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Development Of An Ionic Green Monopropellant 1 Newton Thruster For Small Satellite Application

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DEVELOPMENT OF AN IONIC GREEN MONOPROPELLANT 1 NEWTON THRUSTER FOR SMALL SATELLITE APPLICATION

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Dedication

I want to thank my family, friends, and mentors for illuminating the path in higher education. You give me the strength, motivation, and guidance to persevere in this journey.

DEVELOPMENT OF AN IONIC GREEN MONOPROPELLANT 1 NEWTON THRUSTER FOR SMALL SATELLITE APPLICATIONS

by

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DISSERTATION

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Executive Summary

In effort to advance green monopropellant propulsion using AF-M315E, a 1N thruster was developed to demonstrate the propellant capabilities for low thrust requirement missions. The current study presents the development of a 1 Newton thruster. The thruster was designed based on available literature for initial parameters. In house developed catalyst was packed into the 1 Newton thruster, and mounted onto a thrust stand for testing. Two configurations of catalyst were used for this testing campaign. As a result, temperature peaks from AF-M315E decomposition gases reached up to 1400°C with an iridium loading factor of 25% by mass with both catalyst configurations. Thruster testing shows the current design of the thruster is capable of reaching the desired thrust of 1 Newton.

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Chapter 1 Introduction

1.1 Cube Satellites Background

CubeSats have a very limited volume for payload and flight components, most of the ones deployed in their earlier years did not incorporate any type of propulsion or maneuvering system. It was only until the second decade of the new millennium that CubeSat designs began to integrate attitude control systems [3]. These new innovative maneuvering systems range from cold gas propulsion to electric propulsion thrusters which can provide low cost and high specific impulse options, respectively, but relatively low absolute thrust levels. Since longer missions usually require the combination of both high thrust levels and high specific impulse, the alternative would be to use liquid propulsion systems [4].

On 2019 NASA sent a Buss with the green propellant AF-M315E to gain knowledge on the behavior of the propellant in space. Lunar flashlight (a CubeSat) is planned to launch on the SLS Mission 1. Launch date is set to be on late 2020. It will be one of 13 CubeSats on SLS Mission. This CubeSat was developed by JPL and NASA MSFC. It is a low-cost secondary payload concept, and will map the lunar south pole for volatiles. It will be the first CubeSat to reach the Moon. It will be the first planetary CubeSat mission to use green propulsion. Finally, it will be the first mission to use lasers to look for ice water.

1.2 Project Overview

At the center for space exploration and technology research at the University of Texas at El Paso, a CubeSat propulsion module is being developed. This propulsion module will be part of a 2U CubeSat know as Orbital Factory X or OFX. OFX will have maneuvering capabilities with a monopropellant system. Specifically, the propulsion system will use four 1N monopropellant thrusters and six micro cathode thrusters. The monopropellant used will be AF-M315E. The CAD models of the propulsion module can be seen below in Figure 1 to 4.

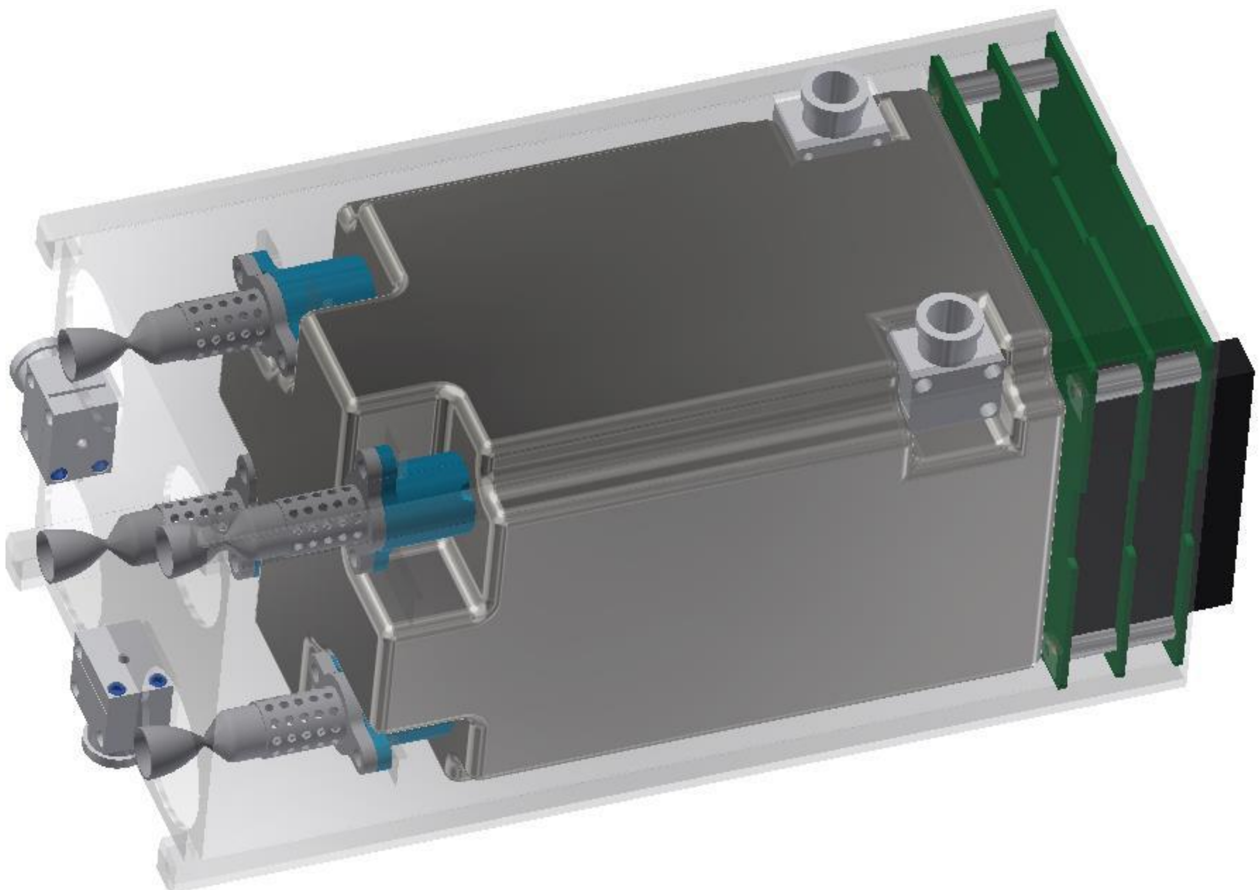

Figure 1 Orbital Factory X Propulsion Module Vertical Design

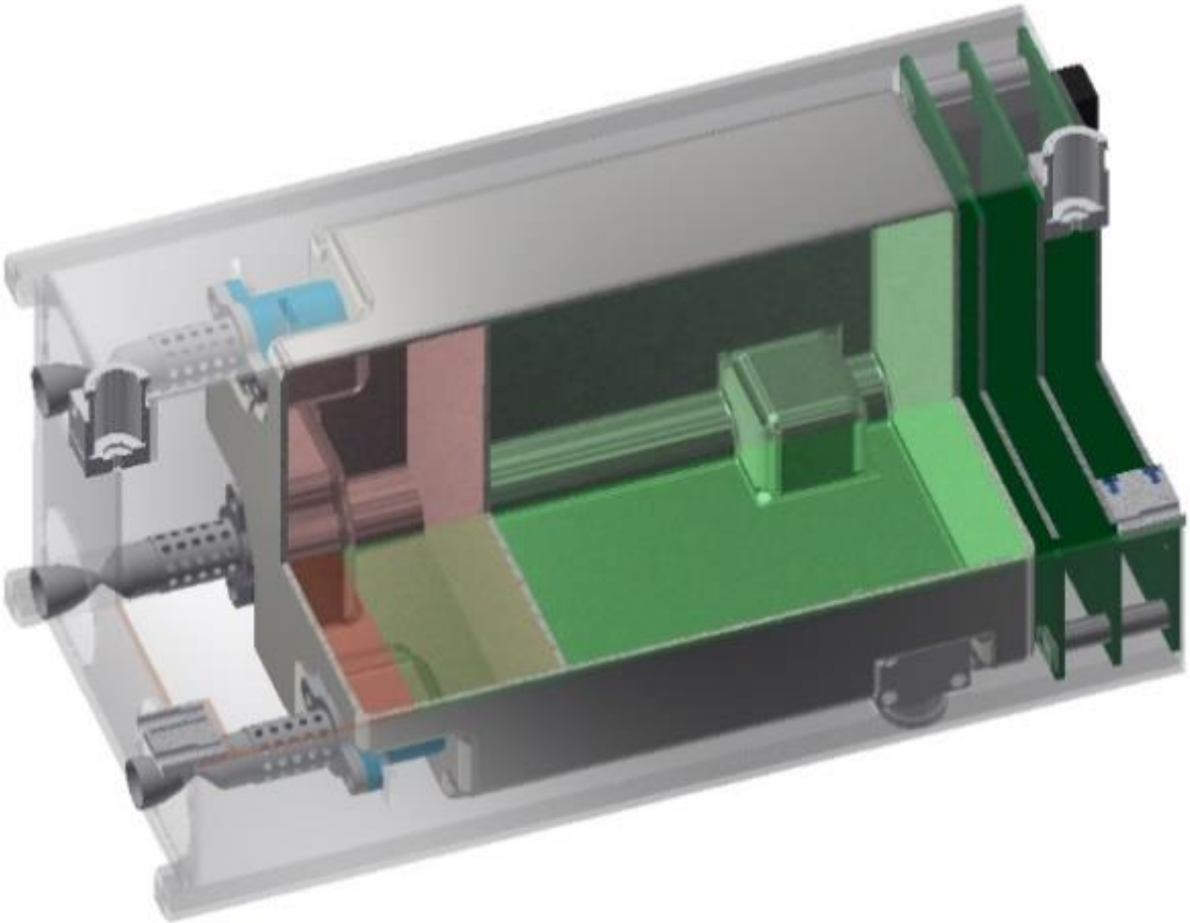

Figure 2 OFX Vertical Design

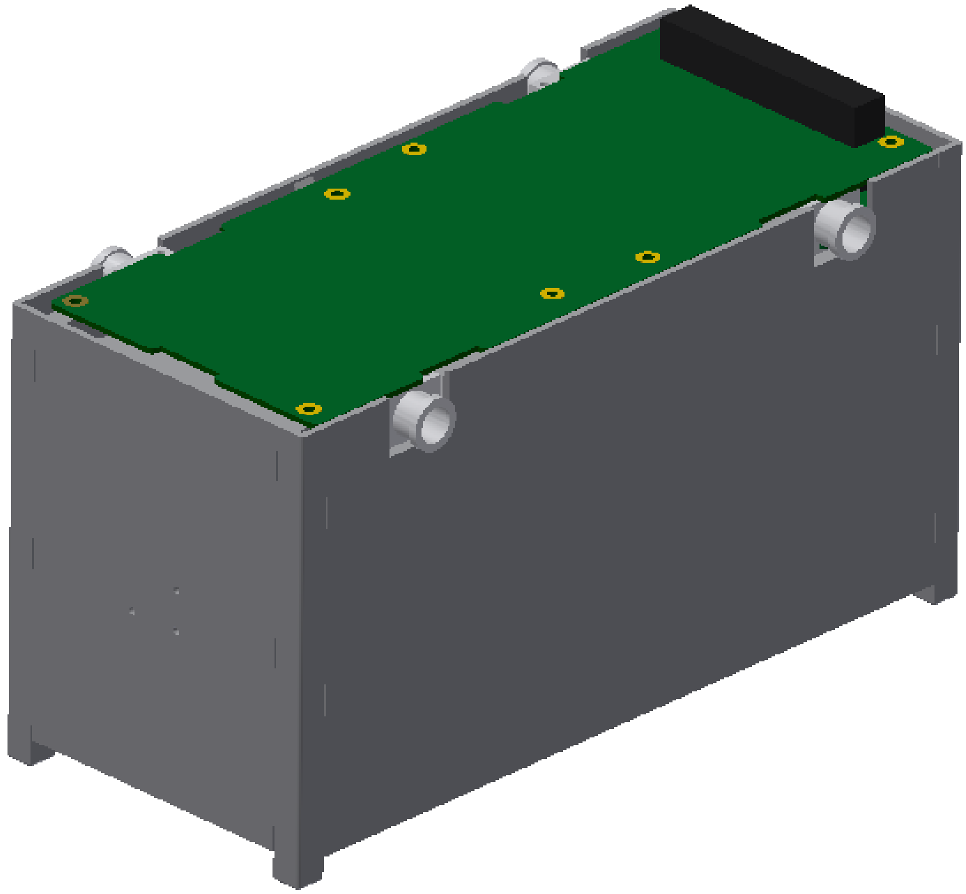

Figure 3 Orbital Factory X Propulsion Module Horizontal Design

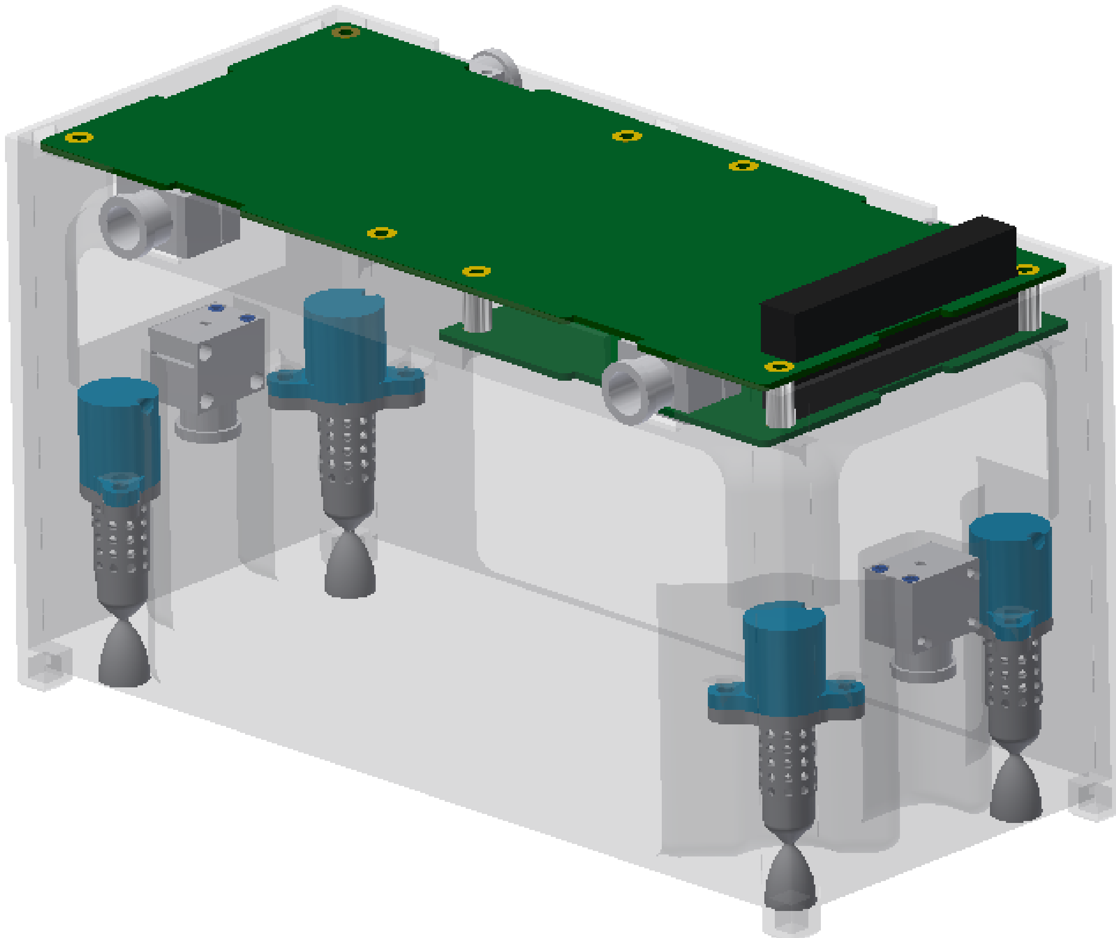

Figure 4 OFX Horizontal Design

Accomplished efforts for the tank are the 1st horizontal tank design has been machined and weld. The tank design has two cavities one for gas and one for propellant. The tank design also has a fill propellant port. This tank has been conventional machined and welded. For future efforts, the weld quality shall be check by performing hydrostatic test on the current tank. The design of the propulsion module includes five solenoid valves. One for each thruster, and one to deliver gas into propellant cavity to pressurize.

The tank was designed to fill the remaining space after integrating the thrusters, valves, and electronics. The horizontal tank has the dimensions of 90 mm x 174 mm x 90 mm and the vertical tank has the dimensions of 174 mm x 90 mm x 90 mm. The CADs were completed in SolidWorks, and astatic simulations were conducted to show that the tanks would maintain an appropriate factor of safety while withstanding the designated pressure of 2.5 MPa. The material of the tanks was assigned to be Ti6Al4V and the wall thickness for the vertical tank is 5.5 mm, and 4.5 mm for the horizontal tank.

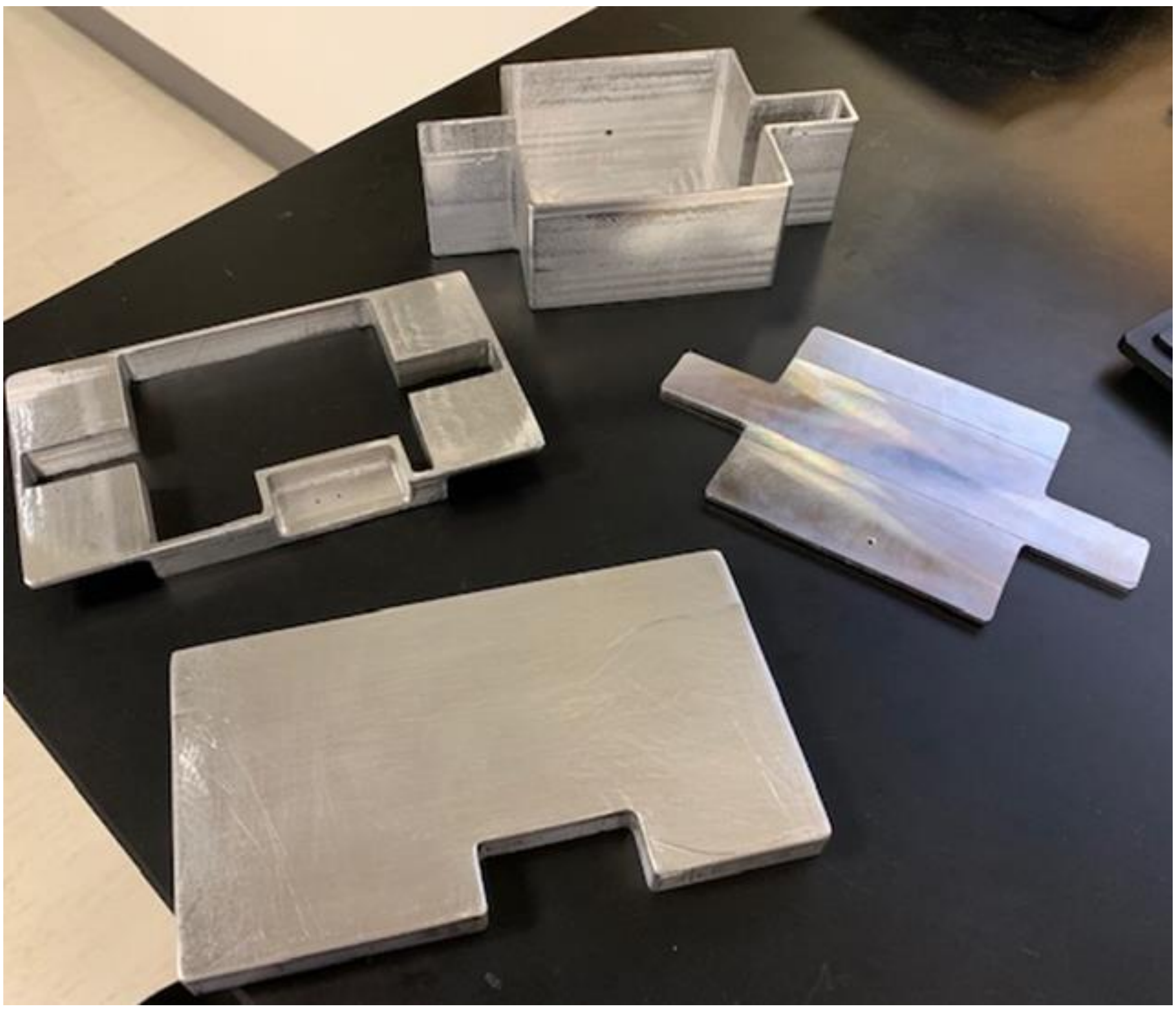

Figure 5 Horizontal Design was Conventional Machined in four Pieces

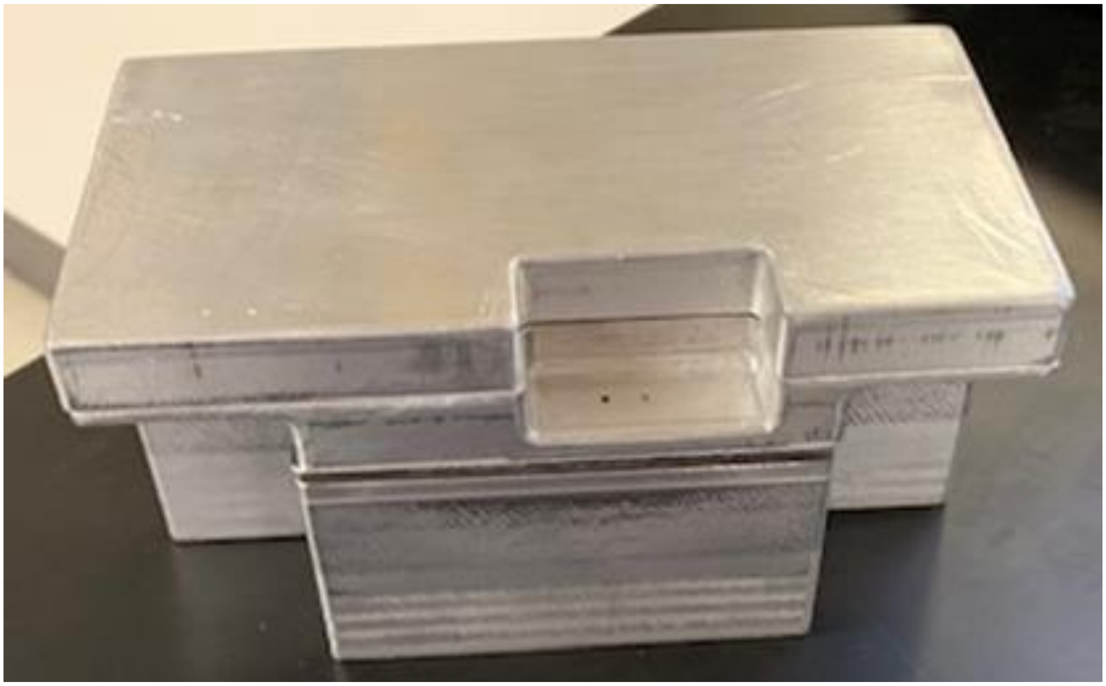

Figure 6 Horizontal Design Welded

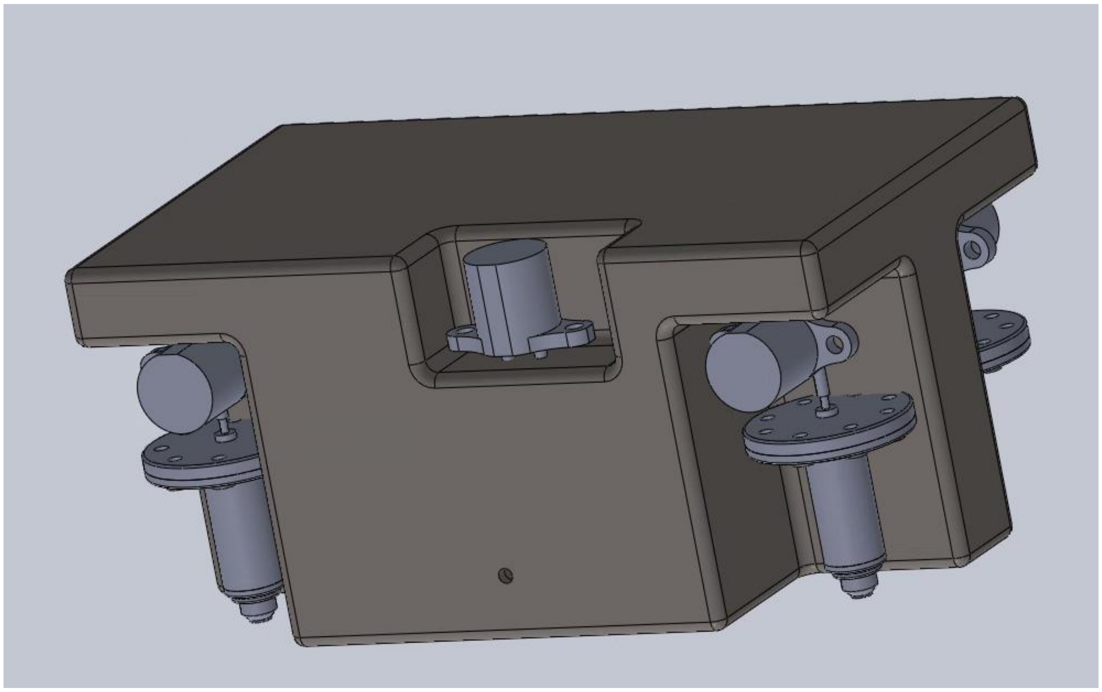

Figure 7 Horizontal Design CAD Model

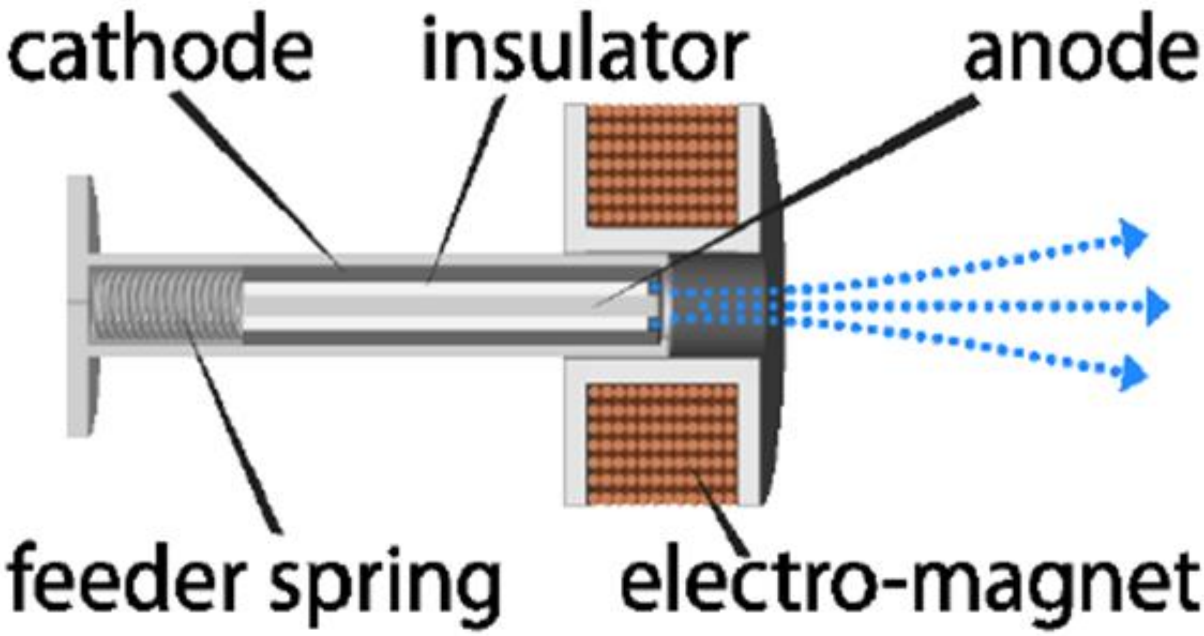

Figure 8 Image of μCAT obtained from 2014 Brochure, The George Washington University.

The figure above illustrates a micro electrical thruster. OFX will have six of these thrusters for maneuvering. These electrical thrusters were designed by The George Washington University.

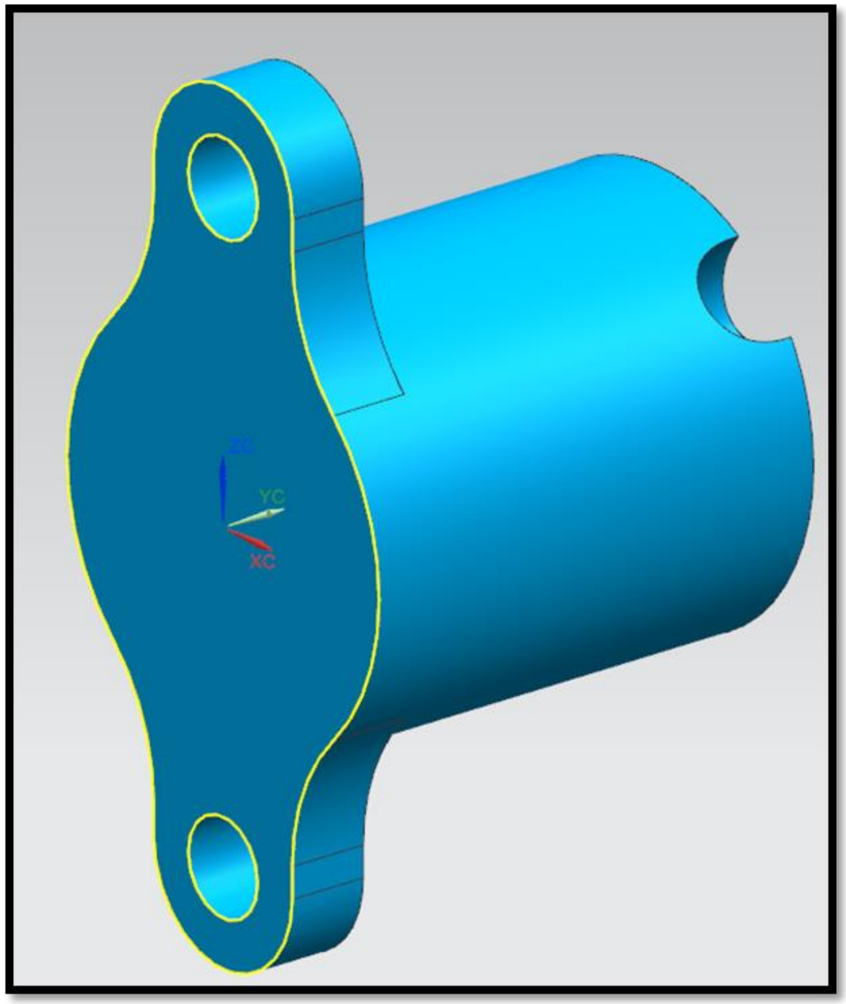

Figure 9 MSFC NASA Valve

The figure above illustrates a thruster valve. OFX will have five of these valves one for each thruster and one to pass gas into propellant cavity. These valves were designed by NASA.

1.3 Monopropellant Systems

Monopropellant systems work on the principal of a chemical decomposition of a single fluid. The fluid is required to be slightly unstable and release high amounts of heat energy when it decomposes. This high amount of heat energy creates the high temperatures inside of a thrust chamber that causes the acceleration and momentum exchange of particles. Decomposition of this chemical or monopropellant is caused by thermal energy, a catalyst, or both. Catalytic decomposition can occur at either room temperature or at elevated temperatures, depending on the type of monopropellant used.

As previously mentioned, hydrazine has been the dominant propellant when it comes to monopropellant systems. It has been an industry standard for several past decades, especially in the 20^{th} century, because it provided the best performance in terms of specific impulse from other monopropellants. Hydrazine however is a very toxic and flammable chemical that poses numerous health hazards when handling and operating. Therefore, much of the effort in the propulsion community has been dedicated to finding a suitable alternative to hydrazine that can provide relatively the same performance without all the hazards.

cSETR has always been interested in green monopropellants. Below is a time line on how the center moved from one green monopropellant to another. In 2013 the center was working on the development of millinewton thrusters using high test peroxide (90%). In 2015 the center was focusing on expanding the facilities. A pressure delivery system was design and assembled. On the other hand, a 20 Newton thruster was developed using LMP-103S. Lastly on 2018 the center started the research on the air force propellant AF-M315E.

Figure 10 cSETR Green Monopropellant Thrusters Time Line

1.4 Project Objectives

The project objective is to design and develop an AF-M315E 1 Newton thruster for small satellite applications integrating cSETR in-house catalyst bed into 1N thruster for testing and performance evaluation.

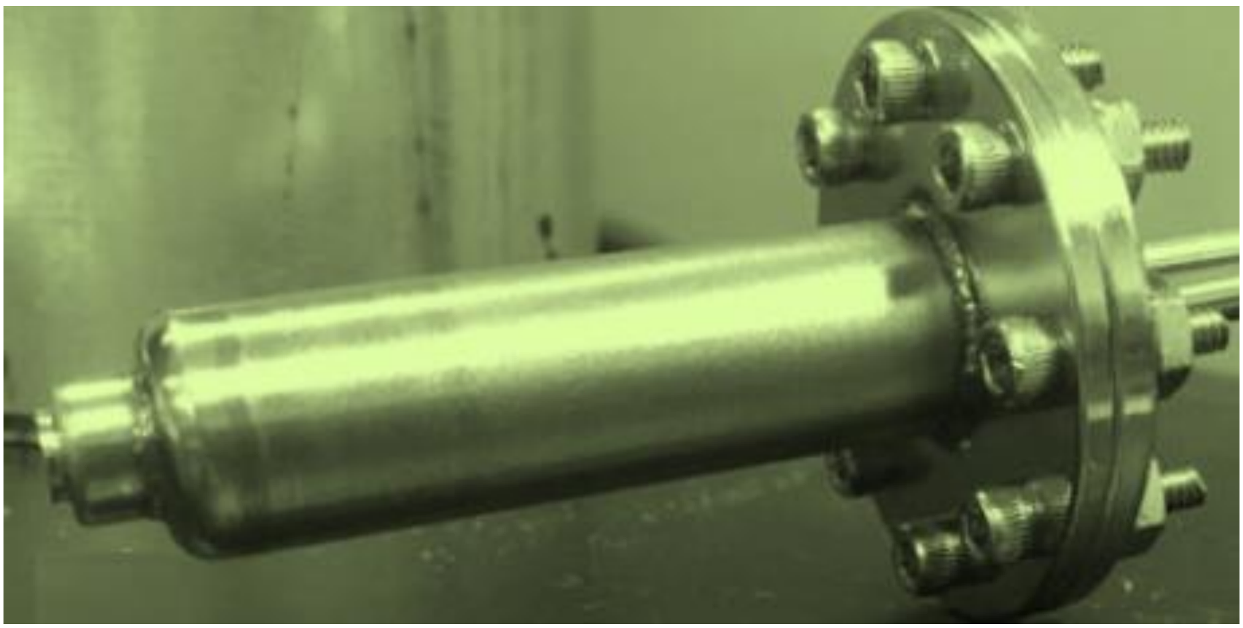

Figure 11 1N Thruster Design

Chapter 2 Methodology

2.1 1N Thruster Test Article Description

The test article was designed in subsections composed of an injector, catalyst chamber, thermal standoff, and chamber-nozzle. The catalyst used in these experiments is Iridium-based catalyst produced by students at the cSETR. The nozzle design is primarily driven by the design process of thrusters illustrated in Modern Design of Liquid Engines. The design process involves assumptions based on the decomposition process of AF-M315E. Using CEA analysis and previous experience with the propellant the design was built.

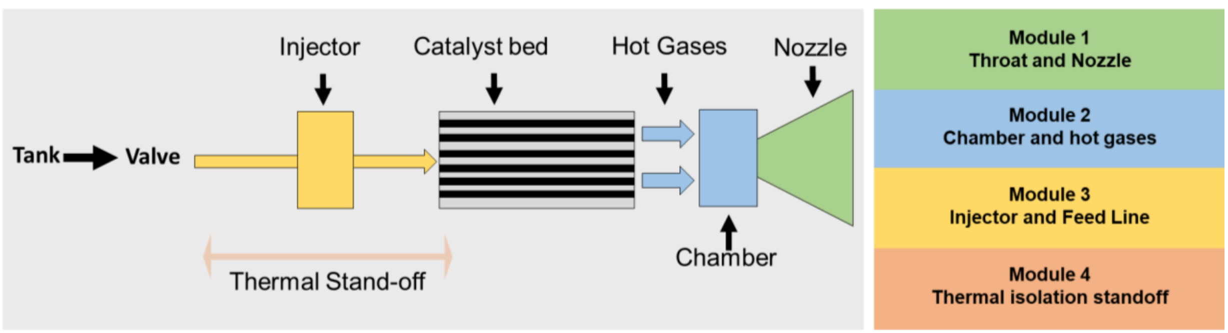

Figure 12 Thruster Design Subsections

Module 1 **Throat and Nozzle**	**Module 2** **Chamber and hot gases**	**Module 3** **Injector and Feed Line**	**Module 4** **Thermal isolation standoff**
• Flow rates • Thruster Efficiencies *C*, Cf, Isp* • Throat *Ac, Pt, Tt* • Nozzle *Ae, Ln, α, β* • Material selection	Catalyst Preheating • Catalyst Heater Selection • Heat transfer calculations • Simulation • Understanding the hot gases • Pressure drop • Pc, Tc • Dimensions *Dc, L*, L cyl, L con, angles* • Material selection • Machining selection	• Dimensions • Pressure Inlet • Pressure drop • Material selection	• Heat transfer calculations • Structural dimension • Simulation • Material selection

Figure 13 Description of Each Module

The thruster was design based on available information for thruster design. The thruster was manufactured using conventional machining process using Inconel 718. The information in Table 2 lists initial design parameters for the 1N thruster. Table 3 lists the parameters calculated for the 1N thruster.

Table 1 Thruster Design Equations

Temperature at the Throat of the Thruster	$T_t = T_c \left[\frac{1}{1+\frac{\Upsilon-1}{2}}\right]$
Pressure at the Throat of the Thruster	$P_t = P_c \left[1 + \frac{\Upsilon-1}{2}\right]^{-\frac{\Upsilon}{\Upsilon-1}}$
Area at the Throat of the Thruster	$A_t = \frac{w_t}{P_t}\sqrt{\frac{RT_t}{\Upsilon g_c}}$
Area at the Exit of the Thruster	$A_e = \frac{A_t}{M_e}\left[\frac{1+\left(\frac{\Upsilon-1}{2}\right)M_e^2}{\frac{(\Upsilon-1)}{2}}\right]^{\frac{\Upsilon+1}{2(\Upsilon-1)}}$
Characteristic Velocity	$C^* = \frac{\sqrt{\Upsilon g_c T_c R}}{\Upsilon\sqrt{\frac{2}{\Upsilon+1}^{\frac{\Upsilon+1}{\Upsilon-1}}}}$
Thrust Coefficient	$C_f = \frac{I_{sp} g_c}{C^*}$
Specific Impulse	$I_{sp} = \frac{F}{\dot{W}_p}$

Table 2 Thruster Initial Design Parameters

Chamber Pressure (kPa)	600
Chamber Temperature (^{o}C)	1700
Thrust (N)	1
Chamber Pressure (kPa)	600
Chamber Temperature (^{o}C)	1700

AF-M315E estimated specific impulse is between 220-250 seconds at sea level and 266 seconds in vacuum. This thruster is designed to be tested at sea level; the optimum specific impulse

of 240 seconds was chosen. By using the specific impulse equation, a flow rate of 4.3E-4 kg/s was calculated to produce 1N of thrust if optimum specific impulse is reached. A key design parameter is 600 kPa for chamber pressure. An analysis estimates decomposition temperature peaks of 1710 °C if ideal decomposition of the propellant is met. Another consideration in the design is the elevation of the testing location which is at 38,000 ft. with an atmospheric pressure of 12.75 psia (El Paso TX). For this reason, the nozzle expansion is only 5 to compensate with the ambient pressure.

Table 3 Thruster Calculated Parameters

Nominal Thrust	1N ± 0.05 N
Target Specific Impulse (Isp)	195 sec
Inlet Pressure (kPa)	600 Kpa
Propellant Flow Rate @ Nominal Inlet Pressure	0.41 g/s
Propellant Inlet Temperature (at FCV Inlet)	< 100 +5°C
Expansion Ratio	3:1
Propellant Inlet Fitting	1/8” Swagelok
Temperatures Sensors (type / qty)	8 Type-K
Maximum Expected Operating Pressure (MEOP)	1000 kPa

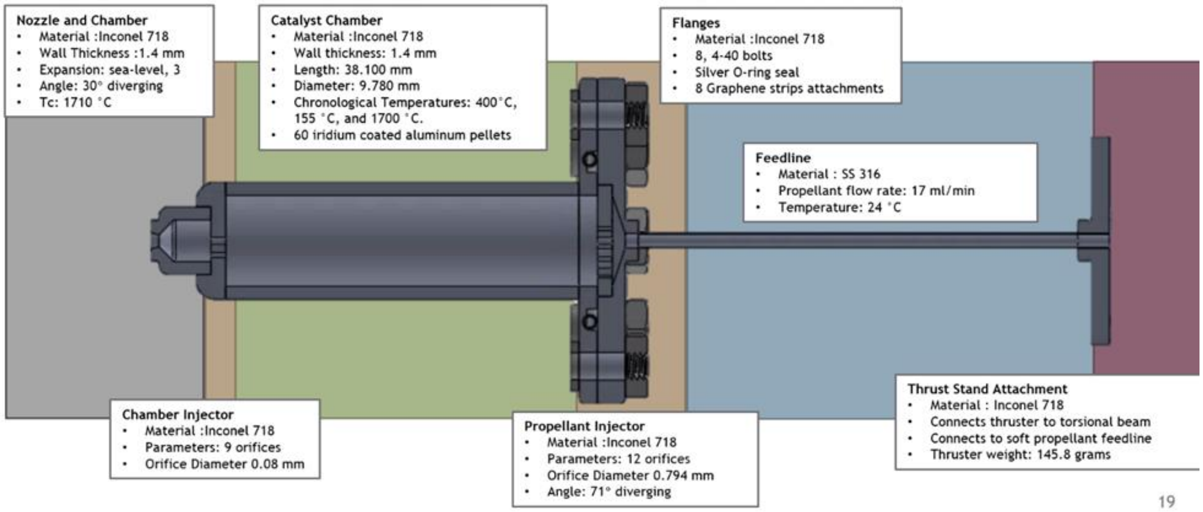

Figure 14 Thruster Description

The thruster development was separated in different section as shown in Figure 14. This method helped to look at each section of the thruster. The thruster design calculations for a 1 Newton thruster were based of the basic thruster equations, and thruster performance equations. An excel document was created to solve the equations and to have the ability to change numbers and, get results faster; if there are changes in the requirements. The excel model was validated with the excel model from NASA engineer Daniel Cavender. A CAD design of the thruster was created using NX. The Thruster was then conventional machined in sections, and finally welded together.

Once the thruster was finalized a passivation process was performed. The passivation process consists of dropping the thruster in a glass beaker. Next, adding nitric acid until the thruster is fully covered. Leaving the thruster submerged in acid overnight, and cleaning it after with distilled water. The process was performed in the chemistry department inside a fume hood, and wearing personal protective equipment.

The thruster was design to have a flange with bolts in order to change the catalyst at any moment since part of the performance of this thruster depends on the catalyst. The injector is a

shower head injector with 37 holes of 1mm in diameter each. The AF-M315E decomposition gases flow through the catalyst chamber, and cross a showerhead gas injector consisting of 9 orifices with dimeters of 0.8 mm before reaching the combustion chamber. This screen holds the catalyst in place. The length of the catalyst chamber is the same from the catalyst holder experiment. The injection line is a 1/16-inch line. In the figure below section B, it can be seen an attachment with four heat dissipation mechanisms. Also, the attachment to hold the thruster to the thrust stand is on the right side of the picture. Clearly it can be seen were the catalyst chamber, and the combustion chamber are. The small nozzle expansion is due to ambient testing conditions.

Figure 15 a) O-ring and Catalyst Placement b) Thruster Configuration.

To prevent propellant leaks, a silver O-ring is place on the flange between the chamber, and the injector as shown in Figure 7a. The iridium coated pellets were packed in the thruster, and can been seen in Figure 7a. The catalyst volume consists of 65 pellets with an iridium loading

factor of 25% by weight. The void percentage between the catalyst chamber, and catalyst material is estimated to be 30%. This catalyst is preheated to 400 °C with a coil heater surrounding the catalyst bed chamber in order for the decomposition of the propellant to start.

AF-M315E thrusters function similarly to other monopropellant thrusters. The propellant is pressure fed into the catalyst chamber. When the propellant, and the catalyst come into contact, a chemical reaction occurs. In this reaction, the green propellant decomposes. The decomposition gases pass through the chamber-nozzle generating thrust. AF-M315E requires the catalyst to be preheated to temperatures above 400 ºC. Iridium has been used as a catalyst by both Aerojet and NASA in different thrusters. cSETR has invested in the creation of an in-house catalyst. After many series of tests, cSETR has found that this in-house catalyst reaches required propellant decomposition temperatures.

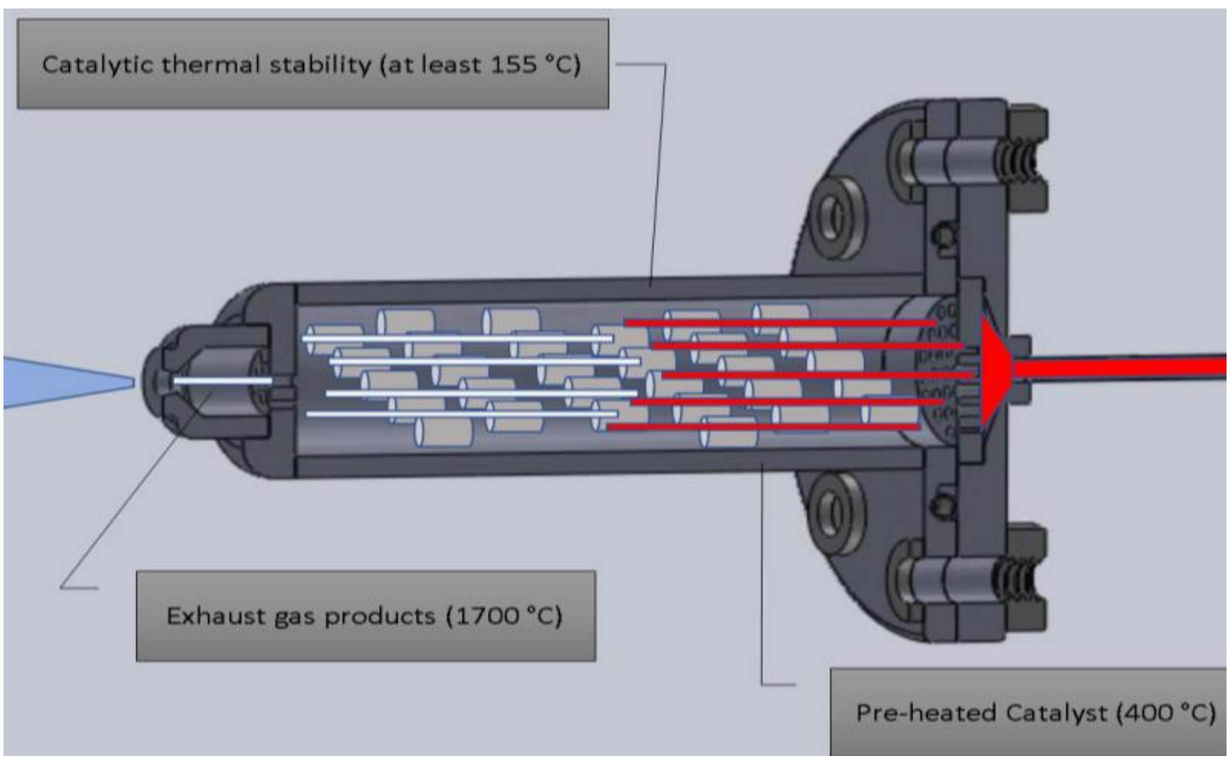

Figure 16 Catalyst and Thruster Configuration

Figure 17 Conventional Machined Thruster Parts

2.2 Heater Development

Before the propellant is introduced into the system, the catalyst chamber is preheated to 400°C. In order to accomplish this temperature a resistance heater was developed in house. Below is a prototype of conceptual design and material for the heater.

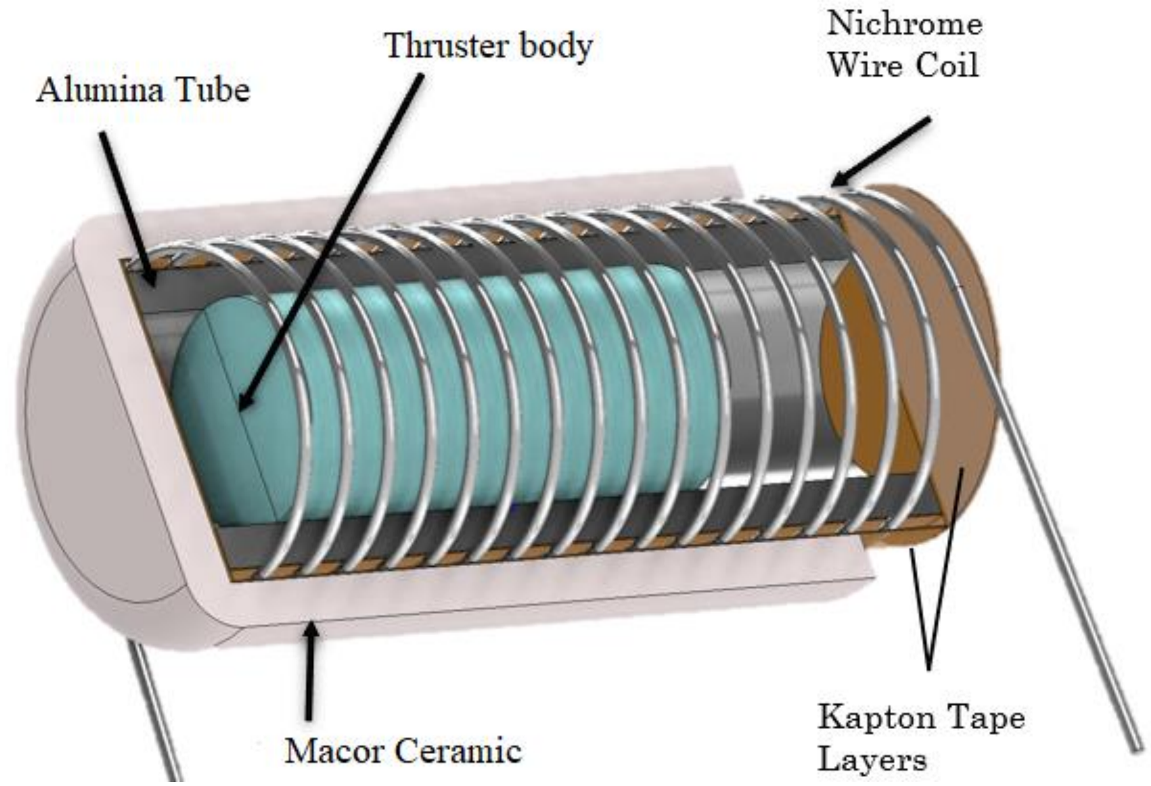

Figure 18 Heater Prototype

Three designs were built and tested to reach the desired temperature. All with the same design concept. The final heater design was designed to meet the catalyst chamber dimensions. This heater consists of an aluminum oxide shell with a coil winding of 18 turns, placed in contact with the catalyst holder. A layer of magnesium oxide is used to coat the electrical resistance to protect the wire and distribute the heat uniformly throughout the chamber length. Finally, a MACOR shell encloses the whole system, preventing heat loss.

Pellets
Inconel tube
Alumina 2hole tube
Nichrome wire
Magnesium Oxide paste
Macor

Pellets
Inconel tube
Magnesium oxide paste
Nichrome wire
Macor

Pellets
Inconel tube
Alumina tube
Nichrome wire
Magnesium oxide paste
Macor

Figure 19 Heater Design Concepts

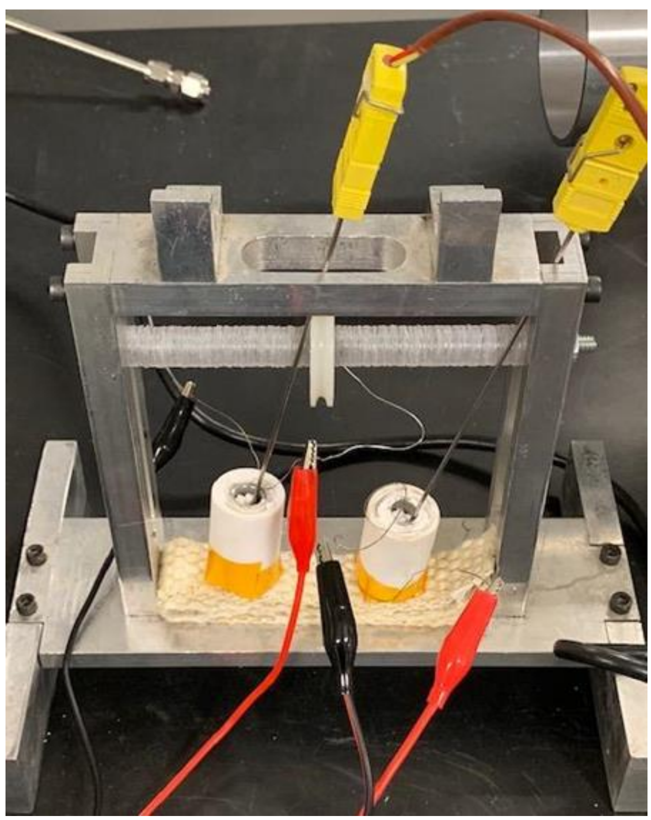

Figure 20 Testing of Conceptual Heater Designs

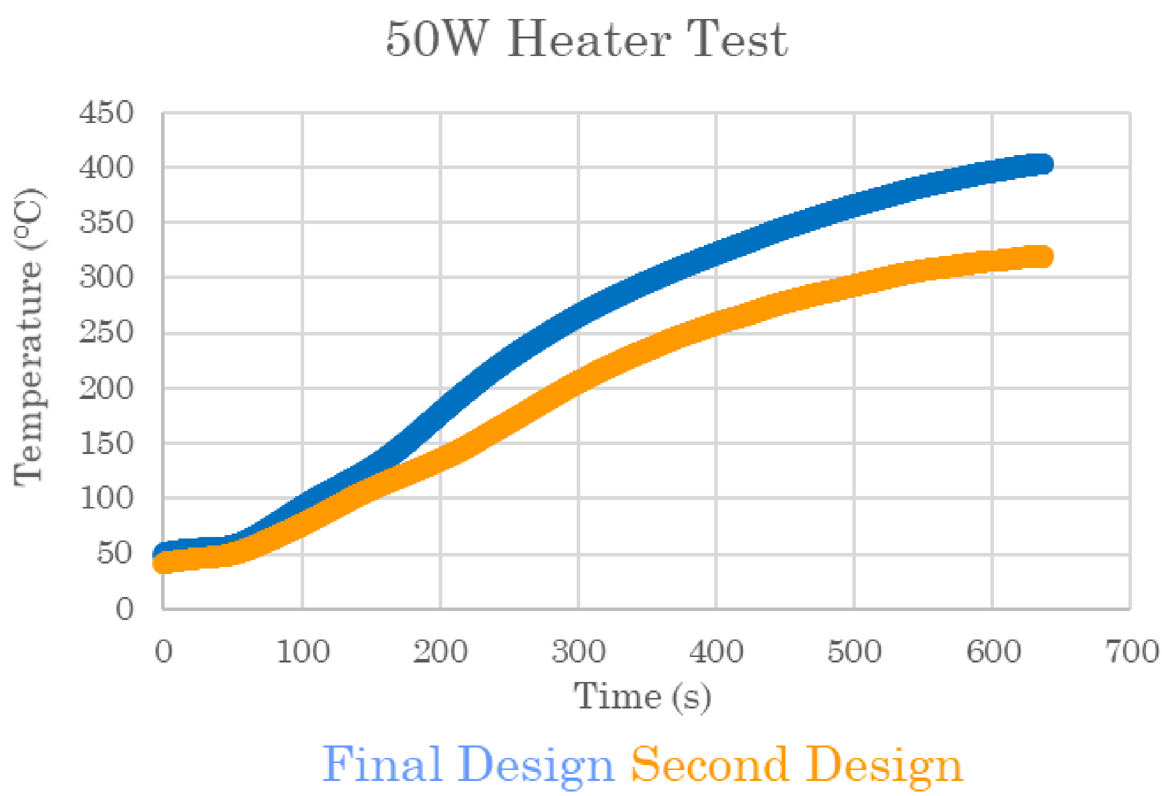

Figure 21 Heater Testing Conceptual Designs

Inconel Thruster Body
Alumina case
Nichrome wire

Magnesium oxide

Macor ceramic case

Figure 22 Final Heater Design Manufacturing Process

Figure 23 Heater Testing under Vacuum Conditions

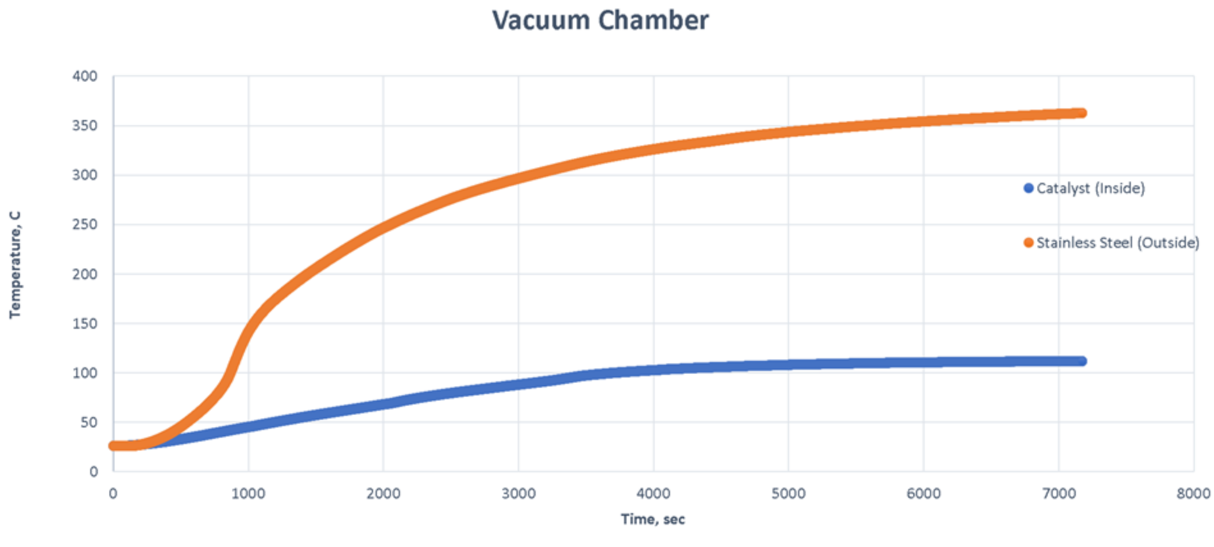

Figure 24 Heater Testing Under Vacuum Conditions Results

2.3 In-house Catalyst Development

At the Center a pellet configuration type catalyst was developed specifically for the decomposition of AF-M315E. The catalyst consists of a substrate material in a form of a pellet then coated with iridium.

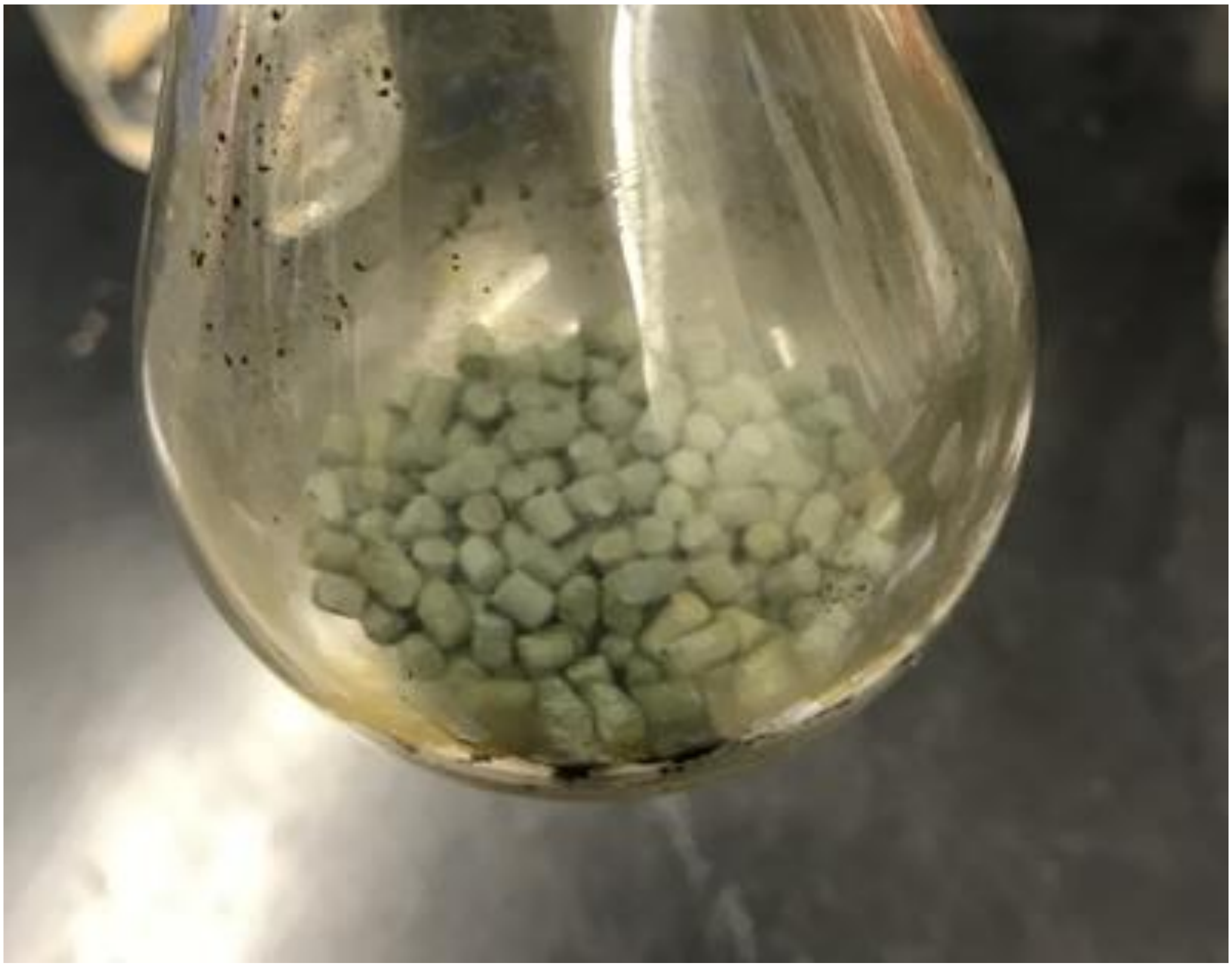

Figure 25 Impregnation of Iridium on Pellets

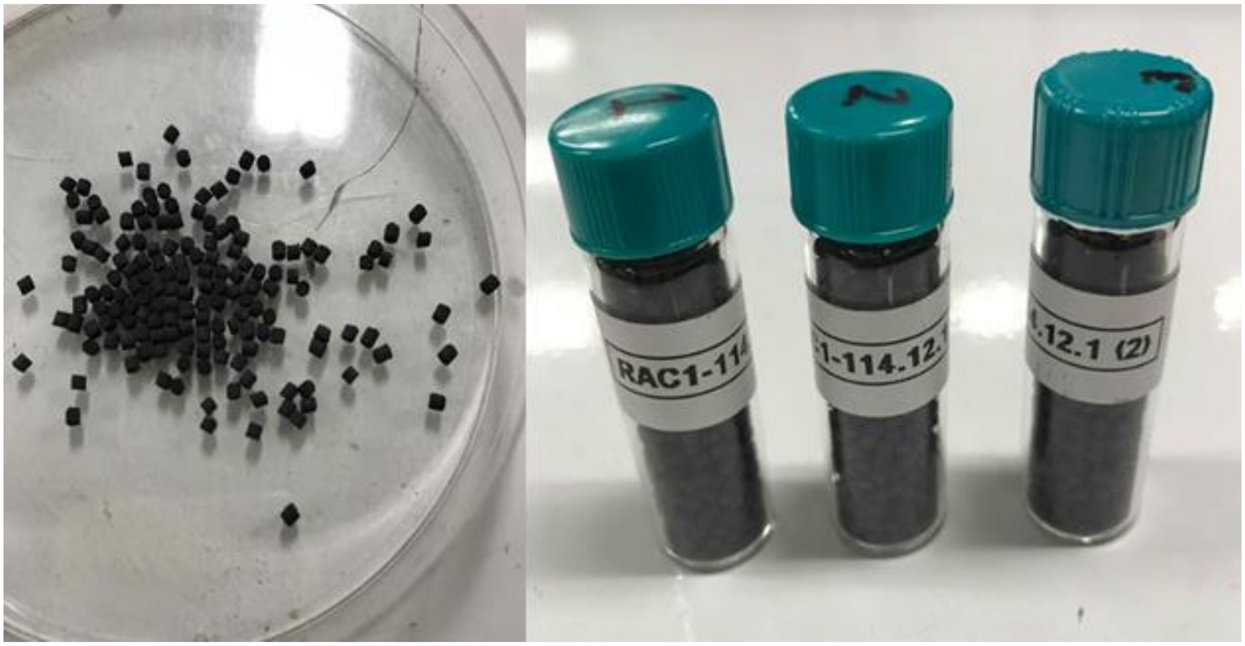

Figure 26 In House Catalyst Coated with Iridium

The two configurations of catalyst produced consists of gamma phase aluminum oxide, about 3 mm in diameter and 3 mm in length, coated with iridium. The second configuration of catalyst produced consist of silicon carbide 1 mm in diameter, and 1 mm in length, coated with iridium. The iridium loading factor of these pellets is of 25% by weight. One type of performance test conducted on this catalyst was the duty cycle testing which investigates the catalyst lifetime. By using these new substrates, it can further improve the development of a more reliable catalyst with a longer resident time. Therefore, 10 ml/min was chosen as the optimal flow rate for the current catalysts configuration. Using this optimal AF-M315E flow rate, the alumina oxide and silicon carbide pellets coated with iridium were tested for catalyst lifetime or duty cycle. The duty cycle testing consists of repeatedly running AF-M315E thorough the catalyst bed in 30 second intervals. Currently, nearly 100 runs have been conducted on the same batch of each catalyst. The duty cycle testing has been conducted only with 10ml/min. After approximate 1 hour of total AF-M315E decomposition, 22% of the alumina oxide pellets used had fractured and 0% of the silicon

carbide used has fractured. The temperatures obtained continued to peak and plateau around 1400°C, indicating no loss in performance.

The test setup to test the catalyst duty cycle consist of heating up catalyst enclosed in the holder to 400 °C. Once at the desired temperature a syringe pump full of AF-M315E injects the propellant into the warm holder at 10 mL/min for 30 seconds. A lab view program is recording the temperature at 25% the length of the holder and at 50% of the length. The setup also has the capability to collect the exhaust gases into two sample cylinders. Another experiment consists of taking the exhaust gases and analyzing them in a mass spectrometer to check the percentage of each gas decomposed. In addition, the holder has the capability of attaching a combustion chamber/nozzle with temperature ports.

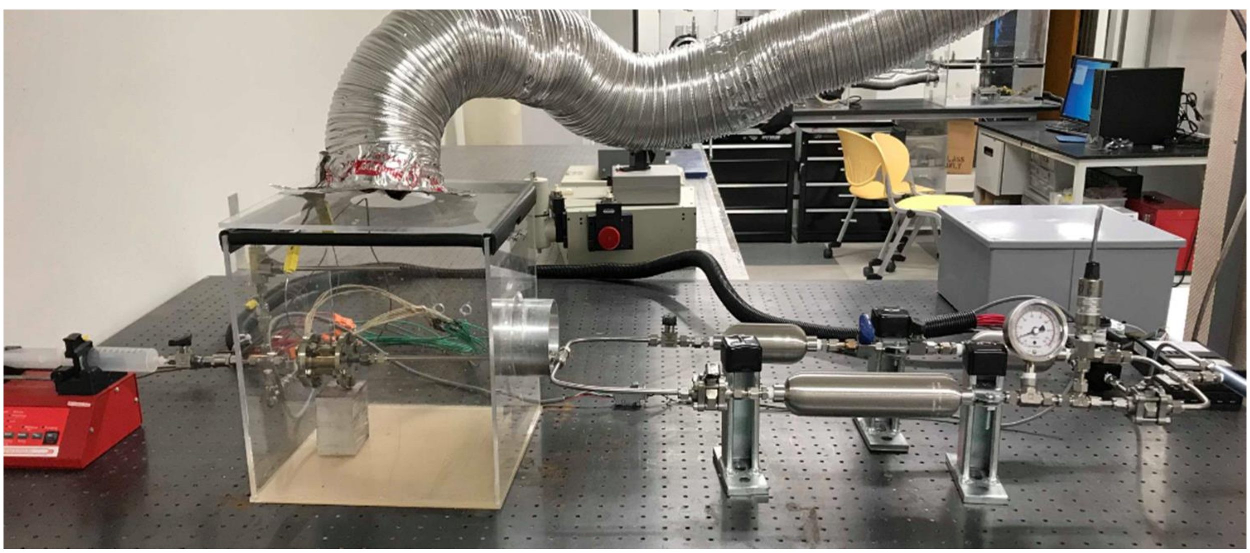

Figure 27 Catalyst Decomposition Test Setup

The Catalyst holder is machined out of Inconel 718. Around the catalyst cavity (in Figure 28, yellow cylinder in figure below), there are 6 cartridge heaters with dimensions of 1 inch in

length and ¼ of an inch in diameter. Each heater provides 100 Watts. The holder also has two flanges covering the top, bottom, and a silver O-ring between the flanges.

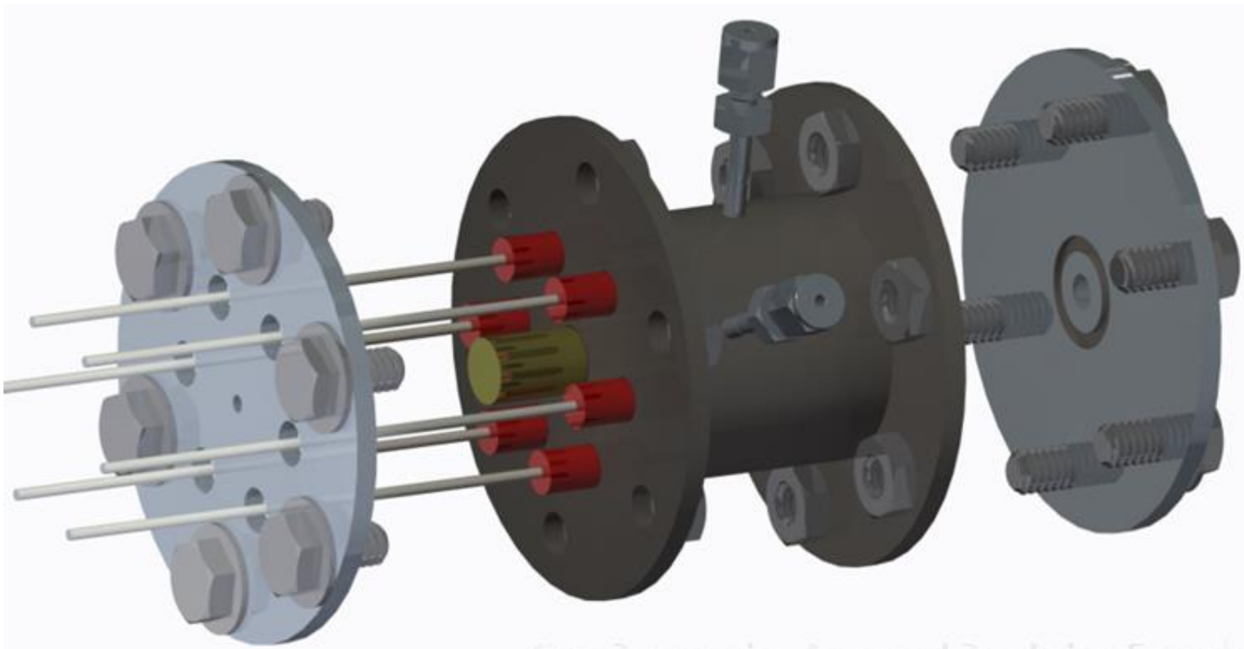

Figure 28 Catalyst Holder Experimental Concept CAD Design

Figure 29 Catalyst Holder

Figure 30 Mass Spectrometer

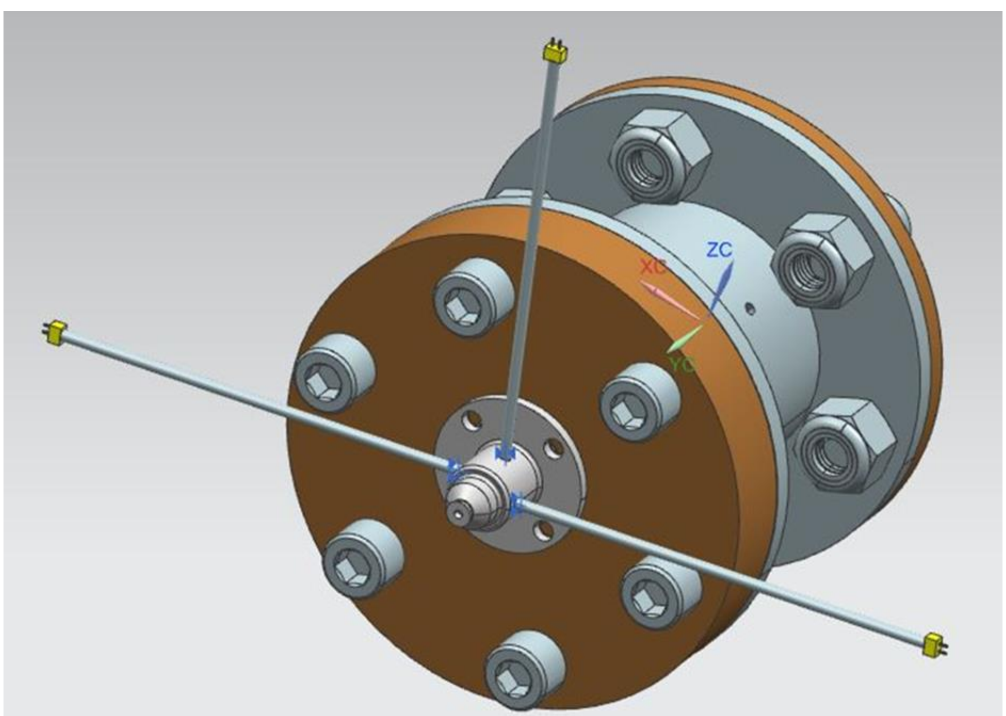

Figure 31 Catalyst Holder with Temperature Ports

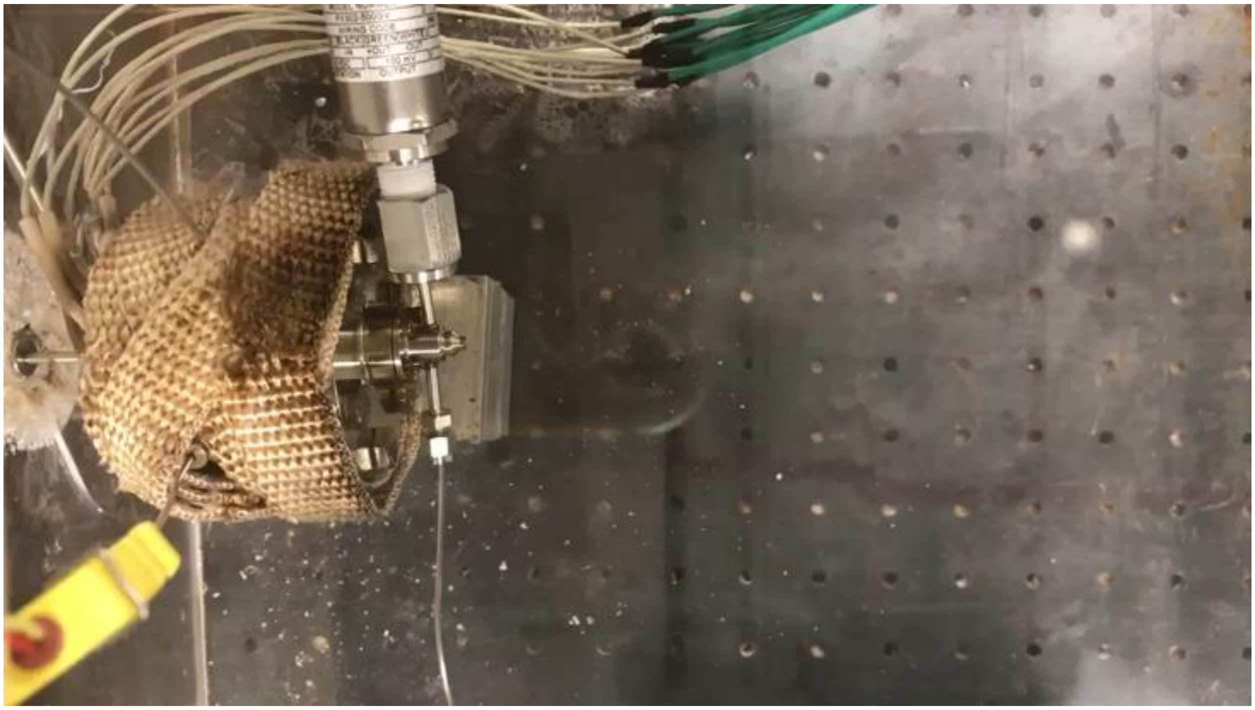

Figure 32 Nozzle Attachment and Holder with Insulation

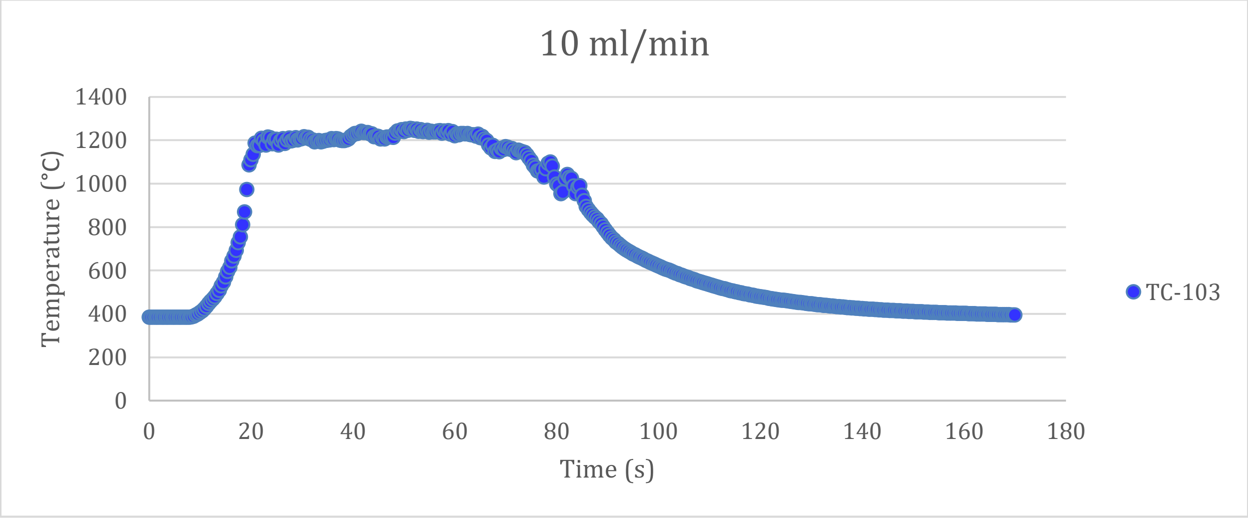

Figure 33 AFM-315E Decomposition Gas Temperature Profile

2.4 Experimental test setup

The experiment was performed at the University of Texas at the Goddard Lab. The facility contains the Multiple Altitude Simulator System (MASS) bunker area, which provides an area with protection to student when testing. The control room is able to control all associated systems remotely. The facility delivery system was design and constructed as part of Jaclyn Mejia's master thesis. The capabilities for this delivery system are the delivery of six different gases, the delivery of three different liquid propellants, the ability to measure, and record data such as, force, temperature, pressure, flow rate, and video. It also has the capability to open and close valves.

Figure 34 Bunker Delivery System

The experimental setup consists of thermocouple instrumentation, catalyst bed heater, gas delivery system, propellant delivery system, thrust stand, laser, load cell, and exhaust systems. AF-M315E is placed in a propellant tank. The catalyst bed heater is turned on until the catalyst bed reaches the preheating temperature of 400 ºC.

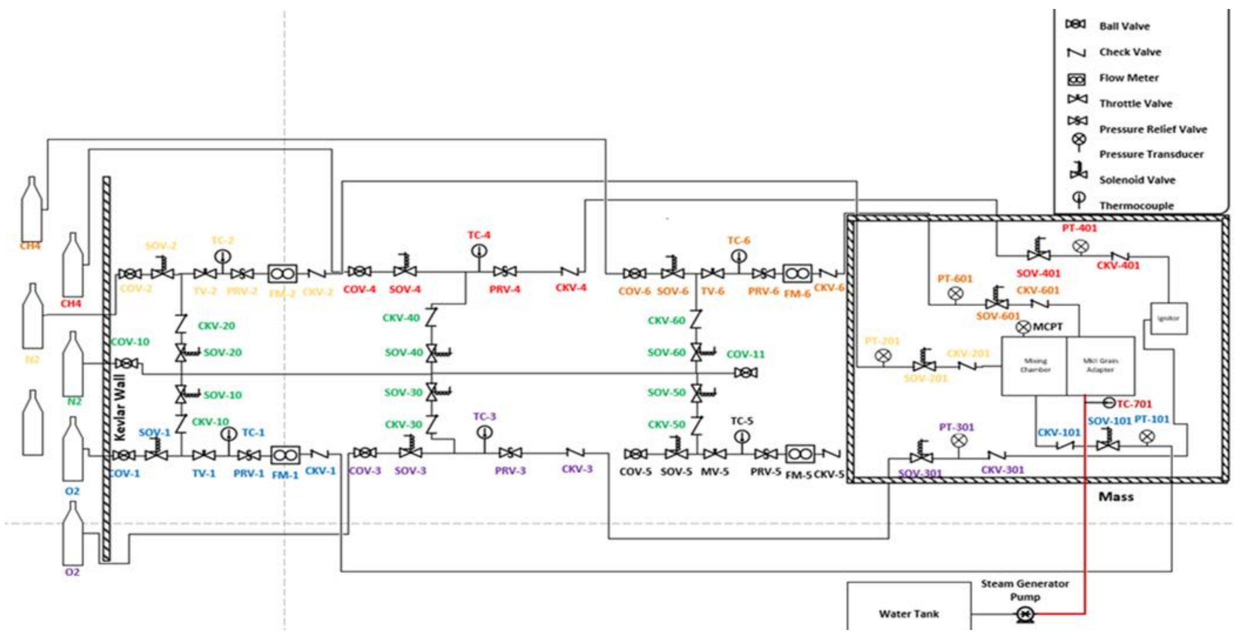

Figure 35 Facility at the cSETR lab

The facility has a gas delivery system that can provide gaseous nitrogen (GN2) and gaseous Helium (GHe2) to the test article. The gas delivery system will regulate and supply helium to the propellant tank. The tank pressure is measured with a pressure transducer It also regulates and supplies high pressure nitrogen for purging. The system was purged with nitrogen and evacuated prior to propellant loading. To load propellant into the system, the propellant was vacuumed into the propellant tank using a venturi vacuum pump. Once the system was ready for testing, the system was fired by opening the valve before the thruster. The component and flow schematic are illustrated below:

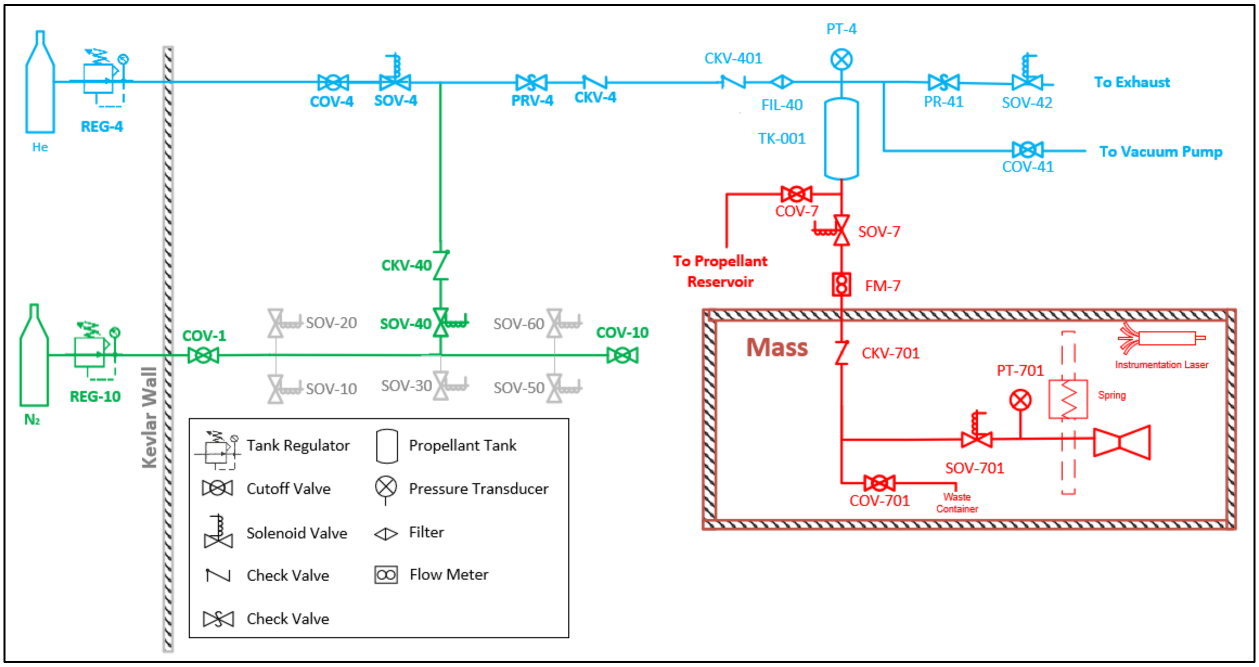

Figure 36 Component and Flow Schematic for Test Setup

Figure 37 Test Article and Instrumentation Inside MASS

2.5 Torsional Trust Stand

Once the propellant reaches the catalyst chamber the decomposition process begins. Thrust is measured using The Torsional Thrust Balance (TTB). The TTTB was constructed at the University of Texas at El Paso. When firing, thrust is produced, and the moment arm moves according to the thrust magnitude. The laser measures the displacement. The TTB is a stand composed of two low friction pivots coupled to a housing attachment for the thruster load transfer beam. This beam is the moment arm and weight counterbalance. As the thruster fires, the moment generated at the center of rotation is recorded by the laser. The TTB capabilities and limitations are illustrated below:

Table 4 Torsional Thrust Balance Parameters

Thrust	25 mN to 4.5 N
Frequency	Steady State to 8 Hz
Thruster Weight	50 gr to .8 kg
Resolution	5% or less of full scale

The laser data is recorded during the firing of the thruster, and it is continuously recorded after. The 1-N thruster has a total weight of 145.8 gr. The is recorded in real time during testing. To minimized error during the test, is important to consider; friction, natural vibration frequency of the system, position inaccuracies, and typical tare on components. They can be different type of tare forces acting on the system during testing such as; deflection, interaction, and gravity. The technical approach to outface error, is to calibration to predict tare loads on the thrust stand.

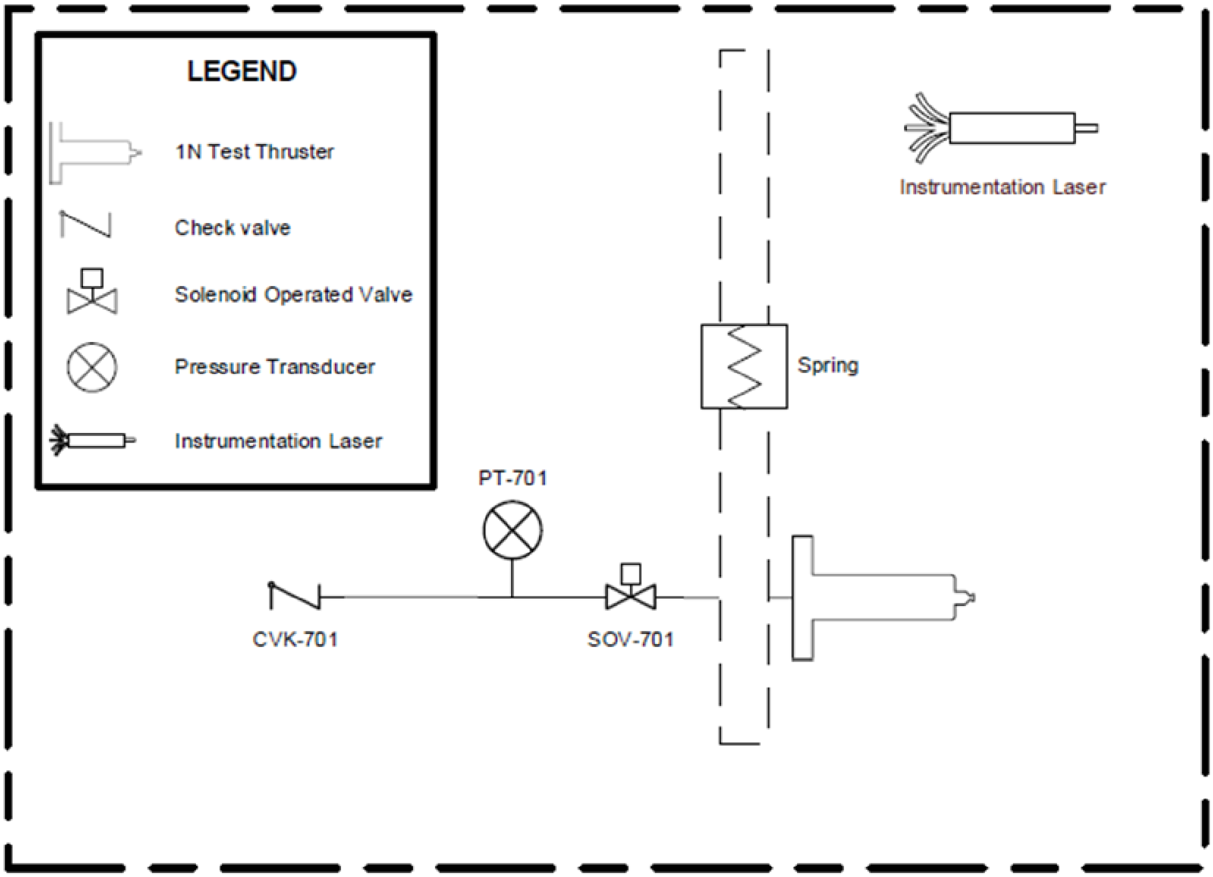

Figure 38 Component Schematic inside the MASS

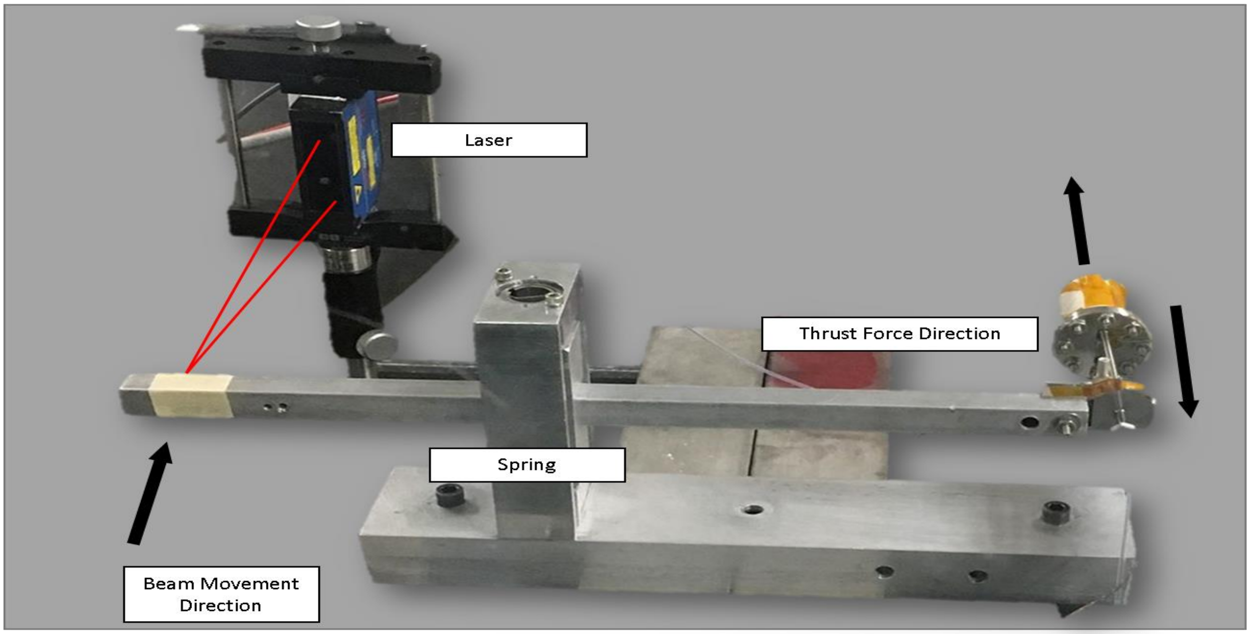

Figure 39 Torsional Thrust Stand Balance with Components

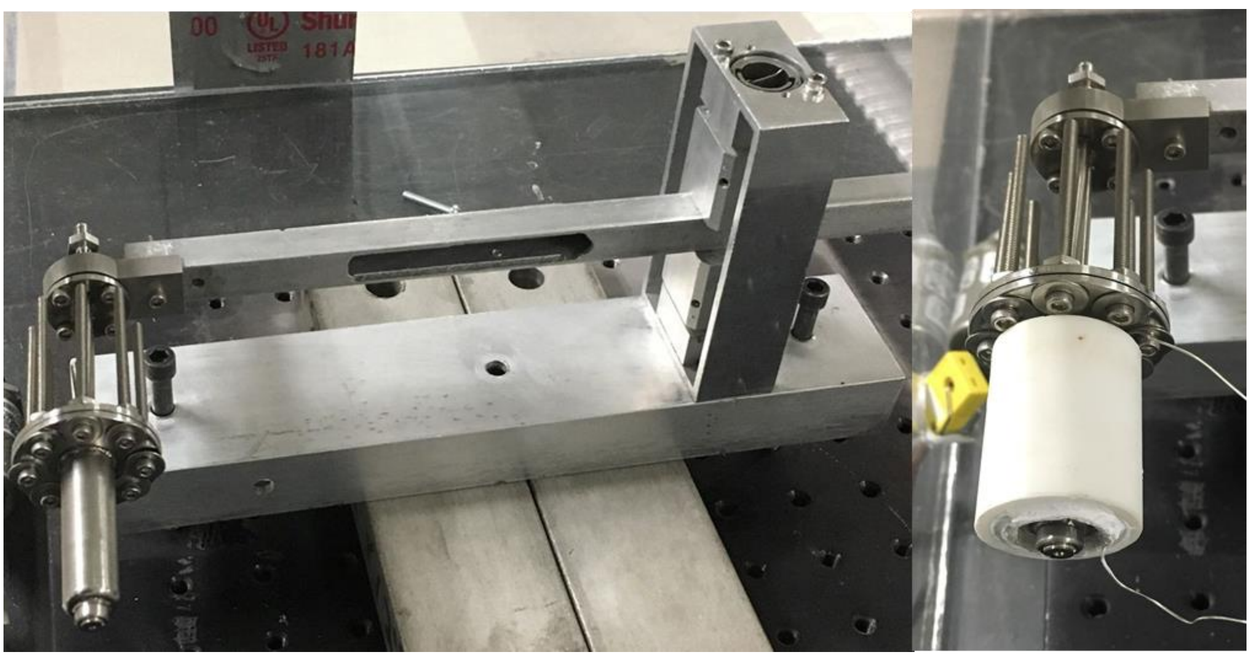

Figure 40 Torsional Thrust Stand Balance Illustrating Thruster and Heater

First, a nylon wire is attached to the moment arm, pass through a pulley, and attached at the end to a dead weight. The dead weight is then attached to the end of the TTB. It provides a force measurement before the system is fired. The system is calibrated before every test series. This is usually done once the system is primed, and pressurized. The calibration set up, and one of the calibration charts are illustrated below:

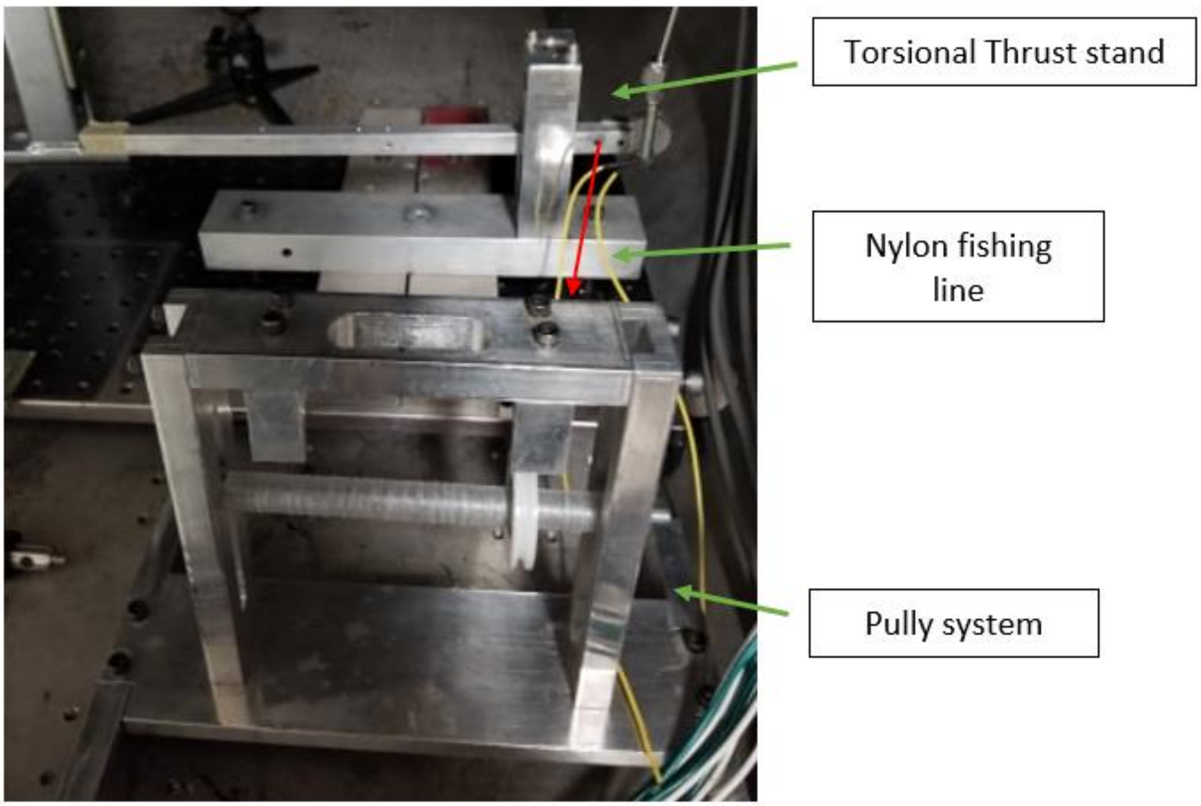

*Figure 41 Calibration Setup (*Red arrow represents the nylon fishing line)*

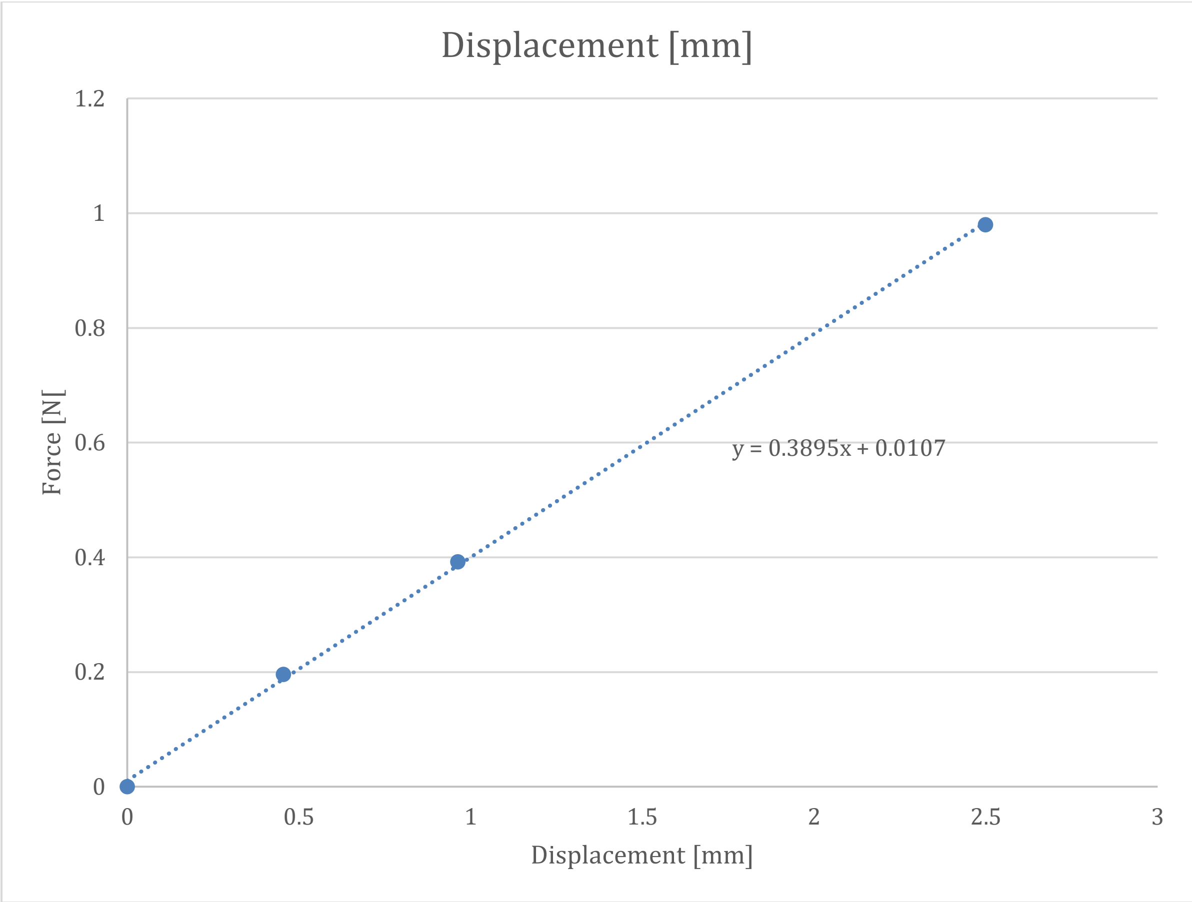

Figure 42 Torsional Thrust Balance Calibration Curve (Test ranges up to 1N)

The laser measurement data is then converted into a force measurement using the linear relationship found in the calibration method by using linear regression on the test data and illustrate the results.

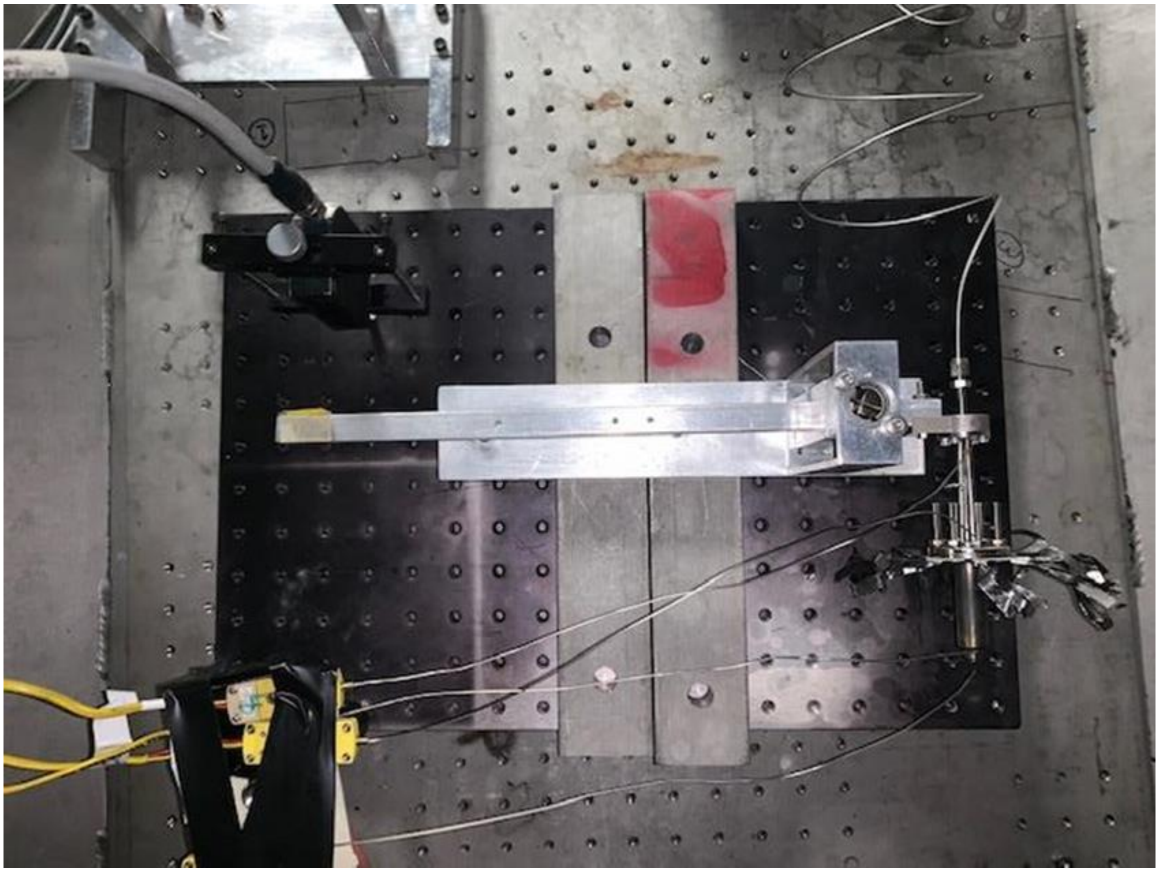

Figure 43 Test Article Integrated in Test Setup

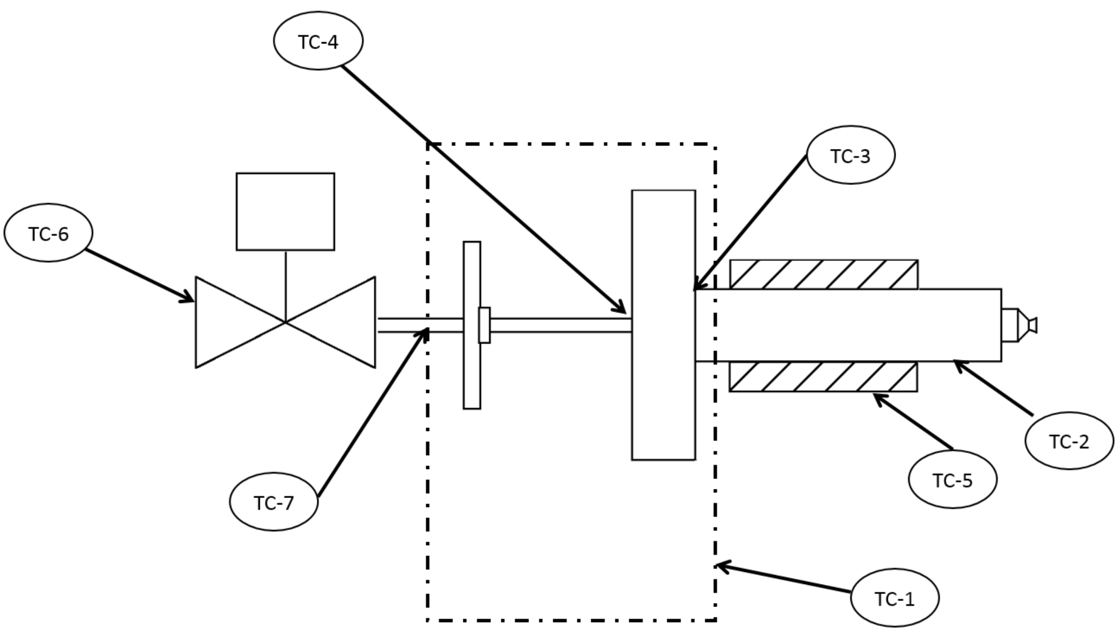

Figure 44 Thermocouple Placement

2.6 Thrust Stand using Load Cell

Once the propellant reaches the catalyst chamber the decomposition process begins. Thrust is measured using NASA Thrust stand. The thrust stand was design by NASA engineers and shared to students at UTEP. Student had the parts conventional machined, and assembled themselves. When firing, thrust is produced, and the system moves according to the thrust magnitude. The load cell measures the force created by the thruster.

Figure 45 Load Cell

The load cell data is recorded during the firing of the thruster, and it is continuously recorded after. The 1-N thruster has a total weight of 145.8 gr. The thrust is recorded in real time during testing. To minimized error during the test, is important to consider; friction, natural

vibration frequency of the system, position inaccuracies, and typical tare on components. They can be different type of tare forces acting on the system during testing such as; deflection, interaction, and gravity. The technical approach to outface error, is to calibration to predict tare loads on the thrust stand.

First, a nylon wire is attached to the thruster valve, pass through a pulley, and attached at the end to a dead weight. It provides a force measurement before the system is fired. The system is calibrated before every test series. This is usually done once the system is primed, and pressurized. The NASA thrust stand is illustrated below. In the figure it can be seen the thruster valve attached to one side of the thrust stand, and the pulley going in the same axis.

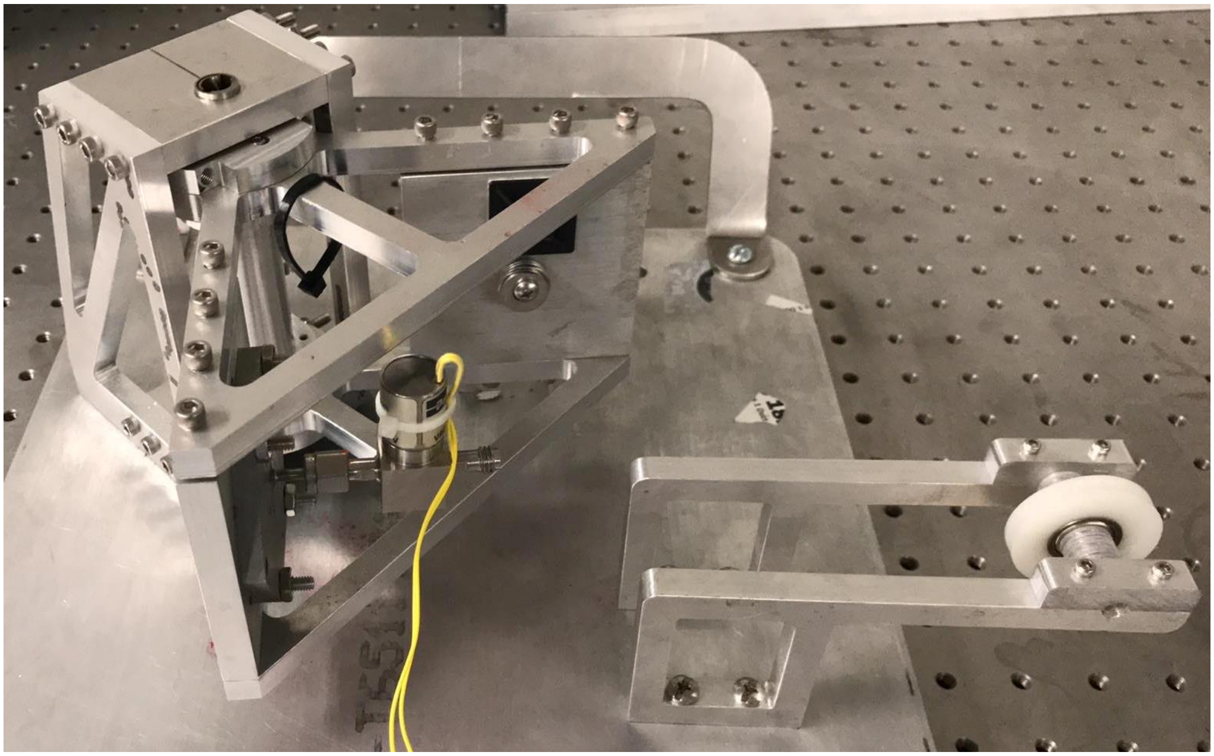

Figure 46 Thrust Stand (design provided by NASA)

The load cell measurement data is then converted into a force measurement using the linear relationship found in the calibration method by using linear regression on the test data.

2.7 NASA Heater

NASA provided a flight approved heater to test the 1N thruster. NASA heater was inserted on to 1N thruster. The heater power output for this heater is 12 Watts. After testing the capability of the thruster in ambient conditions it was notice that the catalyst did not reach 400 ºC in ambient conditions due to radiation. In order to reach the desire catalyst initial temperature some modifications had to be applied. First a layer of aluminum foil between the inner heater body, and outer thruster body was placed in order to reduce convection. The Aluminum made sure all metals were fully in contact, and only conduction was taking place. A second modification was the addition on heating wire covering the NASA heater and most of the thruster body. After the two modification a second test was performed. After testing the capability of the thruster in ambient conditions with the two modifications it was notice that the catalyst did not reach 400 ºC again. A Third modification was applied which was adding layers on insulation tape. After this modification the catalyst finally reached the desired temperature. Since the NASA heater was design for space, Once the thruster is tested in vacuum condition the need for modifications is not necessary.

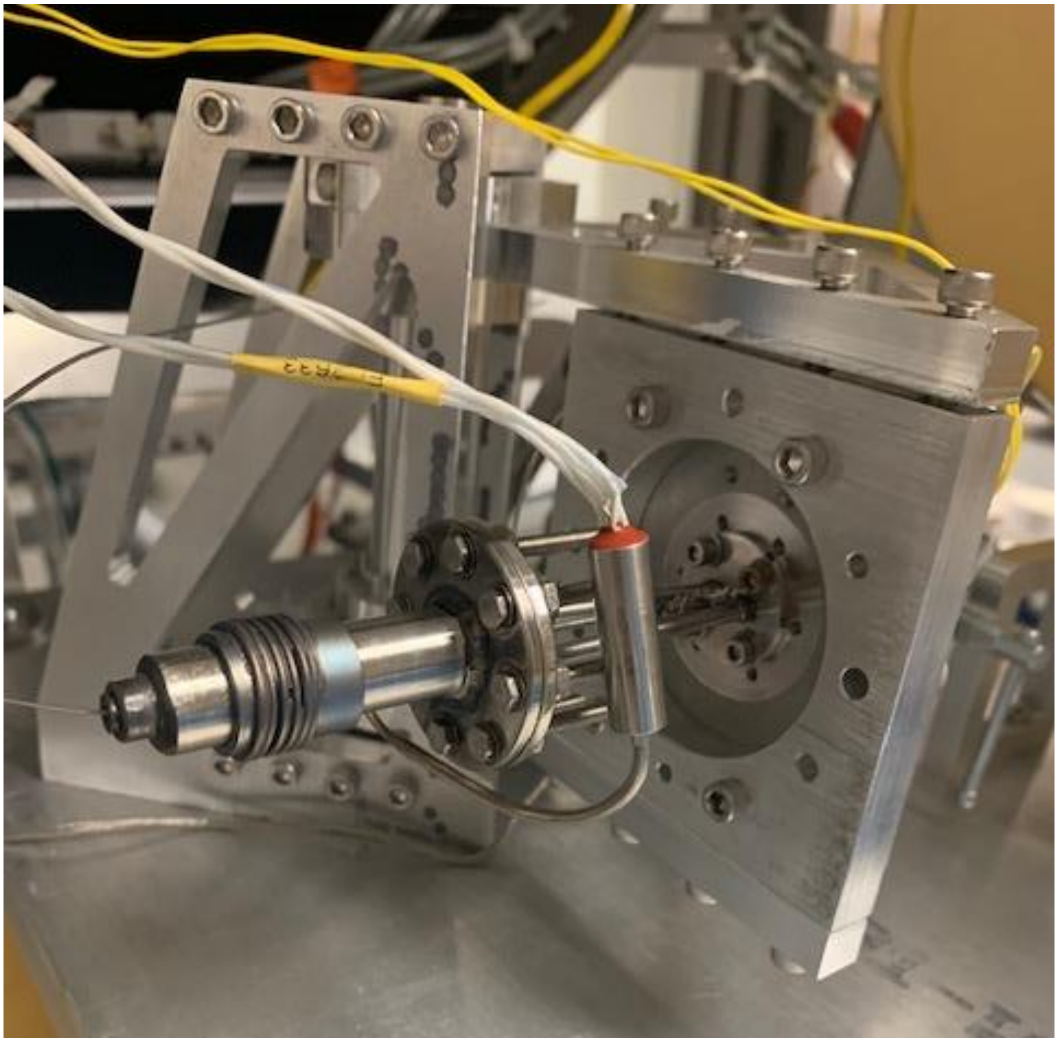

Figure 47 NASA Heater Inserted on to 1N Thruster

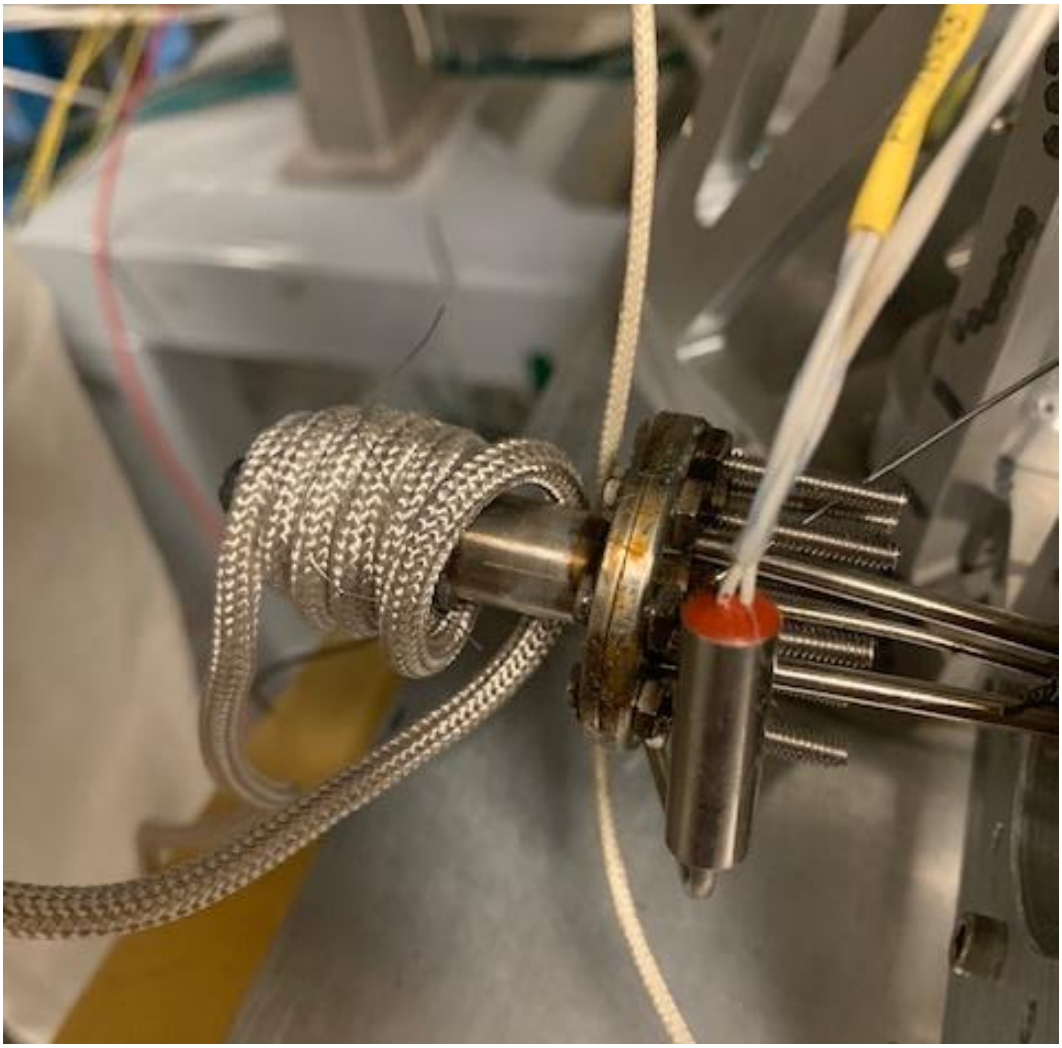

Figure 48 NASA Heater and 1N Thruster Wrapped with Heating Wire

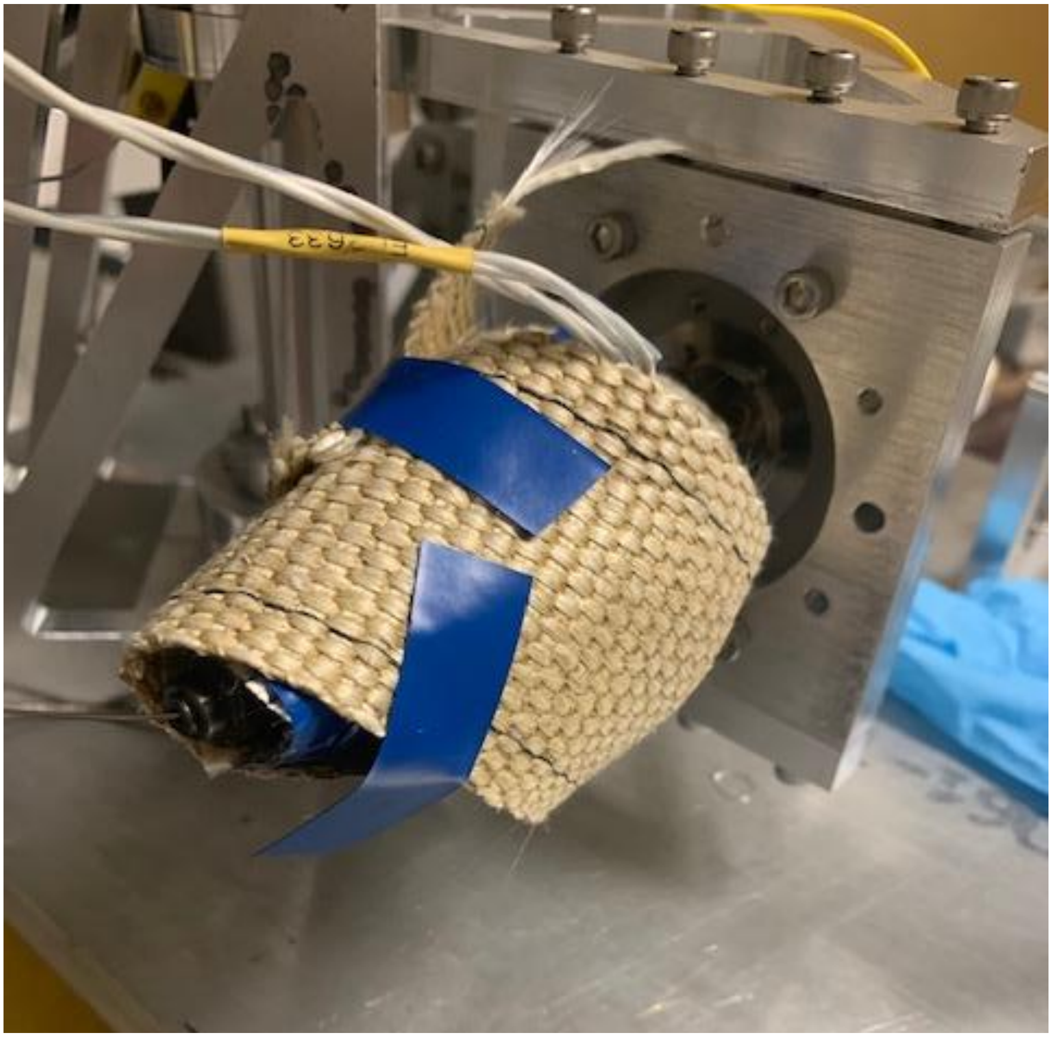

Figure 49 NASA Heater, 1N Thruster, and Heating Wire Wrapped with Insulation

2.8 Test matrix for 1N Thruster Testing with Aluminum Oxide/Iridium Catalyst

The test matrix for this test series is illustrated below. The tests run for 30 seconds, with a catalyst initial temperature of 400 ^{0}C ±10. Calibration of torsional thrust stand is performed before and after completing the test matrix. The catalyst heater was set to 120 Watts, with a wait period below 20 minutes. The catalyst used is the same from all tests. The tests were performed using the torsional thrust stand. The device used to measure thrust was a laser.

Table 5 Test Matrix for AlO_3/Ir

Test #	**Tank pressure (PSIA)**	**Test #**	**Tank pressure (PSIA)**
1.1, 1.2, 1.3	50	6.1, 6.2, 6.3	100
2.1, 2.2, 2.3	60	7.1, 7.2, 7.3	110
3.1, 3.2, 3.3	70	8.1, 8.2, 8.3	120
4.1, 4.2, 4.3	80	9.1, 9.2	130
5.1, 5.2, 5.3	90		

2.9 Test Matrix for 1N Thruster testing with Silicon carbide/Iridium Catalyst

The tests were performed using the NASA thrust stand. The device used to measure thrust was a load cell.

Table 6 Test Matrix for SiC/Ir

Test #	**Tank pressure (PSIA)**
1.1, 1.2, 1.3	50
2.1, 2.2, 2.3	60
3.1, 3.2, 3.3	70

Chapter 3 Experimental Results

3.1 Preliminary Testing

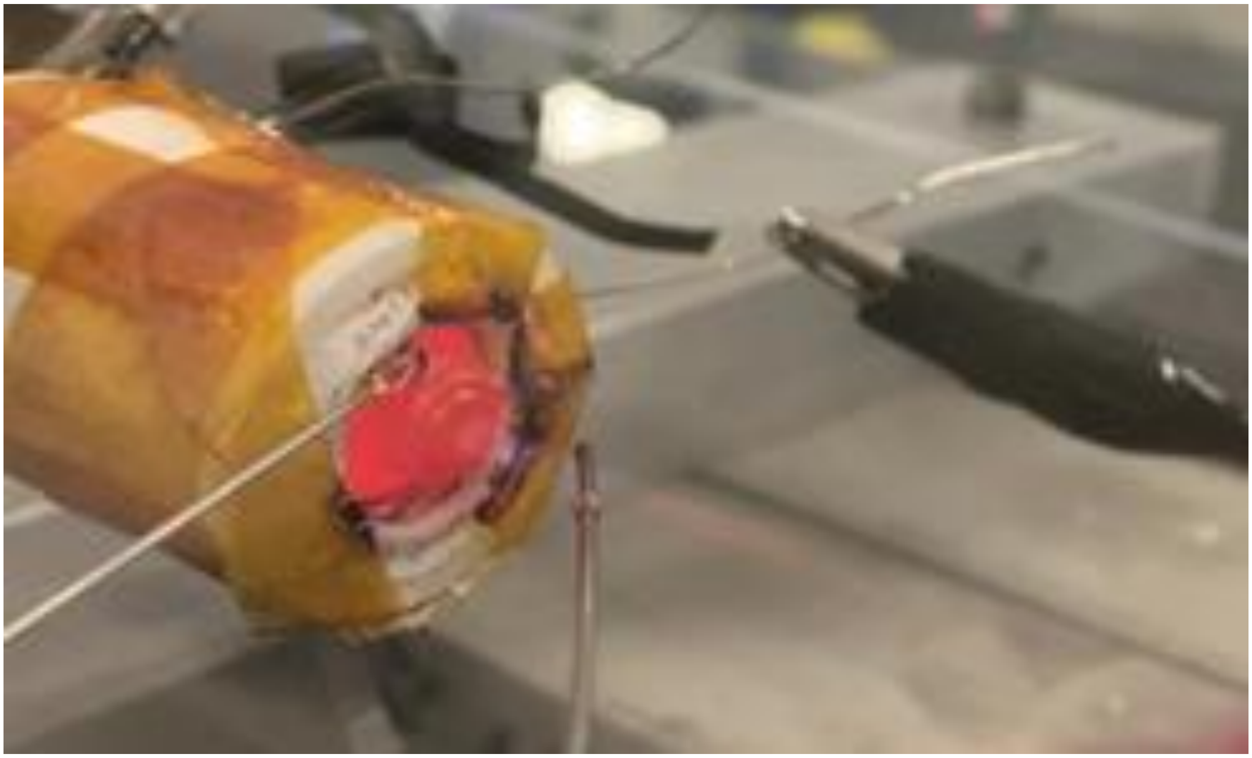

Figure 50 1N Thruster at Preliminary Testing

Preliminary testing was performed using a syringe pump to simulate a delivery system. This test was conducted a low flow rates in order to introduce the reaction of the propellant to test conductors, and other key factors before the actual test matrix.

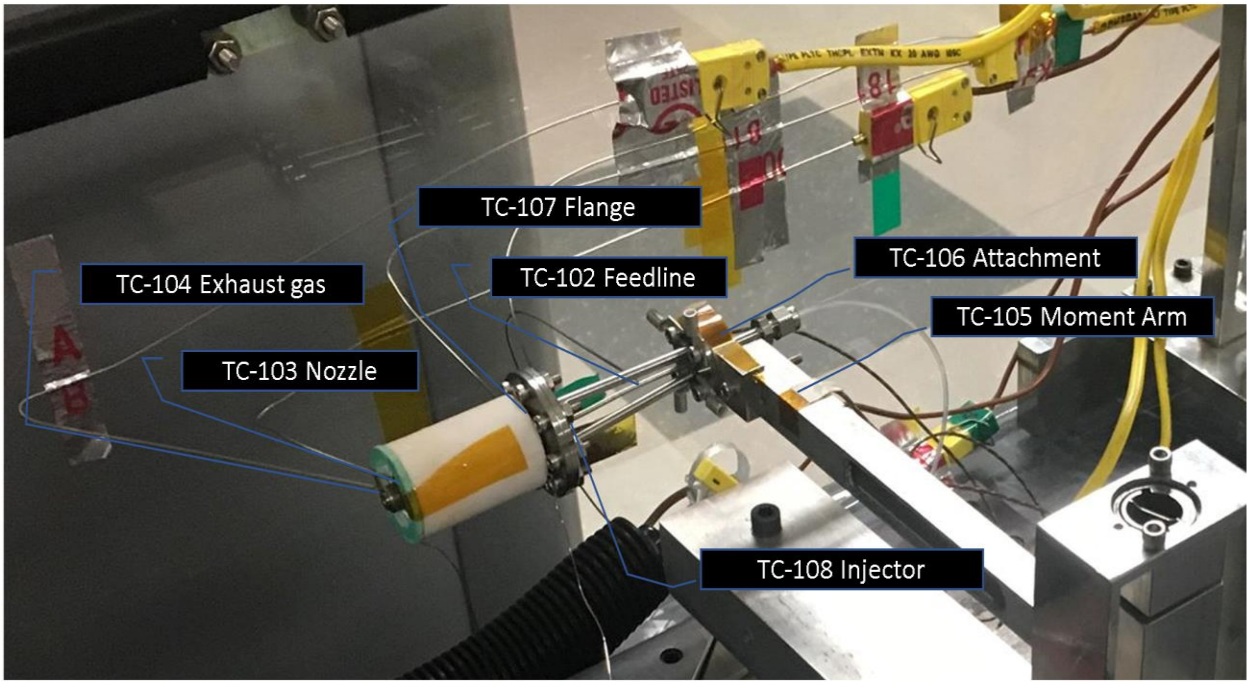

Figure 51 Thermocouple Placement for Preliminary Test

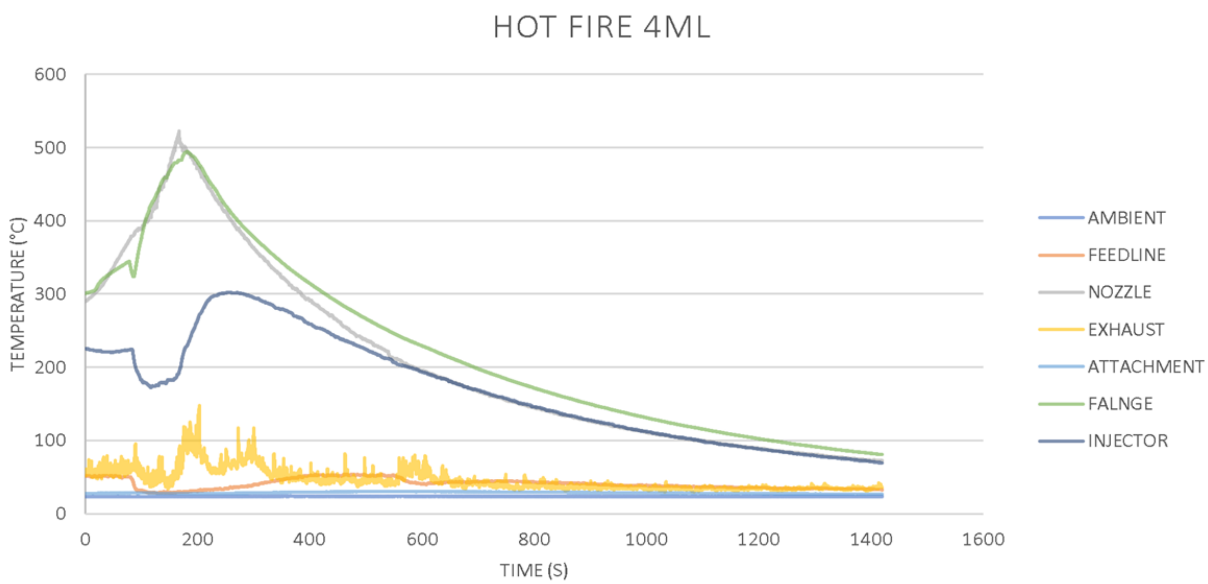

Figure 52 Temperature Results for Preliminary Testing

3.2 1N Thruster testing with Aluminum Oxide/Iridium Catalyst

Test data includes thrust, pressure drop, and temperatures for nozzle. The test below is the test series 9.1 and 9.2, at 130 PSI tank pressure.

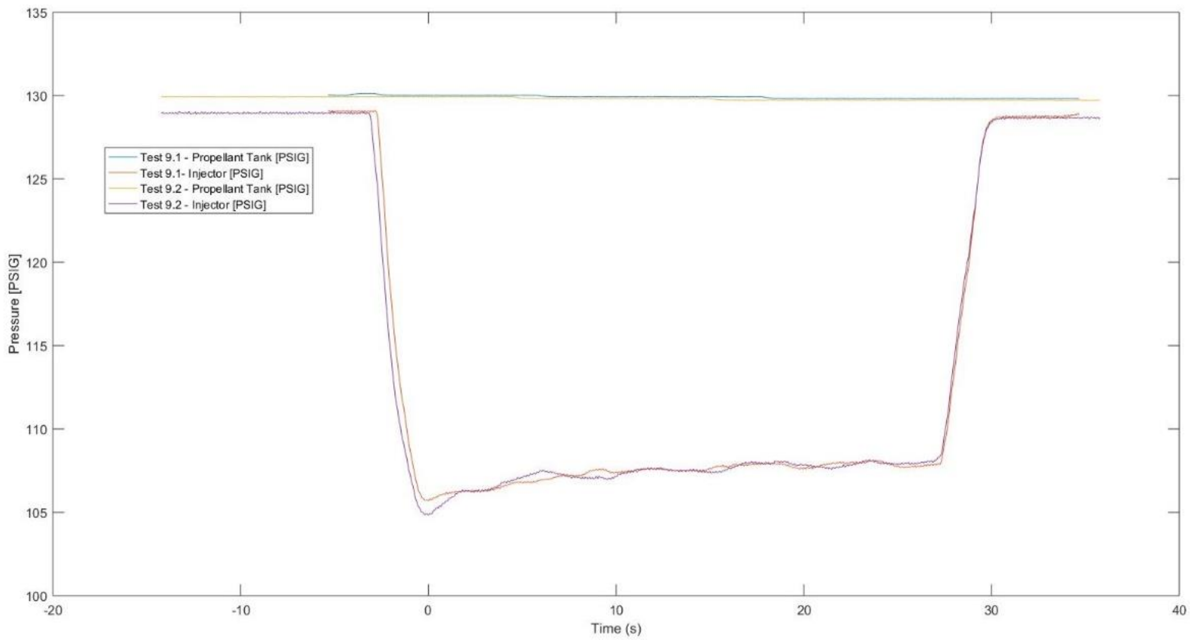

Figure 7 *Test 9.2 and Test 9.2 Pressure at the Tank and Injector*

The pressure behavior from the injector and tank are analogous between different set pressures. The injector begins at a lower pressure from the tank pressure, due to pressure drop. As the decomposition occurs, the pressure of the thruster drops between 105 and 110 PSIG. The pressure drop is associated with the decomposition process and the system becoming an open system to ambient. Assuming a 10% pressure drop between the injector pressure transducer and the injector plate face, an estimated 90 PSIG pressure at the decomposition chamber is estimated. This is higher than the appropriate 87 PSIG desired at the combustion chamber. However, the pressure profile did not match the expected results. Instead of a 1N thrust, the thrust profile was lower than expected.

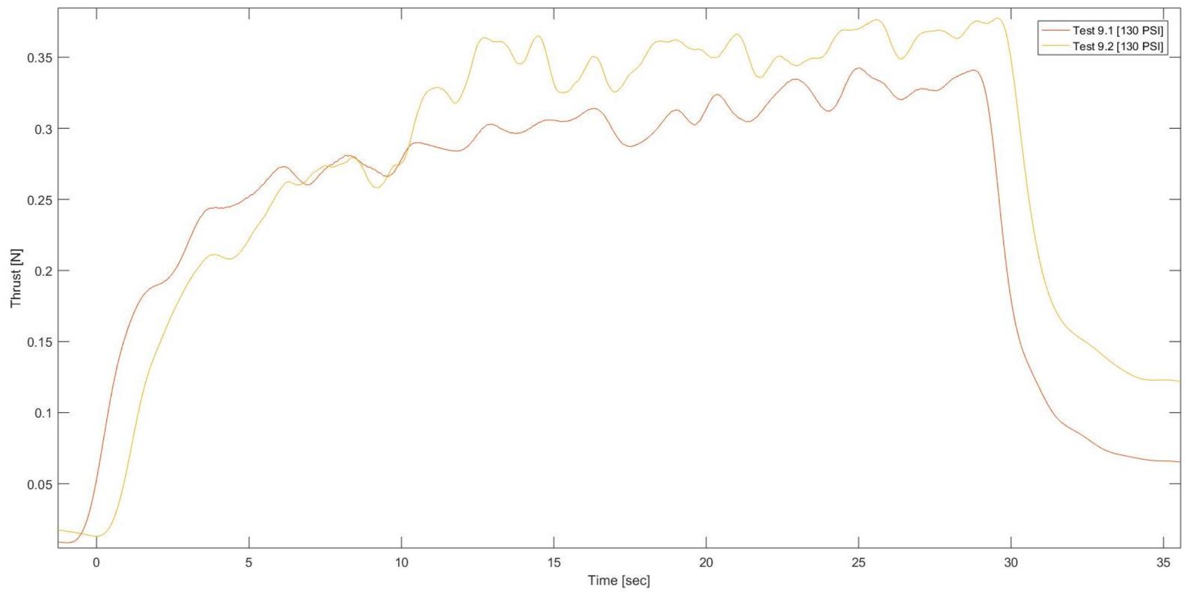

Figure 53 Thrust Profiles for Test 9.1 and Test 9.2

The thrust profile achieves at highest point less than 0.4N. The temperature profiles for the injector, nozzle and exhaust gasses are illustrated below:

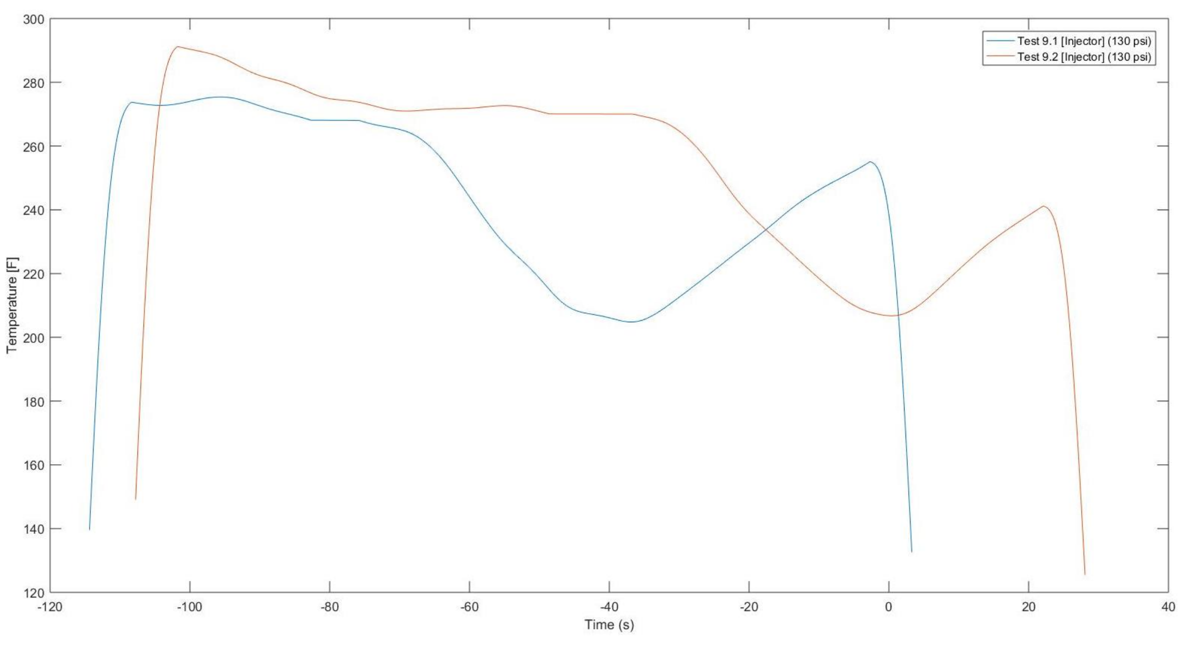

Figure 54 Injector Temperature for Test 9.1 and 9.2

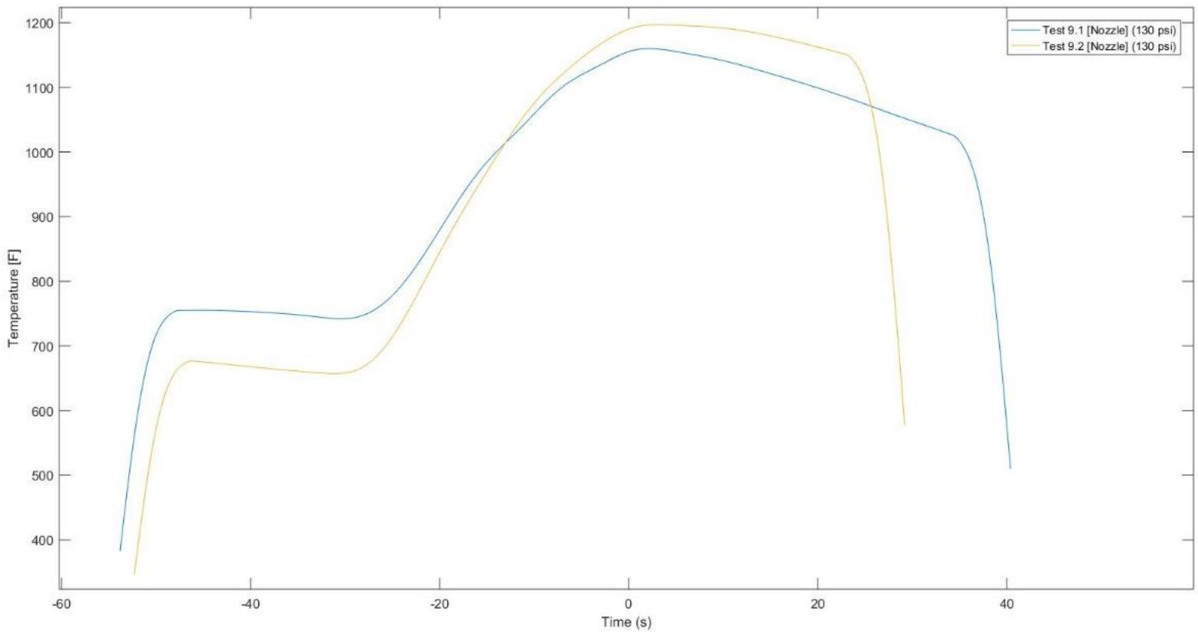

Figure 55 Nozzle Temperature for Test 9.1 and 9.2

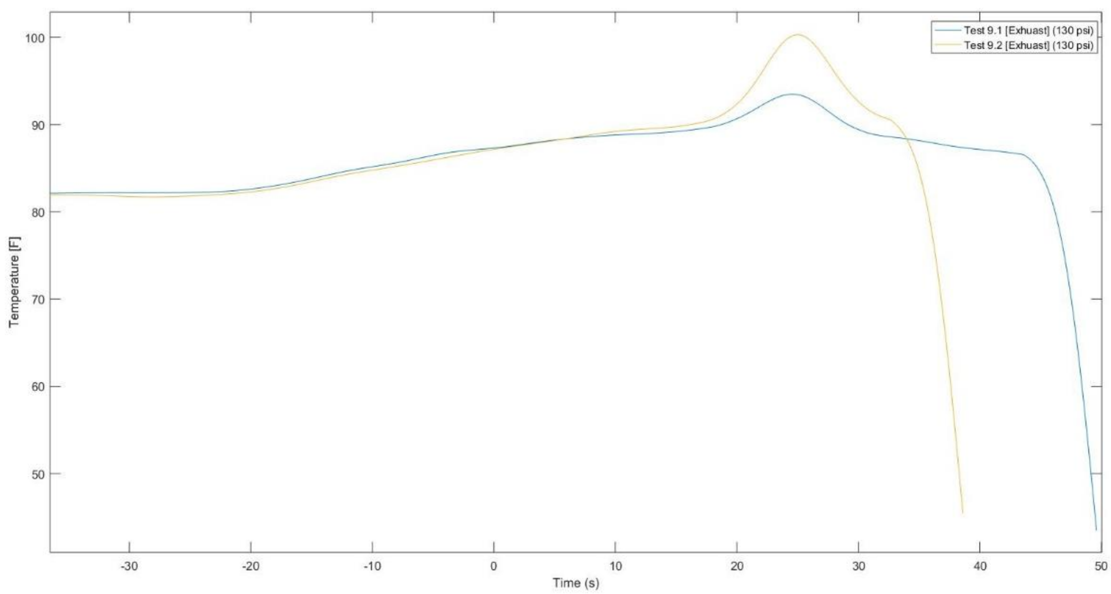

Figure 56 Exhaust Temperature for Test 9.1 and 9.2

Below is the test data for the rest of the test matrix. The initial time of the tests are not at the same point since the tests were controlled manually. The future test series will be done using an automated sequence to have better data analysis results.

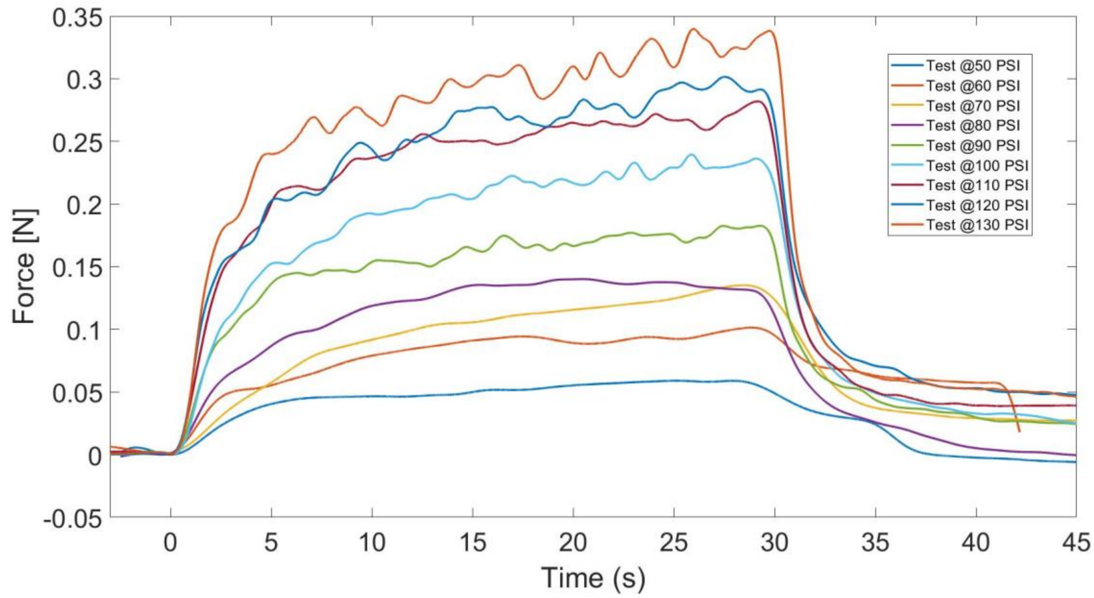

Figure 57 Thrust for Different Tank Pressures (50-130 PSIG)

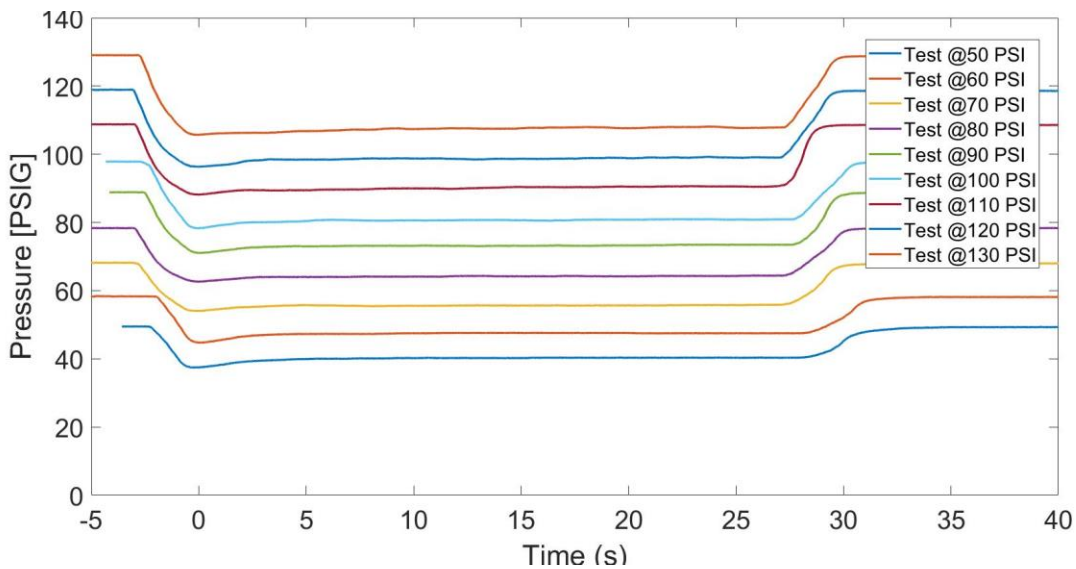

Figure 58 Pressure at the Injector for Different Tank Pressures (50-130 PSIG)

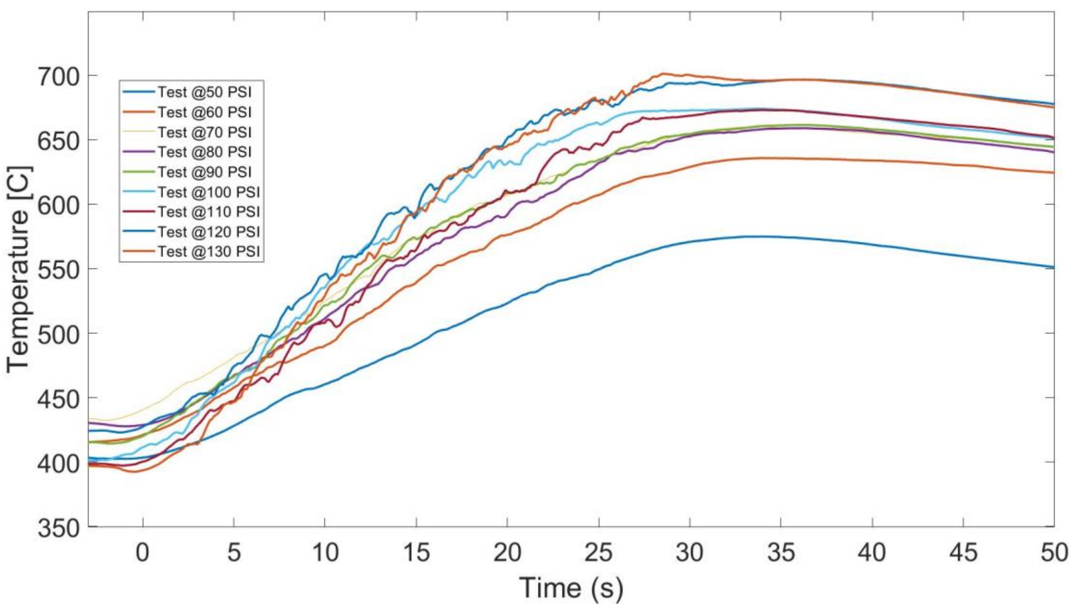

Figure 59 Temperature at the Nozzle for Different Tank Pressures (50-130 PSIG)

3.3 1N Thruster testing with Silicon carbide/Iridium Catalyst

The test data includes thrust, pressure drop, and temperatures for nozzle.

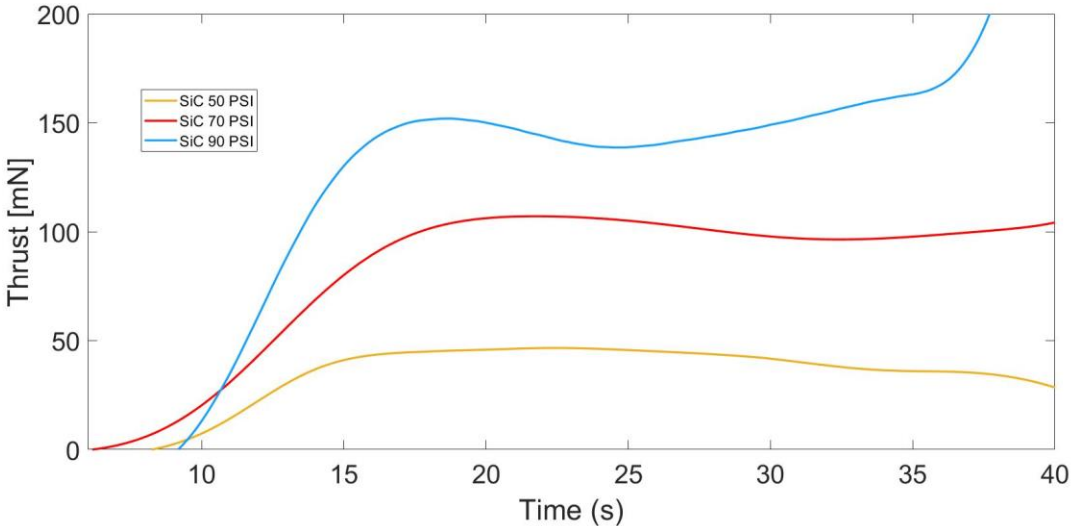

Figure 60 Thrust for Different Tank Pressures (50, 70, and 90 PSIG)

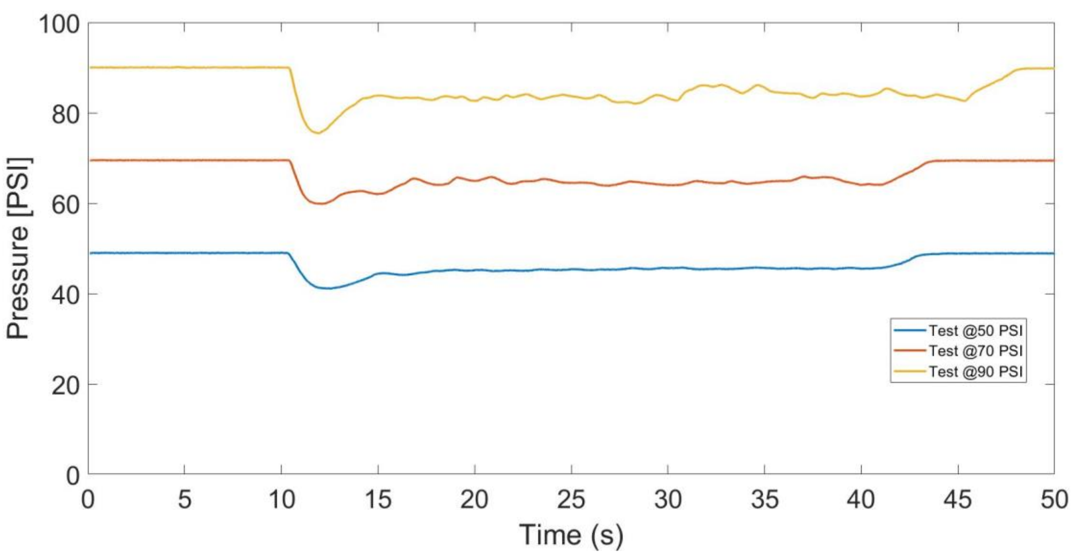

Figure 61 Pressure Drop for Different Tank Pressures (50, 70, and 90 PSIG)

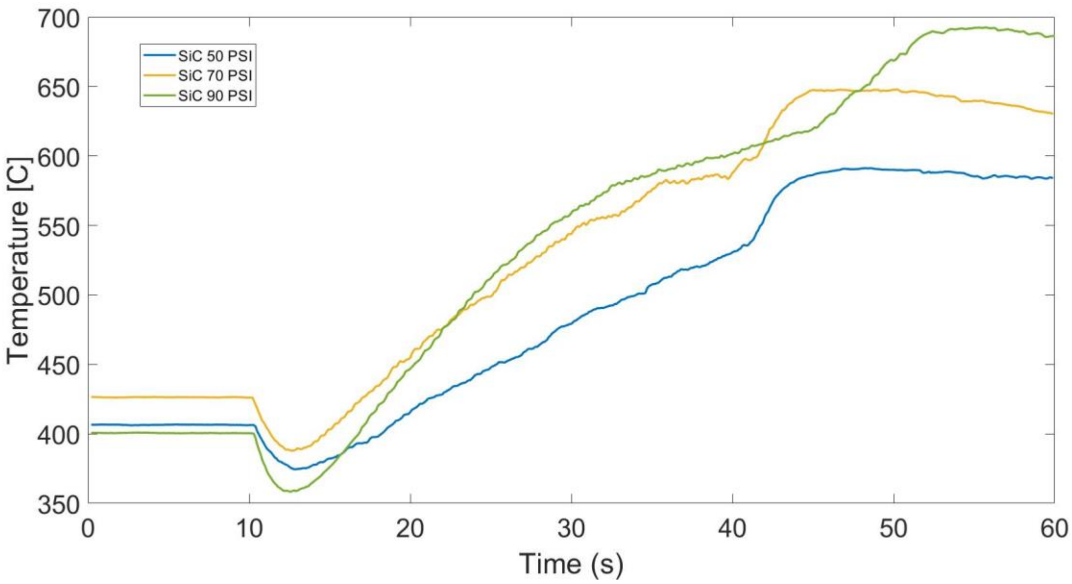

Figure 62 Nozzle Temperature for Different Tank Pressures (50, 70, and 90 PSIG)

Comparison of Results

In this section the comparison of results is shown. The black lines used a laser to take the force measurement, and the catalyst used was aluminum oxide coated with iridium. On the other hand, the red lines used a load cell to take the force measurement, and the catalyst used was silicon carbide coated with iridium.

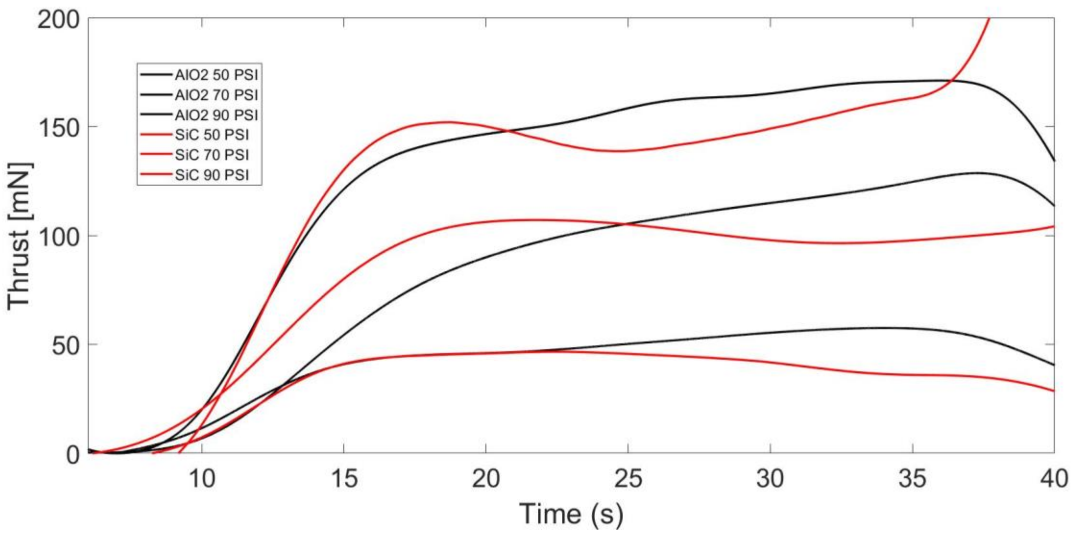

Figure 63 Comparison of Thrust for Different Tank Pressures (50, 70, and 90 PSIG)

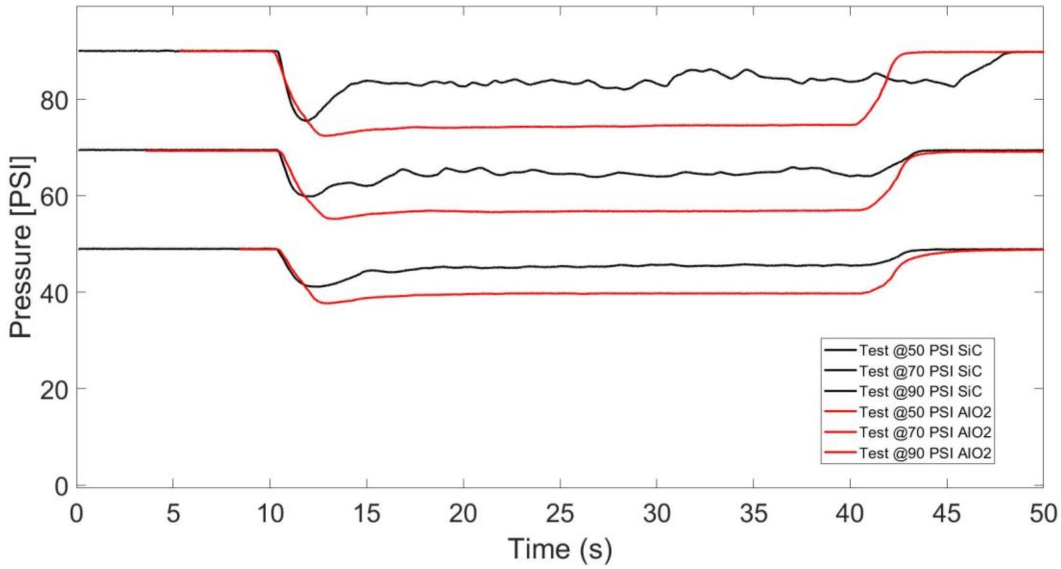

Figure 64 Comparison of Pressure Drop for Different Tank Pressures (50, 70, and 90 PSIG)

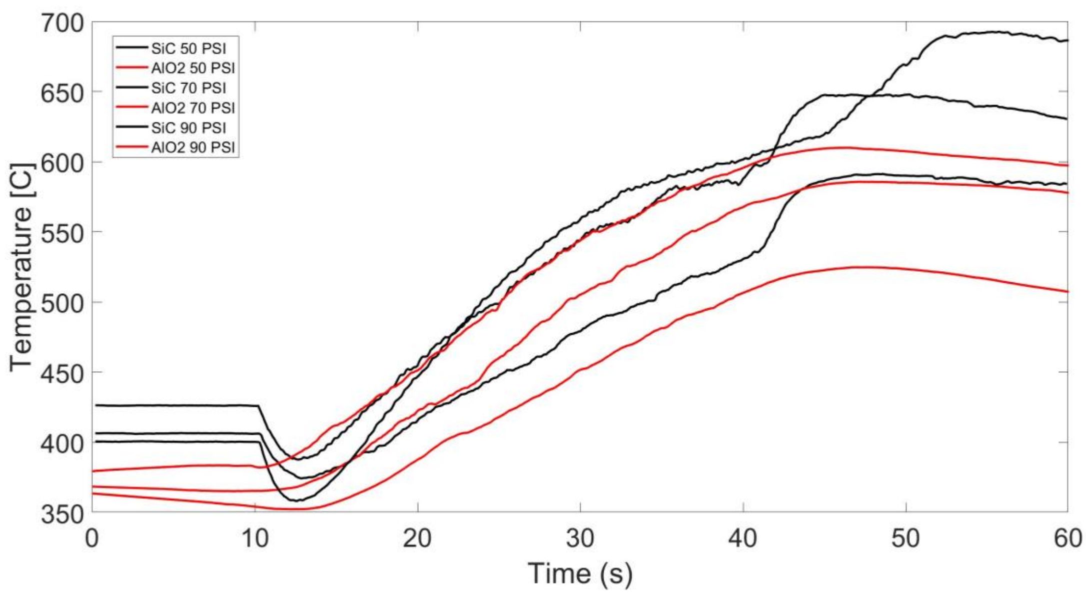

Figure 65 Comparison of Nozzle Temperature for Different Tank Pressures (50, 70, and 90 PSIG

Figure 66 Weld Failed on Thruster after 107 Tests

After 107 test the weld holding the catalyst chamber and the combustion chamber/Nozzle failed. 45 tests were ran using alumina oxide coated with iridium in the torsional thrust stand (laser). 24 tests were ran using silicon carbide coated with iridium in the torsional thrust stand (laser). 38 tests were ran using silicon carbide coated with iridium in the NASA thrust stand (load cell). Half of the catalyst sustained the other half fused to the catalyst chamber thruster body. For this reason, the test matrix using silicon carbide was shortened to only three inlet pressures.

Table 7 Total Number of Tests Using the 1N Thruster

# of hot fire tests	Catalyst used	Thrust measurement device
45	AlO_3/Ir_Batch 1	Laser
24	SiC/Ir_Batch 1	Laser
38	SiC/Ir_Batch 2	Load Cell

Chapter 4 Summary and Conclusion

The center for Space Exploration Technology Research has developed iridium-based catalyst in pellets form. Also, the center developed a 1 Newton thruster using AF-M315E (green propellant developed by the air force). The 1N thruster was design and conventional machined. Hence, the thruster required welds at the joints of each part manufactured. A pressurized delivery system was design and constructed to reach the desired pressures a 1 Newton thruster requires. Test plans, test procedures, and safety assessments were written and approved prior to testing. There were two test campaigns. First a torsional thrust stand with a laser was used to take the force measurement, and the catalyst used was aluminum oxide coated with iridium. Second the NASA thrust stand with a load cell was used to take the force measurement, and the catalyst used was silicon carbide coated with iridium. Both campaigns shown similar results compare to each other.

A second generation was designed and conventional manufactured. It will be tested under vacuum conditions using in-house catalyst. The result of this work will enable promising thrusters for small propulsion modules.

Chapter 5 Future Work

5.1 1N Thruster Testing in Vacuum

The OFX team will travel to NASA Marshall Space Flight Center to test the 1 N thruster in vacuum conditions under the supervision of NASA engineer Daniel Cavender. The result will dictate the performance of the thruster. Also, the result of this effort will dictate if the thruster is ready to be placed on a CubeSat.

Table 8 Proposed Test Matrix

Test #	Feed Pressure		Ton	Toff	Period	CMD Rate	Duty	# Pulses	Acc. On Time	Prop. Flow Rate	Throughput	Acc. Throughput
	(kPa)	(psia)	(s)	(s)	(s)	(Hz)	(%)		(s)	(g/s)	(g)	(g)
1	414	60	0.05	9.95	10	NA	100	1	10	0.45	4.5	4.5
2	414	60	0.05	9.95	10	NA	100	1	20	0.45	4.5	9
3	414	60	0.05	9.95	10	NA	100	1	30	0.45	4.5	13.5
4	586	85	0.05	9.95	10	NA	100	1	40	0.45	4.5	18
5	586	85	0.05	9.95	10	NA	100	1	50	0.45	4.5	22.5
6	586	85	0.05	9.95	10	NA	100	1	60	0.45	4.5	27
7	690	100	0.05	9.95	10	NA	100	1	70	0.45	4.5	31.5
8	690	100	0.05	9.95	10	NA	100	1	80	0.45	4.5	36
9	690	100	0.05	9.95	10	NA	100	1	90	0.45	4.5	40.5
10	793	115	0.05	9.95	10	NA	100	1	100	0.45	4.5	45
11	793	115	0.05	9.95	10	NA	100	1	110	0.45	4.5	49.5
12	793	115	0.05	9.95	10	NA	100	1	120	0.45	4.5	54

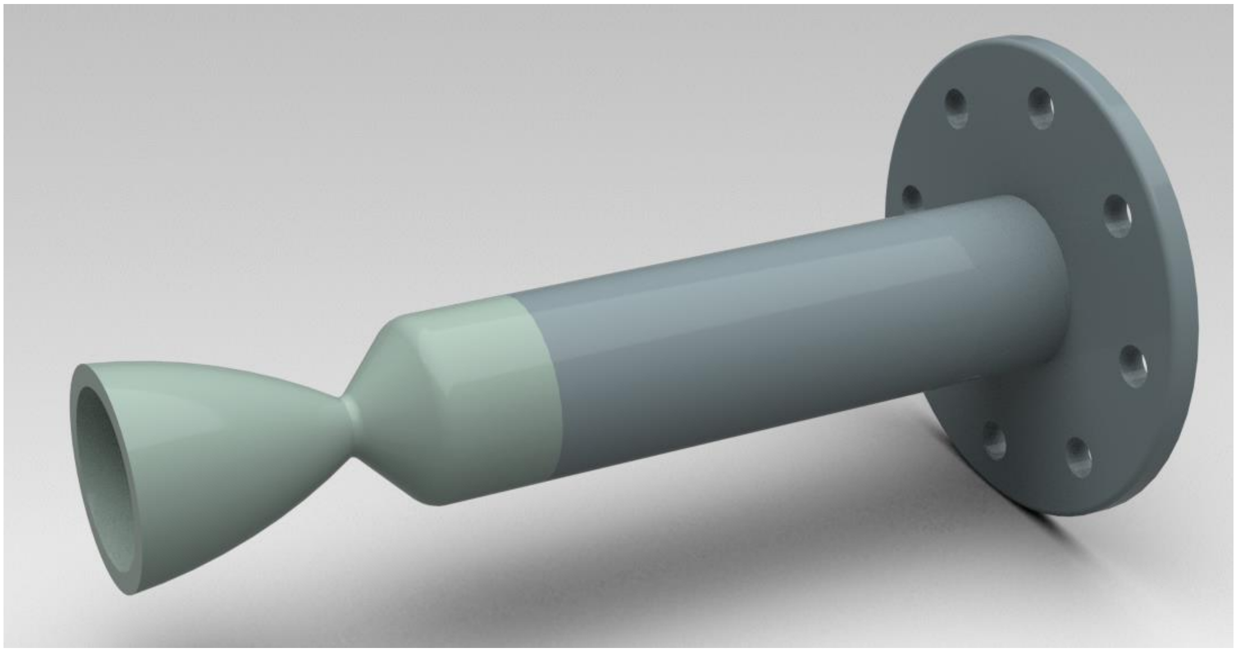

Figure 67 1N Vacuum Thruster CAD Model

Figure 68 1N Vacuum Thruster Conventional Machined

Figure 69 1N Vacuum Thruster Conventional Machined in Three Pieces

Figure 70 1N Thruster for Vacuum Testing

5.2 Lattice Structures

NASA engineer Daniel Cavender has sent seven additively manufacture metal lattice structures. All lattices have different shapes. A pressure drop test will take place using different fluids. Testing fluids are Nitrogen, Air, and a liquid composed of 28% Water and 72% Glycerin (from AFRL match AF-M315E viscosity). The cubes will also be weighted before and after to see loss of material. The placement of the cube will be in different direction to verify the pressure drop of all lattice shape sides.

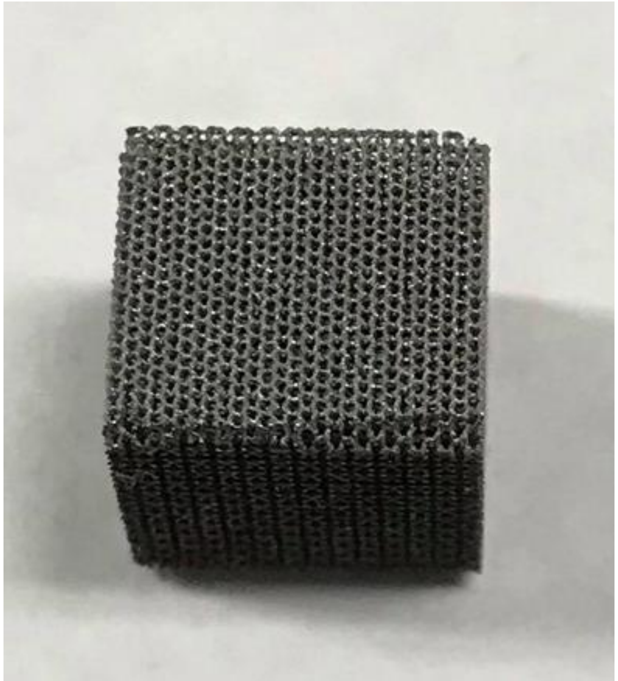

Figure 71 Lattice Cube

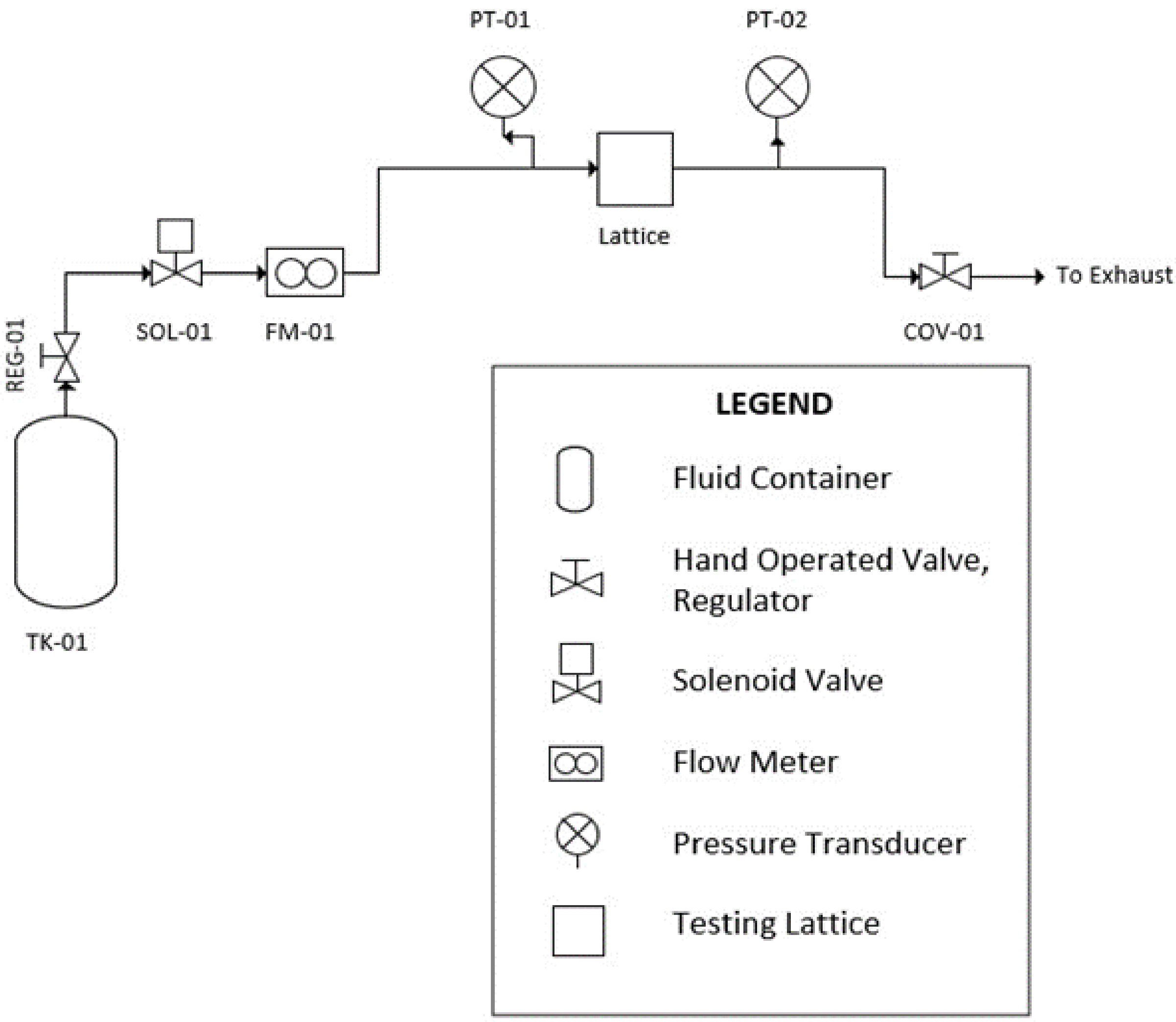

Figure 72 Gas Experimental Setup Component Schematic

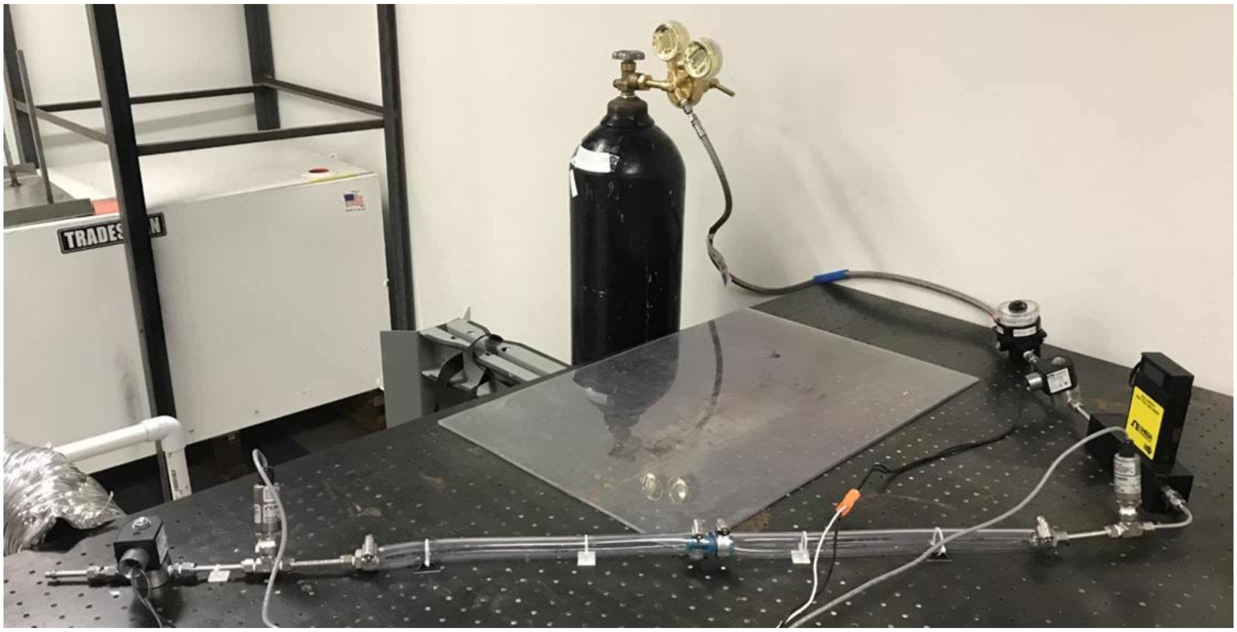

Figure 73 Catalyst Pressure Drop Test Setup Using Gases

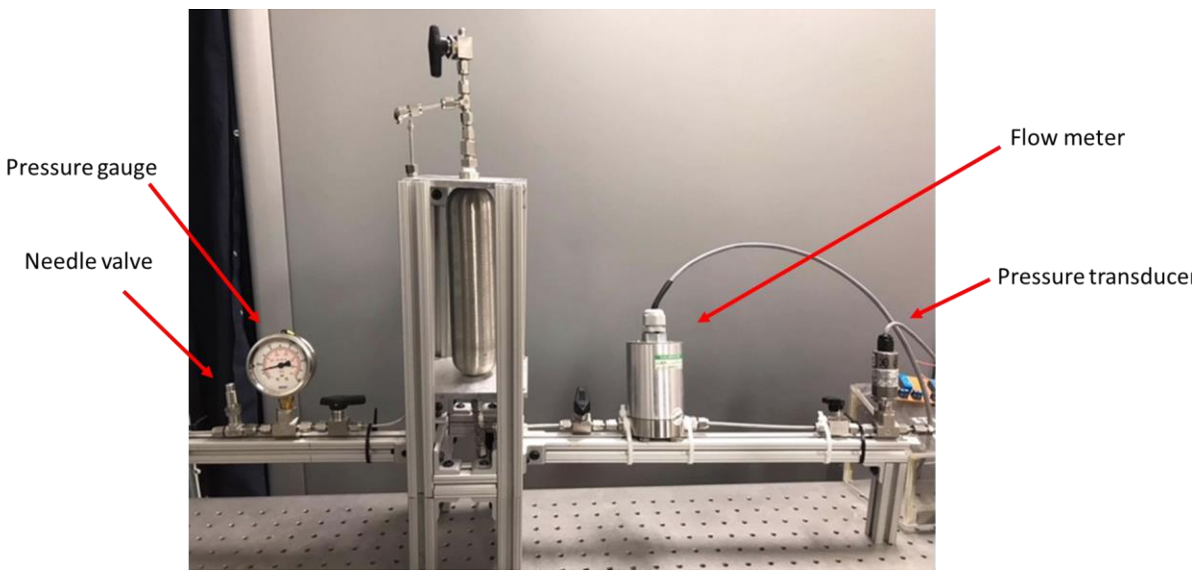

Figure 74 Catalyst Pressure Drop Test Setup Using Liquids

These lattices are an idea of a substrate for catalyst material. The OFX team will impregnate the metal lattices with catalyst material. Then, the lattices will be tested for duty cycles in a holder using AF-M315E. If the results from the duty cycles are successful, the lattices will then be placed inside the thruster to perform hot fire testing, and analyze thruster/catalyst performance.

Test #	Cube Size (mm3)	Lattice Type	Orientatio n	Fluid	Tank Pressure (psi)	Mass Flow (g/s)	Initial Pressure (psi)	Final Pressure (psi)	Pressure Drop (psi)	Mass Change (g)
1	10	Pentacube	XZ	N2	20	0.79	12	6	6	0
2	10	Pentacube	XZ	N2	40	1.46	25	15	10	0
3	10	Pentacube	XZ	N2	60	2.13	40.2	24	16.2	0
4	10	Pentacube	XZ	N2	80	2.81	55.8	35	20.8	0
5	10	Pentacube	XZ	N2	100	3.49	69.8	45	24.8	0

Figure 75 Test Matrix with Results of Preliminary Pressure Drop Testing on One of the Lattices

Tested "Pentacube" lattice in XZ direction with nitrogen from 20 – 100 psi in increments of 20 psi. Allowed nitrogen to flow for 20 seconds per test. Cube weighed 1.20 g before test. Ran each tank pressure test 3 times.

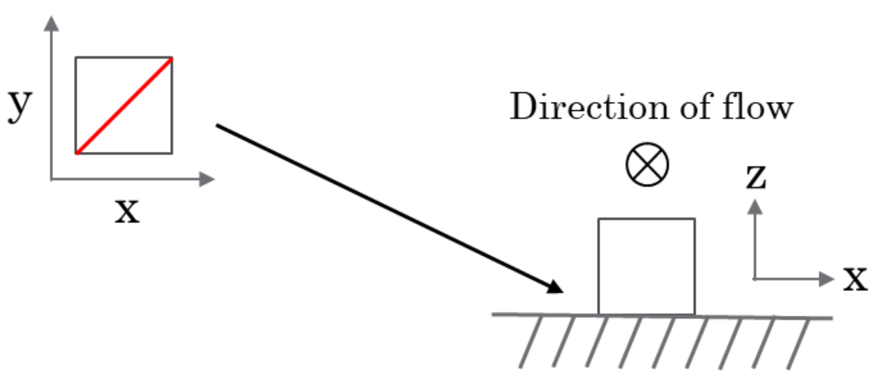

Figure 76 Direction of flow on Lattice Cubes

5.3 Development of a 5 N Thruster

A model was created using thrust design equations from brown book. The model was then compared to Daniel Cavender's (mentor) model to compare parameters and dimensions. A 5N thruster was designed, and additively manufactured. The design has a honeycombed structure inside to hold the catalyst in place. Two ports to take temperature, and pressure measurements during testing. Also, a disk to brace a custom-made heater. The design includes a flange with holes for nuts and bolts. This feature is important to be able to change the catalyst to be tested. The OFX team will incorporate the cSETR in-house catalyst into the thruster to perform hot fire testing, and analyze the performance of both catalyst and thruster. Initial parameters used in the design of this thruster were an Isp of 246 seconds, a chamber pressure of 145 psi, a chamber temperature of 1711 Celsius, an ideal thrust of 5 newtons, and an expansion ratio of 200.

Figure 77 5 N AF-M315E Thrusters

References

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[2] Cortheoux L., Popa, F., Rossignol, S., and Kappenstein, C. ***Improvement of catalysts for the decomposition of HAN-Based monopropellants for space applications - Comparison between aerogels and xerogels***, University of Poitiers, LACCO UMR 6503, 40th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, 2003

[3] ***Green Propellant Infusion Mission Project***. National Aeronautics and Space Administration, FS-2013-07-057-MSFC

[4] Polaha, J., ***Internal Resistive Heating of Catalyst Bed for Monopropellant Catalyst***. Aerojet Rocketdyne Inc., January 2010. Patent.

[5] Fodel, T., Hatorri, T., Hopping, E., Mathews, Jordan. ***Characterization of AF-M315E Monopropellant and 1 Newton Thruster***. NASA MSFC, August 2015

[6] Vazquez, A., Aguilar, D., Mejia, J., Cuevas, R. ***Development of An Iridium Catalyst for The Decomposition of Green Monopropellants.*** University of Texas at El Paso, Joint Army NASA Navy Air Force Conference and Exhibit 2018

[7] Mejia, J., Vazquez, A., Cuevas, R. ***Development of An Iridium Catalyst for The Decomposition of Green Monopropellants and Testing of 1N Thruster.*** University of Texas at El Paso, Joint Army NASA Navy Air Force Conference and Exhibit 2019

[8] Vazquez, A., Cuevas, R., Mejia, J., Valenzuela J. ***Development of an Aluminum Oxide, Zirconia, and Silicon Carbide based Iridium Catalyst for the Decomposition of AF-M315E.*** University of Texas at El Paso, Joint Propulsion Conference and Exhibit, 2019

[9] Valenzuela J, Mejia, J., Vazquez, A., Cuevas, R. ***Development and Testing of a 1N AF-M315E Thruster for Small Satellite Applications.*** University of Texas at El Paso, Joint Propulsion Conference and Exhibit, 2019

Vita

Jaclyn Mona Mejia was born in El Paso TX on April 12, 1992. Her parents are both UTEP graduates, Carlos Mario Mejia, BS 1985 and Raquel Mejia, BBA 1985. Both with master degrees. She was raised as an ex-patriot while her father ran a Multinational Manufacturing company, Jaclyn was raised with community and family values in Chihuahua City, Chih., Mexico during her formative years. She attended Colegio Montessori de Chihuahua for her elementary and middle school education, which focused was on self-motivation, team work and creativity. Then, she attended Tecnologico de Monterrey, Campus Chihuahua, considered a highly demanding prepschool, for part of her high school years while actively participating in the soccer team. She returned home to El Paso, where all her extended family resides and completed high school at Franklin high school as part of the National Honor society, in June 2010. She selected to continue her higher education at the University of Texas at El Paso due to its' excellent engineering program and vast research opportunities. During her sophomore year, Jaclyn began working for Dr. Ahsan Choudhuri at the center for Space Exploration Technology Research. She began with cSETR February 2012 where she worked on several projects under NASA and MDA. In the summer of 2014, Jaclyn had an internship in Houston, NASA's Johnson Space Center. She obtained her Bachelors of Science degree in Mechanical Engineering in Spring 2015, also volunteering in various venues to motivate others to study STEM education. She was accepted into the Master of Science in Mechanical Engineering program where she graduated in spring 2017. Jaclyn decided to continue her education and got accepted into the Doctoral program in Mechanical Engineering at UTEP. Jaclyn has been offered a position as a senior propulsion engineer at Lockheed Martin Corporation under the missile and fire control site at Grand Prairie TX. She will begin her new career right after graduation.