Performance Evaluation Of A LOx-LCH4 Reaction Control System Thruster

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PERFORMANCE EVALUATION OF A LO\textsubscript{X}-LCH\textsubscript{4} REACTION
CONTROL SYSTEM THRUSTER

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Dedication

The following work is dedicated to my parents, Laura and Jose Luis Mena, who have supported me throughout my entire life; always guiding me to follow my dreams and never give up. I would also like to dedicate this work to my sister, Laura Mena-Hanson, who has inspired my life in more ways than one. To my beautiful Barbs, who had the patience to deal with my late nights at the lab and help me de-stress when experiments failed. Finally, I’d like to dedicate this to my friends and co-workers who have made my graduate studies an unforgettable time.
PERFORMANCE EVALUATION OF A LO\textsubscript{X}-LCH\textsubscript{4} REACTION CONTROL SYSTEM THRUSTER

by

JOSE LUIS MENA, B.S. Mechanical Engineering

THESIS

Presented to the Faculty of the Graduate School of
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for the Degree of

MASTER OF SCIENCE

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THE UNIVERSITY OF TEXAS AT EL PASO
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Special recognition is given to Arturo Acosta-Zamora, who not only formed the development and preliminary testing phases of this project, but also mentored me as I took the project to the next phase.

Finally, I’d like to thank all cSETR members and staff, as each and every one made an impression that contributed to the project and my academic career.
Abstract

The following work outlines the assessment and characterization of a 2 to 8 lbf LO\textsubscript{X} - LCH\textsubscript{4} reaction control thruster. The evaluation includes both experimental tests and preliminary computational analysis. Testing was conducted at O/F values ranging from 1.0 to 3.0; however, future testing will evaluate the thruster at higher and intermediate mixture ratios. The test setup includes three main systems: cryogenic propellant delivery system, thrust measurement capability, and automation and controls system. Propellant flow control is achieved through the use of in-house designed cavitating venturis. Autonomous system control is accomplished via a LabVIEW program interface and DAQ system. This allows for the manipulation of system components, as well as real-time data acquisition. The thrust measurement system uses a torsional thrust balance, where the rotational axis is fitted with torsional pivots providing a known resistance to the force provided by the thruster. Testing has concluded significant under performance when compared to theoretical values, as well as near 60\% ignition reliability. Moreover, baseline CFD analyses have been conducted to give an understanding of the injector performance and propellant mixing at the point of ignition. Both test data and CFD analysis will drive the next generation thruster design.
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Chapter 1: Introduction

The following work describes the testing and performance evaluation of a 2-8 lb_f class liquid oxygen-liquid methane (LO\textsubscript{X}-LCH\textsubscript{4}) reaction control system (RCS) thruster. The work herein is a continuation of previous test setup and thruster validation studies which were used as benchmark for testing. The work is divided into 5 chapters which describe the test setup, test results and discussion, and concludes with the analysis performed for future modifications to improve thruster performance.

1.1 Introduction

The thruster, also referred to as the “pencil thruster”, was provided to the center for Space Exploration Technology Research (cSETR) through a partnership with the National Aeronautics and Space Administration (NASA). The pencil thruster is currently in use in the Morpheus Lander and works as the vehicle’s reaction control system.

The thruster was granted to the cSETR for performance validation and operation reliability studies. It offers a unique opportunity of study, as not only does it incorporate major aspects of rocket propulsion, but is also pioneering the use of LO\textsubscript{X}-LCH\textsubscript{4} propellant combination. This is a particular point of interest, as the aforementioned propellant combination may offer a “nontoxic” alternative to its hypergolic predecessors. Moreover, when compared to the widely used liquid hydrogen-liquid oxygen combination, LCH\textsubscript{4} carries several advantages that make it an ideal candidate for future rocket engines. This includes storage and handling feasibility due to methane’s higher density and comparable storage temperatures to LO\textsubscript{X}, comparable specific impulse, and the opportunity of in situ resource utilization [10, 11]. However, despite the apparent advantages there is still a significant lack of knowledge and test data in order to accurately design safe, high performing, and reliable LO\textsubscript{X}-LCH\textsubscript{4} rocket engines.
1.2 Pencil Thruster Overview

The pencil thruster is composed of five major components: (1) combustion chamber, (2) LO$_X$ manifold, (3) LCH$_4$ manifold, (4) nozzle, and (5) igniter. See figures 1.1 and 1.2 below. It should be noted that specific injector and chamber dimensions are not included due to export control regulations.

![Figure 1.1: Pencil thruster CAD assembly and cross-sectional view.](image)

The thruster was designed considering a sea level specific impulse, $I_{sp}$, of 150 seconds. It incorporates a radial injection concept via the LO$_X$ and LCH$_4$ manifolds which sleeve over the combustion chamber body. Both the chamber and nozzle are cooled using fuel film cooling.

The thruster is equipped with a $15^\circ$ conical nozzle of area ratio, $A_e/A_{th}$, value of 7 and throat area of 0.059 in$^2$ [10]. Nozzle expansion analyses (see Chapter 3) demonstrate that this area ratio is over expanded for the 13 psia local atmospheric pressure. The effect of the overexpansion is detailed in Chapter 3.
Figure 1.2: (a) Chamber cross-section showing electrode; (b) Retrofitted sparkplug which serves as thruster igniter.

Figure 1.2 (a) closes in to the cross-section of the chamber and electrode. From the figure, LO\textsubscript{X} (represented with a blue arrow) flows over the electrode head and meets the LCH\textsubscript{4} (represented in red arrows), which is injected 0.02” downstream of the electrode head. The film cooling is also represented in red. It is important to note that the electrode placement is currently under study as it may serve as a point of improvement for the next iteration thruster.

Figure 1.2 (b) shows a retrofitted automobile sparkplug that serves as the igniter for the thruster. Ignition is achieved by creating an arc between the electrode tip and the body of the thruster. This spark is generated through the use of an 8247 MSD ignition coil which transforms 12 V at 11 Amp (provided from a lawnmower battery) to approximately 40 kV. Moreover, the ignition system requires a 5 V\textsubscript{pp} TTL square wave at a frequency of 100 Hz [1]. The signal is provided by a function generator and is the point of control for testing purposes. See Appendix B for more details.

1.3 Objectives

The Morpheus lander feeds the RCS directly from the main engine tanks, challenging the conventional method of using separate less-pressurized tanks specifically for attitude control [10, 11]. This method saves both space and weight within a space craft design; however, limits the RCS propellant inlet conditions to that of the main engine requirements. With this in mind, the
main objective is the evaluation of the pencil thruster performance over a wide range of inlet conditions as to ensure its safe and reliable in-flight operation. The following work covers a mixture range from 1 to 3; however future testing will focus on determining the mixture ratio for optimum performance. The thruster was also evaluated for ignition reliability at the different mixtures.

Thrust, chamber pressure, and propellant conditions were monitored throughout the tests. Baseline CFD analysis have been conducted to give an understanding of the injector performance and propellant mixing at the point of ignition.
Chapter 2: Experimental Setup

The following chapter details the propellant delivery, thrust measurement, and controls and data acquisition subsystems that make the experimental setup. The propellant delivery section also includes the design and validation work taken to introduce a flow control feature into the propellant delivery system.

2.1 Propellant Delivery System

The pencil thruster’s propellant delivery system was previously designed and validated by former cSETR students [13]. The system is composed of 304 stainless steel ½” OD tubing, with the exception of the ¼” OD bypass on the LO\textsubscript{X} supply line. The oxidizer system operates through the actuation of LO\textsubscript{X} compatible solenoid valves [4]. Fluid conditions are monitored through the use of cryogen rated instrumentation. Figure 2.1 below represents the LO\textsubscript{X} supply schematic in blue.

![Figure 2.1: LO\textsubscript{X} delivery system fluid schematic.](image)

From figure 2.1, LO\textsubscript{X} is supplied via a self-pressurizing dewar tank. The line is equipped with both a cavitating venturi and turbine flow meter for flow rate measurement and control. The cavitating venturi flow rate control feature is explained with further detail below. A bypass was amended to the original LO\textsubscript{X} delivery system to protect the turbine flow meter from over spinning during initial start-up, as well as to aid in the chilling process prior to testing. Liquid nitrogen, LN\textsubscript{2}, (black line in Figure 2.1) is utilized to condition the line before introducing LO\textsubscript{X},
reducing the overall LOx consumption. This method was implemented in an effort to avoid an oxygen rich environment within a combustion test. Furthermore, the LN2 can also be utilized for purging posttest. Gaseous nitrogen (GN2) purging (represented in green) is also utilized once the test has concluded to remove the system from any combustibles.

The fuel delivery system carries added complexity as, due to university safety protocol, LCH4 storage is not allowed and thus must be condensed prior to testing. The propellant delivery system was originally designed to house the condensing unit stationed next to the GN2 purge. This section is marked with a red dashed box in Figure 2.2 below.

![Figure 2.2: LCH4 delivery system fluid schematic.](image)

However, due to logistical reasons, a mobile condensing unit (hereafter referred to as the Cryocart) was amended to the LCH4 supply subsystem to serve the thruster requirements. This section is marked by a blue dashed box in Figure 2.2. The inlet originally designed for the condensing unit was replaced by a secondary GN2 purge.

The Cryocart allows for up to 3.2 gallons (12 L) of liquid methane production. Figure 2.3 below illustrates the Cryocart fluid schematic. Eight cryogenic thermocouples line the condensing tank, which allow the LCH4 level to be measured. Once the desired LCH4 quantity is achieved, helium is used to pressurize the Methane Condensing Tank and deliver the LCH4 to the test article. A detailed condensing procedure is included in Appendix A.
Propellant Flow Control

Cavitating venturis were implemented within the propellant delivery system in order to control the flow rate into the thruster. The following section briefly discusses the theory behind the venturi, as well as the design and validation process undertaken.

Propellant Flow Requirements

As previously stated, the primary project goal is to test the thruster under various inlet conditions. In order to accomplish this, the first step was to implement a repeatable and reliable method of propellant flow rate control. Original thruster design criteria called for a sea level Isp of 150 seconds. Assuming the design Isp is achieved at the 8 lbf thrust level, equation 1 below was used to determine the required propellant mass flow and consequent venturi dimensions [2, 17].

\[
Isp = \frac{F}{m_{prop}}
\]  

(1)

Having the required flow rate, a cavitating venturi was designed to passively control the liquid propellant feed into the thruster. The cavitating venturi takes advantage of the Bernoulli principle by directing the flow through a converging nozzle which causes a velocity increase and
subsequent pressure drop. By dropping the pressure below its saturation point at the given operating temperature, cavitation is induced which limits the amount of liquid that can exit the venturi. This phenomenon can be quantified as a one dimensional incompressible flow through equation 2 below [7].

\[
\dot{m} = C_d A_{th} \sqrt{2g_c \rho (P_1 - P_{th})}
\]  

(2)

From equation 1, \(A_{th}\) is the venturi throat area, \(P_{th}\) is the throat pressure, \(P_1\) is the inlet pressure and \(\rho\) is the fluid’s inlet density at liquid state. The \(C_d\) represents the venturi’s discharge coefficient, which was experimentally determined. For mass flow calculations, the throat pressure is set to the fluid saturation pressure at the given operating temperature. An important aspect to note is that the downstream pressure has no effect on the flow rate. This idealized situation, however, is limited to a specific venturi cavitating regime, which can be quantified through a pressure recovery approach through equation 3.

\[
P_{cr} = \frac{C_p C_d^2}{1 - (\beta)^4}
\]  

(3)

From equation 2, \(P_{cr}\) represents the critical pressure ratio between downstream to upstream pressure, \(C_p\) is the pressure recovery coefficient, and \(\beta\) is a ratio of throat to inlet diameters [7]. Equation 2 thus provides a pressure ratio (downstream to upstream pressure) limit, where surpassing the set limit ceases the cavitation process and allows the downstream pressure to diminish the flow rate [7]. Testing determined a critical pressure ratio and pressure recovery coefficient of 0.69 and 0.72, respectively. See Figure 2.5 (a).

For simplicity and economical purposes a single cavitating venturi design was implemented for both the fuel and oxidizer. As a result, LO\(_X\) fluid properties governed the design criteria, given that LO\(_X\) boils at a significantly lower temperature than LCH\(_4\) (162 °R vs 201 °R at atmospheric pressure). A throat diameter of 0.04” was determined to provide flexibility for both fluid’s flow range, as well as machining feasibility. Converging and diverging angles of 15° and 7°, respectively, were utilized to diminish irreversible losses [3]. See Figure 2.4 below.
Flow Control Validation

The cavitating venturi underwent a series of tests to determine the $C_d$ and operating cavitating regime. Both water and LCH$_4$ flow validation tests were conducted. LO$_X$ flow was validated during the actual RCS testing. Figures 2.5 (a) gives the $C_d$ results for the water tests, Figure 2.5 (b) displays the LCH$_4$ tests results, and Figure 2.5 (c) shows the LO$_X$ flow validation.

![Figure 2.4: Cavitating venturi CAD and cross-sectional view.](image)

![Average $C_d = 0.977$, $P_{cr} = 0.69$](image)

Figure 2.5: (a) Water flow validation results; (b) LCH$_4$ flow validation results; (c) LO$_X$ flow under RCS operation.
From Figure 2.5 (a), water tests concluded the venturi upholds cavitation when operated at $P_2/P_1$ (downstream to upstream pressure ratio) below 0.69. In other words, $P_{cr} = 0.69$. When operating under the venturi’s cavitating regime, the venturi upholds an average $C_d$ value of 0.977. This value carries an uncertainty of ±0.0357 with a 90% confidence interval. A similar relationship was found for the LCH₄ tests (Figure 2.5 b), as an average $C_d$ of 0.94 was maintained when operating below the critical pressure ratio. From Figure 2.5 (c), the LOₓ flow data closely matched the theoretical limit set by the cavitating venturi. The sharp overshoot pertains to the transient startup of the turbine as the chamber pressure rose during ignition. Flow rate was measured using a Hoffer Flow Control HO Series liquid turbine flow meter. Temperature data was collected through the use of OMEGA E-Type thermocouple probes. Pressure data was gathered using OMEGA thin film cryogenic pressure transducers. Instrumentation specifications can be found at [8, 14].

2.2 Thrust Measurement System

The following section provides an overview of the previously developed thrust measurement system. This section serves only as reference for understanding of the system setup and results analysis, however, for a detailed study of the torsional thrust balance see reference [5].

![Figure 2.6: Torsional Thrust Balance (TTB) CAD model.](image)

Thrust is indirectly measured through the use of a student design torsional thrust balance (TTB) system. The system is composed of a fixed base which allows a central axis pivot point
for the *Moment* and *Counterweight* arms. See Figure 2.6 above. Moreover, system stiffness is provided by four torsional pivots housed inside the *Central Base* and *Moment Arm Block*. The pivots induce a 7.52 in-lb/deg stiffness, for a total spring rate of 30.08 in-lb/deg [5].

The thruster is attached to the *Moment Arm* using the pre-fabricated holes. Prior to attachment, thruster and interfacing element assembly (valves, lines, instrumentation, etc.) are weighed. A counterweight is then placed on the *Counterweight Arm* to balance the thruster assembly weight. The location within the respective arms is determined though a static moment-balance equation. This step is undertaken as to isolate thrust measurement and avoid external forces acting on the TTB [5, 15].

When the thruster is fired, the thrust creates a moment about the central axis. This causes an equal, yet opposite displacement of the *Moment* and *Counterweight* arms. It should be noted that the TTB operates under a small angle assumption which allows for a linear relationship between the displacement of the *Moment Arm* and the actual thrust generated. As a result, both thruster and counterweight placement is important to avoid arm rotation larger than 5 degrees for any given thrust level. Thruster location can be determined by the expected thrust level and overall torsional spring constant [5, 6, 15].

The TTB provides a repeatable arm displacement to thrust generated relationship; however, the displacement itself is detected through the use of a laser. An Opto-NCDT 1402-100 displacement laser was chosen for the application. This laser has a measurement range of 3.94 in (100 mm) with a resolution of 0.0008 in (0.02 mm). Displacement is measured on the *Counterweight Arm* as shown on Figure 2.7 below. Displacement data is gathered at a frequency 750 Hz however can be gathered at a maximum frequency of 1500 Hz [5]. For best results, the laser should be aimed perpendicular to the *Counterweight Arm* hitting a dull light-colored surface.
A fixed weight calibration process was used to correlate displacement to thrust. Figure 2.8 below illustrates the resulting calibration curve. A linear relationship (R^2 value of 0.9972) is seen between the displacement reading and the load imposed by the calibrated weights. A detailed calibration procedure is included in Appendix A.

![Figure 2.7: TTB and displacement laser assembly.](image)

**Figure 2.8: Displacement-Thrust averaged calibration curve.**

\[
y = 1.7408x - 0.354 \\
R^2 = 0.9972
\]
Given that the TTB functions through the oscillation of the *Thrust/Counterweight Arms*, the system’s natural frequency must be considered in order to filter system vibrations from the actual thrust reading. This is of particular importance under thrust pulsing operation. Previous work determined a loaded TTB natural frequency of 4.7 Hz [1, 5]. This was accomplished by performing a Fast Fourier Transform (FFT) analysis on the system’s raw data. Figure 2.9 illustrates the analysis conducted.

![FFT results](image)

*Figure 2.9: FFT results carried out on a three second RCS test fire data.*

From the figure, the loaded system natural frequency is evident as the amplitude peaks at a frequency of 4.7 Hz. Using a MatLab based program, which implements a combination of Butterworth and Band Stop filters, the systems natural frequency is removed from the thrust data [5].

### 2.3 Data Acquisition & Control System

The following section provides an overview of the previously developed controls, automation, and data acquisition system. This section is included to provide insight to how all subsystems are integrated for the pencil thruster testing, however, for a detailed description of the system see reference [1].
2.3.1 System Control

All valves and ignition system are controlled via LabVIEW software, which implements the use of NI PCI 6521 cards, NI CB-37FH connector blocks, and PCI relays. By pairing the connector block physical channel to its respective virtual channel in the LabVIEW programming, a virtual switch can be created which actuates the respective valve and/or TTL signal (used to operate the igniter) based on a true or false statement. User interaction is achieved through the Graphical User Interface (GUI), by implementing Boolean operators as valve/igniter on-off switches. Figure 2.10 below illustrates the GUI for manual valve and igniter operation.

![Manual Controls Table]

Figure 2.10: LabVIEW graphical user interface for valve and igniter manual operation.

Automation was introduced into the pencil thruster testing operations not only to remove the human error factor, but also to have repeatability within individual test sequences. Automated valve control is achieved through the use of a FOR loop within the LabVIEW program [1]. The loop receives input from a pre design script file as shown in Figure 2.11 below. From the figure, the left-most column dictates the time, in milliseconds, when each valve is either opened or closed. Each remaining column actuates a specific valve based on a binary number system, where the valve will be opened and remain open whenever a 1 is read and be
closed and remain closed whenever a 0 is read. The igniter is controlled in the same manner by turning the TTL signal on and off at the desired time.

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<td>0</td>
</tr>
<tr>
<td>7</td>
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<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
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<td>0</td>
<td>0</td>
<td>0</td>
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</tr>
<tr>
<td>8</td>
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<td>0</td>
<td>0</td>
<td>1</td>
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<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>0</td>
</tr>
</tbody>
</table>

Figure 2.11 LabVIEW test script

2.3.2 Emergency Sequence & Red-Line Activation

Noting the hazardous nature of the pencil thruster tests, both an emergency button and a secondary automated sequence can be accessed in the event of an emergency. The emergency button is hardwired into the electrical system, thus acts independently from LabVIEW software, however must be manually actuated. On the other hand, the secondary automated sequence operates within the LabVIEW program and thus acts independently from the user. The emergency measure functions through continuous monitoring of set instrumentation, typically thermocouples and pressure transducers. The individual instrumentation inputs can then be programmed to act as triggers, known as red-lines, which automatically end the normal test operation and activate the secondary emergency sequence. This allows the user to set both pressure and temperature limits on the specific test. For this particular testing operation, all pressure transducers and K-type thermocouples were set to act as triggers in the event a pressure of 300 psia (2.07 MPa) and/or temperature of 1842 °R (1023 K) were reached.
2.3.3 Instrumentation Data Acquisition

Similar to the valve operation, the physical channel connecting the instruments within the DAQ must match its respective analog input channel in the LabVIEW program. Moreover, individual virtual channels must be created for the different types of input (i.e. voltage, current, etc.). All instrumentation devices and indicators are housed under a WHILE loop, allowing for continuous data feed at the onset of the program. Figure 2.12 provides an example of the E-type thermocouple channel input into the DAQmx, as well as the resulting GUI display [1]. It should be noted that the order of the channel input dictates the order the thermocouples must be connected into their physical channel within the DAQ.

Data acquisition was limited to approximately 5 Hz by the system’s slowest component (USB Thermocouple DAQ); however, the system has been upgraded to sole PCI card use and will allow faster recording rates for future tests. Pressure and temperature data was collected for the individual propellants at upstream locations and just prior to thruster injection. LOX flow rate, chamber pressure and thruster temperature data was also recorded.

Figure 2.12 LabVIEW instrumentation block diagram and GUI display
Chapter 3: Test Results & Discussion

The following chapter discusses the test matrix under study, along with its respective test results. The test procedure is detailed within Appendix A. Nearly 40 liquid-liquid tests were conducted at varying conditions; however, only 5 tests will be discussed in detail.

3.1 Test Matrix

As previously stated, the project objective was to test the pencil thruster over a range of operating conditions. Several tests were conducted between oxygen-to-fuel mixture ratio (O/F) of 1 and 3. Table 3.1 below summarizes the test matrix used. Noting previous testing data, LO\textsubscript{X} and LCH\textsubscript{4} temperatures were set to 212 °R and 249 °R, respectively, to determine propellant flow rate and require pressure.

<table>
<thead>
<tr>
<th>O/F:</th>
<th>LO\textsubscript{X}:</th>
<th>LCH\textsubscript{4}:</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Pressure (psia):</td>
<td>Flow Rate (lb/s):</td>
</tr>
<tr>
<td>1.0</td>
<td>150 ± 1</td>
<td>0.027</td>
</tr>
<tr>
<td>1.5</td>
<td>160 ± 1</td>
<td>0.032</td>
</tr>
<tr>
<td>2.0</td>
<td>165 ± 1</td>
<td>0.036</td>
</tr>
<tr>
<td>2.5</td>
<td>170 ± 1</td>
<td>0.038</td>
</tr>
<tr>
<td>3.0</td>
<td>175 ± 1</td>
<td>0.040</td>
</tr>
</tbody>
</table>

3.2 Thrust & Chamber Pressure Results

Figure 3.1 (a) graphs the chamber pressure and thrust (right y-axis) vs time, while Figure 3.1 (b) illustrates a still frame shot of the thruster under testing. Both figures are representative of Test 1. The test was a 3 second steady state firing. Due to colder than expected temperatures, the propellant pressures were adjusted in an attempt to maintain the test matrix O/F, however, resulted in an O/F value of 1.3. LO\textsubscript{X} line pressure and temperature were maintained at 155 ±1 psia and 205 ±2 °R. LCH\textsubscript{4} line pressure and temperature were maintained at 160 ±1 psia and 221
±2 °R. A maximum thrust value of 1.7 lbf was observed with a maximum chamber pressure of 45.4 psia. Arrows are shown in Figure 3.1 (a) to depict the propellant valves open and close time. It should be noted that the thrust (graphed in red) ends abruptly as it was only filtered up to the end point.

Figure 3.1 (a) Test 1 (O/F ~ 1.3) chamber pressure and thrust data; (b) Test 1 hot fire still frame.

Figure 3.2 (a) and Figure 3.2 (b) summarize the results for Test 2. An average O/F of 1.6 (goal of 1.5) was achieved for the 3 second test. Maximum thrust and chamber pressure of 2.22 lbf and 43.22 psia were observed. Test matrix LOX conditions were attained; however, the LCH₄ pressure was raised to 140 psia due to the warmer methane temperatures observed during condensation.

Figure 3.2 (a) Test 2 (O/F ~ 1.6) chamber pressure and thrust data; (b) Test 2 hot fire still frame.
Test 3 was conducted with an average O/F of 1.9 for the three second duration. LO\textsubscript{X} line pressure and temperature were maintained at 160 ± 1 psia and 207 ± 2 °R. The LCH\textsubscript{4} line pressure and temperature were maintained at 140 ± 1 psia and 246 ± 2 °R. A maximum thrust value of 1.88 lbf was observed with a maximum chamber pressure of 47.8 psia. See Figures 3.3 (a) and (b).

Figure 3.3 (a) Test 3 (O/F ~ 1.9) chamber pressure and thrust data; (b) Test 3 hot fire still frame.

Similarly, Figures 3.4 (a) and (b) summarize the results for Test 4. The test was a 3 second steady state firing, which resulted in an average O/F of 2.2. LO\textsubscript{X} line pressure and temperature were maintained at 165 ± 1 psia and 210 ± 2 °R. The LCH\textsubscript{4} line pressure and temperature were maintained at 118 ± 1 psia and 248 ± 2 °R. Maximum thrust and chamber pressure of 2.64 lbf and 50.81 psia, respectively, were observed.

Figure 3.4 (a) Test 4 (O/F ~ 2.2) chamber pressure and thrust data; (b) Test 4 hot fire still frame.
A much faster chamber pressure initial response was noted on Test 4. The change is thought to be due to the air moisture freezing within the small line orifice (1/8” OD) that connects the pressure transducer to the thruster chamber. This obstructs the line and occasionally causes a much slower response.

Finally, Figures 3.5 (a) and (b) summarize the results for Test 5. The test was a 5 second steady state firing conducted at an average O/F of 2.9. LOX line pressure and temperature were maintained at 180 ±1 psia and 214 ±2 °R. LCH₄ line pressure and temperature were maintained at 160 ±1 psia and 266 ±2 °R. A maximum thrust value of 2.75 lbf was observed with a maximum chamber pressure of 61.59 psia.

Figure 3.5 (a) Test 5 (O/F ~ 2.2) chamber pressure and thrust data; (b) Test 5 hot fire still frame.

Figure 3.5 (a) displays an interesting thrust profile as the thrust rises rapidly then drops before rising again to its steady state operation. This phenomenon might be explained by a slight change in mixture ratio, as this test was conducted after a longer wait period between tests, allowing significant propellant boil off. Table 3.2 summarizes the thrust performance. Theoretical C* values were computed using NASA CEA-RUN software, while the actual values were obtained using equation 4 below [2, 17]. From the equation, P_c is the average chamber pressure, A_th is the throat area, g_c is the gravitational constant and w_p is the propellant flow rate.

\[ C^* = \frac{P_c A_{th} g_c}{w_p} \] (4)
Table 3.2: Pencil thruster test results summary

<table>
<thead>
<tr>
<th>Test</th>
<th>O/F:</th>
<th>Average Thrust (lbf):</th>
<th>Average Pc (psia):</th>
<th>% C*:</th>
<th>Average Isp (s):</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>1.3</td>
<td>1.28</td>
<td>32.6</td>
<td>16.2</td>
<td>16.0</td>
</tr>
<tr>
<td>2</td>
<td>1.6</td>
<td>1.68</td>
<td>30.0</td>
<td>25.4</td>
<td>38.4</td>
</tr>
<tr>
<td>3</td>
<td>1.9</td>
<td>1.46</td>
<td>34.8</td>
<td>17.5</td>
<td>21.6</td>
</tr>
<tr>
<td>4</td>
<td>2.2</td>
<td>2.06</td>
<td>42.9</td>
<td>24.8</td>
<td>35.9</td>
</tr>
<tr>
<td>5</td>
<td>2.9</td>
<td>2.31</td>
<td>44.0</td>
<td>27.6</td>
<td>44.1</td>
</tr>
</tbody>
</table>

3.3 Ignition Reliability

A critical aspect of flight grade RCS thrusters is their ignition reliability. Overall, the thruster upheld a 64% successful ignition rate under liquid-liquid propellant injection conditions. It should be noted that successful ignition was characterized by a sustained steady plume throughout the entire test duration. Table 3.3 below summarizes the number of tests conducted at various O/F ranges, along with their respective ignition success rate.

Table 3.3: Ignition reliability summary

<table>
<thead>
<tr>
<th>O/F Range:</th>
<th>No. of Tests:</th>
<th>Successful Ignition %:</th>
</tr>
</thead>
<tbody>
<tr>
<td>&lt; 1.0</td>
<td>4</td>
<td>50</td>
</tr>
<tr>
<td>1.0 – 1.49</td>
<td>14</td>
<td>64</td>
</tr>
<tr>
<td>1.5 – 1.99</td>
<td>9</td>
<td>78</td>
</tr>
<tr>
<td>2.0 – 2.49</td>
<td>4</td>
<td>50</td>
</tr>
<tr>
<td>2.5 – 3.0</td>
<td>3</td>
<td>67</td>
</tr>
<tr>
<td>&gt; 3.0</td>
<td>2</td>
<td>50</td>
</tr>
</tbody>
</table>

From the table, the majority of the tests were conducted at an O/F between 1 and 2 where, on average, the thruster upheld a 70% successful ignition rate. Overall, there were only 3
tests throughout the entire O/F range which did not ignite at all. The remaining lack of successful trials was characterized by either delayed or fluttering ignition.

3.4 Discussion

The tests data and analysis revealed a significant lack in thruster performance, whose root cause is believed to be a less than adequate camber and cooling design. Moreover, a near 40% of test had either poor ignition or did not ignite at all, further emphasizing the poor mixing within the chamber and/or placement of the igniter. Through both mass balance and simple CFD analysis, the thruster’s fuel injector allots a near 70% film cooling to the chamber walls. Test data reveals average maximum wall temperatures of 825 °R and a maximum observed wall temperature of 905 °R; these temperatures are significantly lower than the maximum allowable working temperate of 1840 °R the SS 316 body material upholds. The propellant’s velocity magnitude contour can be seen in Figure 3.6 below.

![Figure 3.6 Velocity magnitude contour](image)

The CFD analysis was carried out with a 410,000 element mesh. A combination of the k-epsilon viscous model, energy equation, and the species non-premixed combustion model using pure LOX-LCH₄ combustion materials were used. A velocity boundary condition was placed at each individual manifold, thus allowing the continuity equation to distribute the flow across the injection holes. Initial gauge pressure and temperature were specified based on test data at an
O/F of 2.70. The analysis was conducted with a SIMPLE solution scheme using the least squared cell based gradient and first order upwind solution method.

Figure 3.7 Methane mass fraction contour

Looking at the CH\textsubscript{4} mass fraction, the majority of the mixing occurs on the front plane of the igniter electrode, while the spark occurs between the electrode and chamber body. In Figure 3.7, a CH\textsubscript{4} mass fraction of 30% is noted at the center of the electrode tip, however only 10% can be seen at the edge of the electrode. Furthermore, as seen in Figure 3.6 the center of the electrode head acts as a blunt body for the propellant flow and thus creates a point of stagnation, which in turn further degrades the propellant mixing. The combination of these factors might account for a near 40% no-ignition/poor-ignition test trials.

Aside from the chamber and cooling design, another source of underperformance is that the thruster’s nozzle is meant for low-altitude operation. In other words, the A\textsubscript{e}/A\textsubscript{t} value of 7 is over expanded for testing at 13 psia ambient pressure. As a result, thrust performance and consequently Isp value are affected. Using equations 5, 6, and 7 Figure 3.8 graphs the theoretical thrust generated by a thruster with A\textsubscript{th} of 0.059” at P\textsubscript{c} = 13 psia, O/F of 1, 2, and 3 and idealized propellant flow rate.
\[
\frac{A_e}{A_{th}} = \sqrt[\kappa]{k \left( \frac{2}{k+1} \right)^{k+1/k-1}} \left( \frac{P_e}{P_c} \right)^{1/k} \sqrt[\kappa]{\frac{2k}{k-1} \left[ 1 - \left( \frac{P_e}{P_c} \right)^{k-1/k} \right]}
\]

(5)

\[
V_e = \sqrt[\kappa]{\frac{2kRg_c T_c}{k-1} \left[ 1 - \left( \frac{P_e}{P_c} \right)^{k-1/k} \right]}
\]

(6)

\[
F = \frac{\dot{m}}{g_c} V_e + (P_e - P_a) A_e
\]

(7)

From the equations, \( A_e \) represents the exit area, \( k \) is the specific heat ratio of the combustion products, \( R \) is the specific gas constant, \( P_e \) is the exit pressure, \( P_c \) is the chamber pressure, and \( T_c \) is the combustion temperature. All thermodynamic properties (\( k, R, \) and \( T_c \)) were obtained using NASA CEA-RUN software at their respective O/F and at a \( P_c \) of 50 psia. From the figure, both O/F of 2 and 3 carry an optimum \( A_e/A_{th} \) of 3.12, while the O/F of 1 carries an optimum \( A_e/A_{th} \) of 2.57. Nevertheless, poorest performance is seen at an \( A_e/A_{th} \) of 7 for all three mixture ratios.
Figure 3.8 Theoretical thrust generated at Pa = 13 psia and various expansion ratios
Chapter 4: Next Steps

Despite the significant lack of performance, the pencil thruster still carries certain design concepts that are worth investigating further. Moreover, through further understanding behind the LOX-LCH\textsubscript{4} combustion physics, both through theoretical calculations and experimental data, several design improvements can be made for the next iteration design of the RCS thruster. The following chapter outlines the design criteria, methodology and current calculations and analysis for the next generation thruster.

4.1 Design Criteria

In order to fully understand the LOX-LCH\textsubscript{4} propellant combination within an RCS thruster, a new thruster with no vehicle attachment will be developed. In other words, previous testing specifications geared toward the thruster use within the Morpheus vehicle no longer constrain the design and testing. Noting the test facility capabilities, the following design criterion was developed. From Table 4.1, the chamber pressure design requirement of 100 psia formed the baseline for combustion calculations and thruster design.

Table 4.1: Design Criteria

<table>
<thead>
<tr>
<th>Design Parameter:</th>
<th>Value:</th>
<th>Notes:</th>
</tr>
</thead>
<tbody>
<tr>
<td>Oxidizer:</td>
<td>Liquid Oxygen</td>
<td>Provided by self-pressurizing dewar</td>
</tr>
<tr>
<td>Fuel:</td>
<td>Liquid Methane</td>
<td>Provided by Cryocart</td>
</tr>
<tr>
<td>Max Propellant Line Pressure (psia):</td>
<td>230</td>
<td>System limited to 230 psia</td>
</tr>
<tr>
<td>Thrust Range (lbf):</td>
<td>8 ±1</td>
<td>Limited by TTB to 15 lbf</td>
</tr>
<tr>
<td>Chamber Pressure (psia):</td>
<td>100 ± 5</td>
<td>Design chamber pressure that will guide the nozzle chamber, and injector design</td>
</tr>
<tr>
<td>Exit Pressure (psia):</td>
<td>5 - 13</td>
<td>Initial nozzle will be designed for ambient pressure; changes will be made to implement expansion ratio suited for lower pressures</td>
</tr>
</tbody>
</table>
### 4.2 Mixture Ratio Analysis

A maximum specific impulse analysis at varying O/F was conducted for the 100 psia chamber pressure design criteria. Equation 8 along with NASA CEA-RUN software were used for the analysis [2].

\[
I_{sp} = \frac{2kRT_c}{\sqrt{g_c(k-1)}} \left[ 1 - \left( \frac{P_{atm}}{P_c} \right)^{\frac{k-1}{k}} \right]^{(a)}
\]

From Equation 8, \( k \) is the specific heat ratio for the combustion products, \( R \) is the specific gas constant, \( T_c \) is the combustion temperature, \( g_c \) is the gravitational constant, \( P_c \) is the chamber pressure (100 psia), and \( P_{atm} \) is the atmospheric pressure (13 psia was used for the analysis, as \( I_{sp} \) will only improve as \( P_{atm} \) is decreased.). All thermodynamic properties (\( k, R, \) and \( T_c \)) were acquired using NASA CEA-RUN.

![Figure 4.1](image)

Figure 4.1 (a) Maximum theoretical \( I_{sp} \) at O/F range 1.0 – 4.0; (b) Maximum theoretical \( I_{sp} \) at O/F range 2.5 – 3.0.

In Figure 4.1 (a) maximum \( I_{sp} \) is observed at an O/F range between the 2.5 and 3.0. By refining the analysis, Figure 4.2 (b) shows maximum \( I_{sp} \) of 222.9 seconds at an O/F of 2.75. It is important to note that NASA CEA-RUN thermodynamic values are calculated under equilibrium and complete combustion assumptions, giving the upper limit to actual thruster performance.

Having the ideal mixture ratio, along with the respective maximum possible \( I_{sp} \), total and individual propellant flow rates were determined from Equation 1. Furthermore, considering the
propellant delivery system as well as previously observed test temperatures, required propellant conditions were also determined. See Table 4.2 below.

Table 4.2: Propellant Conditions for O/F = 2.75

<table>
<thead>
<tr>
<th></th>
<th>LO\textsubscript{X}:</th>
<th>LCH\textsubscript{4}:</th>
</tr>
</thead>
<tbody>
<tr>
<td>Flow rate (lb/s):</td>
<td>0.0266</td>
<td>0.0097</td>
</tr>
<tr>
<td>Line Pressure (psia):</td>
<td>210</td>
<td>210</td>
</tr>
<tr>
<td>Injection Pressure (psia):</td>
<td>140</td>
<td>140</td>
</tr>
<tr>
<td>Line Temperature (°R):</td>
<td>212</td>
<td>249</td>
</tr>
</tbody>
</table>

4.3 Thruster Redesign

The following section discusses the analysis behind the proposed changes to the pencil thruster. This includes changes to the injector, in particular the film cooling, as well as the combustion chamber and nozzle. It should be noted that although each thruster component is discussed in separate subsections, the actual design process is highly integrated as the design of one part affects subsequent components.

4.3.1 Injector

The current pencil thruster injector maintains simplicity within a compact design. As a result, the radial injection concept will be maintained however with a few modifications to improve atomization and propellant mixing.

In an attempt to optimize the combustion process, a droplet size analysis was conducted using a numerical application of the Sauter Mean Diameter (SMD) method. The SMD, which determines the mean droplet diameter with same ratio of volume to surface area as the entire spray, is often the preferred method for liquid-fuel combustion [12]. According to Lefebvre et al, the atomization mechanism in a high velocity stream, such as in the pressure-jet atomizers used in the pencil thruster, is primarily governed by two forces: aerodynamic forces from the relative velocity between the sprayed liquid and gaseous medium, and hydrodynamic forces caused by
turbulence within the liquid itself. The overall interaction of the forces can be quantified by Equation 9 below [16].

\[
SMD = A\left[\frac{\sigma^{0.5} \mu_L^{0.5}}{\rho_A^{0.5} \Delta P_L} \right]^{0.5} [t \cos \theta]^{0.25} + B\left[\frac{\sigma \rho_F}{\rho_A \Delta P_L} \right]^{0.25} [t \cos \theta]^{0.75}
\] (9)

From Equation 9, the first term represents the turbulence within the liquid sheet, as well as the growth rate of the disturbances within the liquid which ultimately break off to form ligaments. The second term represents the second stage of the atomization process, where the relative velocity between the injected fuel to the surrounding medium causes further breakup of the ligaments into droplets. From the equation, \(\sigma\) is the LCH\(_4\) surface tension, \(\mu_L\) is the LCH\(_4\) viscosity, \(\rho_A\) is the density of the medium into which the fuel is injected (an average density of the combustion gases was used), \(\rho_F\) is the LCH\(_4\) density, \(\Delta P_L\) is the pressure drop across the injector, \(\theta\) is the spray half angle, and \(t\) is the film thickness which is given by Equation 10 below [16].

\[
t = 2.7 \left( \frac{d_o FN \mu_L}{\sqrt{\Delta P_L \rho_F}} \right)^{0.2}
\] (10)

From Equation 10, \(d_o\) represents the orifice diameter while FN is the nozzle flow number, defined by Equation 11 [16].

\[
FN = \frac{\dot{m}_L}{\sqrt{\Delta P_L \rho_F}}
\] (11)

Moreover, constants A and B from Equation 9 are experimentally derived parameters and are given by Equations 12 and 13 below [16].

\[
A = 2.11 [\cos 2(\theta - 30)]^{2.25} \left( \frac{3.4 \times 10^{-4}}{d_o} \right)^{0.4}
\] (12)
\[ B = 0.635[\cos 2(\theta - 30)]^{2.25}(\frac{3.4 \times 10^{-4}}{d_o})^{0.2} \quad (13) \]

Having a way to calculate the droplet diameter, the propellant vaporization time can be determined using Equation 14 below [12].

\[ t_e = \frac{\rho_F D_o^2}{8(k_{cp})_g \ln(1 + B)} \quad (14) \]

From the equation, \( \rho_F \) is the LCH\(_4\) density, \( D_o \) is the droplet diameter, \( k_g \) is the gaseous oxygen thermal conductivity, \( C_{pg} \) is the gaseous oxygen average specific heat at constant pressure, and \( B \) denotes the ratio of enthalpy in the surrounding gas to the heat required to evaporate the fuel, and is quantified by Equation 10 below [12].

\[ B = \frac{c_{pg}(T_f - T_b)}{L} \quad (15) \]

From Equation 10, \( T_f \) represents the flame temperature, \( T_b \) is the boiling temperature of the fuel, and \( L \) is the latent heat of vaporization of the fuel. All properties were gathered using both RefPROP and NASA CEA-RUN software.

Equation 16 was used to determine the injection area as well as the number (\( n \)) and diameter of the orifices [9].

\[ A = \dot{m} \sqrt{\frac{2.238K}{\gamma \Delta P_L}} = \frac{\pi}{4} n d_o \quad (16) \]

From Equation 16, \( \dot{m} \) is the propellant flow rate, \( \gamma \) is the propellant specific weight, \( \Delta P_L \) is the pressure drop across the injector, and \( K \) accounts for minor losses. From literature, a typical \( K \) value of 1.7 was used [9].

Noting that Equations 9-13 are primarily a function of the spray half angle (\( \theta \)) and the pressure drop across the injector (\( \Delta P_L \)), which are both sensitive to the number and diameter of the injection orifices, an iterative process between Equations 9-16 was implemented to determine a system-feasible injector design that would minimize the evaporation time. The analysis
concluded that with a $\Delta P_L$ of 40 psi, nine LCH$_4$ and eight LO$_X$ injection orifices of 0.018” and 0.014” diameter, respectively, would uphold an SMD of 0.0017” and 0.33 ms evaporation time.

4.3.2 Combustion Chamber

Combustion efficiency is a direct function of atomization and consequent propellant droplet size [9,]. Moreover, for thruster applications, the combustion efficiency effect is further emphasized by the allowed residence time within the combustion chamber. As previously stated, an SMD approach was used to determine the droplet size and residence time. Using the calculated time of 0.33 ms, Equation 17 below was used to determine a 0.636 in$^3$ chamber volume.

$$V_c = \frac{\dot{m}t_e}{\rho_A}$$ (17)

From Equation 17, $\dot{m}$ is the propellant flow rate, $t_e$ is the residence time, and $\rho_A$ is the density of the combustion products [9]. By considering a cylindrical geometry, a chamber diameter and length of 0.60 and 2.05 inches, respectively, was found to maintain film cooling within a reasonable range (see Section 4.3.3 below), as well as maintain a compact design.

4.3.3 Film Cooling

Considering a major contributor to the pencil thruster’s poor performance was excessive film cooling, an analytical evaluation was conducted using Equations 18-20 below. All properties were gathered using both RefPROP and NASA CEA-RUN software. A minimum 27% fuel film cooling (0.0026 lb/s) was determined at an assumed 70% film cooling efficiency.

$$H = \frac{c_{pvc}(T_{aw} - T_{wg})}{c_{plc}(T_{wg} - T_{co}) + h_{vap}}$$ (18)

$$G_g = \frac{\dot{P}_{c}A_{th}k g_c}{A_c \sqrt{k g_c R T_c}} \sqrt{\frac{2}{k + 1}} \left(\frac{k + 1}{k - 1}\right)^{\frac{k+1}{k-1}}$$ (19)
\[ G_c = \frac{G_g}{\eta_c} \frac{H}{a \left( 1 + \frac{c_{pvc}}{c_{pcl}} \right)} \]  

(20)

From Equation 18, \( C_{pvc} \) is the vapor methane specific heat at constant pressure, \( T_{aw} \) is the adiabatic wall temperature, \( T_{wg} \) is the maximum allowed wall temperature, \( C_{pcl} \) is the liquid methane specific heat at constant pressure, \( T_{co} \) is the methane initial liquid temperature, and \( h_{vap} \) is the methane heat of vaporization [9].

Equation 19 represents the combustion gas flow rate per unit area of chamber cross-section perpendicular to flow. \( A_c \) is the chamber cross-sectional area, \( P_c \) is the chamber pressure, \( A_{th} \) is the nozzle throat area, \( k \) is the specific heat ratio for the combustion products, \( R \) is the specific gas constant, \( T_c \) is the combustion temperature, and \( g_c \) is the gravitational constant [2].

Finally, Equation 20 represents the liquid film cooling per unit area. From the equation, \( \eta_c \) represents the film cooling efficiency (70% assumption) and parameters \( a \) and \( b \) represent ratios of axial free stream velocity and were obtained from literature [9].

The above calculations carry several assumptions, and are highly dependent on actual combustion temperatures, gas exit velocities, methane inlet conditions, and chamber and injection design, to name a few. Moreover, actual film cooling results in two phase flow, which is inherently difficult to analyze. As a result, a 30% film cooling will be used for design purposes and will be a center point of testing for future work.

**4.3.4 Nozzle Design**

Nozzle design is strictly dependent on the desired operating altitude and consequent ambient pressure. From thermodynamics, the nozzle throat was determined by Equation 21 below.

\[ A_{th} = \frac{F}{P_c C_f} \]  

(21)

From Equation 21, \( F \) represents the thrust, \( P_c \) is the chamber pressure, and \( C_f \) is the thrust coefficient given by Equation 22 below [2].
\[ C_f = \sqrt[2]{\frac{2k^2}{k-1}} \left( \frac{2}{k+1} \right)^{k+1} \left[ 1 - \left( \frac{P_e}{P_c} \right)^{k-1} \right] + \left( \frac{P_e - P_a}{P_c} \right) \frac{A_e}{A_{th}} \] \tag{22}

Setting the design chamber pressure of 100 psia and thermodynamic values determined from NASA CEA-RUN, Equations 21, 22, and 5 were used to determine a throat area of 0.066 in\(^2\) with an expansion ratio of 1.88. Considering the test facility’s capabilities to pull vacuum, several nozzles will be designed with the future intent of testing at altitude. Table 4.3 illustrates the required nozzle parameters for the given altitude of operation.

### Table 4.3: Nozzle design parameters for operation at various altitudes.

<table>
<thead>
<tr>
<th>Approximate Altitude (ft):</th>
<th>( P_a ) (psia):</th>
<th>( A_{th} ) (in(^2)):</th>
<th>( A_e/A_{th} )</th>
<th>( C_f )</th>
</tr>
</thead>
<tbody>
<tr>
<td>3,350</td>
<td>13</td>
<td>0.066</td>
<td>1.88</td>
<td>1.21</td>
</tr>
<tr>
<td>7,800</td>
<td>11</td>
<td>0.064</td>
<td>2.09</td>
<td>1.25</td>
</tr>
<tr>
<td>12,950</td>
<td>9</td>
<td>0.062</td>
<td>2.38</td>
<td>1.29</td>
</tr>
<tr>
<td>19,125</td>
<td>7</td>
<td>0.060</td>
<td>2.81</td>
<td>1.34</td>
</tr>
<tr>
<td>26,950</td>
<td>5</td>
<td>0.057</td>
<td>3.54</td>
<td>1.40</td>
</tr>
</tbody>
</table>

### 4.4 Future Work

Having the proposed thruster dimensions, the next steps will model the thruster injection, mixing, and cooling performance through CFD analysis. This will allow validation of the assumed propellant conditions and further optimize injection and mixing parameters.
Chapter 5: Conclusion

In conclusion, a thruster performance evaluation was carried out for the LO\textsubscript{X}-LCH\textsubscript{4} thruster at O/F values ranging from 1 to 3. Significant lack of performance was observed, which was primarily deemed to the excessive film cooling and further emphasized by the nozzle overexpansion. Moreover, CFD studies have shown that igniter placement may be the source of poor ignition reliability.

Test data and analysis will continue to drive the next generation of thruster design, which is considering both the excessive cooling and propellant mixing issues encountered with the current pencil thruster. Altitude testing will also allow testing of specific nozzle performance at ideal and varying exit pressures. Despite the observed lack of performance, the LO\textsubscript{X}-LCH\textsubscript{4} propellant combination still offers promise for the next generation space exploration and will be the source of future thruster studies.
References


Appendix A

A.1 Role Assignment:

Before any test is conducted, all team members should have a clear understanding of their respective duties during testing operations. For safety purposes, testing requires a minimum of 4 members. The following list outlines the suggested roles and obligations for a four-person team.

- **Test Supervisor** – The Test Supervisor’s role is to ensure everything required for testing, such as gases and equipment has been procured in a timely manner. He/she also ensures that tests are conducted in a safe manner by having all team members wear PPE and ensuring all test procedures are strictly followed. The Test Supervisor has the authority to abort/cancel the test at any time if deemed unsafe.

- **Test Conductor** – The Test Conductor operates the test setup controls through the LabVIEW interface. This includes actuating valves during system check and monitoring instrumentation during laser calibration, condensation, and testing.

- **Test Technician** – The Test Technician is responsible for the manual labor during pre-test operations (calibration, condensation, system check, etc). He/she is also responsible of changing propellant pressures during testing.

- **Test Assistant** – The Test Assistant is a multi-disciplinary role, who assists any of the other three members with any of the testing operations. This includes helping the Test Technician with pre-test operations, acting as a secondary safety manager during testing, and helping the test conductor monitor the testing within the LabVIEW interface.
### A.2 Hazards Analysis:

<table>
<thead>
<tr>
<th>HA Index</th>
<th>Hazard Analysis Descriptions of Fields:</th>
<th>Severity</th>
<th>Likelihood</th>
<th>HA Index</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>System: Description of the system where the hazard may reside.</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>Hazard: Description of the specific hazard that could occur.</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>Severity: measure of the severity of the hazard as listed below (in Excel, menu options exist):</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>1 - Minor</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>2 - Moderate</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>3 - Significant</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>4 - Catastrophic</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>Likelihood: measure of the probability or likelihood of occurrence according to the below listed options (in Excel, menu options exist):</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>1 - Unlikely</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>2 - Infrequent</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>3 - Frequent</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>4 - Imminent</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>HA Index: measure of the degree of consideration needed as follows (calculated automatically In Excel):</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>1 - Minimal Risk: Proceed with Mitigation in mind</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>2 - Moderate Risk: Proceed with Mitigation in mind</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>3 - High Risk: Proceed with PI Concurrence and Mitigation in mind</td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>4 - Severe Risk: Proceed only with PI Approval and Review of all means of mitigation</td>
<td></td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

#### Hazards Analysis for Pencil Thruster

<table>
<thead>
<tr>
<th>#</th>
<th>System</th>
<th>Hazard Description</th>
<th>Severity</th>
<th>Likelihood</th>
<th>Mitigation</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Propellant Lines</td>
<td>Pressure build-up</td>
<td>2 - Moderate</td>
<td>1 - Unlikely</td>
<td>Install relief valves in adequate sections of line, facing safe location</td>
</tr>
<tr>
<td>2</td>
<td>Propellant Lines</td>
<td>LN2 spill</td>
<td>2 - Moderate</td>
<td>2 - Infrequent</td>
<td>Proper ventilation, containment areas, usage of appropriate PPE (gloves, glasses, apron, mask)</td>
</tr>
<tr>
<td>3</td>
<td>Propellant Lines</td>
<td>LOX spill</td>
<td>2 - Moderate</td>
<td>2 - Infrequent</td>
<td>Proper ventilation, containment areas, usage of appropriate PPE (gloves, glasses, apron, mask)</td>
</tr>
<tr>
<td>4</td>
<td>Propellant Lines</td>
<td>LOX spill</td>
<td>2 - Moderate</td>
<td>1 - Unlikely</td>
<td>Proper ventilation, containment areas, usage of appropriate PPE (gloves, glasses, apron, mask)</td>
</tr>
<tr>
<td>5</td>
<td>Propellant Lines</td>
<td>LOX combustion</td>
<td>3 - Significant</td>
<td>1 - Unlikely</td>
<td>Proper maintenance of delivery system (cleaning, filtering), isolation of propellant, Remote operation of LOX system and personnel clearance from affected area</td>
</tr>
<tr>
<td>6</td>
<td>Propellant Lines</td>
<td>Explosion</td>
<td>4 - Catastrophic</td>
<td>1 - Unlikely</td>
<td>Proper ventilation system, close propellant valves, run purge</td>
</tr>
<tr>
<td>7</td>
<td>Propellant Lines</td>
<td>Fire</td>
<td>3 - Significant</td>
<td>1 - Unlikely</td>
<td>Usage of propellant, suppression system on-hand at time of experimentation, stop experimental sequence, emergency stop if necessary</td>
</tr>
<tr>
<td>8</td>
<td>Propellant Lines</td>
<td>Valve Failure</td>
<td>2 - Moderate</td>
<td>1 - Unlikely</td>
<td>Stop experimental procedures, check valve connections, troubleshoot</td>
</tr>
<tr>
<td>9</td>
<td>Control</td>
<td>LaVTFM failure</td>
<td>3 - Significant</td>
<td>2 - Infrequent</td>
<td>Press Emergency Stop button</td>
</tr>
<tr>
<td>10</td>
<td>Power Lines</td>
<td>Short-circuit</td>
<td>2 - Moderate</td>
<td>2 - Infrequent</td>
<td>Stop experimental procedures, check electrical connections</td>
</tr>
<tr>
<td>11</td>
<td>Cryogenics</td>
<td>Skin burn</td>
<td>2 - Moderate</td>
<td>1 - Unlikely</td>
<td>Usage of appropriate PPE and remote operation of system</td>
</tr>
<tr>
<td>12</td>
<td>Ignition</td>
<td>Ignition Failure</td>
<td>3 - Significant</td>
<td>2 - Infrequent</td>
<td>Stop propellant delivery, turn ignition system off, run purge on system</td>
</tr>
<tr>
<td>13</td>
<td>Power</td>
<td>Power outage</td>
<td>2 - Moderate</td>
<td>1 - Unlikely</td>
<td>Isolate test area, stop experimental procedures</td>
</tr>
<tr>
<td>14</td>
<td>Propellant Lines</td>
<td>Gaseous CH4 leak</td>
<td>2 - Moderate</td>
<td>1 - Unlikely</td>
<td>Use of methane detector on test area, stop experimental sequence, troubleshoot</td>
</tr>
<tr>
<td>15</td>
<td>Propellant Lines</td>
<td>Oxygen leak</td>
<td>2 - Moderate</td>
<td>2 - Infrequent</td>
<td>Monitor oxygen level in test area, stop experimental procedure, troubleshoot, Bunker back door open for proper ventilation</td>
</tr>
</tbody>
</table>
A.3 Methane Condensation:

The following section outlines the methane condensation procedure. Test technician handling the Cryocart valves must wear proper protective equipment (cryo-gloves, cryo- vest, and face shield). The use of a walkie-talkie set will also be required to communicate with the test conductor/assistant.

1. Install the Cryocart to LCH4 line interface (See Figure A.1)

![Interface Connection (1/4” Swagelok Nut)](image)

**Figure A.1 Cryocart to LCH4 line interface**

2. Connect all valve wires and pressure transducer feeds to their labeled channel on panel (All wires are labeled as shown on panel).

3. Connect E-type Thermocouples (TC 1-6) to their labeled probe. (See Figure A.2)
4. Verify all instrumentation reads ambient conditions (13 psia ±1 psia and 22 °C ±1 °C).

5. Verify all valves on the Cryocart valves are working. This is done by the Test Conductor opening the valves one by one and the Test Technician confirming a successful opening. The Conductor will indicate which valve is being tested and the Technician will verify that it opens by audio/visual confirmation (i.e. touching the valve and feeling it open).

6. Close hand valves to the methane and helium tanks (See Figures A.3 and A.4)
7. Attach the LN$_2$ flex hose to Cryocart (See Figure A.3)

8. Attach the vacuum pump to the system and begin pulling vacuum in the lines and condensation tank (See Figure A.3). Vacuum is achieved when the pressure transducer reading the condensation tank pressure reads 3 +/- 1 psia.

9. Methane and Helium tanks are opened and regulators are set to the correct pressure. Methane is set to 60 psia and helium pressure is set according to the test session conducted from the test matrix.
10. Set the liquid nitrogen tank to 200 psia and open the CH$_4$ tank regulator and cooling valves to begin cooling the methane condensing tank.

11. Open the hand valve to the methane tank and allow methane to enter the previously vacated condensing tank.

12. Ensure that the condensation tank pressure transducer is reading between 65 and 75 psia and maintain that pressure until the desired amount of methane is condensed.

13. Close the methane hand valve when the desired level is reached. Methane level can be inferred from the temperature readings of the thermocouples attached to the condensation tank (~ -140 °C +/- 2 °C indicates liquid at that level).

14. Once the desired level is achieved, open the helium hand valve. The helium tank pressure can be changed in between test runs to provide various methane line pressures.

15. Close the liquid nitrogen cooling valves (LN$_2$ tank valve remains open for cooling purposes during testing)
A.4 Laser Calibration:

A calibration process must be conducted prior to each test date in order to get an accurate representation of thrust through the laser displacement measurement. This is an intricate process that requires the use of a 3.3 ft (1 m) of 18 gauge stainless steel wire, a basket capable of holding up to 8.8 lb (4 kg), a calibrated weight set, and a pulley stand.

1. The steel wire is looped around the back end of the thruster and then passed through the pulley stand. The other end of the wire is attached to the basket as shown in Figure A.5. For accurate calibration, the steel wire must clear all instrumentation wires and propellant feed lines. Similarly, the hanging weight basket must be free of obstructions. The wire should be perpendicular to the thrust stand (i.e. same axis as the thruster).

![Diagram of laser calibration setup](image)

Figure A.5 TTB laser calibration representation

2. *Test Technician* should load the lines up to the thruster valves with GN₂ at test pressures. This takes into account the added stiffness in the flexible feed lines the thruster would experience under normal operation.

3. *Conductor* zeroes the laser reading when noting is hanging from the pulley, including the basket. The conductor then instructs the *Test Technician* to let the basket hang.

4. The mean displacement is recorded.

5. The basket is loaded with 0.88 lb (400 g) and the mean displacement is taken once again.
6. Step 5 is repeated at every 0.88 lb (400g) increments until 8.8 lb (4 kg) are reached.

7. The process is then reversed in 0.88 lb (400 g) decrements until nothing is hanging from the pulley.

8. Repeat the entire process two more times for a total of six calibration curves; an average calibration curve is then determined and used for displacement to thrust conversion.

A.5 Testing:

The following section includes the current testing procedure used, as well as considerations that have been found useful during testing operations. Laser calibration and methane condensation should have already been completed.

1. Test Supervisor must conduct a secondary check to ensure all pre-test action items have been completed. Furthermore, a secondary safety check is conducted. This includes checking that the bunker light is switched to yellow, Test Technician has placed the safety gates outside the bunker, and fan is set to ventilate gases out of the bunker.

2. Test Technician should check that the igniter battery is connected, Helium ¼ turn valve within Cryocart is opened, and Helium regulator is set to proper test pressure.

3. Test Technician informs the Test Conductor he/she is ready to exit bunker and act as access control.

4. Test Assistant closes all bunker doors; Test Supervisor must switch bunker light to red.

5. Test Supervisor must confirm all members are set in their respective positions. Test Supervisor asks Test Technician (through walkie-talkie) if the area outside the bunker is clear of any pedestrians. If a green light is given by the Test Technician, the Test Supervisor gives the final “Go/No-Go” to the Test Conductor.
6. If “Go” signal given by Test Supervisor, Test Conductor switches igniter manual switch to on, switches LabVIEW to automated control, and runs the desired test sequence.

7. During automated sequence operation:
   a. Test Conductor presses record in LabVIEW and on SONY Go camera. Secondly, Test Conductor keeps an eye on the live camera feed. If ignition is not achieved, the manual switch should be turned off to avoid ignition outside of the thruster.
   b. Test Assistant takes notes of testing operations, including testing time, expected inlet conditions, if ignition was achieved, and anything else that might be helpful during post test data processing.
   c. Test Technician stands as access control to the outside bunker doors. Test Technician should stand guard clear of the bunker doors. Test Technician should wait for further instructions by Test Supervisor.
   d. Test Supervisor stands by the Emergency button ready to push it in the event of an explosion, fire, or any other unexpected firing. Test Supervisor informs the Test Technician when the individual test sequences have been completed.

8. After the automated sequence has completed:
   a. Test Conductor stops the SONY Go camera recording. Test Conductor properly stops the laser and data recording before completely stopping the LabVIEW program. He/she then informs the Test Supervisor of system conditions, including LCH₄ level, temperatures, pressures, etc. Furthermore, he/she runs the data monitoring program and informs the Test Supervisor of chamber pressure and propellant conditions.
   b. Test Assistant makes any changes to LOX pressure (open/close pressure build) dependent on Test Supervisor discretion. Test Assistant ensures both bunker doors are closed if bunker re-entry was necessary.
c. Test Technician must wait for Test Supervisor notice to enter bunker for Helium pressure modification.

d. Test Supervisor decides if and when any pressure changes must be made. If so, the bunker should be purged with GN2 for 30 seconds and allowed to ventilate for at least 2 minutes before entering. Test Supervisor must change bunker light from red to yellow before allowing Test Assistant and/or Test Technician into the bunker; light is switched back to red once the bunker has been cleared.

9. Steps 5-8 are repeated for the remainder of test matrix.
Appendix B

B.1 120V AC Valve Electrical Schematics:

This section discusses the electrical configuration of the delivery system valves and how it was developed to incorporate automated control and emergency shutdown operations. The system operates through the use of three electromechanical relays, six solid state relays and nine PCI relays. See Figure B.1 below.

Under normal operation, 120 V AC are supplied by enabling the system either through LabVIEW or manual control. The normally open Emergency Stop Relay then feeds the Enable DAQ Relay solenoid which closes the circuit. As a result, 120 V AC are supplied to solid state relays SSR 1 and SSR 2, closing their respective circuit and enabling normal operation. See Figure B.2 (a) below.
Figure B.2 Delivery valves electrical schematic normal operation: (a) Enable DAQ electromechanical relay closed and (b) solid state relays SSR1 and SSR2 closed

Figure B.2 (b) forms the continuation from B.2 (a). Depending on manual or automated control (changed through patch panel switch), each valve can be actuated through their respective manual switch on the patch panel or through LabVIEW virtual switch, which actuates the PCI relay.

As a safety measure, an emergency shut off button is located within the patch panel of the control room. When pressed, the Emergency Stop Relay circuit is closed, consequently removing current to the Enable DAQ Relay. As a result, SSR 1 and SSR 2 are opened, closing all valves enabled by these solid state relays. Moreover, the now enabled Emergency Stop Relay provides 120 V AC to T Delay Relay, closing its circuit and enabling SSR 3 – SSR 6. See Figure B.3 (a) and (b) below.
Figure B.3 Delivery valves electrical schematic emergency operation: (a) Emergency Stop electromechanical relay closed and (b) solid state relays SSR3 – SSR6 closed

The SSR’s are connected as to bypass both the manual switch and PCI relays; as a result, $CH_4$ Purge, $CH_4$ Delivery, $LO_X$ Purge and $LO_X$ Delivery valves are opened. It should be noted the $T$ Delay Relay has been programmed so as to re-open after 10 seconds of operation, thus closing the aforementioned valves after a 10 second purge.

B.1 12V DC Valve Electrical Schematics:

Propellant delivery is achieved through the actuation of the normally closed 12 V DC solenoid valves. $LO_X$ compatible GEMS Sensors & Controls cryogenic solenoid valves were selected for the purpose. These valves require a minimum of 9 V DC to actuate while pressurized to 200 psi, but are normally operated at 12 V DC. Under normal operating conditions (150 psi and -150 °F) the valves have an opening and closing time of 15.6 ms and 7.6 ms respectively [1]. The valves are powered through the use of EXTECH 382270 Power Supply. An electrical schematic of the valves is included below.
B.2 Igniter Electrical Schematic:

The ignition system functions through the integrated operation of a B&K Precision 4012A 5 MHz function generator, an MSD 8247 ignition coil, a Champion Spark plug RJ12C (which matches the 14 mm with 0.375” thread requirement), and 12 V lawn mower battery. Figure B.3 below illustrates the ignition system electrical schematic.
From the figure, the function generator sends a 5 V<sub>pp</sub> TTL square wave to the MSD coil at a frequency of 100 Hz by closing both the manual switch and PCI relay. Note that the PCI relay is closed through the LabVIEW interface. The igniter coil also requires a 12 V DC excitation voltage at 11 A current, which is supplied by the lawn mower battery. This is transformed to approximately 40,000 V, creating the arc between the electrode tip and the thruster chamber body. As a result, the thruster, and consequent interfacing conductive materials, must be electrically grounded.

A PCI relay controls the signal generator output and is actuated through LabVIEW software. This is accomplished through the use of a NI CB-37FH connector block and NI PCI-6521 card. As a result, the igniter can be actuated either manually or through the automated control through the LabVIEW interface. As a secondary safety measure, a redundant manual switch was introduced to the system.
Vita

Jose Luis Mena attained his Bachelors of Science degree in Mechanical Engineering from the University of Texas at El Paso in the spring of 2012. He then continued to pursue a Master’s of Science in Mechanical Engineering, where he completed his graduate studies at the center for Space Technology Research. During this time, he completed an internship at NASA Johnson Space Center working under project Morpheus. His graduate research focus has primarily dealt with the liquid oxygen-liquid methane propellant combination. The research has included the propellant flow rate control into thruster systems, thruster evaluation, and thruster design.

Upon graduation, Jose Luis will continue his studies at the University of Maryland where he will pursue a Ph.D. in Mechanical Engineering with a focus in propulsion. His doctoral research will pertain to the design and testing of an Ammonium Dinitramide rocket engine for missile applications.

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This thesis was typed by Jose Luis Mena.