INVESTIGATIONAL HYBRID ROCKET ENGINE DESIGN
FOR RIT M.E.T.E.O.R. ORBITAL LAUNCH SYSTEMS

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ABSTRACT

Hybrid rocket motors are, by far, the safest method of propelling a rocket in terms of handling and controlability. The Rochester Institute of Technology Microsystems Engineering and Technology for the Exploration of Outer Regions (METEOR) project, in seeking a low cost and safe low-orbit satellite launch system, has begun design of a simple hybrid rocket motor with capabilities tailored to their unique airborne launch approach. The preliminary design presented here is intended for safe ground testing, with design goals that would allow the engine to be easily altered for a high-altitude balloon launched system. Due to the simplicity of manufacture and safety of testing, this hybrid rocket design is perfect for any higher learning institution that wishes to begin development of hybrid rocket motors.

INTRODUCTION

The Rochester Institute of Technology METEOR Rocket project is currently in its first phase, which consists of designing an N₂O/HTPB hybrid rocket for ground testing. It is intended to have thrust specifications similar to those that would be used on the final stage of an orbital launch. The 2006 Rocket Design Team is working on the program through the Rochester Institute of Technology multi-disciplinary senior design project and is composed of 4th and 5th year Mechanical engineering students who are enrolled in Senior Design I and II during the Winter 2005-02 and Spring 2005-03 quarters.

The objectives incorporated into this phase of the design project include overall hybrid rocket layout, fuel grain design, oxidizer feed system, rocket nozzle design, ignition system, test stand, instrumentation and data acquisition. The main objective of this project is to gain experience with hybrid rocket engines. Future work will include optimization of the fuel grain pattern and oxidizer injection system, as well as design and production of a truly isentropic exit nozzle. The temperature and pressure data obtained will also be utilized to design a combustion chamber suitable for orbital flight.

METEOR OVERVIEW
Project METEOR is a long term project devoted to making low-orbital satellite development a more economically feasible endeavor for universities and smaller businesses alike. Current launch methods for small satellites are prohibitively expensive and require long lead times. By launching these "pico-satellites" from a high-altitude helium filled balloon, it is proposed that these miniature satellites can be launched at much lower costs, with little lead time, and under much safer conditions [1]. Figure (1) shows a typical METEOR launch profile.

![Figure 1: Typical METEOR Launch Profile](image1)

Benefits of this type of system include the ability to launch without considerable ground infrastructure as well as near vacuum conditions for the duration of powered flight. Because the rocket is launched from around 100,000 feet, where the atmosphere is roughly 1% of that at sea-level, the exit nozzle sustain near isentropic operation without geometry changes, and aerodynamic drag is negligible.

**ROCKET ENGINE LAYOUT**

While the ground test engine did not need to incorporate nitrous oxide tanks into the motor itself, preparations have been made for the eventual integration of light weight oxidizer tanks for a flight configuration. The rocket design team has decided that four oxidizer tanks should be mounted around the circumference of the combustion chamber, as shown in figure (2). This not only decreases the overall length of the rocket engine, but also allows for the possibility to some day preheat the oxidizer tanks using excess heat from the chamber itself. Preheating would increase oxidizer tank pressure which would, in turn, allow higher combustion chamber pressures, thereby increasing thrust and specific impulse.

![Figure 2: Oxidizer Tank Configuration](image2)

The increase in frontal area created by this configuration is not a concern for us, because the rocket will launch from near vacuum and experience little atmospheric drag.

**INJECTOR PLATE/COMBUSTION CHAMBER**

The injector plate was designed and built to provide an interface between the feed system and the combustion chamber, to provide a seal at the front of the combustion chamber, to provide sufficient strength for the stresses seen during operation and to prepare the oxidizer to best react with the fuel grain.

![Figure 3: Nine Hole Injector Pattern](image3)

The feed system connects to the injector plate using a 3/8" NPT thread at the center of the injector plate. The injector plate is made from 304 Stainless steel because of its appealing strength properties and ability to withstand volatile environments such as the inside of a combustion chamber. Because of the high pressure we expect to see in the chamber (550 psi), we
need to make sure there is no leakage between the injector plate and the combustion chamber. This has been accomplished by putting two o-rings on the portion of the injector plate that slips into the combustion chamber.

Once the desired oxidizer flow rate of 0.217 kg/s was determined we were able to figure out the number and size of holes we needed for the oxidizer to pass through. We decided to machine two circular patterns of holes to produce a spray pattern similar to that of a shower head. This will help to atomize the oxidizer, which will make the burning faster and more efficient. One injector plate was machined with 4 holes of 0.070” diameter and one was machined with 9 holes of 0.048” diameter. The 9 hole injector pattern seems to have atomized the oxidizer flow more efficiently to increase thrust output.

The combustion chamber was made of 304 Stainless steel tubing with ½” wall thickness. It was important for our testing that all components to be placed in the chamber (ceramic rings for pre- and post-combustion chambers, nozzle, injector plate and fuel grain) fit snugly, with no gaps between the outer diameter of the component and the inner diameter of the chamber (see figure (4)). If there were gaps we may experience hot-gasing or burning on the outside of the fuel grain which would damage the integrity of the chamber walls and cause a catastrophic failure in a composite chamber. To ensure this fit, we bored out approximately 5” in length of the inside of the chamber to a perfect round for the nozzle and post-combustion chamber and approximately 3” in length for the injector plate and pre-combustion chamber. Doing this we obtained a surface that was circular to .002”. The fuel grain is fairly flexible and it took the shape of the non-bored portion of the combustion chamber. Additional assurance against burning on the outside of the fuel grain was obtained by tightly sandwiching all components inside of the fuel chamber axially.

The outside of the combustion chamber was turned down slightly on a lathe until the chamber was also a perfect round. This operation was important to make sure that all bearings would be perfectly parallel and co-linear with one another once the test stand was fully assembled. This rounding operation was done only where the chamber was to be clamped into place by our “pillow blocks”.

OXIDIZER FEED SYSTEM

The selection and design of the feed system for ground testing was a critical component of the overall rocket design. Safety was the primary consideration when determining proper selection of components and required system redundancies. A major safety concern was the need to ensure that if electrical power was lost that the valve controlling the Nitrous Oxide flow would close. This then led to the selection of a normally closed solenoid valve (Coax KB 15) to provide on / off flow capabilities. Another concern is ensuring that there is a way to extinguish any combustion in the chamber immediately in the event of a mishap. To allow this, a nitrogen purge system was added to the ground test system. The last major concern was of an over pressurization and subsequent explosion of either the feed system lines or the combustion chamber itself. To prevent this from happening relief valves were installed in both the nitrogen supply line and on the rocket injector (the Nitrous Oxide tanks could not reach high enough pressure to damage oxidizer feed system components).
An initial desire for the feed system was the ability to manually adjust the flow rate between tests. An adjustable needle valve was designed into the system to allow this adjustment. The user could select the required position of the valve based on oxidizer tank pressure, feed system losses and required injector inlet pressures as shown in the formula below. This pressure loss can then be looked up on a plot of needle valve position and desired flow rate.

\[ \Delta P_{\text{needle}} = P_{\text{tank}} - P_{\text{losses}} - P_{\text{injector}} \]

Figure 6: METEOR Feed System

Figure 7: Actual METEOR Feed System

The possibility of contamination is a concern of all aerospace propellant feed systems. To prevent any contamination from damaging valve seat surfaces or clogging the oxidizer injector holes a number of filters were installed just prior to these components to capture any of these harmful particles. In addition to these filters, much care was taken prior to assembling the feed system. All components were cleaned in an ultrasonic with IPA (Isopropyl Alcohol). As a result of the cleaning, many potential sources of contamination were eliminated.

**IGNITION SYSTEM**

The ignition system consists of approximately ¼ inch of 40 gauge, high-resistance nickel chromium (NiChrom) wire. This wire will glow red hot at approximately 700 °F with approximately 3-4V and .5-.7Amps. The NiChrom wire is then dipped in a mixture of 70% pyrodex and 30% nitrocellulose laquer. Pyrodex is a high explosive black powder substitute that can be ignited at relatively low temperatures by the NiChrome wire. The nitrocellulose laquer binds the pyrodex together so that it can be attached to the NiChrome. After allowing the pyrodex to dry, this so called “match-head” is molded into a ring of 66% ammonium perchlorate and 33% HTPB, as shown in figure 8. Ammonium perchlorate is an oxidizer that allows the HTPB to burn without the presence of the N₂O oxidizer. The HTPB/Oxidizer mix is easily ignited by the pyrodex and burns at a much slower rate in order to heat up the combustion chamber to a sufficient temperature to dissociate the oxygen from the nitrous oxide.

Figure 8: Igniter Assembly

Figure 9: Complete Igniter Assembly
Using an Ammonium Perchlorate and HTPB mixture for the ignition system of the rocket allows for ignition in any atmosphere. This is important if we hope to one day launch our rocket from a balloon tethered platform at 100,000 feet which can be considered oxygen free. In addition this method of ignition is safer than placing a small tank of oxygen onboard the rocket or using a tank of oxygen for ground testing because of its high combustibility.

**EXIT NOZZLE**

Due to the time and manufacturing constraints imposed upon our team, design of a true isentropic exhaust nozzle was deemed to be infeasible. We were instead forced to construct nozzles with a conical diverging section. Despite this large compromise, we were able to achieve an exhaust velocity of approximately 2080 m/s during ground tests. Further design work would allow for considerably higher exhaust velocity and a resulting increase in specific impulse.

The nozzle itself was machined from 3 inch graphite rod stock. This was done on a hand operated engine lathe with the half angles for the converging and diverging sections being 30 and 15 degrees, respectively. The intake section was further refined by adding a smooth contour over the entire surface.

The nozzle geometry was designed using isentropic flow relations with an estimated specific heat ratio of 1.20. With a mass flow rate of 0.235 kg/s, the optimal sea-level throat and exit diameters were found to be 1.078 and 2.572 cm, respectively. This would result in a theoretical maximum exit Mach number of 2.88.

The nozzle was held inside of the chamber assembly using a steel snap ring, as shown in figure 11.

**TEST STAND DESIGN**

The test stand for our hybrid rocket engine is a low friction slide rail that pushes into a cantilevered beam load cell. The main components of the test stand are the aluminum ground precision plate, rail slides, rail connectors, pillow blocks, infrared sensor mounts, and calibration system. Each rail connector is attached to two linear bearings that glide along freely on their individual rail slides. The pillow blocks are mounted on these rail connectors. The test chamber is placed on the bottom pillow block and the top pillow block is bolted down on top of the test chamber to clamp our engine securely into place. The combustion chamber and pillow block surfaces were both made circular to the same diameter to achieve a firm grip. In addition, this makes certain that the rocket is held straight and not firing at an angle. The top and bottom faces of the pillow blocks were ground to ensure the rocket was perfectly level when firing to prevent flawed load cell readings. Attached to the front rail connector is a threaded rod, which connects to our load cell that is mounted at the end of slide rails. To calibrate our load cell, we developed a pulley weighting system. Through a hook adapter we place in the injector plate, a cable is run through the pulley system and weights are placed on the circular disk as seen in the picture below. This forces the test chamber to move forward into the load cell and we can accurately calibrate our load cell before testing. The infrared sensor mounts are a threaded rod clamping system; which gives us the freedom to move the infrared sensors in any direction necessary.
Each component of the test stand assembly gives us the ability to test different size chambers in the future with minimal alterations to the actual test stand. Future teams will want to test rockets of various sizes and diameters to try and achieve different levels of thrust and burn times. This test stand incorporates these plans and will allow the team to test a rocket up to 5 feet long and a diameter of one to two feet.

DATA ACQUISITION

In order to measure the required test results we had to use the software/hardware from DATAQ Instruments. Each sensor was hooked up to a channel on one of three DATAQ data acquisition blocks. The blocks were then attached to a 100 foot cable that feed out to 3 laptops at the teams location, where each sensor was recorded and monitored during the testing.

The sensors that were used to acquire the required test results were:
- 3 pressure transducers (0-1000 psig)
- 4 infrared sensors (0-400 F)
- 1 high infrared sensor (300-1000 F)
- 1 heat flux/thermocouple sensor
- 1 load cell (0-500 lbs)

The required measurements included:
- Pressure for:
  - Nitrous Oxide tank
  - Just before injector plate
- Temperature for:
  - Surface of the chamber at 4 points, extending from the nozzle to the pre-combustion chamber
  - An additional high temperature infrared sensor was placed at the nozzle
- Thrust for:
  - Load rocket produces
- Heat flux for:
  - Determining inside chamber surface temperature and environment

Each sensor measured at a sampling rate of 240 Hertz divided by 4 channels per DATAQ data acquisition block. Therefore, readings were recorded every 1/60 second.

RESULTS

Feed System Test Data:

Performance data as shown below in Figure 13 show an almost 300 psi pressure drop from the oxidizer tanks to the inlet of the injector. This extremely high pressure loss resulted in a lower than desired combustion chamber pressure of 227 psig. To lower the losses through the feed system, a needle valve that was used to restrict flow was removed from the system.

![Figure 13: Test #1 Pressure Data](image)

As the data below in Figure 14 shows, removing the feed system to only 120 psi. This lower pressure loss allowed a higher combustion chamber pressure of 403 psig. This almost double combustion chamber pressure contributed to an almost doubling of thrust created by the rocket.
Figure 14: Test #2 Pressure Data

Figure 15 shows the thrust data obtained using an 11 inch fuel grain. The maximum thrust fell well short of our design value of 600 Newtons or 134 pounds. This is mainly due to the low chamber pressures resulting from inadequate oxidizer flow. The apparent fluctuation in thrust values was not due to a pulsation of the rocket, but rather to simple electrical noise through our load cell amplification circuit.

Figure 15: Eleven Inch Fuel Grain Thrust Curve

Figure 16 shows thrust measurements from a more successful rocket firing. The thrust averaged to approximately 110 pounds, over 80% of our target thrust. Considering the simplicity of the nozzle profile, this seems to be acceptable for preliminary testing.

Figure 16: Eighteen Inch Fuel Grain Thrust Curve

Figure 17: Eighteen Inch Fuel Grain Flam Plume

CONCLUSION

With further design modifications an HTPB/N₂O hybrid rocket should be capable of achieving necessary specific impulse and thrust requirements necessary for low earth orbit. The redundant weight of the rocket would have to be reduced far beyond the ground test equipment used for our testing for any orbital missions. As with any space mission, this remains to be the biggest hurdle to tackle.

The safety of hybrid rocket engines makes them an appealing candidate for university projects. Through the METEOR project, RIT will demonstrate the attainability of space flight for private institutions. The data presented here, while still somewhat rudimentary, lays the groundwork for future senior design teams to attempt sub-orbital and eventually orbital launches. While the project is very ambitious, to say the least, the mere fact that it is being conducted brings educational opportunities to RIT that would have been previously unimaginable.
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REFERENCES