HOT-FIRE TESTING OF AN AF-M315E 1-NEWTON THRUSTER

by
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Abstract

AF-M315E is a green monopropellant currently being investigated for use as a potential replacement for hydrazine in low-thrust spacecraft propulsion applications. To support development efforts, a series of hot-fire tests were conducted to assess the performance of AF-M315E in a 1N thruster. Preliminary design and analysis shows an increase in performance versus a comparable hydrazine system. Testing was conducted with a 1N TZM (titanium, zinc, and molybdenum alloy) thruster in late July 2016 in collaboration with the Spacecraft Propulsion Systems office (ER23) at Marshall Space Flight Center (MSFC). Testing focused on characterizing the transient behavior of the thruster during initial startup. Consistent startup behavior was observed when the thruster catalyst beds had been adequately heated. Thruster response was found to be particularly sensitive to catalyst temperature. Peak thrust levels of 0.6-0.7N were achieved during 1-second firings.
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1 Introduction

1.1 Overview

This paper describes the hardware, methodology, and results of a testing campaign aimed at understanding the feasibility of using the ionic liquid AF-M315E as a replacement for hydrazine in low-thrust spacecraft propulsion applications. More specifically, the testing described herein focuses on assessing thruster startup behavior as a function of catalyst temperature. A NASA-built TZM thruster (Figure 1) was used for all testing in this paper, with the long-term goal of conducting testing using the Plasma Processes, Inc. (PPI) iridium thruster (Figure 2) to achieve optimal performance values. These thrusters have the potential to be an important cost-saving component due to the increase in performance and density over hydrazine and the reduction in necessary safety equipment required to handle AF-M315E in comparison to hydrazine. This reduction in safety equipment is demonstrated in the MSDS and safety handling documentation covering the use of AF-M315E and comparing it to that of hydrazine. Significant value can be realized if AF-M315E and its associated hardware can meet the requirements to replace hydrazine on future spaceflight missions.

The justification for testing is to demonstrate the capabilities of AF-M315E and its associated hardware to NASA and commercial projects. The test plan and matrix were developed to increase understanding of the behavior of propulsion systems using this propellant. The knowledge gained from these tests will provide guidance on how to modify future requirements and testing to use the propellant and thrusters as efficiently as possible. The testing undergone is creating a baseline for further testing endeavors.

1.2 Motivation to Replace Hydrazine

Hydrazine is currently the most common propellant used for spacecraft attitude control thrusters. Hydrazine thrusters are well suited for in-space propulsion applications due to their inherent simplicity, which usually feature a pressurized propellant tank, an electric solenoid valve, and the thruster
itself. The simple design of these thrusters affords a high degree of reliability. Hydrazine thrusters are capable of providing a vacuum specific impulse (Isp) of around 235 seconds.[1] Hydrazine is also a liquid at room temperature, distinguishing it from cryogenic propellants that require more complex propulsion system designs. This also means that hydrazine can be stored onboard spacecraft for prolonged periods without the risk of boil-off.

Hydrazine also presents a number of significant safety and environmental challenges. In its liquid form, hydrazine can be extremely damaging to living tissues. It has also been shown to lead to increased risk of cancer in animals exposed to the substance.[2] Hydrazine also has a very high vapor pressure at room temperature, and thus produces relatively large amounts of toxic gasses. As a result, Self Contained Atmospheric Protective Ensemble (SCAPE) suits are required in order to safely handle the propellant. Hydrazine also presents a significant flammability hazard, and thus requires even further regulations to mitigate this danger.[3]

The procedures required to safely handle hydrazine are well established. However, as space flight shifts from government organizations to the private sector, the logistical costs of using hydrazine systems is often a burden for companies with limited budgetary resources. Thus, a new propellant is desired to replace hydrazine that will have equal or greater performance while significantly reducing associated handling hazards and costs.

1.3 AF-M315E: A Potential Replacement for Hydrazine

AF-M315E is a spacecraft monopropellant developed by the Air Force Research Lab at Edwards Air Force Base in California. It is significantly less toxic than hydrazine and has virtually zero vapor pressure at room temperature.[4] This characteristic alone eliminates the need for SCAPE suits and significantly reduces the inherent costs of using the propellant. AF-M315E is also much less flammable than hydrazine, which makes it less of a hazard to store, handle, and load aboard spacecraft.[1]

In addition to the significantly reduced costs and hazards, AF-M315E affords a sizeable performance increase over hydrazine. The propellant has a theoretical Isp of 257 seconds (9% greater than that of hydrazine). It also has a 47% higher density than hydrazine (1.47 g/cc vs. 1.00 g/cc for hydrazine).[1] This translates to a 60% increase in density-Isp, and means that a spacecraft using AF-M315E propulsion systems would require significantly smaller fuel tanks than a comparable hydrazine system, reducing the dry mass of the spacecraft. Given the increase in propulsion performance coupled with the significantly reduced handling costs and hazards, AF-M315E has garnered
much interest as a potential replacement for hydrazine in spacecraft propulsion applications.[5]

1.4 Project Objectives

The motivation for the AF-M315E hot-fire testing was to gain a fundamental understanding into how AF-M315E propulsion systems behave. This was a significant test campaign spanning several months. The focus is on generating key performance information that is most beneficial for spacecraft development efforts and mission planning.

The overarching goal of the project is to demonstrate that AF-M315E is a viable replacement for hydrazine in low-thrust spacecraft propulsion applications. As such, it is necessary to show that performance of the propellant matches or exceeds that of hydrazine. In addition, it is necessary to show that AF-M315E propulsion systems can be integrated into new and existing satellite and spacecraft designs without major changes in cost or effectiveness.

In order to satisfy the project goals, a number of objectives were developed. A test setup had to be constructed that allowed firing of an AF-M315E thruster. A propellant delivery system also needed to be assembled in order to feed propellant into the thruster in a reliable and predictable manner. Once the thruster and propellant delivery systems were constructed and operational, a series of hot-fire tests would be performed in order to observe the behavior of the thruster. Items of particular interest were how the thruster behaved during startup and the repeatability of thruster behavior.
2 Preliminary Predictions

2.1 Combustion Analysis

The performance of a spacecraft propulsion system depends heavily on the fuel used. Further, the gas properties of the combustion products inside the thruster’s combustion chamber determine how the gas behaves as it undergoes an expansion process in the thruster’s nozzle. Determining these gas properties is a complicated process, but there are tools commonly available to aid in their calculation. For this analysis, NASA’s Chemical Equilibrium with Applications (CEA) code was used to determine the combustion product composition and resultant properties of the gas mixture.[6] CEA calculates the combustion equilibrium composition by iteratively referencing thermodynamic values in order to minimize the Gibbs free energy in the combustion products. It then prints the gas properties of interest into an output file for reference in subsequent calculations. CEA will also provide theoretical performance values for a rocket propulsion system in which the fuel and certain physical parameters of the system are given as inputs. In all rocket problem calculations, CEA requires that an estimated chamber pressure ($P_c$) be specified. For this analysis, the expansion ratio of the 1N AF-M315E thruster nozzle (26.89) was also input into CEA. Since the chamber pressure was not known prior to this analysis, a range of chamber pressures was input into CEA to see how it would affect the gas properties of interest. It was found that the ratio of specific heats ($\gamma$), combustion temperature ($T_c$), sonic velocity ($a$), and thrust coefficient ($C_F$) all varied less than 1% for chamber pressures of 100 to 10,000 psia. A tabulation of values obtained at their corresponding chamber pressures is given in table 1 below.

Table 1: Gas properties output by CEA at input chamber pressures

<table>
<thead>
<tr>
<th>$P_c$ (psia)</th>
<th>$T_c$ (K)</th>
<th>$\gamma$</th>
<th>$a$ (m/s)</th>
<th>$C_F$</th>
</tr>
</thead>
<tbody>
<tr>
<td>100</td>
<td>2154.65</td>
<td>1.21</td>
<td>999.4</td>
<td>1.7360</td>
</tr>
<tr>
<td>200</td>
<td>2156.23</td>
<td>1.21</td>
<td>999.6</td>
<td>1.7360</td>
</tr>
<tr>
<td>500</td>
<td>2157.65</td>
<td>1.21</td>
<td>999.9</td>
<td>1.7361</td>
</tr>
<tr>
<td>750</td>
<td>2158.10</td>
<td>1.21</td>
<td>999.9</td>
<td>1.7361</td>
</tr>
<tr>
<td>1,000</td>
<td>2158.38</td>
<td>1.21</td>
<td>1000.0</td>
<td>1.7361</td>
</tr>
<tr>
<td>2,000</td>
<td>2158.90</td>
<td>1.21</td>
<td>1000.1</td>
<td>1.7361</td>
</tr>
<tr>
<td>3,000</td>
<td>2159.14</td>
<td>1.21</td>
<td>1000.1</td>
<td>1.7361</td>
</tr>
<tr>
<td>10,000</td>
<td>2159.70</td>
<td>1.21</td>
<td>1000.1</td>
<td>1.7362</td>
</tr>
</tbody>
</table>
2.2 Thruster Performance Prediction

Before thruster testing could be done, an analysis had to be carried out in order to predict the flow rate of fuel into the thruster that would be required in order to produce the desired one newton of thrust. With the thrust coefficient already being known from the CEA analysis, equation (1) from reference [7] can be used to find the thrust \(F\) as a function of chamber pressure and nozzle throat area \(A_t\).

\[
F = C_F A_t P_c
\]  

(1)

Since the throat area is fixed and known, finding thrust becomes a matter of solving for chamber pressure. A modified version of the choked flow equation from [7] that is solved for \(P_c\) can be used to find the chamber pressure.

\[
P_c = \frac{\dot{m}}{A_t \gamma} \frac{\sqrt{\gamma} R T_c}{\sqrt{\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}}
\]  

(2)

The term \(\sqrt{\gamma} R T_c\) in equation (2) is simply the sonic velocity \(a\). Thus, \(\dot{m}\) is the only unknown quantity in equation (2) needed to solve for the chamber pressure inside the thruster. Since the mass flow of fuel into the combustion chamber will equal the mass flow of exhaust gasses out of the nozzle, \(\dot{m}\) can be treated as the the mass flow of fuel into the thruster. Chamber pressure thus becomes a direct function of fuel flow rate into the thruster, as does thrust. It’s then possible to find the desired fuel flow rate in order to achieve the desired one newton of thrust. Figure 3 shows thrust and chamber pressure calculated as a function of fuel flow rate using equations (1) and (2).
3 Methodology

3.1 Test Setup

The work was performed at MSFC in the Propulsion Research and Development Lab Building 4205 Room 104. All testing was performed at altitude conditions with an approximate cell pressure of 0.8 to 2 torr using a single vacuum pump. All testing was done with the vacuum pump running to keep the altitude chamber at test conditions. A single-pass cross-flow heat exchanger was used to cool the propellant to the proper temperature and to help cool the thruster valve so that it would not be damaged from the heat generated by the thruster during firing. Figure 4 is the schematic layout of the entire testing system from the pressure supply panel through the chamber and cooling systems to the vacuum pump and exhaust.

The PPI iridium thruster was already in possession of ER23 from testing done the previous summer. The second thruster, made out of TZM, was procured by ER23 through NASA as a sacrificial thruster to prevent damage to the iridium thruster. The instrumentation required for testing was developed from previous test data and the investigation into the effects of cold propellant on the thruster. Pressure transducers were used in the propellant delivery system as well as inside the vacuum chamber. This allowed for the pressure inside the propellant feed line to be monitored, and also allowed the operators to see if ambient conditions inside the vacuum chamber deviated from test parameters during firing. Thermocouples were introduced into the propellant feed line immediately before the thruster valve to monitor the temperature of propellant entering the thruster. Thermocouples were also introduced to monitor the surface temperature of the thruster combustion chamber as well as the exhaust temperature. A propellant flow meter, force sensor, and optical camera were also used. Figure 5 shows a front view of the propellant feed system and figure 6 shows a back view of the thrust measurement stand with the thruster. Both figures include all

![Figure 3: Thrust and chamber pressure as a function of fuel flow rate into thruster](image-url)
The intention behind the two thrusters was to use the TZM thruster to find optimal system settings and then use the iridium thruster to gather optimal performance data. This was done to protect the iridium thruster from thermal cycling and eventual damage. MSFC supplied instrumentation that was used to gather data for performance calculations. The data gathered from these instruments included fuel tank pressure, metering valve turns and corresponding pressure drop, thrust measurement, valve current and voltage, propellant temperature, heater temperature, ambient temperature and pressure, and flow rate. Flow data would be correlated from both the turbine flow meter and the pressure drop across the metering valve to ensure accurate flow readings. The test operators recorded other instrumentation needed by the facility operators in maintaining the system and providing correct operating conditions to the propellant feed system.

Data was acquired by a single, high-speed data acquisition system (DAQ). The Dewetron system recorded at 10 kHz and allowed MSFC personnel to monitor data in real time as it was being acquired. As the data was acquired from each individual test, it would be analyzed to determine whether or not testing should proceed to the next step. Significant noise was seen when either the
Figure 5: Propellant feed system

Figure 6: Thrust measurement stand
propellant coolant or vacuum pump was turned on. An FFT analysis indicated the most prominent noise to be at 60 Hz, likely due to electrical interference. Steps were taken to dampen mechanical noise significantly, mostly due to pipes vibrating. Post-test filtering of the force sensor data would be required to minimize noise and provide a more accurate force sensor reading, as discussed later in this paper.

3.2 Heater Design

In order to bring the catalyst beds to the temperature required for combustion, a heater was designed and manufactured in-house (Figure 7). The location of the heater is just forward of the injector. The heater was fabricated for a snug fit with the thruster to provide as much thermal contact as possible. The motivation was mainly for testing in a vacuum to simulate flight-like conditions. Since there is no fluid medium for convection, the only available modes for heat transfer are through conduction and a small amount of radiation. To maximize the amount of heat transfer to the thruster, copper was used. Copper has a high thermal conductivity and a reasonably high melting point. This provided for quick temperature rises and decent temperature margins between the expected thruster temperature and the melting point of the copper.

The first heater design had two ports specifically for two small cylindrical cartridge heaters. The holes were also machined to have as much thermal contact as possible in order to maximize conductive heat transfer. This heater design was tested outside of the vacuum chamber and reached approximately 400 degrees Celsius in under 10 minutes. The second iteration of the heater had two additional ports drilled at its top. These ports were designed for the insertion of two type-E thermal probes.
probes to measure the surface temperature of the thruster skin (Figure 8). The thermal probes only measured the temperature on the external surface of the thruster due to the assumption that the temperatures would be approximately the same inside of the combustion chamber prior to firing.

Initial attempts to test the heater at altitude conditions were not successful due to the cartridge heaters not having adequate thermal contact to diffuse thermal energy across the copper material. This resulted in both cartridge heaters used in the initial design becoming dysfunctional due to overheating. A better approach was developed without changing the design of the copper block. Instead of having two cylindrical cartridge heaters, two heating coils created from nichrome wires were used. The wires were coiled with a slightly larger diameter than the heater ports. This was intentional to allow the metal wire to fit tight against the walls of the heater ports for a guarantee of thermal contact. The coils also naturally want to expand due to thermal expansion during heating, further ensuring solid contact between the heating coils and the copper heating block. Another altitude test was conducted to test the modified heater and it was a success. The heater drove the temperature up to approximately 400°C in about 30 minutes. Precaution was taken to reach this temperature in order to prevent the wires from deteriorating.

3.3 Thruster Configuration

The thruster configuration used for testing required careful consideration of the injector characteristics and internal geometry of the thruster. Figure 9 shows the important components and features for bringing the thruster to a test-readiness state. The injector used was designed to evenly distribute propellant onto the front face of the leading catalyst bed. The proof pressure of the thruster valve as well as the desired flow rate were the primary limitations on injector design. A conventional injector that produces an atomized spray was found to require impractically small fea-

![Figure 9: Assembly drawing showing the components and features of interest for properly configuring the thruster for testing](image)

![Figure 10: Thruster along with two catalyst beds and the TZM spacer used for testing](image)
tures. The injector design chosen is satisfactory as long as care is taken to insure that the leading catalyst bed is sufficiently close to the face of the injector. For this test setup, a catalyst-injector clearance of less than 0.010 inches was desired. If the space between the injector and the front face of the catalyst was too large, propellant could potentially begin to pool in the bottom of the thruster. The pool would then undergo a detonation after making contact with the catalyst. There is evidence to suggest this mode was responsible for a previous thruster failure.

The catalyst beds were stacked on top of each other inside the thruster with the first one resting against the catalyst bed seat. The catalyst bed seat was simply a small ledge machined inside the thruster for the catalyst beds to sit against. The length of each individual catalyst bed was measured prior to stacking inside the thruster. It was found that the catalyst bed interfaces would mesh together, and account for approximately 0.005 inches of loss of total length of the stack per interface. In order to accurately measure the stack height, the catalyst beds to be used for testing were stacked in their final test configuration and the length of the total stack was measured. A spacer was then fabricated from TZM that would sit between the catalyst bed seat and the catalyst beds. The spacer helped to bring the catalyst-injector clearance within desired limits. The catalyst bed stack and TZM spacer can be seen in figure 10. It should be noted that although figure 10 shows two catalyst beds, four catalyst beds were used in the final test configuration.

3.4 Flow-Testing AF-M315E Propellant

Several sets of flow tests were conducted in an attempt to verify the calibration of the turbine flow meter and determine the pressure drop and flow characteristics of a needle valve in the system. The first set of flow tests determined that the initial calibration on the turbine flow meter was inaccurate when used with AF-M315E due to the high viscosity of the propellant. Using the traditional Cv equations [8] with the pressure drop across the needle valve also proved inaccurate.

An attempt was made to model the flow characteristics of AF-M315E with mixed results. A second series of flow tests was conducted and a numerical model for predicting flow was built using the methods prescribed in ANSI/ISA-75.01.01-2012.[9] A new turbine flow meter was calibrated for use with viscous fluids and was installed prior to starting this test series. The primary challenges of modeling the flow of AF-M315E propellant in a 1-newton thruster setup arise from a combination of relatively small flow rates and the high viscosity of the propellant. The combination of these two factors provides for very low Reynolds numbers (on the order of 101). As such, industry standard Cv relations do not hold in this setup, and corrections for viscosity must be made before a reasonable
Flow tests were conducted by flowing propellant through a Swagelok S Series needle valve with the Cv set to 0.0029 based on literature from the manufacturer.[10] Pressure transducers were placed immediately before and after the valve (Figure 11). A solenoid valve downstream of the needle valve was opened and the two pressure transducers were monitored until they had reached steady state. Once steady state was achieved propellant was gathered in a beaker while the flow was timed. The propellant gathered was weighed with a precision scale and flow rate was calculated using the time that propellant was flowing into the beaker and the known density of the fluid. Selected values from flow testing are shown in figure 12.

ANSI/ISA-75.01.01-2012 involves several valve-specific parameters, such as a valve-style modifier and internal orifice diameter. Since these were not known values, the flow model was modified until predicted flow rates as a function of pressure drop matched test data obtained (Figure 13). The model was able to be fitted to the test data to a degree of accuracy of ±3%.

Given past difficulties experienced in attempting to model the flow of AF-M315E propellant, initial confidence in the model was high. However, later flow testing showed that the model was consistently under-predicting actual flow rates that were obtained, as shown in figure 14.

One possible explanation for the disagreement between measured values and model predictions is the viscosity changes that could have occurred when propellant in early testing was exposed to ambient air. AF-M315E is a highly hydrophilic ionic liquid, and is known to gain mass by absorbing water from humidity when exposed to ambient air. Given the large discrepancy between the viscosities of water and AF-M315E, it is plausible that water being dissolved into the propellant would cause the viscosity to decrease appreciably. Hence, if this propellant with water dissolved into it were reused for subsequent flow tests (as it was for the data produced in figure 14), the lower

Figure 11: Flow testing setup with pressure transducers before and after needle valve

Figure 12: Data from flow testing showing flow rate as a function of pressure drop across the needle valve
viscosity would cause flow to be higher than predicted.

Future testing is needed to fully characterize the effects of water absorption on flow predictions of the propellant. The creation of a robust model incorporating viscosity changes due to dissolved water is needed, but was not feasible given the time constraints of the project.

### 3.5 Data Filtering

Significant efforts were made to eliminate noise in the test setup. As mentioned previously, some noise was eliminated by securing piping that tended to vibrate in the propellant delivery and cooling system. However, a significant noise floor persisted. Figure 15 shows a frequency spectrum analysis of the test data with a significant peak at 60 Hz. This noise is likely due to a 60 Hz alternating current interfering with the data acquisition setup. In order to reduce the noise seen in the test data,
a smoothing function was applied to the data specifically to average out the 60 Hz noise. Figure 16 shows raw data with the smoothed data overlaid on top. The smoothing function helped to significantly reduce the noise while still accurately representing the shape of the thrust curve. All data presented in the results section has this smoothing filter applied.
4 Results

4.1 Hot-Fire Testing Results

Hot-fire testing was conducted on four separate occasions. Initial tests involved only short-duration pulses between 30 and 100 ms, and showed no appreciable thrust response. The second series of tests involved pulses up to 500 ms (Table 2). After this set of tests the operators noted marks on the plume deflection plate from partially burned propellant, indicating that full combustion was not achieved during parts of testing. Figure 17 shows the thrust profiles from each of the tests on this day. The thruster was fired for pulse lengths of 100, 250, and 500 ms. The t=0 point is set as the time at which the thruster valve was commanded to open. With this thruster and thrust measurement system, pulses of duration less than 250 ms have not shown any appreciable response on the force sensor. The most important part of this data set is the change in the behavior of the thruster over time. During the first 250 ms test, there was essentially no rise in thrust over time. Despite the absence of detectable thrust, a thermocouple in the thruster plume did record an increase in temperature, indicating that combustion was taking place inside the thruster. Similar response was recorded for the second 250 ms pulse. During the third 250 ms pulse, there was a spike in thrust approximately one second after the valve was closed. This was likely caused by unburnt propellant remaining in the catalyst bed until it started to react well after propellant flow had been stopped. During the first 500 ms test, the propellant reacted at a slow rate both during propellant flow and after the valve shut. For the second 500 ms test, there was a short delay followed by a quick rise in thrust.

Table 3 describes the full set of tests on the 27th. Figure 18 shows the thrust profiles of each of the 250 ms pulses in the series. One important feature of this plot is that the first pulse shows significantly reduced reactivity. In that pulse, the propellant seems to slowly burn even after the thruster valve closes. Both of the later pulses show a much faster rise in thrust. Figure 19 shows all of the thrust profiles from 15:39 onwards during this day of testing.

Each thrust profile in this series is remarkably similar even though the pulse durations are very different. In each test: the valve opens, there is a delay before the reaction begins, and there is a period of increasing thrust at a roughly linear rate. The two 250 ms pulses performed at 15:39 show a faster response than the 500 and 1000 ms pulses. This is likely a result of the catalysts beds heating significantly during the early pulses in this series, with 11 pulses of various lengths being performed in the 17 minute window between 15:22 and 15:39, as noted in table 3. In contrast, there was a roughly 16 minute pause between the last 250 ms pulse and the first 500 ms pulse. During this
Table 2: Second series: July 25 tests

<table>
<thead>
<tr>
<th>Month/Day/Year, Time</th>
<th>Pulse Length (ms)</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>07/25/2016,12:38</td>
<td>100</td>
<td>No measured thrust</td>
</tr>
<tr>
<td>07/25/2016,12:38</td>
<td>250</td>
<td>No measured thrust</td>
</tr>
<tr>
<td>07/25/2016,12:41</td>
<td>250</td>
<td>No measured thrust</td>
</tr>
<tr>
<td>07/25/2016,12:48</td>
<td>250</td>
<td>Spike in thrust approximately 1 second after closing valve</td>
</tr>
<tr>
<td>07/25/2016,12:54</td>
<td>500</td>
<td>Slow reaction during and after propellant flow</td>
</tr>
<tr>
<td>07/25/2016,12:55</td>
<td>500</td>
<td>Short delay followed by quick rise in thrust</td>
</tr>
</tbody>
</table>

Table 3: Third Series: July 27 tests

<table>
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<tr>
<th>Month/Day/Year, Time</th>
<th>Pulse Length (ms)</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>07/27/2016,15:22</td>
<td>30</td>
<td>Unable to retrieve meaningful data from short (&lt;250 ms) pulses</td>
</tr>
<tr>
<td>07/27/2016,15:25</td>
<td>50</td>
<td></td>
</tr>
<tr>
<td>07/27/2016,15:25</td>
<td>50</td>
<td></td>
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<tr>
<td>07/27/2016,15:27</td>
<td>100</td>
<td></td>
</tr>
<tr>
<td>07/27/2016,15:27</td>
<td>100</td>
<td></td>
</tr>
<tr>
<td>07/27/2016,15:29</td>
<td>250</td>
<td>Behavior similar to 07/25/2016,12:54 test</td>
</tr>
<tr>
<td>07/27/2016,15:36</td>
<td>100</td>
<td></td>
</tr>
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<td>07/27/2016,15:36</td>
<td>100</td>
<td></td>
</tr>
<tr>
<td>07/27/2016,15:39</td>
<td>250</td>
<td>Behavior similar to 07/25/2016,12:55 in this test and all subsequent tests</td>
</tr>
<tr>
<td>07/27/2016,15:39</td>
<td>250</td>
<td></td>
</tr>
<tr>
<td>07/27/2016,15:55</td>
<td>500</td>
<td></td>
</tr>
<tr>
<td>07/27/2016,16:09</td>
<td>500</td>
<td></td>
</tr>
<tr>
<td>07/27/2016,16:19</td>
<td>1000</td>
<td></td>
</tr>
<tr>
<td>07/27/2016,16:31</td>
<td>1000</td>
<td></td>
</tr>
</tbody>
</table>

Table 4: Fourth series: July 29 test matrix

<table>
<thead>
<tr>
<th>Month/Day/Year, Time</th>
<th>Pulse Length (ms)</th>
<th>Notes</th>
</tr>
</thead>
<tbody>
<tr>
<td>07/29/2016,09:29</td>
<td>1000</td>
<td>Thruster failed ~1.5 seconds into test</td>
</tr>
<tr>
<td>07/29/2016,09:34</td>
<td>5000</td>
<td>Test not performed</td>
</tr>
<tr>
<td>07/29/2016,XXXX</td>
<td>5000</td>
<td></td>
</tr>
</tbody>
</table>
Figure 18: Thrust profiles from all three 250 ms pulses during the third series of test firings

Figure 19: Series 3, late day thrusts

Figure 20: Series 4 thrust profiles
time, it is likely that the catalyst beds cooled due to heat being conducted away from the thruster and into the thruster valve and test stand. This theory is supported by the fact that a thermocouple measuring the surface temperature of the thruster valve registered 47.50°C at the end of the 15:39 test, but had increased to 48.45°C by the beginning of the 15:55 test. Similarly, there were long pauses between tests after 15:55 until the end of testing that day, accounting for the very similar response time in each of those tests.

July 29 was the final day of testing. The goal was to complete a set of 5000 ms burns. Table 4 shows the test matrix for that day. That morning, the heater was brought to temperature and held for 10 minutes prior to the start of testing. Figure 20 shows the thrust profile from each of the tests that day.

Neither of these tests exhibited behavior similar to the later tests on July 27. In both cases, the thruster produced some small initial thrust. In the first test, the thruster continued to provide some small amount of thrust several seconds after the valve closed. This indicates that a relatively large amount of propellant built up inside the thruster and continued to burn after valve shutoff. In the second test, the thrust began to rise shortly before 1 second. Since the valve did not shut off flow at 1 second, it then moved into an exponential increase that caused the thruster to over-pressurize and fail.

### 4.2 Failure Analysis

Figure 21 shows three tests. Each test is a different pulse length and was performed on a different day. However, the thrust response is fairly similar. In all three tests the thrust builds slowly and continues at the same low level for a significant time after the valve closes. None of these tests show response similar to that seen in Figure 19. The thruster exhibited this same behavior, which immediately preceded the failure, until late in the test series each day.

One explanation for this behavior is that the catalyst bed was not fully heated at the beginning
of each test series. On days with a string of short pulses early in a test series, the catalyst bed was heated by combusting propellant during the early pulses. Later tests in a test series displayed fast response of the thruster because the catalyst had been heated during those early pulses. On the day of the failure, there were no early short duration pulses. The operators moved directly to a 1 second and then a 5 second test.
5 Conclusions and Future Work

Testing indicates that the propellant response in a thruster is particularly dependent on catalyst temperature. With this test setup, the heater was capable of bringing the catalyst temperature to a level that would cause propellant to combust, but not at a suitable response rate. This fact is illustrated by the pulses seen in figure 21. In these pulses, it is apparent that combustion is occurring after propellant flow into the chamber has stopped. This indicates that the catalyst was hot enough to cause combustion to occur, but not at a rate fast enough to burn all of the propellant as it entered the thruster. However, the response time of the thruster was significantly better once the catalyst had been heated by the burning of propellant. This effect is illustrated particularly well by the third pulse in figure 18. In this pulse, there is a short pause followed by a fast buildup in thrust. When the valve is closed at 250 ms, thrust begins to decrease immediately. This suggests that the catalyst was hot enough during this pulse to burn the propellant quickly, leaving no residual propellant inside the catalyst to continue combusting after the valve had closed.

Further evidence of the sensitivity of thruster response to catalyst temperature can be seen in the above discussion of how response time varied during the third series of testing based on how long the catalyst had to cool between tests. The next iteration of this test setup will have a focus on thermally insulating the thruster and catalysts from the thruster valve and test stand, thus alleviating the need to worry about heat being conducted away from the catalysts.

A critical component of future work will include developing a method to better understand catalyst bed temperature. The current test setup relies on thruster response to gauge whether the catalyst is hot enough. A more direct way of measuring catalyst temperature will allow for thruster response to be assessed as a function of catalyst temperatures. In addition, future testing using the current setup will include a series of short pulses in order to heat the catalyst beds by combusting propellant. Only after this series of short pulses will longer duration testing be attempted.

Figure 21: Various pulses exhibiting thrust after valve shutoff: 7/25-500 ms pulse; 7/27-250 ms pulse; 7/29-1000 ms pulse
The thruster began to approach a steady state ~0.8 seconds into the one second pulses. Mass flow of propellant into the thruster decreased as thrust and chamber pressure built inside the thruster. An Isp of 98 seconds was recorded just before the valve was closed during both of the one second pulses. It is expected that thrust and Isp will increase as chamber temperature and pressure builds during longer duration firings.

Additional research and testing is needed on the injector design for this thruster. Future work will determine the effect of catalyst coverage and injection velocity on thruster performance and start-up behavior.

During flow testing, the turbine flow meter was accurately able to measure the flow during steady state operation. However, due to the very low flow rates, the flow meter was operating at frequencies below 10 Hz. This caused significant delays in the flow meter readings and reduced the ability of the flow meter to measure low flow rates. A Coriolis flow meter will be installed into the flow system in order to better understand the flow system during low flow rate tests.
6 Acknowledgements

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7 References


