DESIGN AND TEST OF A LAB-SCALE 
N$_2$O/HTPB HYBRID ROCKET

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Abstract
The MaCH-SR1 project involved the design of a 1000-lb thrust hybrid rocket with the effort being undertaken during two semesters by 8 senior aerospace engineering students at the University of Colorado at Boulder. The 2002-2003 team was the second year for the project, as the previous team built two static engines and performed test burns on both. The hybrid implemented a liquid O$_2$/HTPB oxidizer/fuel combination, but this year’s team switched to a N$_2$O/HTPB combination due to the oxidizer’s simplicity and safety. Four lab-scale chambers were constructed and hot-fired in order to determine the fuel regression rate, data acquisition, and to verify the overall design for the 1000-lb thrust hybrid rocket chamber. The full-scale rocket is to be test fired in the spring of 2003 and data will be obtained for the thrust, chamber and feed pressures, and the instantaneous oxidizer mass flow rate. The lab-scale portion of the project resulted in a well-documented design and test program, which laid the foundation for future teams to build larger and more powerful engines, leading to the eventual goal of CU’s first student built sounding rocket.

Introduction
The MaCH-SR1 project aims at building a sounding rocket to deliver a 10-lb payload to the edge of space, about 125 km above the Earth’s surface. A single stage hybrid design has been chosen for the task, since it is safer than either a purely liquid or solid rocket. It is true that the hybrid has decreased performance than a liquid engine, but is less complicated and actually performs better than solid rockets. The hybrid’s solid fuel will only burn in the presence of a liquid oxidizer, which can be throttled to vary thrust. A solid rocket has both the fuel and oxidizer mixed in a solid fuel grain, and once burning begins, it cannot be controlled until burnout of the engine. A liquid engine uses highly flammable liquids, which pose safety and storage problems in addition to added complexity of the feed system.

Designing a hybrid rocket is not an easy undertaking; therefore the project was divided into various subsystems, with each team concentrating on the subsystem/s of interest. The fuel team mixed and cast the fuel into the chambers; the feed system team acquired parts for the gas feed and related plumbing; the injector team designed and machined injector plates; the nozzle and chamber team optimized the nozzle for altitude and built the chambers to withstand thermal and structural stresses; the data acquisition team focused on the sensors recording data; and the system integration lead made sure all the components fit and worked together.

The full scale, 1000-lb thrust design required the construction of small, lab-scale chambers to give the group an idea of what the design process for a
hybrid engine involved, and to provide a learning curve. In the process, the regression rate for the fuel/oxidizer combination was determined, data was acquired of the chamber and feed pressures, and the effort involved in successfully testing the engines was realized.

**Design**

Most of the design work for the lab-scale rocket was done in SolidWorks, a popular 3D CAD/Solid Modeling package. In addition, analysis of the chamber properties was carried out in Cosmos/Works, an add-on module that is an integrated mesher/solver/post-processing software package.

**Performance**

The main consideration for the chamber design was thrust. Once this quantity was decided upon, other performance parameters could be determined. For the thermodynamic parameters, GDL ProPEP from the former Martin Marietta Corporation (Lockheed Martin) was used to provide initial estimates of the combustion products mass, ratio of specific heats, and specific impulse. An equation summary is presented below:

Coefficient of Thrust:

\[ C_F = \sqrt{\left(\frac{2}{k+1}\right)^{2k/(k-1)} + \left(\frac{2}{k+1}\right)^{k/(k-1)}} \cdot \frac{p_a}{p_c} \]  \hspace{1cm} (1)

Impulse: \( Ft_b \) \hspace{1cm} (2)

Specific Impulse: \( \frac{C_F c^*}{g_0} \) \hspace{1cm} (3)

Characteristic exhaust velocity: \( c^* = \frac{c}{C_F} \) \hspace{1cm} (4)

Propellant mass flow rate: \( m = \frac{p_c A_t}{c^*} \) \hspace{1cm} (5)

Fuel mass: \( m_f = \frac{m_p}{MR + 1} \) \hspace{1cm} (6)

In the above equations, \( k \) is the ratio of specific heats \( c_p/c_v \), \( p_a \) is atmospheric pressure while \( p_c \) is chamber pressure, and \( c \) is the effective exhaust velocity. In addition, \( A_t \) is the throat cross-sectional area, \( m_p \) is the total propellant mass, and \( MR \) is the oxidizer to fuel mixture ratio. All parameters were determined at a chamber pressure of 500 psi. The calculations were also performed under the assumption of frozen flow through the nozzle to approximate the low end of performance. In reality, performance is between frozen and shifting flow assumptions, since the combustion products change composition during expulsion from the nozzle, leading to slightly higher performance than frozen flow.

Knowing the fuel and oxidizer combination, a graph of the specific impulse could be made as a function of the mixture ratio. Figure 1 shows the specific impulse as a function of mixture ratio, created with GDL ProPEP.

![Figure 1 – Theoretical Performance](image)

**Nozzle**

The nozzle was designed for a single altitude, as the engines are being static tested and do not need to be optimized for a flight trajectory. All the testing is being conducted in Denver, Colorado, so the design atmospheric pressure is 12.1 psi. A converging nozzle implementing a straight plug section and a 45° converging section was used. The nozzle throat section was gradually rounded to provide a smooth transition for gas flow and to minimize erosive burning. Graphite was chosen as the material of choice, since it is very tolerant of high temperatures due to its low thermal expansion and high thermal conductivity, in addition to having been successfully used in past nozzles. Graphite’s ablative properties are also attractive for the application since some of the heat is taken away during operation of the chamber. A standard plug design was chosen because other similar lab-scale engines also employed this method, and measuring thrust was not a goal for the lab-scales. To maintain chamber pressure, this design sufficed and kept the design simple. The following equations in detail the nozzle design:

Nozzle throat area: \( A_t = \frac{F}{C_F p_c} \) \hspace{1cm} (7)

Nozzle exit velocity: \( V_e = \frac{F}{m} \) \hspace{1cm} (8)

where \( F \) is the thrust exerted by the chamber. The resulting converging nozzle is illustrated in Figure 2.
Fuel

Hydroxyl-Terminated Poly-Butadiene (HTPB) was the primary component of the fuel, the exact formula having been worked out by the previous year’s team. Other ingredients included Isophorone Diisocyanate (IPDI), Carbon Black, and Castor Oil. The approximate component percentages are shown in Table 1.

<table>
<thead>
<tr>
<th></th>
<th>Percentage</th>
</tr>
</thead>
<tbody>
<tr>
<td>HTPB</td>
<td>84.57%</td>
</tr>
<tr>
<td>IPDI</td>
<td>10.34%</td>
</tr>
<tr>
<td>Carbon Black</td>
<td>0.09%</td>
</tr>
<tr>
<td>Castor Oil</td>
<td>5.00%</td>
</tr>
<tr>
<td>Total</td>
<td>100.00%</td>
</tr>
</tbody>
</table>

The fuel port was sized by the oxidizer mass flux, with the length determined by preliminary regression rate exponents and the desired oxidizer to fuel mixture ratio. Due to the uncertainty of the exponents, a safety factor of 2 was employed for the worst-case regression rate in the design equations. This and other equations governing the fuel grain are listed below:

Fuel regression rate: \( \dot{r} = aG_{ox}^n \) (9)

Fuel port length: \( L = \frac{m_f}{N} \frac{r_0}{2\pi R \rho f} \) (10)

The above equations show \( a \) and \( n \) as the regression rate exponents, \( G_{ox} \) as the oxidizer mass flux, \( R \) the fuel port radius, \( \rho \) as the fuel density, and \( N \) as the number of fuel ports. Due to simplicity and available tubing sizes, the fuel port diameter was chosen as 1.01 inches, with the port length being about 9.25 inches. The remaining parameters were determined accordingly.

Combustion Chamber

The combustion chamber was designed to be a pressure vessel capable of withstanding the thermal and dynamic stresses placed upon it during firing. Due to the unknown nature of this particular hybrid, stainless steel 304 tubing was used for the lab-scale chambers, with a .125-inch wall. The length of fuel cast inside, plus a 0.5 length to diameter ratio for both the pre and post combustor sections, determined the chamber’s length. The fuel is designed to act as insulation during each burn due to its material properties. However, the pre and post combustors were both exposed to the steel of the chamber. Therefore, phenolic spacers were designed to form an ablative barrier between the wall and the hot gases undergoing combustion. Glass phenolic has a very low thermal conductivity, and is therefore well suited for insulation. The lab-scale chamber diameter was chosen to provide a significant factor of safety for short-duration burns and to also minimize the cost of components. To calculate the failure modes and safety factors of the chambers, the following equations were employed:

Tangential Stress: \( \sigma_t = \frac{r_t^2 p_i}{r_0^2 - r_i^2} \left( 1 + \frac{r_i^2}{r_t^2} \right) \) (11)

Radial Stress: \( \sigma_r = \frac{r_r^2 p_i}{r_0^2 - r_i^2} \left( 1 - \frac{r_i^2}{r_r^2} \right) \) (12)

Allowable working pressure: \( p = \frac{2t\sigma_{r,\text{max}}}{d_i + t} \) (13)

Bolt Stress: \( \sigma_{\text{bolt}} = \frac{F_b}{A_f} \) (14)

The calculations showed that even with a 1000-psi pressure spike in the chamber, a reasonable factor of safety is maintained. Pressure spikes are an important consideration, since a badly designed rocket will suffer from combustion instabilities, resulting in pressure spikes and possible failure. Table 2 lists the failure scenarios of the lab-scale chamber.

<table>
<thead>
<tr>
<th>Failure</th>
<th>( p_\text{f} ) (psi)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Tangential</td>
<td>3328</td>
</tr>
<tr>
<td>Radial</td>
<td>40000</td>
</tr>
<tr>
<td>Bolt</td>
<td>44763</td>
</tr>
<tr>
<td>Seal</td>
<td>4732</td>
</tr>
</tbody>
</table>

A thermal analysis was carried out of the lab-scale assembly assuming the fuel port burn area reached halfway to the chamber, and the thickness was found to be adequate to contain combustion. An area of concern however was the nozzle, but with the addition of EPDM insulation, the system was capable of being safely operated.

Ignition

The ignition of the rocket chambers was decided upon after doing research on other similar hybrids, and which method was the most reliable and simplest. The idea of preheating HTPB by applying...
current to steel wool inside the fuel port seemed like an attractive option. Electrical leads were loosely wrapped with steel wool, which was then placed snugly up the fuel port. The leads were then connected to a car battery. The pre-combustion chamber was then filled up with gaseous oxygen for several seconds to ensure an oxygen-rich environment. Then current was applied across the leads, causing the wool to burn and heating the surrounding HTPB fuel sufficiently for combustion to occur. In all, four tests were conducted to validate this ignition method, with success in every trial.

**Injector**

The injector plate was designed to provide the desired flow rates into the thrust chamber at the same time turning the liquid oxidizer into gas (atomization). For this purpose, the injector holes were ordered in a radial showerhead pattern to distribute the flow evenly across the fuel port. Separate holes were added to accommodate gaseous oxygen feeds and pressure transducers. A cavity was designed in the center of the plate to provide a seat for a pipe nipple, which would hook to the feed system. The following equations\(^1\) drove the design:

\[
\dot{m} = C_d A_i \sqrt{2 \rho \Delta p} \quad (15)
\]

\[
V_o = C_d \frac{2 \Delta p}{\rho} \quad (16)
\]

Where  \(\rho\) is the oxidizer density, \(\Delta p\) is the pressure drop, \(A_i\) is the total injection hole cross-sectional area, and \(C_d\) is the discharge coefficient. Injector test pieces were tested with water to determine the mass flow rate and the resulting pressure drop. Two lab-scale injectors were designed, for 0.15 kg/s and 0.25 kg/s. The following diagram in Figure 3 illustrates the injector pattern with varying hole sizes that was tested.

![Figure 3 - Lab-Scale Injector Pattern](image)

The tests were carried out after attaching an injector piece to a pressurized tank assembly, which also had a gauge to measure the difference between the tank and atmospheric pressures. See Figure 4 for the injector test.

Figure 5.a shows the mass flow rate as a function of the pressure drop. The mass flow follows an almost linear behavior, allowing for extrapolation of pressure drops for higher mass flows. Figure 5.b shows the Discharge Coefficient versus pressure drop. Note the non-linearity of the \(C_d\) as the pressure drop hits 60 psi. This is a result of using two different pressure gages, one relatively accurate to 60 psi, and the other less accurate to 100 psi. This is a possible source of error for the pressure drop prediction at 0.15 and 0.25 kg/s of expected flow rate. However, the data was extrapolated to a predicted drop of 300 psi for 0.15 kg/s and 180 psi for 0.25 kg/s. Due to the measurement of \(C_d\) using two gages, no conclusive \(C_d\) value could be determined, and as a result a value of 0.65 was taken from Sutton\(^1\) for an orifice size of ~1mm (0.025 in). A second injector design had a different hole size of 0.032 inches, but hydraulic testing was not performed due to time constraints.

![Figure 4 - Injector Hydraulic Testing](image)

![Figure 5.a - Mass Flow Rate](image)
The lab-scale injector is shown below in Figure 6 and two holes are clearly seen adjacent to the main injection holes. The first hole was used for the injection of gaseous oxygen to facilitate ignition of the fuel, and the second hole was for the pressure transducer fitting.

![Figure 6 - Lab-Scale Injector Plate](image)

The injector pattern itself was decided to be a simple radial showerhead design with 12 holes of similar size spread out on two bolt circles with diameters of 0.2 and 0.4 inches, adequate for clearance on a standard 1–inch pipe nipple, and coverage of the fuel port as well.

**Feed System**

The feed system for the static firings was composed of a series of valves, pressure regulators, transducers, check valves, and stainless steel tubing. The feed system provided oxygen for the ignition, oxidizer for the thrust chamber during firing, and Nitrogen to purge the system after each test. The main concern was the pressure drop across the valves and the injector. To design for the chamber pressure, the properties of Nitrous Oxide (N₂O) were examined at the expected temperatures of testing. The graph in Figure 7 shows the liquid density of N₂O at a range of temperatures.

![Figure 7 - Properties of N₂O](image)

Also, Figure 8.a illustrates the feed system diagram with all associated plumbing and a legend explaining the symbols.

![Figure 8.a – Feed System Schematic](image)

![Figure 8.b – Feed System Solid Model w/o Tanks](image)
Figure 8.b is a solid model of the chamber hooked up to the feed system tubing without tanks attached.

**Data Acquisition**

The sensors involved in taking measurements for the lab scale engines were two pressure transducers, one measuring the feed pressure, and the other reading the chamber pressure. In addition to the real time pressure readings, the oxidizer tanks were weighed prior to and after each firing, so the average mass flow rate could be measured. The primary data acquisition system consisted of a Hewlett Packard OmniBook laptop computer with a data acquisition card, connected to an SC-2345 Signal Conditioner. Eight channels were available for use, with the 2 pressure transducers providing real-time data. See Figure 9 for a schematic.

![Figure 9 – Data Acquisition System for Testing](image)

Voltage output from the transducers was fed into LabView, where the voltages are calibrated to correspond to different pressures. The data was then imported into an Excel spreadsheet for analysis.

**Test Stand**

The design for the lab-scale test stand was basically driven by the need to just clamp the chamber down during firing, preventing any movement. Originally, the design called for a uni-strut tube configuration, with the chamber bolted on with tube fasteners. To interface between the chamber and the test stand, a metal plate was designed to not only adhere to the contour of the chamber bottom in a horizontal fashion, but to also bolt down onto the top of the test stand. See picture in Figure 10 for initial design.

![Figure 10 – Initial Test Stand Design](image)

Figure 11 shows the culmination of the design effort in a lab-scale hybrid rocket chamber. The cross-sectional view allows all parts of the system to be seen.

![Figure 11 – Final Chamber Design](image)

**Construction**

**Nozzle**

The nozzle was machined out of medium grade graphite, with relatively large grain size. A single 3-inch rod of graphite was cut into four pieces, representing each lab-scale nozzle. After mounting on the lathe, each piece was faced off and the inner contour was milled out. Using heavy grit sandpaper, the throat was rounded. Then a splitter was used to cut the finished nozzle from the graphite stock. Since graphite is a hazardous material to work with, respirators had to be worn during machining. See Figure 12.

![Figure 12 – Nozzle Machining](image)

**Fuel**

Preliminary testing of the fuel mixtures in small amounts was done to determine the best mandrill release agent. After several variations, it was determined that saran wrap was the best medium between the fuel and the mandrill. The casting involved the construction of fuel plugs, which provided the sealing of the chamber bottom and insured the mandrill...
stayed in place. Two lab-scale engines were cured at the same time, after fuel was poured into each and the distance from the fuel top to the chamber top was measured. The first batch of two chambers unfortunately contained the wrong mixture ratios of the fuel and had to be scrubbed out using Methyl Ethyl Ketone (MEK). The next batches were successful, curing in the ovens for about 4 days at 165°F. It was discovered at the testing site however, that the fuel in the chambers cracked longitudinally along the fuel port, due to the high temperatures during curing, which placed considerable stress into the fuel. It was decided that for the full scale, the curing temperatures would be around 135°F for a week or more. Primary obstacles included casting the fuel without including air bubbles in the mixture and finding a suitable means of keeping the fuel surface smooth during curing. The saran wrap had a slight tendency to wrinkle, no matter how tightly it was wrapped around the mandrills. Due to the high bonding strength of the fuel, the mandrills had to be taken out using an industrial press, and the fuel port was sufficiently smooth and round.

**Combustion Chamber**

The chambers for the lab-scale engines were built by purchasing 304 stainless steel tubing, which was cut into sections to provide 4 engines. The chambers had Con-Flanges welded onto the ends, allowing for a seal interface for the injector on one end, and the nozzle on the other. The welding was done on a turntable using a TIG welder. The flanges were eyeballed to make sure they were perpendicular to the tube. First they were spot welded, then turned upright, so the full welding between the Con-flange and the steel tube could be finished. Pressure testing was performed on one chamber, using a hydraulic pump to provide the pressure. Pressures of 1500 psi were reached, validating the safety of the welding technique and providing for an operating pressure safety factor of 3 (based on a 500-psi chamber pressure).

**Injector**

The injector plates were machined out of solid Con-Flat blanks, eliminating a significant amount of machining time, since the bolt holes were already in place. First, a small seat for the nipple hookup to the feed system was machined into the center of the injector. For the lab-scale, the injector holes were very small (0.025 and 0.032 in diameter), requiring EDM (Electron Discharge Machining) to be implemented. Sealing of the injector plates to the chamber was accomplished using copper gaskets crushed between Con-Flat flanges. The gaskets placed provided a metal-metal seal, ensuring a leak free system.

**Ignition**

The ignition system required minimal construction, as the main components were readily available. This included a car battery, some electrical leads, and steel wool, which were sold in packages available at the local hardware store.

**Feed System**

The feed system involved the purchase of numerous fittings. The lab-scale involved 0.25 inch tubing for the Nitrous Oxide since the flow rates were not significant. The gaseous Oxygen and Nitrogen used 0.125 tubing and were connected in the system as shown in Figure 8.a. All the gases were bought from a local supplier and the necessary safety precautions were observed in addition to completion of the needed paperwork with the University of Colorado. Solenoid valves from the previous year’s team were used, and regulators were placed onto the Nitrogen, Helium, and gaseous Oxygen feeds. The N₂O flow was regulated with a Helium head pressurized to 1400 psi, and thus, no regulator was needed for the oxidizer. See Figure 13 for the tank setup.

**Test Stand**

The test stand was a store bought half-shelf unit or (Gorilla Rack) that provided adequate support. The wooden planks were screwed into the metal girders, and the girders were welded together at each joint. The metal interface plate was squared on the mill, and the CNC mill then cut out the contour on the aluminum plate. See Figure 14 for a view of the test stand with one of the hybrid rocket chambers attached.
Testing

Testing of the lab-scale rocket chambers took place at Pioneer Astronautics in Lakewood, Colorado during the 2002/03 winter break. The test coordinators at the facility were Mark Caviezel and Gary Snyder. Their facility had a test cell that has been previously used for other rocket engine firings, and the unique concrete-bunker style of the cell was adequate for the lab-scale firings. See Figure 15.

Mission Control was set up inside the Pioneer Astronautics building with a window providing a link to the test cell by accommodating the wiring and other cables for the various electronics. Two video cameras were used: one recording the test fires (Chamber Cam), and the other was focused on the regulator and pressure gauge (Gauge Cam) in the feed system to verify transducer measurements. The pressure transducer readings were initially set for a 1000 samples/second, but for the later firings, a sampling rate of 250 samples/second was deemed adequate due to excessive data points. See Figure 16 for pictures of Mission Control and the Chamber Setup in the test cell.

Four Lab-Scale chambers were tested and video of each firing along with pictures of the final two burns were obtained. See Figure 17.

The equipment setup and testing was completed over a period of 5 days in late December and early January, during which time all four chambers were tested, with three providing sufficient data to move onto the full-scale design. The team was satisfied with the progress made in the fall semester that led to the successful testing of the hybrid rocket chambers and the team took a moment to pose for a group picture in Figure 18.
Results

The objective of the testing was to fire four identical chambers, but with two different mass flow rates and two different burn times. This would allow the solving of simultaneous regression rate equations, thus determining the regression rate exponents, a and n.

Initial difficulty with the electrical system, the ignition system, and the N₂O solenoid valve delayed testing. The issue with the ignition involved the steel wool having been too densely packed around the electrical leads, thus when current was applied, the spark was spread out over a much larger surface area. This problem was resolved by hooking up the leads to a 120-Volt Variac, which provided instantaneous spark capability, as opposed to the 12-Volt car battery tried earlier. The problem with the solenoid valve initially was that it would freeze open after actuation. It wasn’t discovered until later that the solenoid used DC current and the switch in the control box was AC. As a result, the first chamber firing lasted longer than the planned 5 seconds, burning fuel for a full 27 seconds before reaching the chamber wall at the nozzle end, and then burning for another 21 seconds a mixture of stainless steel and residual fuel upstream of the nozzle. After depletion of the fuel, the oxidizer tank was manually turned off and Nitrogen purged the system for about a minute. Upon inspection, the chamber was thoroughly cleaned of fuel and the worst-case scenario was illustrated without any chamber failure, to the relief of the team. Unfortunately, the pressure readings were abruptly stopped 8 seconds after ignition due to confusion among the team as to why the chamber didn’t stop firing after the planned 5 seconds. In addition, the Chamber Cam was wrapped with bubble wrap to provide stability on a slightly damaged tripod, and during the first burn the bubble wrap melted, causing the camera to tilt down after about 15 seconds. The Gauge Cam however was used to determine the total time the system was pressurized, thus allowing for an approximate determination of the burn time and the feed and chamber pressures as well. Disconnecting the solenoid wire from the switch and manually putting it on a car battery terminal resolved the valve problem. The remaining three tests were carried out without any problems. A summary of the test results versus what was planned is presented in Table 3.

Table 3 – Test Summary

<table>
<thead>
<tr>
<th>Chamber</th>
<th>D</th>
<th>C</th>
<th>B</th>
<th>A</th>
</tr>
</thead>
<tbody>
<tr>
<td>Burn Time (s)</td>
<td>10</td>
<td>5</td>
<td>5</td>
<td>10</td>
</tr>
<tr>
<td>Oxidizer Flow Rate (kg/s)</td>
<td>0.15</td>
<td>0.15</td>
<td>0.25</td>
<td>0.25</td>
</tr>
<tr>
<td>Average Mixture Ratio</td>
<td>6</td>
<td>6</td>
<td>6</td>
<td>6</td>
</tr>
<tr>
<td>Average Chamber P. (psi)</td>
<td>500</td>
<td>500</td>
<td>500</td>
<td>500</td>
</tr>
<tr>
<td>Oxidizer Flow Rate (kg/s)</td>
<td>0.15</td>
<td>0.21</td>
<td>0.17</td>
<td>0.16</td>
</tr>
<tr>
<td>Average Mixture Ratio</td>
<td>5.7</td>
<td>4.6</td>
<td>4.8</td>
<td>5.2</td>
</tr>
<tr>
<td>Average Chamber P. (psi)</td>
<td>193</td>
<td>280</td>
<td>280</td>
<td>253</td>
</tr>
</tbody>
</table>

Even though the results for the burn times and mass flow rates were off from design values, the average regression rate for each chamber was still determined. Since the first burn (Chamber D) was unreliable for various reasons, corresponding data was not used in the fuel regression analysis. See Table 4 for a summary and Figure 19 for the observed regression curve.

Table 4 – Regression Results

<table>
<thead>
<tr>
<th>Chamber</th>
<th>r-dot (in/s)</th>
<th>L (in)</th>
<th>G₂₉₂₅ (lbm/s-in²)</th>
<th>G</th>
</tr>
</thead>
<tbody>
<tr>
<td>C</td>
<td>0.0675</td>
<td>9.174</td>
<td>0.578</td>
<td>0.879</td>
</tr>
<tr>
<td>B</td>
<td>0.0548</td>
<td>9.325</td>
<td>0.468</td>
<td>0.55</td>
</tr>
<tr>
<td>A</td>
<td>0.0406</td>
<td>9.475</td>
<td>0.44</td>
<td>0.517</td>
</tr>
</tbody>
</table>

Figure 19 – HTPB Regression Curve

The regression curve in Figure 19 illustrates that as the oxidizer mass flux increased along the fuel port, so did the average regression rate of the fuel. This was as expected, based on other hybrid results presented in
Sutton and Larson. Putting the results of Table 4 into matrix form and solving for a 1000-lb thrust, 500-psi chamber pressure hybrid design, the following parameters were determined in Table 5.

Table 5 – Full-Scale Hybrid Regression Parameters

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>a</td>
<td>0.1160</td>
</tr>
<tr>
<td>N</td>
<td>0.9874</td>
</tr>
</tbody>
</table>

Table 6 shows that the maximum chamber pressures attained in each firing did not exceed about 300 psi, not quite the 500 psi originally designed for.

Table 6 – Tank Pressures

<table>
<thead>
<tr>
<th>Chamber</th>
<th>D</th>
<th>C</th>
<th>B</th>
<th>A</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max Tank Press. (psi)</td>
<td>1000</td>
<td>1400</td>
<td>1400</td>
<td>1400</td>
</tr>
<tr>
<td>Max Chamber Press. (psi)</td>
<td>220</td>
<td>297</td>
<td>305</td>
<td>312</td>
</tr>
<tr>
<td>Max Feed Sys. Press.(psi)</td>
<td>774</td>
<td>1131</td>
<td>781</td>
<td>792</td>
</tr>
</tbody>
</table>

The following graphs in Figure 20 present the feed system and chamber pressure graphs as recorded by the pressure transducers on the system. Each graph is presented in the order that it was recorded.

The data is cut off for the first chamber firing as mentioned earlier, but the extrapolated data lines based on the Gauge Cam overlap the transducer data and complete the data set. The chamber pressure was seen as peaking around 774 psi, which was lower than the 1000 psi in the oxidizer tank. This was the result of a more than 200-psi pressure drop through the solenoid valves and tubing leading up to the transducer in the feed system. The CGA-326 fitting on the N₂O bottle is also partly responsible for the drop, since the orifice size on the fitting is only 0.096 inches in diameter. The pressure on the chamber transducer recorded a peak value of 220-psi, indicating a large loss through the injector. This was because the injector inlet may already have seen gas due to the expansion of the oxidizer from 0.25 inch tubing to a 1-inch diameter by 4-inch long pipe section. This was unanticipated and will be taken into account for the full-scale design. This loss is on top of the restricting orifices of the injectors contributing another ~300-psi in losses. The 0.032-inch orifices were not as restrictive as losses vs. between the highest chamber pressures were reached with that injector hole size.
An initial concern the team had was combustion instability, as some hybrids have experienced this in past experiments. But according to the graphs in Figure 20, the pressure profiles are relatively smooth curves, with no adverse pressure spikes present. Also note in the graphs that as the burn progresses, the pressures decay gradually in the feed system. This is due to the fact that the Helium head on the N$_2$O produced essentially a blow-down system that will naturally decay as gas expelled over time. The initial plan called for a regulator on the N$_2$O tank, but one couldn’t be obtained for the needed pressure and flow rates to be experienced during testing.

**Conclusion**

The fall semester of ASEN 4018: Senior Design Projects came to a successful conclusion for the MaCH-SR1 team and their lab-scale hybrid rocket effort. The complete engineering process was carried out in the semester, starting with research and moving through design, construction, and testing of actual hardware. The main goal of characterizing the oxidizer/fuel combination was achieved, in addition to exposing the group to all aspects of a team-oriented project. A lesson of primary importance was the integration of various subsystems and how they interface with each other. Not everything went according to plan, but now a more thorough understanding of a complex system such as a rocket propulsion system has been gained. The completion of this phase of the project allowed the team to go ahead with the full-scale 1000-lb thrust hybrid engine with experience already gained for the task.

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**References**


