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An Investigation on Pulsing Performance of mN Class Bi-Propellant Thrusters

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AN INVESTIGATION ON PULSING PERFORMANCE OF mN CLASS BI-PROPELLANT THRUSTERS

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Dedication

I dedicate this thesis to my grandfather. May he rest in peace.

AN INVESTIGATION ON PULSING PERFORMANCE OF mN CLASS BI-PROPELLANT THRUSTERS

by

CARLOS F. GOMEZ , B.S. Mechanical Engineering

THESIS

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Chapter 1: Introduction

With the advancement of technology and machining methods, considerable attention has been focused in manufacturing micro thrusters for micro spacecraft applications (Yetter et al, 2003). Micro propulsion systems are currently being investigated for Miniature Kill Vehicles (MKV) and Micro Satellite Systems (MSS) with an emphasis in reducing weight and cost but without affecting performance. Reducing the weight seems practical when reducing the size of the propulsion system, but problems are encountered with system design.

In addition, propulsion requirements strongly influence choice in propellant and complexity. A thorough understanding of stabilizing thrust levels and the ability to control the propulsion system is important in order to meet the requirements of the specific propulsion application. Certain applications might require long thrust durations, while others require small thrust pulses. Choice in propellant can influence hardware selection needed to achieve reliable thrust levels. Various micro propulsion systems have be proposed but only a few have been investigated.

This paper investigates the pulsing characteristics of a mN Bi-propellant thruster for small spacecraft applications. The thruster was completely design to have its own pressure and valve control system in order to control thrust levels and pulse lengths. The propellants tested where gaseous Methane/Oxygen and Hydrogen/Oxygen which allowed the system be simple. Each individual component of the thruster was fabricated in-house with materials available to the public.

1.1 Project Purpose and Objectives

The thruster is an assembly of other parts that have been researched: combustion chamber, thruster injector and nozzle. The integration of all these parts have been tested previously with gaseous Methane/Oxygen and Hydrogen/Oxygen under vacuum conditions. The variables recorded were thrust, mass flow rates, back pressure(in vacuum), and line pressure, which were all under steady state conditions for 5 seconds(Flores, 2009). It was proven that the thruster can be ignited in vacuum and sustain combustion. However, pulsing at short pulse widths and actual chamber pressure readings were not investigated for this specific thruster.

The Micro Thruster presented in this thesis was designed to investigate pulsing at different thrust levels while maintaining a simple control system. The propellants used for testing were Methane/Oxygen and Hydrogen/Oxygen to compare result from the previous work done. The variables recorded were mass flow rates, chamber pressure, and chamber temperature. Thermodynamics was used to predict rocket performance parameters for this micro thruster. Pressure regulators and solenoid valves where used in sync with LabVIEW in order to provide the time sensitive pulsing and ignition for the micro thruster. Theoretical performance values were compared to previous values using the same thruster at steady state thrust. More detail was focused in the fabrication and integration of the ignition source for the thruster. In addition, other options are explored for the significant use of this micro thruster.

Chapter 2: Experimental Setup

The experimental setup was a miniature rocket engine test stand, which was designed for testing various propellant mixtures with a control system. The thruster consist of a four piece integrated system: swirl injector with igniter, rocket nozzle, propellant feed system, and LabVIEW control system. The rocket nozzle, injector and igniter system was completely designed and fabricated in house. The propellant feed system was assembled using off the shelve parts such as tubing, valves and regulators. The thruster and injector was designed so that practical tools can be used to assemble and disassemble the system. All of the hardware was attached to a 12” x 24” optical table beside the DAQ instrumentation. The thruster performance data and valve control system was controlled and monitored using LabVIEW from National Instruments. It was beneficial to keep the experimental setup as simple as possible since the main goal was to reduce weight and size. More details are described in the next sections.

2.1 INJECTOR AND IGNITER

The design of the injector had gone through several iterations in order to find the best configuration for a co-axial swirl pre-mixer. A fundamental part of the design was to insure easy manufacturability while keeping the injector modular for different propellants. The injector used for these experiments is shown in Figure 1 One of the first designs incorporated an injector orifice plate that was used to further atomize the mixture of liquid propellants. However, in this paper, only gaseous propellants were used for testing: eliminating the use of the orifice plate.

The injector allows fuel to be injected into four elements that are tangent to the oxidizer flow at the core of the center of the body. Figure 2 shows an illustration on how the injector functions. The total mixing length in the injector body was ¼ in which led directly to the combustion chamber of the thruster. The body of the injector was fabricated from 1” x 1” Stainless Steel bar stock. The four tangential fuel inlets ports where machined at 1/32 in. The single oxidizer inlet diameter was machined at 3/16 in. The inlet of the oxidizer and fuel were threaded for a 1/8 in NPT adapters. The propellant feed system was directly connected to these adapters.

Figure 1-Co- axial Swirl Injector with Orifice plate

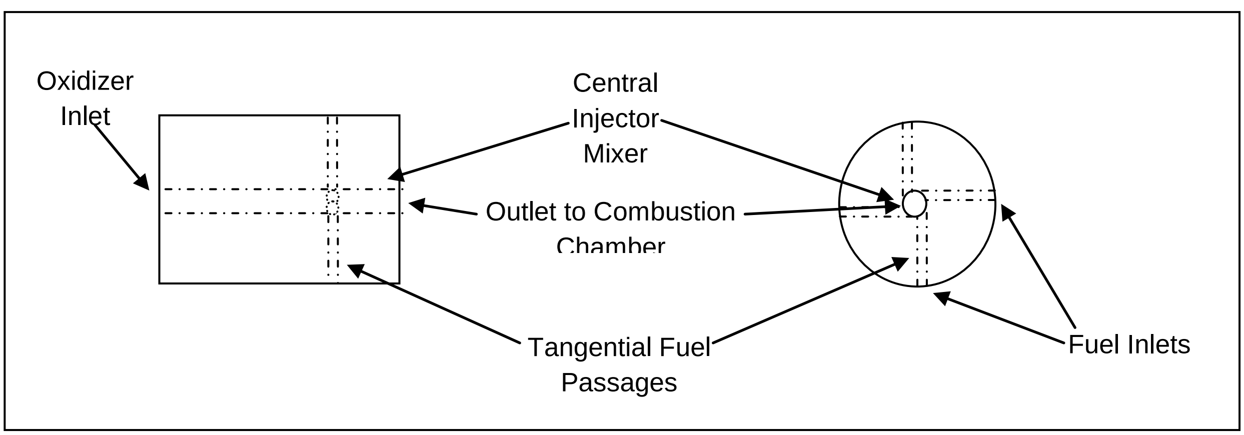


Figure 2- Injector Illustration

Two additional ports were machined on to the injector body: the igniter and pressure transducer ports. The igniter port was specifically designed to be at an angle of 5 degrees toward the exit of the injector. The purpose of this angle was to have the tip of the igniter as close as possible to the combustion chamber for a more stable ignition. This provided improved combustion at the chamber since the chamber had a larger volume than the injector. Figure 3 below, depicts the design of the igniter angle.

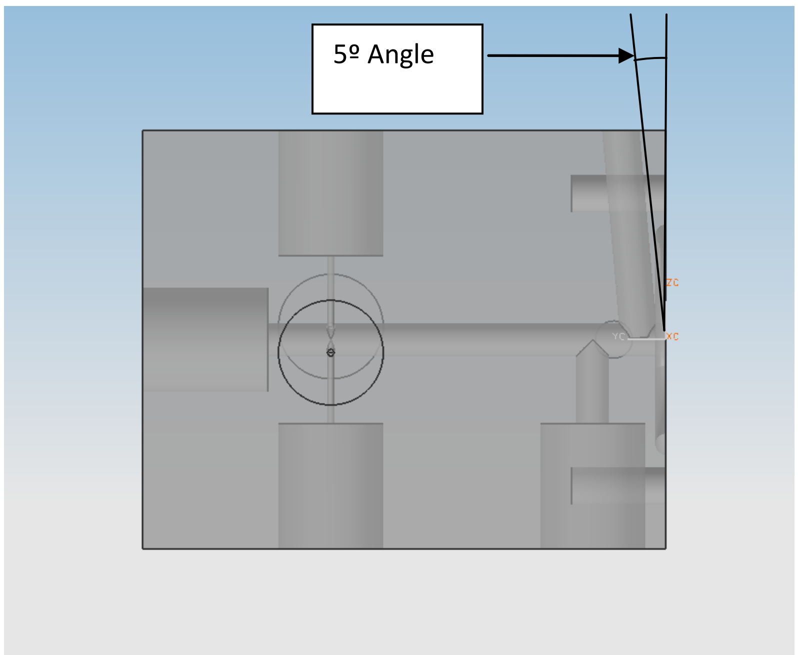


Figure 3- Injector with Igniter Angle

The igniter was made out of a steel tube with a .125 in outer diameter and a .085 in inner diameter. An alumina tube was inserted inside which insulated a single platinum wire. The platinum wire was bonded on to the alumina tube using Resbond 919, a high resistance ceramic compound with a high dielectric strength. The platinum wire protruded .01 in at one end of the igniter. This perturbation was necessary in order to create a high voltage arc within the wall of the injector. A voltage adapter was incased at the other end of the igniter to connect the high voltage transformer. The steel body of the igniter was threaded in order to seal the injector body from any leaks due to spikes in chamber pressure.

2.2 CONVERGING DIVERGING NOZZLE

The thruster nozzle was machined together with the combustion chamber out of Titanium stock Ti-6%Al-4%V. The thruster was fabricated using a CNC mini lathe. The combustion chamber length is 3/8 in with a with .45 in chamber diameter which converges into a .75-mm throat then diverges into an exit diameter of .3 in at a diverging section half angle of 15 degrees. Since all the testing was at altitude pressure, the area ratio of the nozzle was designed to be 3 in order for the exit plane pressure to match the ambient pressure of 14.7 psi. The thruster was attached to the injector body using four 10-34 Allen head bolts. The complete assembly of the thruster system is shown below in Figure 4.

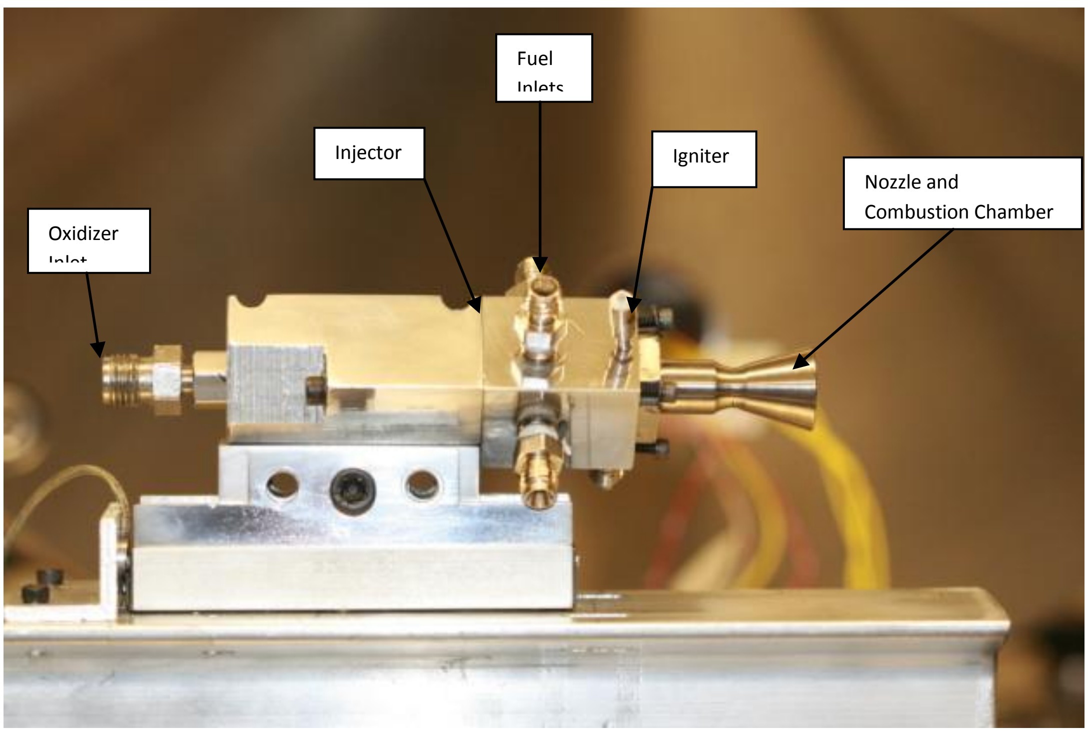


Figure 4- Thruster Assembly

2.3 PROPELLANT FEED SYSTEM

The oxidizer and fuel supply lines were constructed identical to each other to keep the system simple. Two pressurized tanks supplied the oxidizer and fuel to stainless steel pipelines that led to a miniature pressure regulator. The pipelines were then reduce from ¼ in to 1/8 in and then connected to a pressure transducer and finally to ASCO solenoid valves. The connections between the valves and the injector body were of polyurethane tubing to allow the thrust stand to be flexible when connections had to be replaced or moved. As mentioned before, the propellant feed system was kept simple but it was also designed to the have the minimum amount of pipeline between the solenoid valves and the injector in order to reduced dribble volume from the oxidizer and the fuel. The propellant feed system is shown in Figure 5 below

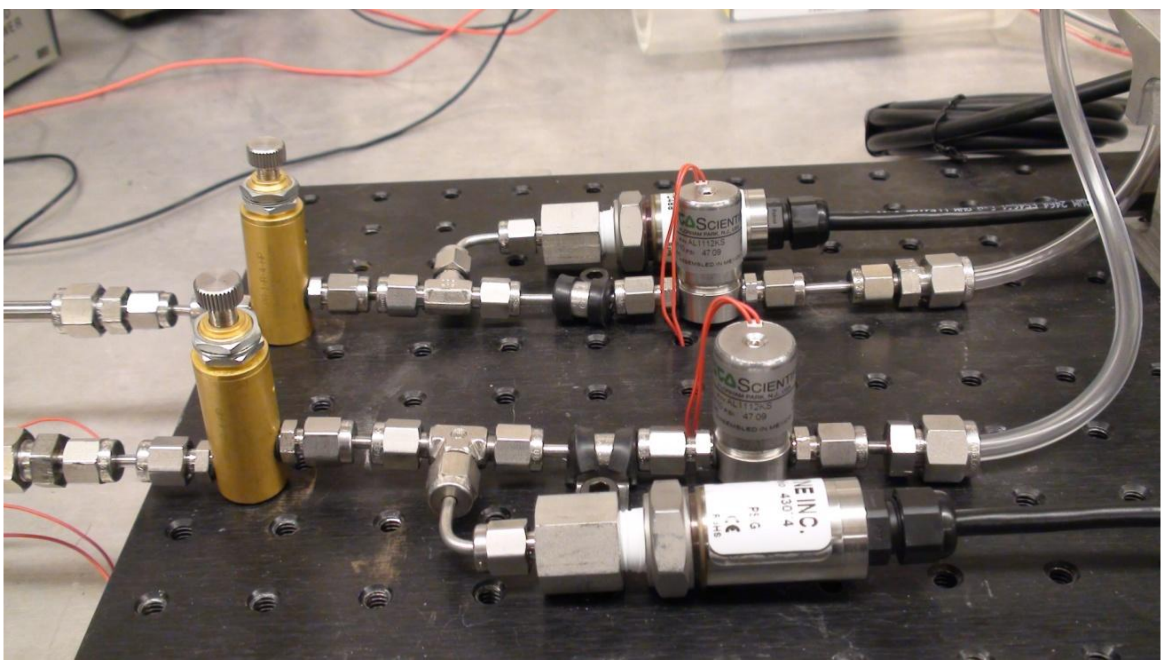


Figure 5- Propellant Feed System

2.4 VALVE AND IGNITER CONTROL SYSTEM

The valve and igniter system was completely controlled and monitored using LabVIEW in conjunction with a Solid State Relay (SSR). Two Asco solenoid valves, one from each line, were connected in a closed loop to a power supply and then to the NI USB 6525 SSR. The igniter connection sequence was identical to the valve connections. The SSR internal switches were always in a normally open state when the LabVIEW program was not running. Power would only be supplied to the valves and igniter when a digital signal from the LabVIEW program was sent using a binary text file. Figure 6 below shows the 9 columns of text needed to send a time dependent digital on/off signal. Four different note pad binary text files were made for different burn times. Burn times were increase by 250 milliseconds on each test.

0	0	0	1	0	0	0	0	0
250	1	1	1	0	0	0	0	0
500	0	0	1	0	0	0	0	0
750	1	1	1	0	0	0	0	0
1000	0	0	1	0	0	0	0	0
1250	1	1	1	0	0	0	0	0
1500	0	0	1	0	0	0	0	0
1750	1	1	1	0	0	0	0	0
2000	0	0	0	0	0	0	0	0

Figure 6- Digital Binary Text

The first column is time in milliseconds while the others to the right are digital signal outputs. Five of the columns to the right were left at zero in order for the program to run. The second and third columns controlled the valve actuation. It can be seen that at time zero the valves are maintained closed and the igniter is on throughout the entire test until the end. Valves were closed for only 250 milliseconds in between pulses for all test. The LabVIEW control block diagram can be seen in Figure 7.

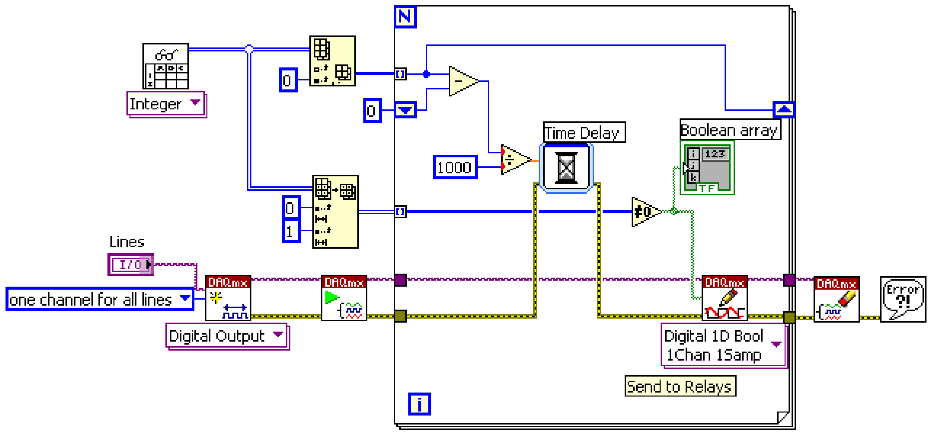


Figure 7-Valve and Igniter Block Diagram

2.5 DATA ACQUISITION

Three pressure transducers were connected to a NI SCC-68 DAQ Board and programmed in LabVIEW. The block diagram is shown in Figure 8. All pressure transducer were calibrated using a reference pressure gage and adjusted for minimal error. The frequency for data collection for each pressure transducer was set at 1000 hertz. The oxidizer and fuel line each had a pressure transducer to measure the line pressure before the inlet of the valves. The valve pressure was used to measure propellant mass flow. A single pressure transducer was connected on the injector near the mixture exit to measure chamber pressure. Pressure, mass flow, O/F, Isp, and thrust measurements were recorded to an excel spread sheet at the same time the valve control system started its sequence.

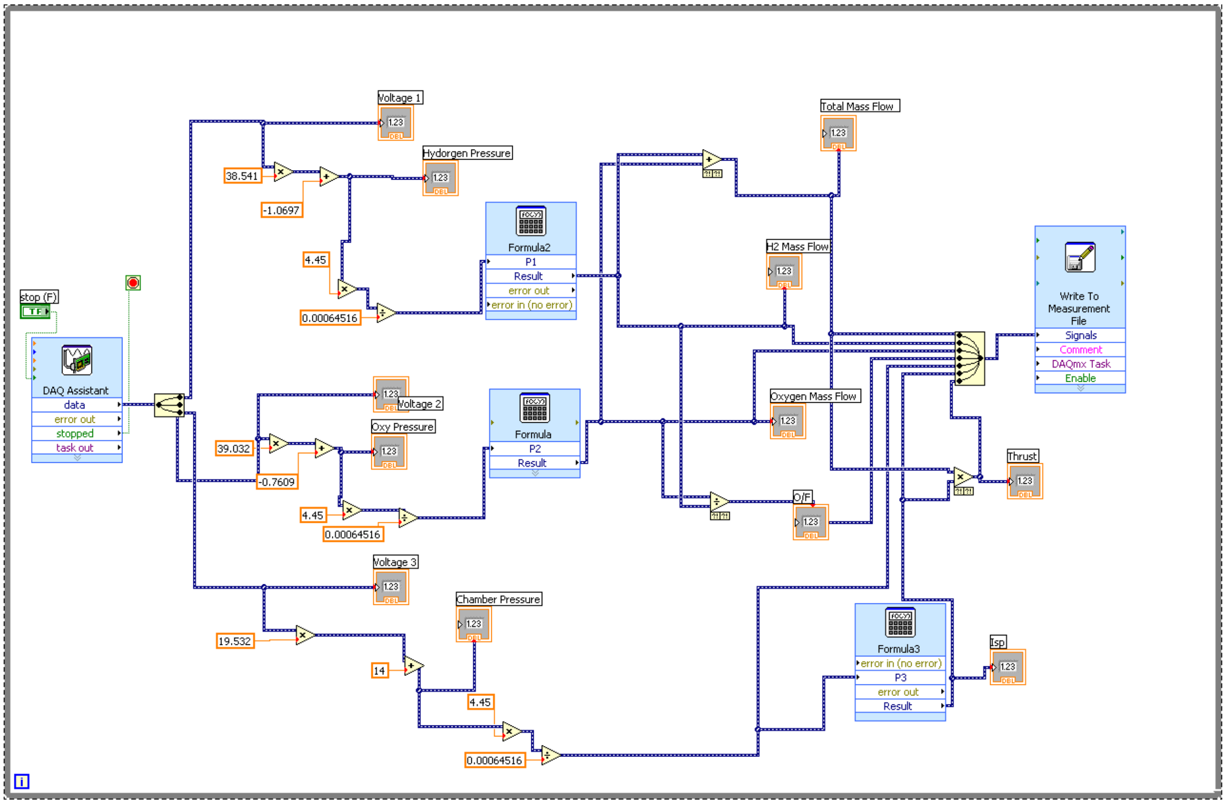


Figure 8- Data Acquisition Block Diagram

Chapter 3: Experimental Procedure

All tests were performed in ambient pressure of 14.7 psi with a nozzle area ratio of 3. Cold fire test were done for oxidizer and fuel to insure no leaks in the system. Previous work had been done where a stability map was generated for gaseous Methane/Oxygen using the same injector (Shirsat, 2009). The stability map showed that at an equivalence ratio of 1.7 there was a stable flame and this was taken as a starting point to find the best mixture of Methane/Oxygen to work with this thruster design. The thruster was allowed to cool in between test so that all data reflected the same starting conditions. The data collected was total propellant mass flow, mixture ratio, chamber pressure, Isp, and thrust. Pre and post fire data was recorded to show pressure ramp and decay in the combustion chamber.

3.1 Mass Flow Rate

The mass flow rate for the propellants were essential in order to sustain combustion in the combustion chamber. The valves came designed with a specific orifice diameter, which was used to calculate the choked mass flow. Choked flow occurs when sonic conditions are met at the orifice. Mass flow can be controlled by increasing the upstream pressure, but velocity is maintained at sonic. The mass flow rate can be determined with the orifice area and stagnation properties T_0 and P_0 found in the following relationship:

$$\dot{m} = A * P_O \sqrt{\frac{k}{RT_O}\left(\frac{2}{k+1}\right)^{\frac{k+1}{2(k-1)}}} \qquad (1)$$

Previous data (Shirsat, 2009) indicated that the proper mixture ratio needed to sustain combustion of was a MR=2.3. After several test with the thruster and igniter system, a mixture ratio of MR=3.4 performed better overall. Each test was calibrated using LabVIEW to reach this mixture ratio before ignition. Its important to note that this equations is true when the downstream pressure is maintained at variable lower pressures. The mass flow would be affected if the downstream pressure was increased.

The choked flow equation was used to measure the mass flow rate of the propellants to insure the proper mixture ratios before each test, however, due to the increase in chamber pressure with ignition downstream, the equation is no longer valid since the flow is not choked at the valves. In addition, no flow meter was available to measure the mass flow at the fast pulse rate. The mass flow was corrected by using the choked flow equation with the thruster stagnation properties.

As mentioned before, a pressure transducer was attached to the injector body, which was located near the combustion chamber. The pressure spike due to combustion of the propellants choked the flow at the throat of the thruster. The chamber pressure and temperature where the main properties that were being recorded. The throat area, molecular weight, and specific heat values of the mixture were then used to measure the total choked mass flow of the propellants.

3.2 IGNITER

After the flow rates were adjusted the power supply for the igniter was turned on. The voltage supplied to the igniter was provided by a Ultra Volt adjustable transformer. The transformer required its own power source and an additional variable voltage input, which could not exceed more that 5 Vdc. The transformer was calibrated to output 6k V to the igniter. An image of the transformer is shown on Figure 8.

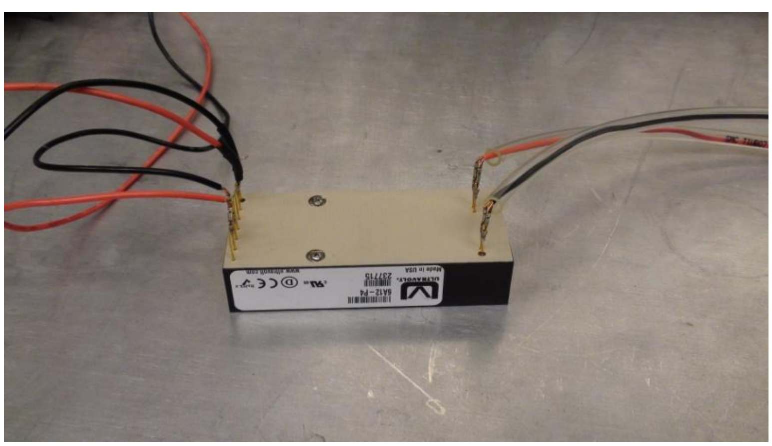
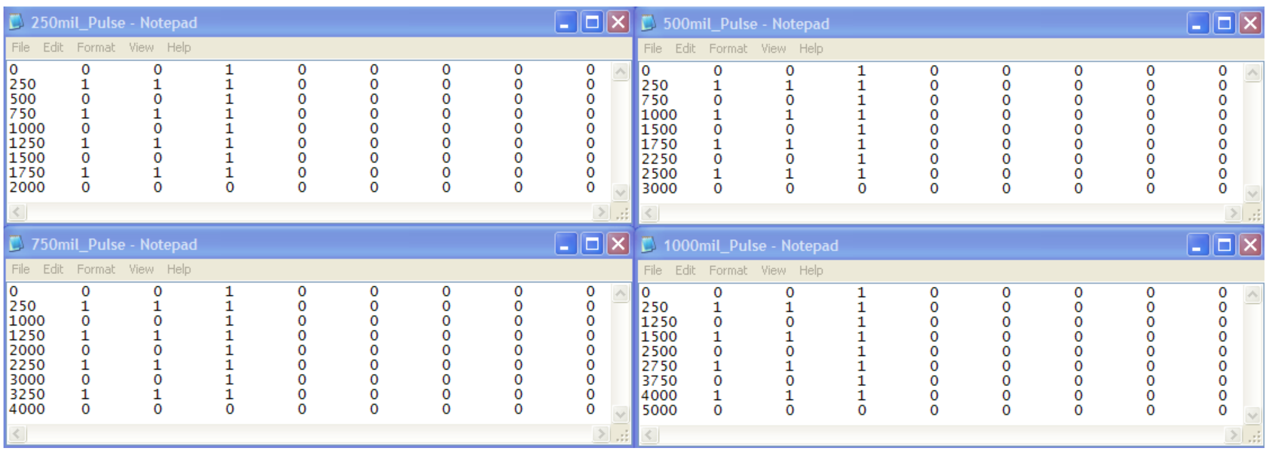


Figure 8-High Voltage Transformer

3.3 PULSE TESTING

The pulse length of the thruster was controlled using a binary text file that was written in Microsoft NotePad. The LabView program was created to read these types of text files in order to import different pulse lengths for a specific test. When the start button was selected, LabView would prompt the user to select a text file before the test started. During the selection process, all data had already started recording and would only stop until the test ended. The binary text files are shown in Figure 9 below.



250mil_Pulse - Notepad

0	0	0	1	0	0	0	0	0
250	1	1	1	0	0	0	0	0
500	0	0	1	0	0	0	0	0
750	1	1	1	0	0	0	0	0
1000	0	0	1	0	0	0	0	0
1250	1	1	1	0	0	0	0	0
1500	0	0	1	0	0	0	0	0
1750	1	1	1	0	0	0	0	0
2000	0	0	0	0	0	0	0	0

500mil_Pulse - Notepad

0	0	0	1	0	0	0	0	0
250	1	1	1	0	0	0	0	0
750	0	0	1	0	0	0	0	0
1000	1	1	1	0	0	0	0	0
1500	0	0	1	0	0	0	0	0
1750	1	1	1	0	0	0	0	0
2250	0	0	1	0	0	0	0	0
2500	1	1	1	0	0	0	0	0
3000	0	0	0	0	0	0	0	0

750mil_Pulse - Notepad

0	0	0	1	0	0	0	0	0
250	1	1	1	0	0	0	0	0
1000	0	0	1	0	0	0	0	0
1250	1	1	1	0	0	0	0	0
2000	0	0	1	0	0	0	0	0
2250	1	1	1	0	0	0	0	0
3000	0	0	1	0	0	0	0	0
3250	1	1	1	0	0	0	0	0
4000	0	0	0	0	0	0	0	0

1000mil_Pulse - Notepad

0	0	0	1	0	0	0	0	0
250	1	1	1	0	0	0	0	0
1250	0	0	1	0	0	0	0	0
1500	1	1	1	0	0	0	0	0
2500	0	0	1	0	0	0	0	0
2750	1	1	1	0	0	0	0	0
3750	0	0	1	0	0	0	0	0
4000	1	1	1	0	0	0	0	0
5000	0	0	0	0	0	0	0	0

Figure 9-Pulse Length Text File

The pulse lengths ranged from 250, 500, 750, 1000 milliseconds. Each test was also unique to a particular propellant pressure shown in the test matrix in Table 1. It can be seen that the fuel pressure was increased by 2 psi for each set of test. The oxidizer pressures vary and where adjusted in maintain the proper mixture ratio. After every test, the thruster was allowed to cool down to sustain the initial condition of ambient temperature of 25° C and pressure of 14.7 psi. The line pressures of the propellants were measured in gauge while the chamber pressure was measured in absolute.

Table 1-Test Matrix

Igniter on 8 psi Methan and 19 psi Oxygen								
Pulse time	Total Mass Flow (Kg/s)	O/F	Pc (Pa)	Thrust (N)	Isp Theo	C* (m/s)	C* theo (m/s)	η
250 mili sec pulse								
500mili sec pulse								
750mili sec pulse								
1000 mili sec pulse								

Igniter on 10 psi Methane and 24 psi Oxygen								
Pulse time	Total Mass Flow (Kg/s)	O/F	Pc (Pa)	Thrust (N)	Isp Theo	C* (m/s)	C* theo (m/s)	η
250 mili sec pulse								
500mili sec pulse								
750mili sec pulse								
1000 mili sec pulse								

Igniter on 12 psi Methane and 28 psi Oxygen								
Pulse time	Total Mass Flow (Kg/s)	O/F	Pc (Pa)	Thrust (N)	Isp Theo	C* (m/s)	C* theo (m/s)	η
250 mili sec pulse								
500mili sec pulse								
750mili sec pulse								
1000 mili sec pulse								

3.4 SPECIFIC IMPULSE (ISP)

Specific Impulse is defined as the thrust per unit weight flow of propellant. It is the premiere measurement to compare propellants, propellant combinations, and efficiency of rocket engines. The theoretical Isp equation is proportional to chamber temperature and pressure and exit pressure. Assuming ideal expansion for a nozzle of an area ratio of 3, the theoretical Isp for a propellant mixture was attained using the following equation:

$$I_{sp} = \sqrt{\frac{2kRT_c}{g_c(k-1)}\left[1-\left(\frac{P_e}{P_c}\right)^{\frac{k-1}{k}}\right]} \qquad (2)$$

3.5 Thrust Measurement

Thrust is defined as the momentum exchange between the exhaust and the vehicle and by the pressure imbalance at the nozzle exit. All test were done at ambient pressure of 12.7 psi and by assuming Pe to equal Pa for optimum expansion, the pressure thrust term is zero and can be dropped. The complete thrust equation is shown below:

$$F = \frac{w_p}{g_c} V_e + (P_e - P_a) A_e \qquad (3)$$

Due to the rigidity of the test stand, an accurate thrust measurement using a load cell could not be obtained. Thrust was measured using the theoretical Isp equation and the total propellant mass flow. Isp and total mass flow were derived using the chamber pressure (Pc). The following equation shows the relationship of Isp and thrust:

$$F = I_{sp} * \dot{w}_p \qquad (4)$$

3.6 Characteristic Velocity

Characteristic velocity is an empirical rocket parameter which measures combustion performance by indicating how many pounds of propellant must be burned to maintain chamber pressure.(Brown, 1996) The performance measurement is dependent on Pc, mass flow of the propellants, and the throat area of the thruster which is found in the relationship:

$$C^* = \frac{P_c A_t g_c}{w_p} \qquad (5)$$

In addition, a maximum C* was derived using the theoretical equation with the adiabatic flame temperature of the propellant mixture. The theoretical C* was compared to the actual value from test data to measure the efficiency of the performance. The theoretical C* is shown in the formula:

$$C^* = \frac{\sqrt{k g_c R T_c}}{k\sqrt{\left(\frac{2}{k+1}\right)^{\frac{k+1}{k-1}}}} \qquad (6)$$

Chapter 4: Results and Discussion

The Micro Thruster presented in this thesis was designed to investigate pulsing at different thrust levels while maintaining a simple control system. As mentioned before, the only data being recorded is the pressure at the inlet of the valves to measure mass flow for mixture ratio and the chamber pressure of the thruster to measure Isp, total mass flow, C*, and thrust. The test data was recorded at a frequency of 1 kilo hertz for each set of test. The following results are only for methane/oxygen propellant mixtures with 6, 8, 10, 12 psi methane line pressure while maintaining a mixture ratio of 3.2. Finally, Due to complications with the design of the igniter, hydrogen/oxygen propellant mixture could not be tested.

4.1 METHANE-OXYGEN THRUSTER

4.1.1 Pressure Measurement Recorded with DAQ

Chamber pressure was measured at short pulses and increased by 250 milliseconds for each additional test. The goal was to see if chamber pressure could be maintained for that small duration of time. In order to maintain constant chamber pressure, a self-propagating flame had to be kept in the combustion chamber, however, due to the high heat loss rate to the wall of the combustion chamber a flame could not be sustained. The chamber wall would act as a heat sink, which absorbed all the heat energy that was generated from combustion of the propellants. To maintain a stable flame, the igniter had to be powered throughout the duration of the test in order to provide the additional heat to sustain combustion. The integration of the igniter and injector allowed the premixed propellants to combust whenever the valves where open. Combustion would cease when the valves closed. Although combustion was sustained, the pressure was not constant throughout the test. Figure 11 shows the pressure reading for a 1000 millisecond pulse at a 8 psi methane line pressure. Figure 12 is a zoomed in image of the pressure reading which shows the pressure fluctuations.

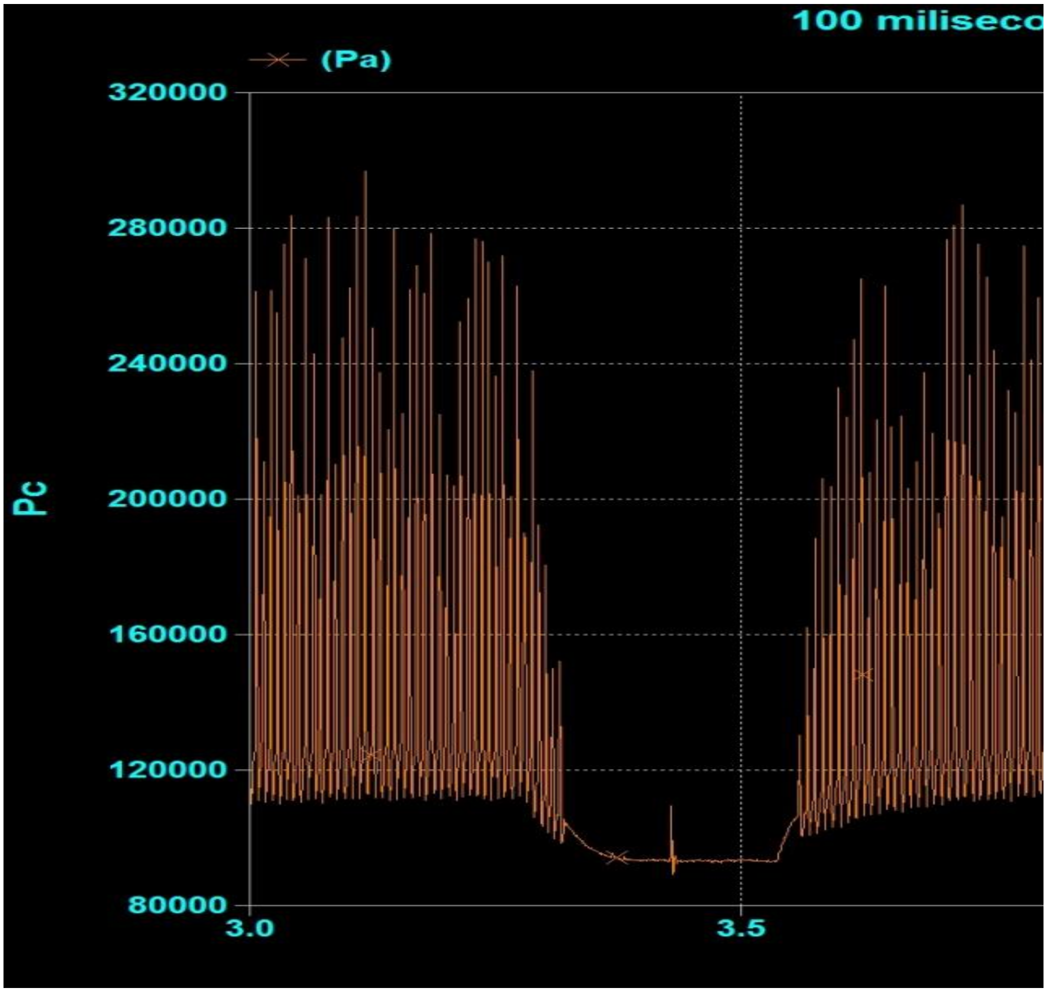


Figure 11- Pressure at a 1000 millisecond pulse

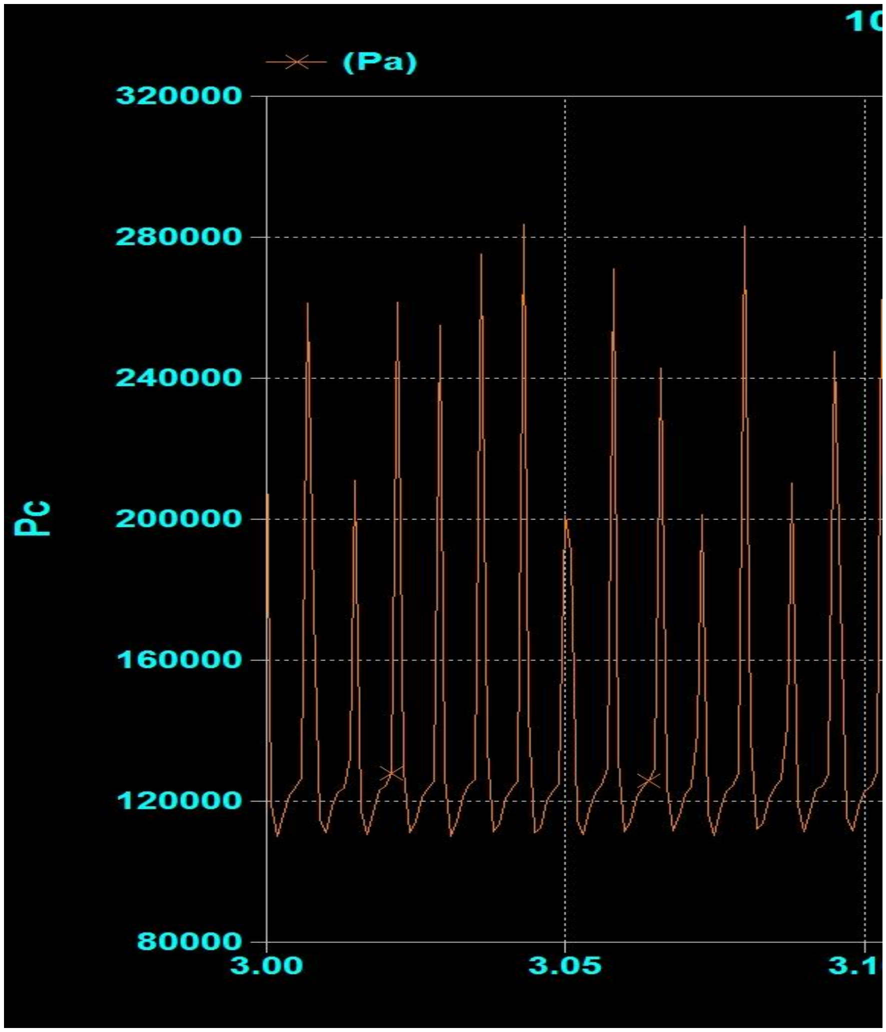


Figure 12- Zoomed Pressure Fluctuation

The pressure considered in all the test data was an average of the highest recorded pressure. Pressure was assumed constant at this average pressure for each pulse, which was later used to calculate the total propellant mass flow, Isp, and thrust. Since the sample rate was 1K hertz, pressure data exceeded more than 5000 points which increased as pulse length increased. WinPlot was used to graph the large amount of data.

4.1.2 Test Matrix Results

As expected, the results show that as propellant line pressure increased the chamber pressure increased. Table 2 shows the results for Methane/Oxygen test matrix. It was difficult to duplicate the same chamber pressure for each set of test since combustion was based on mass flow and mixture ratio, which was affected by valve actuation and calibration. Chamber pressures had slight fluctuations when pulse lengths were increased resulting in similar theoretical specific impulse for each set of test.

Table 2-Summary Table for Methane-Oxygen

Igniter on 6 psi Methan and 14 psi Oxygen								
Pulse time	**Total Mass Flow (Kg/s)**	**O/F**	**Pc (Pa)**	**Thrust (N)**	**Isp Theo**	**C* (m/s)**	**C* theo (m/s)**	**η**
250 mili sec pulse	0.000151649	3.43	226695	0.1585	106.56	1320.15	2106.94	0.63
500 mili sec pulse	0.000151582	3.46	226595	0.1584	106.54	1333.04	2106.94	0.63
750 mili sec pulse	0.000151421	3.23	226354	0.1582	106.48	1362.81	2106.94	0.65
1000 mili sec pulse	0.000154512	3.36	230974	0.1630	107.54	1388.97	2106.94	0.66

Igniter on 8 psi Methan and 19 psi Oxygen								
Pulse time	**Total Mass Flow (Kg/s)**	**O/F**	**Pc (Pa)**	**Thrust (N)**	**Isp Theo**	**C* (m/s)**	**C* theo (m/s)**	**η**
250 mili sec pulse	0.000162769	3.46	243318	0.1760	110.19	1341.98	2106.94	0.64
500 mili sec pulse	0.000162492	3.66	242904	0.1755	110.11	1367.88	2106.94	0.64
750 mili sec pulse	0.000160972	3.57	240631	0.1731	109.63	1350.82	2106.94	0.66
1000 mili sec pulse	0.000159894	3.61	239020	0.1714	109.29	1347.71	2106.94	0.67

Igniter on 10 psi Methane and 24 psi Oxygen								
Pulse time	**Total Mass Flow (Kg/s)**	**O/F**	**Pc (Pa)**	**Thrust (N)**	**Isp Theo**	**C* (m/s)**	**C* theo (m/s)**	**η**
250 mili sec pulse	0.00016964	3.42	253589	0.1868	112.25	1341.98	2106.94	0.64
500 mili sec pulse	0.000168603	3.38	252039	0.1852	111.95	1369.93	2106.94	0.65
750 mili sec pulse	0.000166429	3.47	248790	0.1817	111.31	1338.72	2106.94	0.64
1000 mili sec pulse	0.000165452	3.39	247329	0.1802	111.01	1334.23	2106.94	0.63

Igniter on 12 psi Methane and 28 psi Oxygen								
Pulse time	**Total Mass Flow (Kg/s)**	**O/F**	**Pc (Pa)**	**Thrust (N)**	**Isp Theo**	**C* (m/s)**	**C* theo (m/s)**	**η**
250 mili sec pulse	0.000183637	3.6	274513	0.2091	116.05	1341.98	2106.94	0.64
500 mili sec pulse	0.000183273	3.45	273969	0.2085	115.96	1355.5	2106.94	0.64
750 mili sec pulse	0.000182366	3.39	272613	0.2070	115.73	1354.52	2106.94	0.64
1000 mili sec pulse	0.000178054	3.36	266167	0.2002	114.59	1328.08	2106.94	0.63

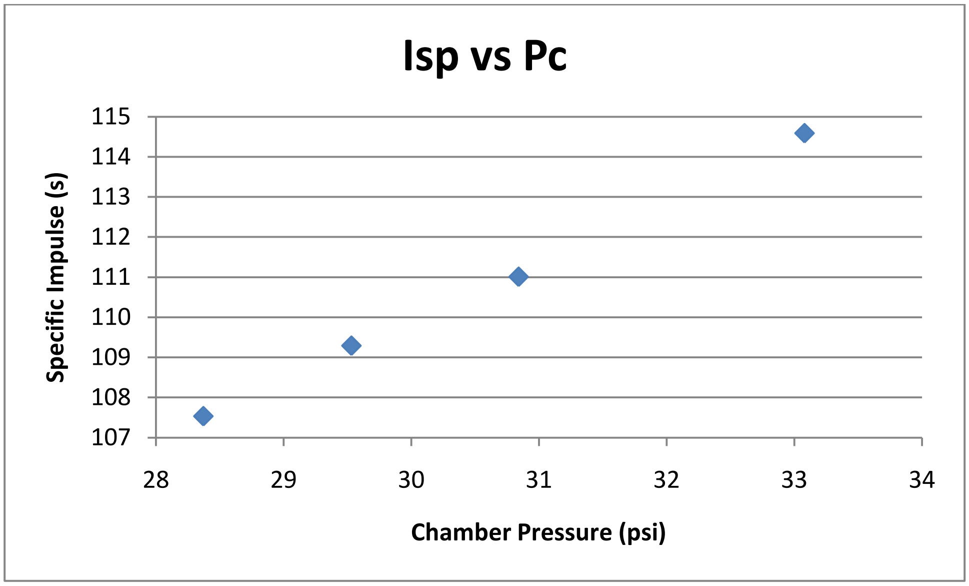


Figure 13-Specific Impulse vs Chamber Pressure

Specific impulse was also expected to increase with chamber pressure since the formula is strongly dependent in that property. Figure 13 shows Isp vs chamber pressure in psi. Any further increase in chamber pressure will result in higher Isp values. An increase in chamber temperature can also increase Isp, but the temperature was kept constant at 1500ºK for every test. This temperature was found by igniting the thruster without the nozzle and measuring the plum temperature with a thermal couple after a 1-minute steady state burn.

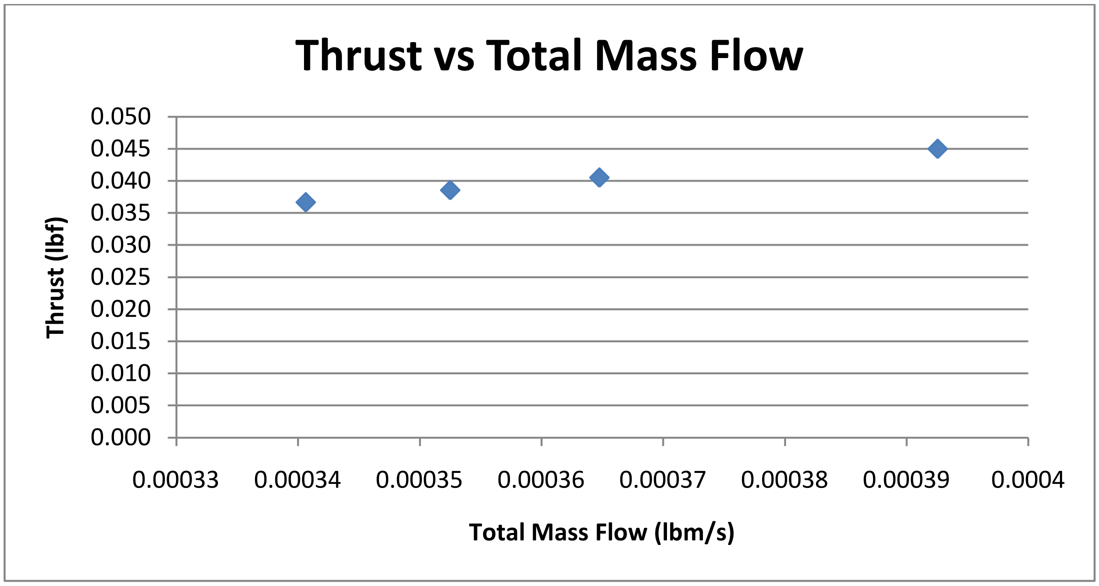


Figure 14-Thrust vs Total Mass Flow

An increase in total propellant mass flow resulted in an increase in chamber pressure which in essence increased the thrust. Thrust is dependent in mass flow and Isp values which are shown to increase with chamber pressure. Figure 14 shows an increase in thrust with an increase of total mass flow rate.

The data shows that the duration of the pulse does not affect the performance of the thruster. In fact, the thruster performance is measured using characteristic velocity efficiency. The C* efficiency was obtained by comparing the C* of the test data, which is dependent on total propellant mass flow and chamber pressure, to the theoretical C* star using adiabatic flame temperature. This formula can be seen in equation 6. The C* efficiency varied from 63% to 67% throughout the entire test.

4.2 HYDROGEN-OXYGEN THRUSTER

Hydrogen and oxygen was very difficult to test and measure due to the low ignition energy needed to start ignition and because of the high temperatures that it would produce. The spark igniter could not sustain the rapid pressure and temperature spike after one of two pulses. As mentioned before, the spark igniter would arc to the wall of the injector face which would leave the platinum wire exposed to the propellant mix when combustion occurred. The low mass flow needed for these test plus the high combustibility of the mixture would damage or destroy the platinum wire before the test would finish its cycle. It was difficult to repair and to replace a new spark igniter after every test. The igniter worked well with the methane/oxygen mixture but due to the violent nature of the hydrogen/oxygen, a new injector-spark igniter system will have to be designed

Chapter 5: Conclusion and Future Work

5.1 CONCLUSIONS

The conclusions that can be drawn from this body of work are:

- Compared to other ignition sources, the integrated arc ignition design worked very well for methane/oxygen mixtures. The tolerance for the distance that the platinum wire and the wall need to have is very important to have ignition. Spark stability is the only method a self-propagating flame can exist in the combustion chamber.
- The LabVIEW program was successful in controlling the valve actuation and igniter. Two block diagrams were both able to run in sync to send digital ON/OFF signals while recording pressure data.
- Having short and long pulse lengths does not affect the performance of the thruster. The pressure recorded did not fluctuate by much for each test set.
- With the chamber pressure data and theoretical Isp equation, thrust measurements from 90mN to 113 mN were achieved. Once again, this is based the thermal dynamic properties at the time of combustion.
- Mass flow rates were very difficult to measure at fast pulse rates. Standard turbine based flow meters could not read the change in mass flow at pulses under one second when combustion occurred. Using the chocked flow equation at the state when the thruster chamber had combustion gave better mass flow data. Chamber pressure was successfully recorded using LabVIEW at 1000K hertz.
- A pressure spike for a 250-millisecond burn time was achievable with the current test setup, however, shorter burn times could be possible with improved solenoid valves.

5.2 FUTURE WORK

The following are suggestions on how to improve the technique of experimentation:

- The integration of the igniter and injector can be improved to work for other propellant mixtures. A modular approach should be taken when designing an igniter in order to allow the user to change and replace the platinum wire inside the casing. A more comprehensive study should be done on ignition of liquid fuels.
- The LabVIEW program can be improved to continuously run without prompting the user to load the SSR digital signal. This would help in calibrating mass flows and adjusting pressures before testing.
- A block diagram should be made specifically for manual valve actuation control. This could improve pre-test procedures when valves are being tested.
- The thruster system should be used as a torch igniter for larger rockets. The igniter can be fabricated with the injector to serve as an ignition source for 50 lbf- 100 lbf rockets.

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Vita

Carlos Gomez was born January 3, 1982, in Los Angeles, California. Carlos graduated with a high school degree from Ysleta High School and attended The University of Texas at El Paso (UTEP) shortly after in the fall of 2002. During his time as an undergraduate student, he participated in activities such as Shell Eco Marathon, Solar Car Project, and the Aerodesign Competition. In December of 2007 he received his Bachelor of Science degree in mechanical engineering and continued with his education at UTEP under the supervision of Dr. Choudhuri. Carlos is currently doing research under Dr. Choudhuir investigating meso scale propulsion systems.

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