricted the gas flow between the proportional control valves (PCV) and the entrance to the cathode and the anode of the thruster. The gas pressure transducers measured the gas pressure upstream of each orifice, while the gas pressure at the entrance to the PCVs was kept at ~ 482633 Pa. Two gas feed lines are provided to the hollow cathode: a high flow rate line for ignition, and a low flow rate line for steady state operation. The anode gas feed line has an orifice with an intermediate hole diameter.

Testing began with bench tests to calibrate the gas flows of the three gas feed lines as shown in Fig. 14. The calibration bench test was conducted at ExoTerra’s facility using argon. MKS mass flow meter sensors were added downstream of the orifices in each of the gas supply lines during the calibration bench test, but removed from the gas feed setup in the Orion chamber. The pumping of the gas feed system during the calibration bench testing was provided by the Robinair 2-stage rough vacuum pump, which provided pressures downstream of the orifices less than 2068 Pa. The obtained testing results were compared to the ab-initio Monte-Carlo simulation of the gas flow through thin orifice into vacuum [3,4] and choked flow estimation ns, assuming the gas velocity through the hole in the orifice is equal to the speed of sound velocity at gas temperature T=300K upstream of the orifice. At choked flow, the mass flow rate at the choke point across the orifice hole exit plane can be increased only by increasing the gas density upstream of the orifice.
orifice, which is proportional to the gas pressure measured by the gas transducers installed between the orifices and PCVs. The choked flow of Ar through the orifice hole can be estimated by the following expression [5]:

\[
m_{Ar} = c_{Ar} \rho_{Ar} A_a \frac{P}{P_0},
\]

(1)

where \( c_{Ar} = 319 \text{ m/s} \), \( \rho_{Ar} = 1.78 \text{ kg/m}^3 \), \( P_0 = 10^5 \text{ Pa} \) and the area of the orifice hole of 0.08mm diameter in the anode gas feed line \( A_a = 1.5 \text{E-8m}^2 \). The comparison of Ar flowrates through 0.08mm orifice calculated by the Monte-Carlo simulation [3,4] and choked flow vs. the measured Ar flowrates are presented in Fig.15 showing near ideal agreement between measured flowrates and Monte-Carlo simulation, while the choked flow exceeds the measured values by nearly a factor of 2. This discrepancy can be attributed to the fact that the average gas velocity through the orifice is about 2 times lower than speed of sound [5]. This calibration was further used for finding a gas flowrate towards the anode and cathode via pressures upstream of the orifice: \( P_{\text{in-a}} \) and \( P_{\text{in-c}} \) respectively while the pressure upstream of the cathode orifice for ignition the cathode-keeper discharge, \( P_{\text{in-ci}} \), was typically greater than that during the cathode stationary operation. The linear

**Figure 14.** Gas feed subsystem scheme.

**Figure 15.** Ar flowrates through the thin 0.08mm orifice into vacuum calculated by Monte-Carlo simulation, and choked flow vs. the measured flowrates.
interpolation of the Xe flowrate through the 0.08mm orifice gives the following expression which was used in the testing of Halo-E with a built-in gas feed system shown in Fig.3:

$$m_{\text{Xe}} = 0.0002p,$$  \hspace{1cm} (2)

were $m_{\text{Xe}}$[sccm] is mass flowrate and $p$[Pa] is pressure.

The calculated Xe flowrates based on Monte-Carlo simulation shows excellent agreement with measured flowrates through the 0.08mm orifice toward the anode as illustrated in Fig. 16.

During the integrated testing, the Halo E model was installed inside of the Orion chamber together with the PPU electronics board and gas feed subsystem in a 9U configuration. Power was supplied to the PPU from an external supply. Initially, the PPU operated the thruster and propellant feed system and the data generated in the testing was recorded by LabView data acquisition (DAQ) software operating from a laptop computer positioned outside of the vacuum chamber. A technician error damaged the PPU midway through testing. Afterwards, the anode power was provided from an external supply.

During the integrated testing, data was acquired using a graphite planar current density probe, affixed to a four-axis motion stage system, both shown in the photo of Figure 17. The probe’s collector disk had an outer diameter of 0.80 cm, which translates to a collector area of 0.50 cm$^2$. The probe has a closely-spaced guard ring, with an outer diameter of 1.90 cm.

Data was acquired with both the collector and guard ring of the planar current density probe biased 60V negative of chamber ground as shown in Fig.18. A Keithley 6517A was used to directly measure collector ion current through the instrument’s internal ammeter. The 6517A’s voltage source was used to bias the collector (bipolar capable to ±1000V). A second lab power supply was used to apply the guard ring bias.

Figure 16. Monte-Carlo simulation vs. experimental measurements of the Xe flowrates through the 0.08mm orifice toward the anode of the Halo E thruster installed in the ORION chamber.

Figure 17. Planar current density probe made of graphite (left) and four-axis motion stage system (right).
5.1.1 Electrostatic Analyzer (ESA)

An electrostatic analyzer made by Plasma Controls was used to measure the ion beam energy of the Halo thruster. The ESA was placed on the beam centerline at an axial distance of 1.3 m from the thruster. The ESA consisted of two spherical sectors nested in a 120° arc with a centerline radial axis of 5.0 cm, and a segment factor of 3.554 [6,7]. The ESA had 1.5 mm collimators at each end of the arc to allow for very narrow solid angle acceptance of ions moving toward the detector as shown in Fig. 8. The collector electrode was located downstream of the exit collimator. To yield the ion energy distribution function (IEDF), a constant voltage bias was applied to the spherical segments and the entrance and exit collimators were swept with respect to the plasma. At each voltage bias setting, a picoammeter was used to measure the ion current that flowed to the collector electrode.

5.2 Integrated Test Results

The latest testing of the Halo thruster was performed as part of a NASA Phase I SBIR for a high impulse, low power CubeSat propulsion system. The configuration was designed to provide >36 kNs of impulse from within a 9U propulsion module while operating at 135 W. The system holds up to 4 kg of Xenon. A smaller version of the system fits 2 kg of Xenon the EP module within a 4U volume. The primary Phase I objective was to demonstrate the performance of an integrated system, including thruster, PPU, and propellant feed system.

During the build up of the system, we performed a functional check of the thruster using Krypton. The testing included measurements of the ion momentum flux to compare against measured thrust. The theoretically estimated averaged ion momentum flux generated by the thruster which can be expressed by the following expression

\[ \Upsilon \cdot I_{\text{anode}} \cdot \sqrt{V_{\text{anode}}} \]

in which the estimated flux of the ions in the thruster’s plume is estimated as proportional to the anode current \( I_{\text{anode}} \) and the characteristic ion velocity is estimated via anode voltage \( V_{\text{anode}} \) as \( \sqrt{V_{\text{anode}}} \). The formfactor \( \Upsilon \) characterizes the divergence of the plume and the corresponding spread of the ion velocity vectors generated by the thruster [13]. The relationship between the thrust and the theoretically estimated averaged ion momentum flux generated by the thruster can be expressed by the following expression.

\[ T \approx \frac{1}{2} \rho \Delta V \cdot \Upsilon \cdot I_{\text{anode}} \cdot \sqrt{V_{\text{anode}}} \]

in which \( T \) is the thrust, \( \rho \) is the mass density of Xenon, \( \Delta V \) is the change in plasma velocity, \( \Upsilon \) is the formfactor, \( I_{\text{anode}} \) is the anode current, and \( V_{\text{anode}} \) is the anode voltage.

Figure 18. Planar current density probe schematic

Figure 19. Photograph of the ESA and electrical schematic used for obtaining the IEDF
thruster shown in Fig.20 have demonstrated almost ideal linear proportionality which indicates that the formfactor $\Upsilon$ of the plume of the thruster and the corresponding beam divergence angle $\theta$ were almost unchanged in these test points with anode flowrates ranging from 10.25 through 13.4 sccm. The Isp and anode discharge efficiency $\eta_a$ are also increasing with the increase of the ion momentum flux.

The current density distribution in the plume of the thruster operating in Kr measured by the current density probe at a distance of 15cm from the thruster’s output flange is shown in Fig.21a, while Fig.21b shows the integrated (total) ion current as a function of the distance from the thruster’s output flange, demonstrating the narrow plasma plume generated by the thruster with relatively small beam divergence in agreement with the integral measurements parameters shown in Fig.20.

After functional checkout, we installed the components on a representative 9U chassis and mounted it to the thrust stand inside of the ORION vacuum chamber as shown in Fig.13. Mass of the thruster was measured at <.65 kg, while the PPU was measured at <.45 kg. Integrated testing was performed using lab supplied Xe propellant. During testing, PPU efficiency ranged between 96%-98.5% depending on the operating voltage. The testing successfully demonstrated the objective of using the PPU to accept commands and set the thruster and flow control system to the desired operating point within a vacuum environment.

Once performance at 135 W was demonstrated to meet the Phase I objective, the system was operated over a range of power and voltage settings to expand the integrated test envelope. The key performance parameters vs. Xe flowrate are shown in Fig. 23. The parameters include voltage, anode power, the anode discharge efficiency and Isp with increase of the Xe flowrate shown in Fig.20 have demonstrated almost ideal linear proportionality which indicates that the formfactor $\Upsilon$ of the plume of the thruster and the corresponding beam divergence angle $\theta$ were almost unchanged in these test points with anode flowrates ranging from 10.25 through 13.4 sccm. The Isp and anode discharge efficiency $\eta_a$ are also increasing with the increase of the ion momentum flux.

The current density distribution in the plume of the thruster operating in Kr measured by the current density probe at a distance of 15cm from the thruster’s output flange is shown in Fig.21a, while Fig.21b shows the integrated (total) ion current as a function of the distance from the thruster’s output flange, demonstrating the narrow plasma plume generated by the thruster with relatively small beam divergence in agreement with the integral measurements parameters shown in Fig.20.

After functional checkout, we installed the components on a representative 9U chassis and mounted it to the thrust stand inside of the ORION vacuum chamber as shown in Fig.13. Mass of the thruster was measured at <.65 kg, while the PPU was measured at <.45 kg. Integrated testing was performed using lab supplied Xe propellant. During testing, PPU efficiency ranged between 96%-98.5% depending on the operating voltage. The testing successfully demonstrated the objective of using the PPU to accept commands and set the thruster and flow control system to the desired operating point within a vacuum environment.

Once performance at 135 W was demonstrated to meet the Phase I objective, the system was operated over a range of power and voltage settings to expand the integrated test envelope. The key performance parameters vs. Xe flowrate are shown in Fig. 23. The parameters include voltage, anode power, the anode discharge efficiency and Isp with increase of the Xe flowrate shown in Fig.20 have demonstrated almost ideal linear proportionality which indicates that the formfactor $\Upsilon$ of the plume of the thruster and the corresponding beam divergence angle $\theta$ were almost unchanged in these test points with anode flowrates ranging from 10.25 through 13.4 sccm. The Isp and anode discharge efficiency $\eta_a$ are also increasing with the increase of the ion momentum flux.

The current density distribution in the plume of the thruster operating in Kr measured by the current density probe at a distance of 15cm from the thruster’s output flange is shown in Fig.21a, while Fig.21b shows the integrated (total) ion current as a function of the distance from the thruster’s output flange, demonstrating the narrow plasma plume generated by the thruster with relatively small beam divergence in agreement with the integral measurements parameters shown in Fig.20.
anode flowrate. The data shows significant efficiency improvements in the Halo-E thruster versus the prior Halo-C data at power levels below 200 W. As shown in Figure 23, we see an improvement in thrust at ~250 V. Thrust improved by 30-40%, reaching 8 mN at 100 W.

We noted a drop in the anode Isp within the measured power range. At 100 W, the Isp decreased from 800 s to 713 s. At 200 W, the Isp shift was less pronounced. Some of the shift is tied to a slightly lower operating voltage between the two tests. However, we believe we are also seeing a reduction in double and triple ions.

We noted an increase in the overall efficiency between the two tests. The anode discharge efficiency reaches 30% at 200 W and improves to 22% at 100 W. This compares to 25.5% efficiency and 16% efficiency respectively in Halo-C testing. We believe the design improvements are leading to a high mass utilization efficiency and less energy being lost to the channel walls.

The ion energy distribution function (IEDF) of the Halo E thruster operating on Xe was measured by the electrostatic deflecting energy analyzer at different flowrates corresponding to different thrust values (Figure 20). Fig. 25 shows the IEDF in relation to the thrust when the Xe flowrates increased from 10.25 to 13.4 sccm. It can be seen that the ion energy is slightly decreasing with the increase of the thrust which can be attribute to the loss of momentum in ion-neutral collisions when the number density of neutral increases with increase of the Xe flowrate.

By estimating the outcoming flux of the ions in the plume of the thruster as proportional to the anode current $I_{\text{anode}}$, we can compare the ion momentum flux and thrust to the theoretical estimation of the thrust by the generated outcoming ion momentum flux similar to the one shown previously in Fig.20. The comparison of the thrust and outcoming ion momentum flux estimated as proportional to the $I_{\text{anode}} \sqrt{E_i}$, where $E_i$ is the average ion energy measured by the ion analyzer in relationship

Figure 22. Critical performance parameters of the thruster vs. Xe anode flowrates.

Figure 23. Thruster Isp and Thrust vs. Anode Power.

Figure 24. Anode Efficiency Comparison
to the theoretical estimation of the thrust based on anode current and anode voltage [13] are shown in Fig.26, demonstrating almost ideal agreement between experimental measurements and theoretical prediction.

Figure 25. IEDF of the ions in the HALO E plume in relationship to the thrust.

Figure 26. The comparison of the thrust vs. measured ion momentum flux as a function of the theoretical estimation of the ion momentum flux via anode current and anode voltage.
6 Mark II Halo Testing

The final set of testing was functional testing with the Mark II Halo. The thruster represents a flight qualification unit and incorporates improvements to magnetic shielding, arc mitigation, erosion mitigation, thermal management and coil efficiency. Total mass of the thruster was measured at .67 kg. Figure 27 shows the Mark II mounted to the thrust stand.

6.1 Mark II Halo Test Setup

Mark II Testing was performed at ExoTerra’s facilities using our 4’ diameter x 6’ long vacuum chamber. The chamber’s Leybold cryo pump has a pumping speed of 10,000 L/s (N_2). At start of testing, the chamber was 3E-6 Torr and maintained 5E-5 Torr during operation. The chamber is equipped with a thrust stand with +/- .5mN accuracy. The stand uses an optical system to measure deflection of the cantilevered stand. The stand is calibrated prior to each test cycle using a series of calibrated masses that are hung from the stand to measure a known force.

During testing, the thrust stand showed a steady upward drift in thrust measurement of ~ 3mN/hr when the thruster coils were turned on. When the magnets turned off, the thrust stand would drift back at a similar rate. The root cause is not yet determined, though is currently thought to be related to magnetic material on the thrust stand. To take thrust measurements, we measured the instantaneous thrust change between an operating thruster and turning the thruster off. As the change was a much shorter timescale than the drift, it provided fairly accurate thrust which was in line with prior testing as shown later.

Flow is measured using a set of MKS flow meters. The cathode flow meters is accurate to .01 sccm while the anode flow meter is accurate to .05 sccm. Power is supplied to the anode and cathode keeper using TDK Lambda power supplies.

Control is provided via a LabView interface. Power, thrust, temperature and flow rate data is recorded in real time through a National Instruments data acquisition system.

6.2 Mark II Halo Test Results

Figure 27: Mark II Halo Thruster

Figure 28: ExoTerra Vacuum Chamber and Thrust Stand
Testing began with optimizing the magnet coil ratios. Testing started at 135 W and 150 V. We performed a sweep of the magnet coil current ratios and power levels to determine the maximum thrust point. Coil current ratios were varied from 0.9 to 1.7. Total magnet power was swept from 5-40 W. As shown in Figure 29, at 135 W, we found the ideal ratio at 1.7 and a magnet power level of 7.4 W. Ratios at 1.3 and 1.5 showed peak thrust within the error measurement of the thrust stand, though at a ~25% higher total magnet power.

We then conducted a power sweep at 150 V by increasing flow rate and current up to 400 W. At each point, we re-evaluated the magnet ratio to confirm the optimum point. The optimal thrust point was reached at each setting between 1.5-1.7.

The process was then repeated at 200 V. Testing is still ongoing and further testing will measure 250 V, 300 V, 350 V and 400 V.

Figure 30 compares the 2019 Mark II thrust measurements against the prior 2017 testing at 150 V. The data shows nearly identical thrust results with the prior testing and well within the +/- .5 mN error range. This provided further confidence that our method of recording thrust was also reasonable.

The Mark II unit demonstrated a significant improvement in $I_{sp}$ versus prior testing at 150 V. The $I_{sp}$ improvement ranged from 7-17%. Figure 31 provides the resulting Anode $I_{sp}$ v. Anode Power.

As expected from the higher $I_{sp}$ values, we found the total efficiency of the thruster also improved. This was aided by improvements to the coil design as well. Figure 31 provides the resulting Efficiency v. Anode Power. Total efficiency improved to 20% at 135 W, and to 40% at 400 W. Efficiencies improved by up to 8 percentage points in some cases.

Further testing of the Mark II is in work. Plans are to complete the full sweep of the operating envelope. The thruster will then be exposed to anticipated launch and environmental loads. Following this, we plan a 1000 hr life test.
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Figure 33: Mark II Halo Thruster Test Fire
References

17. V. Gorokhovsky, Modeling of DC Discharges in Argon at Low Pressures, poster presentation at the 12th Intern. COMSOL Conf., Boston 2012.
18. V. Gorokhovsky, S. Robertson, Low pressure cascaded arc discharge, poster presentation at the ICOPS45, Denver 2018.
20. V. Phelps. ftp://jila.colorado.edu/collision data/.