Design And Fabrication Of A Full-featured Labscale Hybrid Rocket Engine

2006

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DESIGN AND FABRICATION
OF A FULL-FEATURED
LABSCALE HYBRID ROCKET ENGINE

by

KYLE I. PLATT
B.S University of Central Florida, 2002

A thesis submitted in partial fulfillment of the requirements
for the degree of Master of Science
in the Department of Mechanical, Materials and Aerospace Engineering
in the College of Engineering and Computer Science
at the University of Central Florida
Orlando, Florida

Spring Term
2006
ABSTRACT

The design, development, integration and testing of a full-featured, Lab-Scale Hybrid Rocket Engine was not only envisioned to be the chosen method of putting student payloads into space, but to be an invaluable teaching resource. The subject of the present thesis is the analysis, design, development, integration and demonstration of a lab-scale hybrid rocket motor. The overarching goal of this project was to establish a working developmental lab model from which further research can be accomplished. The lab model was specifically designed to use a fuel source that could be studied in normal laboratory conditions. As such, the rocket engine was designed to use Hydroxyl Terminated Polybutadiene as the fuel and Liquid Nitrous Oxide as the oxidizer.

Developing the rocket engine required the usage of several electronics modules and a software package. The custom-designed electronics modules were a Signal Conditioning & Data Amplification Interface and a Data Acquisition Network. The software package was coded in Visual Basic (VB).

A MathCAD regression rate computer model was designed and written to geometrically constrain the engine design. Further, the computer model allowed for the "what-if" situations to be evaluated. Using ProPep, solutions to the Equilibrium Thermodynamics Equations for the fuel and oxidizer mixture were obtained. The resultants were used as initial input to the computer model for predicting the lab-scale rocket's Chamber Pressure, Chamber Temperature, Ratio of Specific Heats and Molecular Weight. Details on the model, the rocket hardware, and the successful test firing are provided.
For all of her technical, moral, and financial support, I dedicate this thesis, and the rest of my life, to Samara Warman.
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Mr. Matthew Stephens
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## NOMENCLATURE

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Definition</th>
</tr>
</thead>
<tbody>
<tr>
<td>HTPB</td>
<td>Hydroxyl Terminated Polybutadiene</td>
</tr>
<tr>
<td>$N_2O$</td>
<td>Nitrous Oxide</td>
</tr>
<tr>
<td>$P_a$</td>
<td>Ambient Pressure</td>
</tr>
<tr>
<td>$P_c$</td>
<td>Chamber Pressure</td>
</tr>
<tr>
<td>$P_e$</td>
<td>Exit Plane Pressure</td>
</tr>
<tr>
<td>$F_t$</td>
<td>Thrust Force</td>
</tr>
<tr>
<td>$\rho_{htpb}$</td>
<td>Density of Hydroxyl Terminated Polybutadiene</td>
</tr>
<tr>
<td>$\rho_{fuel}$</td>
<td>Density of Nitrous Oxide / HTPB fuel mixture</td>
</tr>
<tr>
<td>N</td>
<td>Number of ports drilled through fuel grain</td>
</tr>
<tr>
<td>$R_i$</td>
<td>Initial Combustion Port Radius</td>
</tr>
<tr>
<td>$R_f$</td>
<td>Final Combustion Port Radius</td>
</tr>
<tr>
<td>$OF_i$</td>
<td>Desired Initial Oxidizer / Fuel Ratio</td>
</tr>
<tr>
<td>$\theta$</td>
<td>Nozzle Half Angle</td>
</tr>
<tr>
<td>n</td>
<td>Regression Rate Exponent</td>
</tr>
<tr>
<td>a</td>
<td>Regression Rate Coefficient</td>
</tr>
<tr>
<td>$\eta_e$</td>
<td>Combustion Efficiency Coefficient</td>
</tr>
<tr>
<td>$D_p$</td>
<td>Combustion Port Burn Diameter</td>
</tr>
<tr>
<td>$C_f$</td>
<td>Thrust Coefficient</td>
</tr>
<tr>
<td>$A_t$</td>
<td>Nozzle Throat Area</td>
</tr>
<tr>
<td>Symbol</td>
<td>Description</td>
</tr>
<tr>
<td>--------</td>
<td>---------------------------------</td>
</tr>
<tr>
<td>$A_e$</td>
<td>Nozzle Exit Plane Area</td>
</tr>
<tr>
<td>$D_t$</td>
<td>Nozzle Throat Diameter</td>
</tr>
<tr>
<td>$\dot{m}_{fuel}$</td>
<td>Fuel Mass Flow Rate</td>
</tr>
<tr>
<td>$\dot{m}_{oxi}$</td>
<td>Oxidizer Mass Flow Rate</td>
</tr>
<tr>
<td>$\varepsilon$</td>
<td>Nozzle Expansion Ratio</td>
</tr>
<tr>
<td>$L_{req}$</td>
<td>Combustion Port Required Length</td>
</tr>
<tr>
<td>$R_u$</td>
<td>Universal Gas Constant ($8.314 \frac{J}{kg \cdot K}$)</td>
</tr>
</tbody>
</table>
1.0 INTRODUCTION

1.1 Hybrid Rocket Engine History

Working to create an easily storable, intrinsically safe, and environmentally sound propulsion system has been an important goal of aerospace designers for years. Early development models demonstrated the need for a throttleable and controllable chemical rocket. Liquid rocket engines possess these attributes but often require the storage of cryogenic fuels and oxidizers and require complex turbo-pump systems. The Solid Rocket Engine is storable and proven to be a very profound and “Standard” method of getting payloads into space, but it cannot be throttled. Although effective, the combustion process cannot be stopped without the complete depletion of the fuel/oxidizer mixture. As shown in Figure 1, Hybrid Rocket engines differ from Solid Rocket engines by controllable injection of oxidizer.

![Figure 1 - Hybrid Rocket Engine](image)
Since the earliest recorded rocket hybrid rocket experiments in the 1930’s\(^1\), engineers experimented with different ways to combine the throttling capability of a liquid rocket engine with the excellent storing characteristics of solid rocket engines. Early development of hybrid rockets started with the notion of being able to throttle a liquid oxidizer with a solid grain of fuel. Though currently there are some developmental engines which apply the combination of liquid fuel with a solid oxidizer\(^1\), the former has proven to be a desired research and development baseline.

In working with hybrid rocket engines, it is important to rationalize the terms of operation. As shown in Figure 2, many of the characteristic definitions remain similar between all rocket engines; which are the chamber diameter, throat diameter, and nozzle exit diameter. Though it is specific to Hybrid Rocket Engines, the fuel mass flow rate, \( \dot{m}_f \), is the rate at which the fuel vaporizes from the fuel grain. Thus, the system mass flow rate for a Hybrid Rocket Engine, \( \dot{m}_{sys} \), is the additive of the oxidizer mass flow rate, \( \dot{m}_o \), and \( \dot{m}_f \).
From a university viewpoint, Hybrid Rocket Engines are extremely attractive. As in this thesis, many of the fuel and oxidizer combinations can be considered inert at Standard Temperature and Pressure conditions (STP). With the underlying guiding principle of deriving safe laboratory experiments for university aerospace engineering students, choosing a Hybrid Rocket Engine remains an attractive alternative.

1.2 Project Goals and Safety

The primary objectives in developing a lab scale Hybrid Rocket Engine can be easily described by the following:

- Proof of Concept
- Safe enough to use in a laboratory environment
- Laboratory setup must be scalable for upgraded instrumentation
- End product must be easy to understand and use by students

Environmental safety is one major concern for all rocket engine projects. Most launch authorities require an exit species analysis to prevent government and private organizations from using toxic propellants in their launch vehicles. GDL ProPep was used to complete the Engineering Thermo-equilibrium calculations. These calculations were completed and resulted in a far less toxic engine plume then other similar scale liquid and solid rocket engines, as shown in the following paragraph.

Hydroxyl Terminated Polybutadiene (HTPB) and Liquid Nitrous Oxide were chosen as the propellant reactants primarily due to their non-restrictive physical handling requirements. At room temperature, these two reactants pose no immediate physical danger. Furthermore, their combusted exhaust species do not immediately pose any
substantial environmental harm, as shown in Table 1 for an Oxidizer to Fuel ratio (OF) of 7.5. The most prominent exhaust constituents consist of molecular nitrogen, water, carbon dioxide, carbon monoxide, hydroxyl radicals, nitrogen oxide, molecular hydrogen, molecular oxygen and atomic hydrogen.

Table 1 – Exhaust Species of a HTPB/N20 Rocket at OF=7.5

<table>
<thead>
<tr>
<th>Exit Species</th>
<th>No. of Moles</th>
<th>Molecular Name</th>
<th>Exit Species</th>
<th>No. of Moles</th>
<th>Molecular Name</th>
</tr>
</thead>
<tbody>
<tr>
<td>N2</td>
<td>0.16821</td>
<td>Molecular Nitrogen</td>
<td>N</td>
<td>3.48E-06</td>
<td>Atomic Nitrogen</td>
</tr>
<tr>
<td>H2O</td>
<td>0.04531</td>
<td>Water</td>
<td>NH3</td>
<td>1.60E-06</td>
<td>Ammonia</td>
</tr>
<tr>
<td>CO2</td>
<td>0.03816</td>
<td>Carbon Dioxide</td>
<td>NHO2</td>
<td>1.59E-06</td>
<td>Nitrous Acid</td>
</tr>
<tr>
<td>CO</td>
<td>0.03358</td>
<td>Carbon Monoxide</td>
<td>NHO2</td>
<td>1.43E-06</td>
<td></td>
</tr>
<tr>
<td>HO</td>
<td>5.47E-03</td>
<td>Hydroxyl</td>
<td>NH2</td>
<td>1.36E-06</td>
<td>Amide</td>
</tr>
<tr>
<td>NO</td>
<td>4.87E-03</td>
<td>Nitrogen Oxide</td>
<td>NH</td>
<td>1.23E-06</td>
<td></td>
</tr>
<tr>
<td>H2</td>
<td>4.84E-03</td>
<td>Molecular Hydrogen</td>
<td>CNHO</td>
<td>7.00E-07</td>
<td>Hydrogen Isocyanate</td>
</tr>
<tr>
<td>O2</td>
<td>2.71E-03</td>
<td>Molecular Oxygen</td>
<td>CNH</td>
<td>4.21E-07</td>
<td></td>
</tr>
<tr>
<td>H</td>
<td>1.15E-03</td>
<td>Atomic Hydrogen</td>
<td>CH2O</td>
<td>2.07E-07</td>
<td>Formaldehyde</td>
</tr>
<tr>
<td>O</td>
<td>7.41E-04</td>
<td>Atomic Oxygen</td>
<td>CNO</td>
<td>1.13E-07</td>
<td></td>
</tr>
<tr>
<td>HO2</td>
<td>2.78E-05</td>
<td>Hydroperoxyl Radical</td>
<td>CN</td>
<td>2.00E-08</td>
<td>cyanide</td>
</tr>
<tr>
<td>NO2</td>
<td>9.09E-06</td>
<td>Nitrogen Dioxide</td>
<td>NO+</td>
<td>1.84E-08</td>
<td>Nitric Oxide Ion</td>
</tr>
<tr>
<td>NHO</td>
<td>7.86E-06</td>
<td></td>
<td>O3</td>
<td>1.80E-08</td>
<td>Ozone</td>
</tr>
<tr>
<td>N2O</td>
<td>5.65E-06</td>
<td>Nitrous Oxide</td>
<td>CO2-</td>
<td>1.74E-08</td>
<td>Ionic Carbon Dioxide</td>
</tr>
<tr>
<td>CHO</td>
<td>3.95E-06</td>
<td></td>
<td>H3O+</td>
<td>9.13E-09</td>
<td>Hydronium Ion</td>
</tr>
</tbody>
</table>

1.3 **Design Implementation and Approach**

This thesis concentrates on the more abundantly found hybrid rocket engines, the Solid Fuel with Liquid Oxidizer. The “Spiral Design Technique” was chosen to aid in the management of the design and development activity. This technique can be best illustrated from Figure 3.
In using this technique it is very important to define the baseline for comparison. Without this, further research and case studies are unsubstantiated. It was found that in order to properly develop all the rocket subsystems, the iterative design processes was necessary. The main subsystems, described in detail in the chapters to follow, were the Electronics, Hardware and Software. The various subsystems are shown in Figure 4.
Figure 4 - Engineering Discipline Breakdown
2.0 METHODOLOGY

2.1 System Overview

To clarify the subsystem components, it is important to graphically outline the major system components that are discussed in this section. Figure 5 defines all of the subsystems required for this rocket project.

![System Diagram](image)

Figure 5 - System Diagram

2.1.1 Electronics Design

As presented in Figure 4 and Figure 5, there were many electronics design projects that all needed to interface correctly to ensure a successful, functional, and scalable end product. Succeeding discussions will disseminate each of components in the Electronics Design.
2.1.1.1  **Pressure Transducer**

One of the empirical outputs of a Hybrid Rocket Engine is the System Chamber Pressure. During the Rough Order of Magnitude (ROM) calculations, a chamber pressure of 500 psi was assumed. This assumption proved to be the most beneficial method for obtaining structural loading and thermodynamic calculations, as in Krauss\textsuperscript{11}.

The pressure transducer was a Honeywell Pressure Transducer which provided a 4-20 mA output based upon a pressure range of 0 psig to 500 psig. A bridge resistor was added to the circuitry, enabling a 0 – 10V output. The Data Acquisition network required data inputs of 0 – 2.5 V, 0 – 5 V or 0 – 10 V. The pressure transducer had calibration potentiometers, enabling the zeroing and range output.

2.1.1.2  **Data Processing and Signal Conditioning [DPSC]**

The Data Processing and Signal Conditioning Electronics were designed, developed, integrated into the Acquisition Network, and tested for functionality. Due to their inherently stable and scalable features, future design iterations should implore Commercial-Off-The-Shelf (COTS) components.

The primary element of the DPSC was the INA125P integrated circuit (IC). This IC enabled easy manipulation of the raw data transferred through the Data Acquisition Network.

2.1.1.3  **Cabling**

Various cabling was used throughout the overall design. The primary power cabling was a 22 AWG, 4-conductor, foil-shielded PP/PVC and PVC/PVC 300 volt
cable. This shielded, 4-conductor core cable allowed for the powering of multiple objects using a single cable. For safety reasons, the rocket testing was to be held 100 feet from the Data Acquisition Host Interface, and as such, a cable integrity requirement was set to mitigate any voltage drop anomalies.

Every data connector used in the system was a high-voltage, 9-Pin, D-Sub Connector. The connector continuity requirement allowed for commonality and ease of replacement.

2.1.1.4 Electrically Actuated Ball Valve

The high flow rate conditions of this rocket engine necessitated the use of an electro-pneumatically actuated ball valve. The final design consisted of a high-torque, 12-VDC motor connected to the mounting tongue of a 2000-psi working pressure, stainless steel ball valve. A special connecting sleeve was designed to translate the rotational energy from the motor to the ball valve. Like most rocket projects, a remote control terminal is required for safety. As such, the sleeve design was critical in enabling the remote actuation of the oxidizer flow into the rocket engine.
A simple 120VAC powered oxidizer flow switch was developed to control the ball valve actuation. Refer to Figure 6 for the designed test setup.

2.1.1.5 **Load Cell**

Measuring the produced thrust was outlined as one of the primary goals of this thesis. As shown in Figure 6, the load cell was mounted to the aft end of the rocket thrust stand. It was designed to be cantilevered off the rear end with a directive force element translating the force developed by the engine to the load cell.

The primary software engineering goal was to convert the mV output signal from the load cell into a usable amplified signal, and then send it through the data acquisition network. As discussed previously, the load cell output was connected directly through
the DPSC, and then was allowed to traverse through the acquisition network. This cabling philosophy allowed for effortless processing of the 0 – 10 V output signal.

2.1.1.6  **Data Acquisition Device (DI-148U)**

A low cost alternative to data acquisition was purchased through DATAQ Instrumentation (http://www.dataq.com). This Universal Serial Bus (USB) device was not only remotely powered, but allowed a direct data pipeline to the Engine Data Acquisition Host Computer. The specifications for the device easily allowed for the transducer required +/- 10V inputs. The manufacturer provided Software Development Kit (SDK) allowed for 100 Hz sampling rate with 10-bit accuracy.

2.1.2  **Hardware Design**

As discussed in Figure 4, there were many hardware design projects that all needed to interface correctly to ensure a quality, functional, and scalable end product. In the following sections, each of the systems that fall into the Hardware Design heading.

2.1.2.1  **Thrust Stand**

The final design was chosen due to the availability of low-cost metal tables. The final, unmodified table was donated by NUMEK Inc. The dynamic nature of this project required for “at-home” design and rework. Common metal cutting jigsaw blades, coupled with adequate cutting oil made the manufacturing of this custom thrust stand easily attainable. Stainless Steel flange brackets were bolted to the legs of the thrust stand. This enabled for a proper, shear resistant ground connection.
2.1.2.2 **Directive Force Element**

In order to properly translate the axial thrust force from the engine to the load cell, a structural-rod member was incorporated into the system. This adjustable, threaded rod rested freely on the AFT portion of the rocket engine. While the engine was running, the Directive Force Element would then apply force to the load cell. As is discussed later, there exists an excessive amount of “chatter” in the data output of the load cell. Some of this noise in the systems is due to the physical “spring-like” nature induced by the load cell flexing as force is applied.

2.1.2.3 **Oxidizer Reservoir**

The oxidizer reservoir was a 10-lb race car Nitrous Oxide tank. Using this device allowed for easy replenishment at any race car speed shop in the area. Since this exercise did not necessitate the proof of concept on the engine’s throttling capabilities, the oxidizer tank was run at full output for the duration of the test. Consequently, and assuming minimal losses, the Nitrous Oxide tank permitted for two cycles before a recharge was required.

Nitrous Oxide has a vapor pressure at ambient temperatures that allows two phase storage and efficient oxidizer delivery through a self-pressurizing process\(^2,3\). Furthermore, the two phase storage of the oxidizer allowed for post-combustion gas phase blow down, which allowed for extinguishing of any non desired combustion.
2.1.2.4 **Rocket Chamber**

As is discussed in the computer model portion of this paper, the rocket chamber was designed and manufactured using the initial assumptions and data that were available. Though there was not much room for scalability, the tasked manufacturing staff was skilled enough to accommodate any and all design iterations that were made. The final schematic can be found in the appendix.

The final product was designed from 3.0-inch I.D, 3.5 inch O.D, 304 Stainless Steel pipe. Since the estimated chamber pressure was based on calculations and empirical results, the stainless steel pipe allowed for tremendous margins of safety. The critical design calculation for this rocket subsystem was the stress imposed on the chamber of combusted gases. Tangential and radial stresses were accounted for in the rough stress calculation.

2.1.2.4.1 **Fuel Grain**

The fuel grain was designed using the quantities listed in Table 2.\textsuperscript{2, 11}. The end product resulted in a misformation. Note that the initial fuel iteration included too much Carbon-Black and not enough MDI curative. Future designs should only allow approximately 1% Carbon-Black\textsuperscript{5, 6, 11}, with the balance added to the MDI contribution. However, the excessive amount of Carbon-Black, utilized herein did not hinder the proof-of-concept, but rather the engine efficiency. The fuel pouring and curing processes were easily attained and facilitated by the use of standard 3.5 inch I.D PVC tubing. As the designed and developed single port stainless steel mandrel was placed in the fuel grain\textsuperscript{5, 6, 7, 11}, a PVC reducing cap was used to securely place the device.
The curing time was estimated, based upon previous research, to be approximately 48 hours\textsuperscript{11}. This cure time is normally a function of the room temperature and humidity, but did not impact the final result, as the lab was not environmentally controlled.

Table 2 - Fuel Grain Percentages

<table>
<thead>
<tr>
<th>Engine Number</th>
<th>Quantity (gm)</th>
<th>Percent</th>
</tr>
</thead>
<tbody>
<tr>
<td>HTPB</td>
<td>690.5</td>
<td>83.900</td>
</tr>
<tr>
<td>MDI</td>
<td>106.3</td>
<td>12.916</td>
</tr>
<tr>
<td>Carbon</td>
<td>26.2</td>
<td>3.183</td>
</tr>
<tr>
<td></td>
<td>823</td>
<td>100.000</td>
</tr>
<tr>
<td>HTPB</td>
<td>694.1</td>
<td>84.011</td>
</tr>
<tr>
<td>MDI</td>
<td>107.06</td>
<td>12.958</td>
</tr>
<tr>
<td>Carbon</td>
<td>25.04</td>
<td>3.031</td>
</tr>
<tr>
<td></td>
<td>826.2</td>
<td>100.000</td>
</tr>
</tbody>
</table>

2.1.3 **Software Design**

As discussed in Figure 4, there was one subsystem which was essential to the overall effectiveness of the project: the data acquisition interface. Further details are provided as follows.

2.1.3.1 **Engine Data Acquisition Host**

The Engine Data Acquisition Host was designed using Microsoft Access 2003, the DATAQ Software Development Kit (SDK) and Visual Basic 6.0. It is important to understand the flow of data so as to properly analyze the information\textsuperscript{13}. For this purpose, an outline of how the data traversed through the data pipeline is provided. As shown in Figure 7, the simplistic nature of the data flow enabled a straightforward software interface, shown in Figure 8.
Due to the limitations of the Data Acquisition device that was purchased, tradeoffs were made in the software design to allow for proper functionality. The rate at which that data were retrieved from the Rocket Engine was limited to 100 Hz. Using the DI-148U’s included SDK, the software GUI, shown in Figure 8, only required the following user input:
- Number of Channels
- Desired Sample Rate (Governed by low cost hardware)
- USB Simulated Serial Communications Port Number
- Communications Port Baud Rate

The software source code has been provided in the appendix.

![Engine Data Acquisition Host GUI](image)

**Figure 8 - Engine Data Acquisition Host GUI**

### 2.2 Hybrid Rocket Engine Theory

In a traditional liquid-supplied rocket engine, chamber pressure and all inlet conditions of the reactants are readily available or can be easily calculated. However, one of the major initial design conditions not available a priori in a hybrid motor is the chamber pressure. In a Hybrid Rocket Engine, one needs to estimate what the chamber
pressure will be, and then run the calculations. The design process then becomes iterative in nature, and normally requires multiple engine firings to determine the proper chamber pressure for a given motor design.

Though the computer model is included in the appendix, it is necessary to step through the quantitative design considerations. The proper initial flow rate conditions, we required first. The sample calculations, as follows, are based upon the initial Oxidizer/Fuel Ratio, referred to as $OF_{\text{init}}$. These numbers are to facilitate the calculation of the constant oxidizer mass flow rate and nozzle throat geometry we will use through out this experiment.

First, an expression for the Thrust Coefficient, $C_f$, is generated in terms of chamber pressure and the coefficient of specific heats, $\gamma$.

$$
C_f = \left[ \frac{2 \cdot \gamma^2}{\gamma - 1} \cdot \left( \frac{2}{\gamma + 1} \right)^{\gamma + 1} \left( \gamma - 1 \right) \right] \cdot \frac{1 - \left( \frac{P_e}{P_c} \right)^{\gamma - 1}}{\gamma - 1}
$$

Once the trust coefficient has been calculated, the required throat area to achieve a choked flow can be calculated.

$$
A_t = \frac{F_t}{C_f \cdot P_c}
$$

To establish the remaining unknown geometrical constraints of the nozzle, the flame temperature must be calculated. Since it is impossible for one actually attain an ideal flame temperature, it was assumed that the combustion efficiency was 90%. The adiabatic flame temperature was iteratively calculated using the Thermo-equilibrium software, GDL ProPep.
Calculating the total system mass flow rate will allow the fuel and oxidizer specific flow rates to be calculated. This can be done by identifying a desired “Ideal” OF ratio\textsuperscript{11,18}. It is important to remember that this OF ratio is a desired OF ratio. Since the oxidizer flow rate is not throttled, the OF ratio will change as a function of time due to the inherent physics of a hybrid rocket engine.

\[
T_c_{\text{actual}} = \eta_c \cdot T_c_{\text{ideal}}
\]

\[
\dot{m}_{\text{sys}} = A_i \cdot P_c \cdot \gamma \cdot \sqrt[\gamma - 1]{\frac{1}{\gamma + 1} \left( \frac{2}{\gamma + 1} \right) \left( \frac{\gamma + 1}{\gamma - 1} \right) \left( \frac{R_y}{MW} \right)}
\]

\[
\dot{m}_{\text{fuel}} = \frac{\dot{m}_{\text{sys}}}{OF_i - 1}
\]

\[
\dot{m}_{\text{oxi}} = \dot{m}_{\text{sys}} - \dot{m}_{\text{fuel}}
\]

Once the fuel and oxidizer specific flow rates are known, we shall now calculate the expansion ratio can be calculated so that the design requirements for the nozzle can be further refined. For this experiment, it is important to point out that the design is for a “lab scale” engine requiring a thrust level of approximately 100 lbf.

\[
\varepsilon = \left( \frac{\gamma + 1}{2} \right)^{-1} \cdot \left( \frac{P_e}{P_c} \right)^{\frac{1}{\gamma}} \cdot \sqrt[\gamma - 1]{\left( \frac{\gamma + 1}{\gamma - 1} \right) \left( 1 - \frac{P_e}{P_c} \right)}^{-1}
\]

\[
A_{\text{exit}} = A_i \cdot \varepsilon
\]

\[
D_{\text{exit}} = \sqrt[\pi]{\frac{4 \cdot A_{\text{exit}}}{\pi}}
\]
The above calculations are not initial design values. These values are for a point in time during the burn. For simplicity of the experiment, only one oxidizer mass flow rate has been chosen. This value relies on the fact that the aforementioned amount of fuel is being vaporized in the combustion chamber.

\[
A_b = \frac{\pi \cdot D_b^2}{4} \quad \text{(10)}
\]

\[
G_a = \frac{\dot{m}_{\text{oxi}}}{A_b} \quad \text{(11)}
\]

It is helpful at this time to model the burn diameter of the combustion chamber as a function of time.

\[
\dot{r} = a \cdot G_a^n \quad \text{(12)}
\]

Since the change in port diameter can be simplified as twice the regression rate, the following can be seen\(^{18}\).

\[
\dot{r} = \frac{1}{2} \cdot \frac{d}{dt} D_b = a \cdot \left( \frac{\dot{m}_{\text{oxi}}}{\pi D_b^2} \right)^n = a \cdot \left( \frac{4 \cdot \dot{m}_{\text{oxi}}}{\pi} \right)^n \cdot D_b^{-2n} \quad \text{(13)}
\]

The relation can be further simplified by invoking the separation of variables technique:

\[
\int_{D_b}^{D_p} D_b^{2n} \cdot dD_b = 2 \cdot a \cdot \left( \frac{4 \cdot \dot{m}_{\text{oxi}}}{\pi} \right)^n \cdot \frac{1}{0} dt \quad \text{(14)}
\]

Further integrating the equation, results in.
\[ D_b(t) = \left[ a \cdot t \cdot (4 \cdot n + 2) \cdot \left( \frac{4 \cdot \dot{m}_{ox}}{\pi} \right)^n \right] + \left(2R_i\right)^{2n+1} \]  \quad (15)

From this relation, the OF ratio as a function of time can be determined.

\[ \dot{r}(t) = a \cdot \left( \frac{4 \cdot \dot{m}_{ox}}{\pi \cdot D_b(t)} \right)^n \]  \quad (16)

From the mass production equation for cylindrical ports\textsuperscript{11,18}, a definition for combustion port length can be found. This value was for the design goal of \( OF_i \). This value was decided upon, since it is not the stoichiometric point, and serves as a good intermediate point. Although the rocket engine will attain the stoichiometric point at some point in time, the oxidizer flow rate for that situation may require a non-lab scale length.

\[ \frac{\dot{m}_{fuel}}{\rho_{fuel} \cdot \dot{r}} = L \cdot \frac{2 \cdot \pi \cdot D_b}{2} \]  \quad (17)

\[ L_{req} = \frac{\dot{m}_{fuel}}{\rho_{fuel} \cdot \dot{r}(0 \sec) \cdot 2 \cdot \pi \cdot R_i} \]  \quad (18)

It is intuitive to see that the mass flow rate of the fuel will “slow” down, and the OF ratio will increase as a function of time. This is due to the fact that the regression rate is slowing down as a function of time. To discuss this, it is useful to should show how the OF ratio varies with time such that one can demonstrate how the engine will perform.

\[ \dot{m}_{fuel}(t) = L_{req} \cdot 4 \cdot \pi \cdot R_i \cdot \rho_{hpb} \cdot \dot{r}(t) \]  \quad (19)

\[ OF(t) = \frac{\dot{m}_{ox}}{\dot{m}_{fuel}(t)} \]  \quad (20)
Since the system mass flow rate must be equivalent to the choked flow rate, we shall compare the two to find out how long we really have for the system to burn and produce the desired amount of thrust.

It is now necessary to create the ratio of specific heats, Molecular weight, and Flame temperature as functions of time.

\[ \gamma(t) = \gamma(OF(t)) \]  
\[ MW(t) = MW(OF(t)) \]  
\[ T_c(t) = \eta_c \cdot T_c(OF(t)) \]

These values are called upon in the computer model and are curve fits to the excel data points seen in the appendix.

As shown in Figure 10, the system mass flow rate varies with time according to the following relationship.
\[ \dot{m}_{\text{sys}}(t) = \dot{m}_{\text{ext}}(OF_t) + \dot{m}_{\text{fuel}}(t) \]  

Figure 10 – System Mass Flow Rate vs. Time

As seen in Figure 11, the exit velocity of the system varies with time. The following relationship describes the nature of the variance\textsuperscript{11, 18}.

\[ V_{\text{exit}}(t) = \left( \frac{2 \cdot \gamma(t)}{\gamma(t) - 1} \right) \left( \frac{R_u}{MW(t)} \right) T_e(t) \cdot \left[ 1 - \left( \frac{P_e}{P_c} \right)^{\frac{\gamma(t)-1}{\gamma(t)}} \right] \]  

For ideal flow, the thrust can be defined by the following relationship, and can be seen, graphically, in Figure 12.

\[ F_i(t) = \dot{m}_{\text{sys}}(t) \cdot V_{\text{exit}}(t) \]
Figure 11 – Exit Velocity vs. Time

Figure 12 – Thrust vs. Time
The exit Mach number and its associated velocity of sound become:

\[ a_e(t) = \sqrt{\frac{\gamma(t) \cdot T_e(t) \cdot R_g}{MW(t)}} \]  

(27)

\[ M_e(t) = \frac{V_{exit}(t)}{a_e(t)} \]  

(28)

Furthermore, \( c^* \) and specific impulse can be calculated as a function of time.

\[ c^*(t) = \frac{F_i(t)}{m_{sys}(t) \cdot C_f(t)} \]  

(29)

\[ I_{sp}(t) = \frac{c^*(t) \cdot C_f(t)}{g} \]  

(30)

From the above initial conditions and the assumption of constant oxidizer mass flow rate, the following describes what the expansion ratio should be to output the desired thrust level. As shown in Figure 13, the varying thrust could be accounted for by designing a nozzle skirt or throttling the oxidizer flow rate.

\[ \varepsilon(t) = \left[ \frac{\gamma(t) + 1}{2} \right]^{\frac{1}{1 - \gamma(t)}} \cdot \left( \frac{Pe}{Pc} \right)^{\frac{1}{\gamma(t)}} \cdot \sqrt{\frac{\gamma(t) + 1}{\gamma(t) - 1}} \cdot \left[ 1 - \left( \frac{Pe}{Pc} \right)^{\frac{\gamma(t) - 1}{\gamma(t)}} \right]^{-1} \]  

(31)

\[ A_{exit}(t) = A_e(t) \cdot \varepsilon \]  

(32)

\[ D_{exit}(t) = \sqrt{\frac{4 \cdot A_{exit}(t)}{\pi}} \]  

(33)
Further, if we were to solve the equation for diameter as a function of time, we can output a relationship for the maximum burn time allowed$^{18}$.

$$
time_{\text{burnout}} = \frac{(2 \cdot R_f)^{2n+1} - (2 \cdot R_i)^{2n+1}}{a \cdot (4 \cdot n + 2) \cdot \left( \frac{4 \cdot \dot{m}_{\text{ox}}}{\pi} \right)^n}
$$

(34)
3.0 SYSTEM AND COMPONENT TESTING

3.1 Testing Objectives

The primary objectives for both the system and component level testing were to demonstrate the level of safety and usability of a Lab Scale Hybrid Rocket Engine. One of the a priori assumptions is that the regression rate of the fuel is a known value\textsuperscript{11}.

One test burn was completed, and its results are very convincing of the fact that the University of Central Florida could, in a very short period of time, have its own payload delivery system at its disposal. Details of the component testing are provided in this chapter.

3.2 Component Level Testing

3.2.1 Hydraulic Needle Valve Calibration

The precise control of the quantity of oxidizer flowing through the rocket engine was required. Since throttling control was not one of the initial assumptions, a constant flow rate needed to be applied to verify the computer model.

The test apparatus dictated the use of a Hydraulic Needle Valve to control the mass flow rate of oxidizer. Since this device was purely analog, and control is done by loosening or tightening a hand-controlled internal piston, several tests needed to be done to define the Needle Valve’s operating conditions. As shown in Table 3, the Hydraulic Needle Valve required 2.5 turns to establish the proper flow rate output of the device.
Table 3 - Mass Flow Testing

<table>
<thead>
<tr>
<th>Test No.</th>
<th>Start Weight (lb)</th>
<th>End Weight (lb)</th>
<th>Start Pressure (psi)</th>
<th>End Pressure (psi)</th>
<th>Duration (sec)</th>
<th>No. of Turns</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>25.58125</td>
<td>21.525</td>
<td>950</td>
<td>900</td>
<td>10</td>
<td>2.5</td>
</tr>
<tr>
<td>2</td>
<td>21.525</td>
<td>17.8125</td>
<td>950</td>
<td>875</td>
<td>10</td>
<td>2</td>
</tr>
</tbody>
</table>

3.2.2 Data Acquisition Network

Before any live firings of the rocket engine, several tests needed to be completed to ensure all connections and software protocols were set correctly. After its calibration, the Shear Beam Load Cell connection between the Signal Conditioning and Data Amplification box needed to be verified and tested. The same testing and sound systematic engineering practices needed to be implemented to ensure the Pressure Transducer, Electrically Actuated Ball Valve, and Remote Ignition source were fully functional from the Data Acquisition Host Computer.

Previously, all tests had been completed in a laboratory environment. In an effort to ensure minimal voltage drops across the long signal and power lines and proper signal-to-noise ratios, field testing was required.
Figure 14 - Data Collection

Figure 14 discusses how the computer model was derived. The computer model will be used to analyze the data gathered from the live firing.

### 3.2.3 Load Cell Testing and Calibration

The Signal Conditioning and Data Amplification box was designed to be multipurpose. This was done to ensure a universal solution to any transducer application. Normally, transducers are rated to produce a certain milli-Volt (mV) to every Volt (V) output. This would define the amplification resistance required. However, in this situation, a precision potentiometer was used.

To calibrate the Signal Conditioning and Data Amplification box, dead weight needed to be applied to the Shear Beam Load Cell. While the load cell was removed from the Thrust Stand, the gain resistance on the Signal Conditioning and Data
Amplification Box was adjusted. The box was calibrated such that the maximum dead weight applied to the load cell equated to 10 VDC. As such, the output voltage equivalent to the no-load situation is 0 VDC.

Due to the limitations of using a single-source power supply, a voltage offset was created by the instrument amplification integrated circuit. This offset, rather than redesigning the circuitry, was accounted for in the data analysis.

Once the load cell and the Signal Conditioning and Data Amplification box had been properly calibrated, the task of installing the load cell into the fully populated system was undertaken. This effort was deemed necessary to establish a force offset multiplier. Due to frictional effects of the Rocket Engine sitting on the Thrust stand, coupled with the U-Bolts that were used to clamp the engine to the table, a significant force offset value was evident. As such, a force multiplier was added to the resulting data.

3.2.4 Remote Ignition Testing

The process of testing the remote ignition was intentionally simplistic. By setting the “match” at an approximate distance, correlating to the distance it would see in a field test, an evaluation could be made to ensure that the voltage drop across the long power transmission lines would not negatively impact the current requirement of the “match”. If the lines were too long, the match would not light.
3.3  

**System Testing**

3.3.1  

**Engine Thrust Validation**

As discussed previously, the raw voltage data passed along the Acquisition Network was un-scaled. Using Figure 15 as a reference, the timing of certain events that occurred during the burn process can be discussed. At approximately 24 seconds, the Hybrid Rocket Engine was ignited, and the combustion process began. Furthermore, at approximately 35 seconds, the Electrically Actuated Ball Valve was closed.

As the Data Acquisition Host cycled its data, we can see that the Load Cell reflected a negative force being applied. This was attributed to the Directive Force Element not being secured to the AFT bulkhead on the engine. The excessive amount of “chatter” can be graphically seen, and a repercussion of this event resulted in the over stressing of the Load Cell.

The engine was designed to output 100 lbf of thrust at an OF ratio of 7.5. As shown in Figure 15, this occurred between 26 and 30 seconds.
3.3.2 Chamber Pressure Validation

The chamber pressure, as previously discussed, is an empirical output of the testing process. Initially, a chamber pressure was assumed. This enabled the regression model to output physical requirements that enabled the engine design. As can be seen in Figure 16, the maximum pressure reading was approximately 240 psi. This is approximately half of the initial design pressure.

We can now investigate what is graphically explained in Figure 16, as time progressed. At approximately 22.4 seconds, the electrically actuated ball valve was engaged. The burn lasted from approximately 24 seconds to 33 seconds. We can also
further assimilate the point at which the engine reached its design OF ratio. At 27 seconds, it can be assumed that the rocket engine reached an OF ratio of 7.5.

![Pressure vs. Time Graph](image)

**Figure 16 - Chamber Pressure vs. Time**

Throughout the entire burn process, pressure anomalies appeared. Although, previous research states that well tested Hybrid Rocket Engines, with expensive fuel systems, do not normally exhibit extreme cases of combustion instability\(^1,2,3,4,5,8,11,15,16,17,18,19\), it was initially suspected. A Fast Fourier Analysis was done to quantify any frequencies that may be a fault mode of the engine design.

### 3.3.2.1 Fast Fourier Analysis

Figure 17 displays the frequency domain Fast Fourier Transform (FFT) Analysis. Since the Acquisition Network was restricted to a 100 Hz sampling rate, the only folding frequencies that could be calculated were 50 Hz and below. Though it is possible for
high frequency pressure fluctuations, the qualitative analysis of the FFT does not agree with the original suspected cause. The cause of the extreme pressure fluctuations is currently unknown, and a detailed investigation is beyond the scope of this thesis.

Figure 17 - Chamber Pressure FFT Analysis

3.3.3 Regression Model Verification

The fuel grain was designed to have a maximum burn time of 60 seconds. Furthermore, from the above figures, it can be seen that the rocket engine ran for approximately 8 seconds. The logical conclusion is that there should be a tremendous amount of sliver residing in the combustion chamber. As shown in Figure 18 and Figure 19, during the engine breakdown exercise, the fuel casing was shown to have very little fuel sliver remaining. Though this phenomenon can not be completely explained at this time, there are several assumptions that could be made to justify the outcome. The main design flaw which could account for the miscalculation was the Electrically Actuated
Ball Valve. As previously discussed, the gasketing internal to the ball valve prohibited the valve from completely disengaging. Furthermore, the fuel percentage misformulation may have resulted in the fast-burning fuel grain.

![Figure 18 - Post Burn Chamber](image18)

When the ball valve was actuated to the closed position, there was a tremendous amount of oxidizer escaping into the combustion chamber. This resulted in an uncontrolled burn situation. In fact, during the testing process, the fail-safe Nitrous Oxide cylinder gate valve was closed to exhaust the flame.

![Figure 19 - Fuel Sliver](image19)
4.0 RECOMMENDATIONS

From a student’s perspective, real world experiments and tangible physics are imperative to the broader understanding of fundamental engineering principles.

Throughout the development of this project, various industry professionals were all interested to help, listen and encourage the additional development of such activities at the University. The Data Analysis portion of this thesis has shown that many improvements can be made to further verify the far-reaching capabilities of Hybrid Rockets. Future design iterations should strongly consider using COTS hardware.

Specifically, such hardware and design recommendations include:

1. Allow for higher volumetric percentage of HTPB Curative in Fuel Grain
2. Allow for lower volumetric percentage of Carbon-Black in Fuel Grain
3. Research and utilize a properly calibrated Pressure Transducer
4. Research and utilize a Data Acquisition Device with a higher sampling rate
5. Research and utilize a certified rocket engine test and burn facility
6. Research and utilize an improved flow shut-off valve
7. Research and utilize an improved flow regulation valve
8. Research and utilize an improved Thrust Stand
5.0 SUMMARY

It has been shown that, not only was the design, development, integration and testing of a Lab Scale Hybrid Rocket Engine safely feasible to due in a laboratory environment, but that with further engineering and monetary investment, the University of Central Florida will have a method of delivering student designed payloads to space. As such, the intrinsic safety and economy of such a rocket motor program would be an excellent asset to any thermo-fluids laboratory.

The MathCAD computer model developed, has proven to be the focal point of the presented experiment, and further, can be ported to any software package; enabling imbedded system functionality and calibration testing.

Throughout the spiral design process, several electronics and mechanical subsystems were developed. As discussed in the presented thesis, these subsystems will require additional research. Though the design has proven the ease of use, several design iterations are required before any university designed space payload can be carried into space.
APPENDIX A – MATHCAD COMPUTER MODEL
Units, Constants and Worksheet References

\[ \text{kmol} := 10^3 \cdot \text{mol} \quad \text{Conversion to kilomol} \]

\[ R_u := 8314 \frac{\text{J}}{\text{kmol} \cdot \text{K}} \quad \text{Universal Gas Constant} \]

\[ \text{kJ} := 10^3 \cdot \text{J} \quad \text{Conversion to kilojoule} \]

Problem Statement & Given Constants

\[ P_a := 14.7 \text{psi} \quad \text{Ambient Pressure} \]

\[ P_c := 300 \text{psi} \quad \text{Chamber Pressure (Assumed)} \]

\[ F_t := 100 \text{-lbf} \quad \text{Desired Thrust Output} \]

\[ \rho_{htpb} := 0.900 \frac{\text{gm}}{\text{cm}^3} \quad \text{Density of Hydroxyl Terminated Polybutadiene (Ref. PL-TR-96-3018)} \]

\[ \rho_{fuel} := 1.737 \frac{\text{gm}}{\text{cm}^3} \quad \text{Density of burned fuel/oxidizer mixture at stated Thermodynamic condition} \]

\[ N := 1 \quad \text{Number of Ports Drilled Through Fuel Grain} \]

\[ R_1 := 1 \text{-in} \quad \text{Assumed Value for initial Combustion Port Radius. This value is based upon a burn distance of 1.5 inches \((r_{dot} \cdot t_b)\) while utilizing a simple to manufacture mandrel - Same as } D_p/2 \]

\[ R_f := 1.50 \text{-in} \quad \text{Design Requirement of 3.0 inch diameter rocket (internal)} \]

\[ OF_1 := 9 \quad \text{Desired oxidizer to fuel ratio (From GUIPEP adiabatic flame calculations @ 500 psi) NOTE: The aforementioned calculations only consider fuel and oxidizer with no additives.} \]

\[ \rho_{htpb} := 0.900 \frac{\text{gm}}{\text{cm}^3} \quad \text{Density of HTPB} \]

\[ \theta := 15 \text{-deg} \quad \text{Chosen Nozzle Half Angle} \]

\[ n := 0.9874 \quad \text{Regression Rate exponent (Univ. of Colorado - Mach SR1)} \]

\[ a := 0.1160 \text{-lb} \cdot \text{in}^{-1} \cdot \text{in}^{1+2n} \cdot \text{sec}^{-n-1} \quad \text{Regression Rate coefficient (Univ. of Colorado - Mach SR1)} \]

\[ \eta_c := 0.90 \quad \text{Combustion Efficiency} \]

\[ P_e := P_a \quad \text{Ideal Rocket Assumption} \]

\[ t := 0 \text{sec., 1sec., 10 sec} \quad \text{Burn time (vector)} \]

\[ D_b := 2 \cdot R_1, 2.01 \cdot R_1, 2 \cdot R_f \quad \text{Burn Diameter (vector)} \]
First we must calculate some initial (back of the envelope) numbers. These numbers will be based upon the \(OF_{\text{init}}\). These numbers are to facilitate the calculation of the constant oxidizer mass flow rate & nozzle throat geometry we will use throughout this experiment.

\[
C_{d}(OF_i) := \sqrt{\frac{2\cdot\gamma(OF_i)^2}{\gamma(OF_i) - 1} \left(\frac{\gamma(OF_i) - 1}{\gamma(OF_i) + 1}\right) \left[1 - \frac{P_e}{P_c}\right]}
\]

\[C_{d}(OF_i) = 1.433\]

\[
A_t(OF_i) := \frac{F_t}{C_{d}(OF_i) \cdot P_c}
\]

\[A_t(OF_i) = 0.233 \text{ in}^2\]

\[
D_t(OF_i) := \sqrt{\frac{4 \cdot A_t(OF_i)}{\pi}}
\]

\[D_t(OF_i) = 0.544 \text{ in}\]

\[
T_{c\text{ actual}}(OF_i) := \eta_c \cdot T_c(OF_i)
\]

\[T_{c\text{ actual}}(OF_i) = 2931.962 \text{ K}\]

\[
m_{\text{sys - 1}}(OF_i) := A_t(OF_i) \cdot P_c \cdot \gamma(OF_i) \cdot \frac{\sqrt{\left(\frac{2}{\gamma(OF_i) + 1}\right)}}{\sqrt{\gamma(OF_i) - 1}} \cdot \frac{R_u}{\gamma(OF_i) - 1} \cdot T_{c\text{ actual}}(OF_i)
\]

\[m_{\text{sys - 1}}(OF_i) = 0.209 \text{ kg/s}\]

\[
m_{\text{fuel}}(OF_i) := \frac{m_{\text{sys - 1}}(OF_i)}{OF_i - 1}
\]

\[m_{\text{fuel}}(OF_i) = 0.026 \text{ kg/s}\]

\[
m_{\text{oxy}}(OF_i) := m_{\text{sys - 1}}(OF_i) - m_{\text{fuel}}(OF_i)
\]

\[m_{\text{oxy}}(OF_i) = 0.183 \text{ kg/s}\]

\[
\varepsilon(OF_i) := \frac{1}{\left(\frac{\gamma(OF_i) + 1}{2}\right)^{\frac{\gamma(OF_i) - 1}{\gamma(OF_i)}} \cdot \frac{P_e}{P_c} \cdot \frac{\gamma(OF_i) + 1}{\gamma(OF_i) - 1} \left[1 - \frac{P_e}{P_c}\right]}
\]

\[\varepsilon(OF_i) = 4.053\]

\[
A_{\text{exit}}(OF_i) := A_t(OF_i) \cdot \varepsilon(OF_i)
\]

\[A_{\text{exit}}(OF_i) = 0.943 \text{ in}^2\]
\[ D_{\text{exit}}(\text{OF}_i) := \sqrt{\frac{4 \cdot A_{\text{exit}}(\text{OF}_i)}{\pi}} \]

\[ D_{\text{exit}}(\text{OF}_i) = 1.096 \text{ in} \]

It is important to understand here these values are NOT initial values, but in a sense are values for a point in time during the burn. For simplicity of the experiment, only one oxidizer mass flow rate has been chosen. That value relies on the fact that the aforementioned amount of fuel is being vaporized in the combustion chamber. This preliminary section does not give detail as to when this chemical balance will occur. The succeeding will shed light on this.

\[ A_b(D_b) := \pi \cdot \frac{D_b^2}{4} \quad \text{Area as a function of Dia.} \]

\[ G_o(D_b) := \frac{m_{\text{oxy}}(\text{OF}_i)}{A_b(D_b)} \quad \text{Mass Flux eqn.} \]

\[ r_{\text{dot}}(D_b) := a \cdot G_o(D_b)^n \quad \text{Regression Model} \]

It is helpful at this time for us to model the Diameter of the combustion chamber as a function of time.

As discussed above, the regression rate, \( r_{\text{dot}} \), can be described as

\[ r_{\text{dot}} = a \cdot G_o^n \]

Since the change in port diameter can be simplified as twice the regression rate

\[ r_{\text{dot}} = \frac{1}{2} \cdot \frac{d}{dt} D_b = a \cdot G_o^n \]

If we are to simplify the right hand side of the above equation, we are to see the following

\[ r_{\text{dot}} = \frac{1}{2} \cdot \frac{d}{dt} D_b = a \cdot \left( \frac{m_{\text{oxy}}}{\pi \cdot D_b^2} \right)^n = a \cdot \left( \frac{4 \cdot m_{\text{oxy}}}{\pi} \right)^n \cdot D_b - 2^n \]

If we separate the variables

\[ \int_{D_b}^{D_b(t)} \frac{2^n}{D_b^2} \, dD_b = 2 \cdot a \cdot \left( \frac{4 \cdot m_{\text{oxy}}}{\pi} \right)^n \int_0^t 1 \, dt \]

Integrating the above equation yields the following result

\[ D_b(t) := \left[ a \cdot (4 \cdot 9.9874 + 2) \cdot \left( \frac{4 \cdot m_{\text{oxy}}(\text{OF}_i)}{\pi} \right)^{0.9874} \cdot t \right] + (2 \cdot R_t)^{2.9748} \]
If we now work backwards, and establish a relationship of regression rate as a function of time, we can then determine what the OF ratio is as a function of time.

If we use the above derived equation for $D_b$, we can now see the following

$$r_{dot}(t) := a \times \left( \frac{4 \cdot m_{oxi}(OF)}{\pi \cdot D_b(t)^2} \right)^{9874}$$

From the Mass Production equation for cylindrical ports, we can see that we now need a definition for combustion port length. This value will be a for the design goal of $OF_i$. This value was decided upon, since it is not the stoichiometry point, and servers as a good intermediate point to solve for. The rocket will attain the stoichiometric point at some point in time, but if you design the above oxidizer flow rate for that situation, the rocket may require a non-lab scale length due to the relationship between OF and length. Remember that the necessary chamber length required to reach a particular OF ratio in time will vary because the regression rate is changing. So we must pick only one length (for simplicity). From the cylindrical port mass production equation

$$\frac{m_{fuel}}{\rho_{fuel}r_{dot}} = L \cdot 2 \cdot \pi \cdot \frac{D_b}{2}$$

$$L_{req} := \frac{m_{fuel}(OF)}{\rho_{fuel}r_{dot}(OF) \cdot \pi \cdot 2 \cdot R_i}$$

$$L_{req} = 9.562 \text{ in}$$

It is intuitive to see that as the mass flow rate of the fuel will slow down, and the OF ratio will increase as a function of time. This is due to the fact that the regression rate is slowing down as a function of time (oxidizer flux is slower). To discuss this, we should show how the OF ratio varies with time such that we can demonstrate how the engine will perform (in general) over time.

Utilizing the mass production equation, we can extract a value for $m_{fuel}$ as a function of time

$$m_{fuel}(t) := L_{req} \cdot 4 \cdot \pi \cdot R_i \cdot \rho_{htpy}r_{dot}(t)$$

$$OF(t) := \frac{m_{oxi}(OF)}{m_{fuel}(t)}$$

It is now necessary to show how the system mass flow rate will vary with time, and contrast that with the geometrically set values from above. Since the system mass flow rate must be larger or equivalent to the choked flow rate, we shall compare the two to find out how long we really have for the system to burn and produce the designed amount of thrust.

$$\chi(t) := \gamma(OF(t)) \quad \text{Specific Heat ratio as a function of time}$$

$$MW(t) := MW(OF(t)) \quad \text{Molecular Weight as a function of time}$$

$$T_e(t) := \eta_c \cdot T_e(OF(t)) \quad \text{Flame Temp as a function of time}$$

$$m_{\ldots}(t) := m_{\ldots}(OF) + m_{\ldots}(t) \quad \text{System mass flow rate as a function of time}$$
Moving on to exit velocity of the system, such that we can characterize the flow exiting the nozzle as a function of time

\[
V_{\text{exit}}(t) := \sqrt{\frac{2\cdot\gamma(t)}{\gamma(t) - 1} \cdot \frac{R_u}{\text{MW}(t)} \cdot T_c(t) \cdot 1 - \left(\frac{p_e}{p_c}\right)^{\frac{\gamma(t) - 1}{\gamma(t)}}}
\]

For Ideal Flow (which is what will assume), thrust can vary by time.

\[
F(t) := m_{\text{sys}}(t) \cdot V_{\text{exit}}(t)
\]

As such, so can the exit Mach number and sonic velocity at the exit plane.

\[
a_l(t) := \sqrt{\frac{\gamma(t) \cdot R_u \cdot T_c_{\text{actual}}(t)}{\text{MW}(t) \cdot \text{sec}}}
\]

\[
M_c(t) := \frac{V_{\text{exit}}(t)}{a_l(t)}
\]

Furthermore, we can calculate cstar, and he engines Isp as a function of time

\[
c_{\text{star}}(t) := \frac{F(t)}{m_{\text{sys}}(t) \cdot C_f(t)}
\]

Now, moving on to the initial specific impulse of the engine

\[
I_{\text{sp}}(t) := \frac{c_{\text{star}}(t) \cdot C_f(t)}{g}
\]

From the above initial conditions, if we are to assume constant oxidizer mass flow rate, the following describes what the expansion ratio needs to be as a function of time to validate the required force output.

\[
\varepsilon(t) := \frac{1}{\left(\frac{\gamma(t) + 1}{2}\right)^{\frac{\gamma(t) - 1}{\gamma(t)}} \cdot \left(\frac{p_e}{p_c}\right)^{\frac{\gamma(t) - 1}{\gamma(t)}} \cdot \left(\frac{\gamma(t) + 1}{\gamma(t) - 1}\right)^{\frac{\gamma(t) - 1}{\gamma(t)}} \cdot 1 - \left(\frac{p_e}{p_c}\right)^{\frac{\gamma(t) - 1}{\gamma(t)}}}
\]

\[
A_{\text{exit}}(t) := A_l \left(\frac{t}{\text{sec}}\right) \cdot c_e(t)
\]

\[
D_{\text{exit}}(t) := \sqrt{\frac{4 \cdot A_{\text{exit}}(t)}{\pi}}
\]
If we were not to take into consideration flow rate issues, as this traditionally would not be the case if throttling was done. We can look at the following chart to see how the fuel would have regressed.

Regression Model at t = 10 seconds

Further, if were to solve the equation for diameter as a function of time, for time, we could result in a maximum burn time to reduce the amount of sliver left for $R_f$. (The burn time will increase as a function of regression rate because the slower the oxidizer flux, the slower the HTPB will erode.)

$$\text{time_{burnout}} := \frac{(2 \cdot R_f)^{2.9748} - (2 \cdot R_f)^{2.9748}}{a \cdot (4n + 2) \cdot \left(\frac{4 \cdot m_{\text{oxid}}(OF)}{\pi}\right)^{0.9874}}$$

$$\text{time_{burnout}} = 51.55 \text{ s}$$
As mentioned above, one of the reasons needed to calculate system mass flow rate, is to ensure adequate flow rate to choke the nozzle without a reduction in chamber pressure. The preceding chart shows that for time = 0 until approximately time = 9 seconds, the chamber pressure will increase, and likewise will decrease after the time in union.
Figure 20 - Data Acquisition Graphical Interface
Table 4 - Test Procedure

<table>
<thead>
<tr>
<th>Item No.</th>
<th>Description</th>
<th>Criteria</th>
<th>Result</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>Establish rocket engine test site, insert 12” stakes to adhere engine to ground.</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>2</td>
<td>Connect Load Cell to J1 on Signal Conditioning &amp; Data Acquisition Box.</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>3</td>
<td>Connect Pressure Transducer to J3 on Signal Conditioning &amp; Data Acquisition Box.</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>4</td>
<td>Connect Signal Conditioning &amp; Data Acquisition Box J4 to DI-148U</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>5</td>
<td>Using 100’ USB extender, connect DI-148U to the Acquisition Host Computer</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>6</td>
<td>Using the valve controller power link, connect the motorized valve to the valve controlling interface.</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
</tbody>
</table>

NOTE: FOR THE FOLLOWING STEP, TAKE PROPER SAFETY PRECAUTIONS. DO NOT CONNECT POWER SOURCE TO THE E-MATCH POWER LINK.

NOTE: IF THIS NOTE IS NOT HEADED, PERSONAL INJURY CAN RESULT

<table>
<thead>
<tr>
<th>Item No.</th>
<th>Description</th>
<th>Criteria</th>
<th>Result</th>
</tr>
</thead>
<tbody>
<tr>
<td>7</td>
<td>While power source is not connected, install e-match. Ensure that power leads are connected.</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>9</td>
<td>Ensure all proper electrical and data connections are made at Acquisition Host.</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>10</td>
<td>Using supplied scale, weigh NO2 tank</td>
<td>Input Weight (lb, gm)</td>
<td></td>
</tr>
<tr>
<td>11</td>
<td>Connect NO2 tank to Engine</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>Step</td>
<td>Action Description</td>
<td>Result</td>
<td></td>
</tr>
<tr>
<td>------</td>
<td>------------------------------------------------------------------------------------</td>
<td>--------</td>
<td></td>
</tr>
<tr>
<td>12</td>
<td>While test assistant is standing by engine, engage valve controller interface to ensure proper actuation.</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>13</td>
<td>While test assistant pulls on test bed with the supplied force gauge, verify the previously calibrated scale force. In doing so, further ensure that the Data Acquisition Host Interface graphically displays the force being applied to the engine.</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>14</td>
<td>Load and Execute the Data Acquisition Host</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>15</td>
<td>Initiate NO2 Flow</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>16</td>
<td>Initiate Ignition</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>17</td>
<td>Open Valve Controller</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>18</td>
<td>Initiate Stop Watch</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>19</td>
<td>Close Valve Controller</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>20</td>
<td>Shutoff Stop Watch</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>21</td>
<td>Using the supplied stop watch, calculate Engine Run Time.</td>
<td>Time input (seconds)</td>
<td></td>
</tr>
<tr>
<td>22</td>
<td>Stop Data Acquisition Host Interface</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>23</td>
<td>Ensure quality non-corrupted acquisition results. Export MS Access table to MS Excel for computing.</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>24</td>
<td>After the mandatory 60 minute waiting period, follow preceding procedures for breakdown.</td>
<td>PASS/FAIL</td>
<td></td>
</tr>
<tr>
<td>25</td>
<td>Using supplied scale, weigh NO2 tank</td>
<td>Input Weight (lb, gm)</td>
<td></td>
</tr>
<tr>
<td>26</td>
<td>Using procedure steps 10 and 25, calculate the difference in weight.</td>
<td>Weight Difference (lb, gm)</td>
<td></td>
</tr>
</tbody>
</table>

**CAUTION! ENGINE IS RUNNING**

**NOTE:** Rocket Engine should NOT be running.

**NOTE:** There is a mandatory 60 minute waiting period before test personnel are allowed to touch the Rocket Engine.

**NOTE:** Serious injury can result if warning is not heeded.
### Table

<table>
<thead>
<tr>
<th>Step</th>
<th>Description</th>
<th>Note</th>
</tr>
</thead>
<tbody>
<tr>
<td>27</td>
<td>Using procedure steps 21 and 26 calculate the average oxidizer mass flow rate.</td>
<td>$m_{\text{oxidizer}} ,(lb/\text{sec})$</td>
</tr>
<tr>
<td>28</td>
<td>Completely disassemble rocket engine, and take photographs.</td>
<td>PASS/FAIL</td>
</tr>
<tr>
<td>29</td>
<td>While rocket engine is dismantled, scour the entire engine assembly using denatured alcohol.</td>
<td>PASS/FAIL</td>
</tr>
</tbody>
</table>

**NOTE:** The following steps should be completed 24hrs AFTER Rocket Ignition
DATA ACQUISITION SOFTWARE CODE

Option Compare Database

Public presrow As Long    'present row to fill
Public numrows  As Long   'number of rows
Public T As Double
Public numchn As Integer  'number of channels
Dim buffer(10000) As Integer   'must be big enough to handle incoming device data
Dim dbs As Database
Dim sql As String
Dim table_name As String   'every time the program is run, and new table will be
created with today's date and time
Dim A(10000) As Integer
Public SampleTime As Double
Dim Comport As Integer
Dim Baudrate As Integer
Dim Device As Integer
Dim Device_String_1 As String
Dim Device_String_2 As String
Private Sub cmdStart_Click()
    Comport = TxtComport
    Device = 148
    Baudrate = TxtBaud
    Device_String_1 = "com" & Comport
Device_String_2 = Device_String_1 & " " & Device & " " & Baudrate

    presrow = 1  'init first row

    numrows = 10000000  'init total rows to write to

    With DataqSdk0

        .DeviceID = Device_String_2  'sets which device to use, see

        "DataqSdkDevice.vbp" or the ActiveX help documentation (dataqxc.chm) for more info

        .ADChannelCount = cmbChannelNumber.Value  'sets ADChannelCount

        .SampleRate = 500  'allows user to select attempted SampleRate

        .EventPoint = 20  'sets EventPoint

        .MaxBurstRate = 10000  'sets MaxBurstRate

    End With

    numchn = DataqSdk0.ADChannelCount  'number of channels

    DataqSdk0.EventPoint = 1  'NewData routine will be fired every 1 sample or higher

    DataqSdk0.Start  'start acquisition device

    Txt_SampleRate = Format$(DataqSdk0.SampleRate, "0.00 Samples/sec")  'display SampleRate

    DataqSdk0.GetDataEx A(0), Count

    DQChart1.ChartEx A(0), 1, Count

    DQChart1.Ymax = 19661  'sets max y to 6 volts

    DQChart1.Ymin = 0  'sets min y to 0 volts

    Txt_Driver = DataqSdk0.DeviceDriver
Txt_DeviceID = DataqSdk0.DeviceID
Txt_Info = DataqSdk0.InfoBoardID
Txt_Firm = DataqSdk0.InfoRev
Lbl_Status.Caption = "Status - Acquiring Data"
Lbl_Status.BackColor = RGB(255, 255, 0)
Txt_SampleRate.BackColor = 12632256
Txt_Driver.BackColor = 12632256
Txt_Firm.BackColor = 12632256
Txt_DeviceID.BackColor = 12632256

'create a new table to record data into, create a new row for each channel
(ADChannelCount)

table_name = Format(Now, "ddd") & Format(Now, "mmm") & Format(Now, "d") & 
"_" & Format(Now, "Hh_Nn_SsAM/PM")

'create a new column for each channel, type varchar

Select Case numchn

Case 1: sql_rows = "channel_1 varchar, Sample_time varchar"

Case 2: sql_rows = "channel_1 varchar, channel_2 varchar, Sample_time varchar"

Case 3: sql_rows = "channel_1 varchar, channel_2 varchar, channel_3 varchar, 
Sample_time varchar"

End Select

Set dbs = CurrentDb

'create the new table and set the primary key as column 1, auto-increment integer
sql = "CREATE TABLE " & table_name & "( row_id int," & sql_rows & ")"

dbs.Execute (sql)

sql = "CREATE UNIQUE INDEX index_row ON table_name (row_id)"

End Sub

Private Sub cmdStop_Click()

    DataqSdk0.Stop 'stop acquisition device

    Lbl_Status.Caption = "Status - Terminated"

    Lbl_Status.BackColor = RGB(255, 0, 0)

End Sub

Private Sub DataqSdk0_NewData(ByVal Count As Integer)

    Dim i As Integer

    i = 0

    DQChart1.Chart (DataqSdk0.GetData) 'display the data in Chart1

    If Count > 10000 Then Count = 10000 'to prevent buffer overflow

    Count = Count - Count Mod numchn 'count evenly divisible by numchn

    DataqSdk0.GetData

    DataqSdk0.GetDataEx buffer(0), Count 'put the data into the array

    Set dbs = CurrentDb 'set the current database

    For i = 0 To Count - 1 Step numchn

        sql = "INSERT INTO " & table_name & " VALUES(" & presrow

        For j = 0 To numchn - 1

            sql = sql & "," & Format((buffer(i + j) And &HFFFC) / 32768 * 10, "0.00")

        Next j

        sql = sql & ")"

        Set presrow = presrow + 1

    Next i

Next j
'Adds time to data bas

T = (presrow - 1) / DataQSdk0.SampleRate

sql = sql & "," & T

sql = sql & "")"

dbs.Execute (sql) 'fill database columns with acquired data

r = r + 1

presrow = presrow + 1

If presrow > numrows Then

presrow = 1 'continuous mode

End If

Next i

End Sub

Private Sub Form_Load()

MsgBox ("Make sure you check the Communications Port Number & The Communications Port Baud Rate")

End Sub
## Table 5 - Bill of Material

<table>
<thead>
<tr>
<th>Drawing Level</th>
<th>Part No.</th>
<th>Description</th>
<th>UOM</th>
<th>QTY</th>
<th>Vendor</th>
<th>Cost</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>26-05-01-001</td>
<td>Hybrid Rocket Assembly</td>
<td>EA</td>
<td>1</td>
<td>Metters</td>
<td>$437.60</td>
</tr>
<tr>
<td>2</td>
<td>9452K226</td>
<td>Buna N O-Ring ASS68A Dash Number 238</td>
<td>EA</td>
<td>2</td>
<td>McMaster</td>
<td>$11.78</td>
</tr>
<tr>
<td>2</td>
<td>92196A542</td>
<td>18-8 Stainless Steel Socket Head Cap Screw 1/4&quot;-20 Thread, 1&quot; Length</td>
<td>EA</td>
<td>16</td>
<td>McMaster</td>
<td>$13.47</td>
</tr>
<tr>
<td>2</td>
<td>90101A230</td>
<td>18-8 SS Hex Thin Nylon-Insert Locknut 1/4&quot;-20</td>
<td>EA</td>
<td>16</td>
<td>McMaster</td>
<td>$5.66</td>
</tr>
<tr>
<td>2</td>
<td>92196A267</td>
<td>18-8 Stainless Steel Socket Head Cap Screw 10-32 Thread, 3/8&quot; Length</td>
<td>EA</td>
<td>2</td>
<td>McMaster</td>
<td>$6.69</td>
</tr>
<tr>
<td>2</td>
<td>26-05-01-003</td>
<td>Nozzle, Graphite</td>
<td>EA</td>
<td>2</td>
<td>Metters</td>
<td>$150.00</td>
</tr>
<tr>
<td>2</td>
<td>26-05-01-005</td>
<td>Bulkhead, Aft Nozzle Mounting</td>
<td>EA</td>
<td>1</td>
<td>Metters</td>
<td>$50.00</td>
</tr>
<tr>
<td>2</td>
<td>26-05-01-007</td>
<td>Bulkhead, Forward</td>
<td>EA</td>
<td>1</td>
<td>Metters</td>
<td>$50.00</td>
</tr>
<tr>
<td>2</td>
<td>N/A</td>
<td>Load Cell</td>
<td>EA</td>
<td>1</td>
<td>Ebay</td>
<td>$61.00</td>
</tr>
<tr>
<td>2</td>
<td>N/A</td>
<td>Valve, Solenoid, High Pressure</td>
<td>EA</td>
<td>1</td>
<td>SkyCraft</td>
<td>$50.00</td>
</tr>
<tr>
<td>2</td>
<td>N/A</td>
<td>Pressure Transducer</td>
<td>EA</td>
<td>1</td>
<td>Aerocon</td>
<td>$100.00</td>
</tr>
<tr>
<td>1</td>
<td>26-05-01-001</td>
<td>Amplification Box - Two Channel</td>
<td>EA</td>
<td>1</td>
<td>N/A</td>
<td>$151.27</td>
</tr>
<tr>
<td>2</td>
<td>N/A</td>
<td>Integrated Circuit, Instrument Amplifier</td>
<td>EA</td>
<td>2</td>
<td>Newark</td>
<td>$10.00</td>
</tr>
<tr>
<td>2</td>
<td>270-1806</td>
<td>Project Box</td>
<td>EA</td>
<td>1</td>
<td>Radio Shack</td>
<td>$4.99</td>
</tr>
<tr>
<td>2</td>
<td>272-135</td>
<td>.1 micro-Farad Capacitor</td>
<td>EA</td>
<td>2</td>
<td>Radio Shack</td>
<td>$1.29</td>
</tr>
<tr>
<td>2</td>
<td>276-1998</td>
<td>16 pin Dip sockets</td>
<td>EA</td>
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<td>Radio Shack</td>
<td>$1.29</td>
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<tr>
<td>Drawing Level</td>
<td>Part No.</td>
<td>Description</td>
<td>UOM</td>
<td>QTY</td>
<td>Vendor</td>
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<td>-----</td>
<td>-------------</td>
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<tr>
<td>2</td>
<td>276-1499</td>
<td>Circuit Board</td>
<td>EA</td>
<td>1</td>
<td>Radio Shack</td>
<td>$4.29</td>
</tr>
<tr>
<td>2</td>
<td>276-1535</td>
<td>Etchant</td>
<td>EA</td>
<td>1</td>
<td>Radio Shack</td>
<td>$4.29</td>
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<tr>
<td>2</td>
<td>23-875</td>
<td>9 Volt Battery</td>
<td>EA</td>
<td>2</td>
<td>Radio Shack</td>
<td>$6.58</td>
</tr>
<tr>
<td>2</td>
<td>276-011</td>
<td>Led, Bulkhead Mount</td>
<td>EA</td>
<td>1</td>
<td>Radio Shack</td>
<td>$2.59</td>
</tr>
<tr>
<td>2</td>
<td>271-1321</td>
<td>1kOhm Resistor Hook-up Wire, 22 AWG, Solid Core, Multicolor</td>
<td>EA</td>
<td>1</td>
<td>Radio Shack</td>
<td>$0.99</td>
</tr>
<tr>
<td>2</td>
<td>278-1221</td>
<td>1kOhm Potentiometer w/ Knobs</td>
<td>EA</td>
<td>2</td>
<td>SkyCraft</td>
<td>$10.00</td>
</tr>
<tr>
<td>2</td>
<td>275-601</td>
<td>SPST Switch</td>
<td>EA</td>
<td>1</td>
<td>Radio Shack</td>
<td>$4.99</td>
</tr>
<tr>
<td>2</td>
<td>276-195</td>
<td>Standoffs, Circuit Board Non-Conductive</td>
<td>EA</td>
<td>1</td>
<td>Radio Shack</td>
<td>$1.59</td>
</tr>
<tr>
<td>2</td>
<td>270-325</td>
<td>Battery Connector, 9 Volt</td>
<td>EA</td>
<td>6</td>
<td>Radio Shack</td>
<td>$1.99</td>
</tr>
<tr>
<td>2</td>
<td>29-05-01-003</td>
<td>1 kOhm Potentiometer w/ Knobs</td>
<td>EA</td>
<td>2</td>
<td>SkyCraft</td>
<td>$10.00</td>
</tr>
<tr>
<td>2</td>
<td>N/A</td>
<td>Extender Cable, USB</td>
<td>EA</td>
<td>5</td>
<td>Ebay</td>
<td>$20.00</td>
</tr>
<tr>
<td>2</td>
<td>276-1538</td>
<td>D-subminiature, 9way, female connectors</td>
<td>EA</td>
<td>10</td>
<td>Radio Shack</td>
<td>$15.90</td>
</tr>
<tr>
<td>2</td>
<td>DI-148U</td>
<td>Data Acquisition Box</td>
<td>EA</td>
<td>1</td>
<td>Dataq</td>
<td>$55.00</td>
</tr>
</tbody>
</table>

| Total         |           |                                                                              |     |     |             | $1,346.12 |
1. INTERPRET PER ASME Y14.5–2001

2. MATERIAL: 3/80 IN. THICK 304 STAINLESS STEEL PLATE OR EQUIVALENT.

3. MATERIAL: 3.50 IN. O.D., 3.10 IN. I.D. 304 STAINLESS STEEL PIPE OR EQUIVALENT.

4. UTILIZING FN 5 AND FN 8, CREATE THE SHOWN FUEL CROSS SECTION WITH A 2.0 IN. DOWEL ROD AS A MAIN REL.
Error! Objects cannot be created from editing field codes.

Figure 22 – Amplification Box, 2 Channels; DWG 29-05-01 Sheet 2
ROCKET ASSEMBLY

Figure 23 – Assembly, Hybrid Rocket; DWG 26-05-01 Sheet 1
Figure 24 – Assembly, Hybrid Rocket; DWG 26-05-01 Sheet 2
Figure 25 – Assembly, Hybrid Rocket; DWG 26-05-01 Sheet 3
Figure 26 – Assembly, Hybrid Rocket; DWG 26-05-01 Sheet 4
Figure 27 – Assembly, Hybrid Rocket; DWG 26-05-01 Sheet 5
ROCKET CHAMBER ASSEMBLY

Figure 28 – Assembly, Chamber; DWG 27-05-01 Sheet 1
Figure 29 – Assembly, Chamber; DWG 27-05-01 Sheet 2
Figure 30 – Assembly, Chamber; DWG 27-05-01 Sheet 3
Figure 31 – Assembly, Chamber; DWG 27-05-01 Sheet 4
<table>
<thead>
<tr>
<th>Chamber Pressure (Pa)</th>
<th>Adiabatic Flame Temperature (Kelvin)</th>
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</thead>
<tbody>
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</tr>
<tr>
<td>50</td>
<td>245,204</td>
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<tr>
<td>100</td>
<td>285,990</td>
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<td>700</td>
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<td>750</td>
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Table 6 - Chamber Temperature Equilibrium Calculation
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<td>Ratio of</td>
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<td>Specific Heat</td>
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</table>

Table 7 - Ratio of Specific Heat Equilibrium Calculation
Table 8 - Molecular Weight Equilibrium Calculation

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<th>Chamber Pressure (ps)</th>
<th>Molecular Weight kg/kmol</th>
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<tr>
<td>500</td>
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</tr>
</tbody>
</table>

... (Continued)
6.0 LIST OF REFERENCES


