Design of 'N' Size Rocket For ESRA Rocket Competition to Launch Truman I Rocket to 10,000 ft. AGL

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ABSTRACT

A rocket has been designed by to take a functional 8 lb payload to 10,000 ft. Students at the University of Missouri constructed the rocket with a 7.5 inch body primarily from fiberglass. The design focuses on reliability and simplicity. Simulations place the rocket's apogee at around 11,000 ft, which is likely to produce altitudes closer to 10,000 ft in actual launch in non ideal situations. It uses 4.5 in motor tube filled with ammonium perchlorate propellant. The rocket recovers using a redundant altimeter and timer design. Both fire independent black powder charges that eject a 10 ft parachute at apogee. The 8 lb payload consists of an array of electronics packages that include a homemade and retail accelerometer, a thermal couple, and a GPS locating device.

1. INTRODUCTION

The AIAA University of Missouri Student Branch elected to pursue the 3rd Annual ESRA Rocket Competition. The competition is sponsored by ESRA (the Experimental Sounding Rocket Assoc.) and requires students to design, construct, and launch an experimental payload closest to 10,000 ft AGL (above ground level) while excelling in the areas of safety and professionalism as demonstrated through this paper and an oral presentation. This paper presents the design and manufacture of the entire Truman I rocket.

2. Preliminary Design

Before any construction was begun, it was necessary to understand what type of force and duration would be needed to launch certain size and weight rockets to 10,000 ft AGL. This was done through the use of a commercially available software package Rocksim. This software has a preloaded database of common rocket parts and components as well as thrust profiles for common commercial motors. The weights were varied from 60-100 lbs while the aerodynamic profile varied from a 4" diameter body tube to an 11" diameter body tube. During each simulation different commercial motors were placed in the simulated rocket and the altitude at apogee was recorded. This process was repeated until all variations were covered. The recorded altitudes were then compared and a design goal was chosen.

The preliminary design of the rocket consisted of an 11" body tube diameter, 12' tall, powered by an 'N' motor whose average thrust was approximately 625 lbs for 5.2 seconds. This design was then discussed with the local rocketry club. The team was then informed that from experience an 11" rocket powered by an 'N' motor could only reach about 8,000 ft. They then told us however that if the body tube was reduced to 7.5" the altitude goal of 10,000 ft was achievable on an 'N' motor. The local rocketry club also said they had some spare 7.5" body tube that we could have. This discussion and donated material then changed our preliminary design. We also decided to increase our design specific impulse from the previous 3,300 $lbf \cdot s$ to approximately 3,600 $lbf \cdot s$. This was decided upon in case of any extra weight that we were not accounting for. It is always easier to add weight than to take it away was the theory behind this design change. In order to reach this goal we desired an average thrust of 900 lbs for approximately 4 seconds.

3. PROPULSION SYSTEM

3.1 Motor Casing

The motor casing chamber is used as a container to hold and burn one or multiple propellant grains under conditions of high pressure and temperature. In general, motor cases are cylindrical and slender in shape. This allows for a decreased cost of machining to the material and due to the shape, a reduction in the overall aerodynamic drag during flight of the rocket body.

The motor casing chamber size is first determined by the volume of the propellant grain required to reach a desired specific impulse. In our design, the desired average thrust output is 900 *lbf* with a desired burn time of approximately 4 *seconds*. To reach the desired specific impulse, out motor casing needed to be 42 *inches* in length which allows for a six-grain motor. Therefore, the total impulse was found using

$$I_t = FI_{sp} \tag{1}$$

Knowing this criterion, an N-size motor was chosen due to its range of total impulse being 2300 to 4600 $lbf \cdot s$.

Once the motor size was determined, a motor case material had to be chosen that could adequately withstand the high internal chamber pressure and temperature during the rocket's burn time. Taking into account material cost and availability, Aluminum 6061 was chosen as our motor case material due to its high ultimate tensile strength and high melting point.

Another critical factor is determining a reasonable and safe wall thickness for the motor tube to prevent any catastrophic events from happening during firing. First a safety factor of 1.5 is taken into account with regards to the internal chamber pressure. Aluminum's 6061 yield strength σ was used in conjunction with the maximum chamber pressure and the motor casing's internal dimensions to find the minimum wall thickness t of 0.06324 *inches* the equation

$$\sigma = \frac{Pr}{t} \tag{2}$$

where r is the internal chamber radius. Due to our high interest in safety and the fact that it was readily available, our final motor casing wall thickness was chosen to be 3/8 *inches* and therefore doubles our safety factor of 1.5 for the wall thickness.

Other than high pressure, during the firing of the rocket, there is an issue of high combustion temperature reaching the motor casing wall. For thermal protection, the placement of high temperature silicon is put in between every grain. Following, every grain is placed inside phenolic tubing which carries a high thermal resistance. Finally, large amounts of lithium grease is applied on the outside of the phenolic tubing and inserted into the motor casing.

Future possibilities for the motor casing are endless. There is a broad range of materials available to use for rocket motor casings today. They range from certain metals like aluminum, steel or titanium which offer a favorable ratio of mechanical strength to density, all the way to the modern rocket motors which are made of composite materials. There are a vast number of composite materials that have mechanical properties superior to those of metals. The only downfall is cost of composite materials are quite high and they are not as readily available as the many different types of metal.

3.2 Propellant

3.2.1 Types of Propellants

Propellant is a material that burns rapidly, releasing energy to propel an object. It consists of combined chemicals located inside of the motor case to create a reactive force propelling the rocket forward. There are six different forms of propellant available and each had its advantages and disadvantages. Some of the more common propellants used are solid, liquid and hybrid where nuclear, ionic, and electromagnetic are mainly used for interplanetary travel.

Solid propellant is the simplest of all the various types to work with, most cost effective and most commonly used for model rocketry. It contains the fuel and oxidizer mixed together prior to launch in the form a grain. Inside the grain is a core, which by the size and shape of the core determines the rocket's performance. Figure 12 shows a few different grain cross-sections available. The performance is also based on the core surface area and the specific impulse. Specific impulse is the impulse given to the rocket by the weight of propellant [5]. Ideally a high specific impulse would be desired because it burns more rapidly. The specific impulse for solid propellant ranges from 200 to 480 seconds. Another factor of performance is the shape of the core and that depends on the desired initial thrust. The advantage to solid propellant is that it can be easily stored which is great for model rocketry. The disadvantages are there is no shut off or thrust adjustment and it may not have a good a performance as other propellants [6].

Liquid propellant is a complex system comprised of pipes, valves and pumps connected to a combustion chamber which mixes and burns the fuel and oxidizer to produce thrust. Even though this system is complicated, it has its advantages. The flow of the propellant can be controlled so that the engine can be stopped, restarted or throttled [5].

Hybrid propellant has the advantages of solid and liquid propellant rockets combined. Inside the combustion chamber is the solid chemical, which is the fuel, and above it is the liquid chemical, the oxidizer. When the liquid is injected into the combustion chamber holding the solid propellant, an ignition occurs and the thrust is produced [8]. The advantages to having a hybrid propellant rocket is they provide high energy, they can be stored for longer periods of time and can be stopped and restart by simply varying the rate of liquid injected. The downfall of hybrid propellant is that they generate less energy per unit mass of propellant and the system is more complex than the solid rockets.

Nuclear, ionic and electromagnetic propellant, not used in model rocketry, is complex method and found in rockets traveling to outer space. Nuclear rocket engines have been studied for delivering heat to a working fluid producing high exit velocities. This method is similar to liquid propellant except the heating of the gas is due to energy being derived by transformations in the nucleus of the atom. While this method seems interesting, it is not fully developed and not yet used today [7].

3.2.2 Solid Rocket Fuel

Solid propellant was chosen for Truman I because of its cost efficiency, simplicity and time constraints. There are many different types of fuel that can be used for solid propulsion. Because we were working with the two members of the Columbia Rocketry Club, we decided to use Ammonium Perchlorate for our fuel since that is what they were familiar with using.

3.2.3 Grain Core Profile

The grain core profile as mentioned above can affect the thrust and burn time of the solid propellant. There are many core profiles available including a hollow core, hollow core with a rod, star, along with many various others. These profiles can also be combined to provide very different burn characteristics during the burn. For our rocket a hollow core was chosen due to manufacturing constraints. When the propellant cures it seizes to the mold making it very difficult to remove. Sometimes the grains need to be twisted off of the mold. This process would not allow for any other shape than a circular profile.

The diameter of the core is also very important. If the diameter is too small erosive burning can occur creating a large pressure increase that will in the end fail the motor casing. This is obviously undesirable. A grain core diameter of 1.5" was chosen because it has been proven to work in a 6-grain 'N' motor by the Columbia Rocketry Club.

3.3 Nozzle

3.3.1 Nozzle Contour

The nozzle is essentially the heart of any propulsion system. The nozzle determines how a rocket will perform by channeling exhaust gases from combusted propellant through an orifice to increase the velocity, consequently creating thrust which propels the rocket.

Initial designs considered the use of one of two types of nozzles. The first type is a bell-shaped nozzle; the second type is a conical nozzle. Both types were examined to determine the best choice. Bell shaped nozzles are capable of producing good thrust while preventing flow separation. Also, because of their contour, bell shaped nozzles can be made shorter, and thus lighter. However, bell shaped nozzles do not perform as well at lower altitudes (below 40,000 ft.) as shown in Fig. 1 In addition, shaped nozzles require complex gas dynamics analysis to design them.

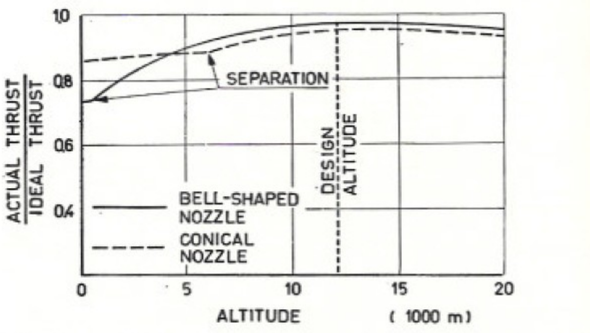


Fig. 1. Ratio of actual thrust to ideal thrust of bell-shaped and conical nozzle as a function of altitude. [1]

Conical nozzles also produce good thrust, but due to their contour are prone to flow separation at higher altitudes. Since the diverging portion of a conical nozzle is shaped, it must be made longer in order to prevent flow separation. Conical nozzles, however are easily designed using simple isentropic equations and perform better at low altitudes than bell shaped nozzles as shown above in Fig. 1.

A third type of nozzle, a plug nozzle, was briefly considered as well. Plug nozzles, or expansion deflection nozzles provide an alternate way to prevent thrust loss caused by overexpansion. It was quickly decided, however that this was not a practical design for this project due to the fact that plug nozzles are only efficient at very high speeds and at very high altitudes.

Several factors were considered when selecting a nozzle design, primarily design altitude and time constraints. The design altitude for the competition is 10,000 ft. AGL (above ground level.) At this altitude, a conical is more efficient than a bell shaped nozzle. Secondly, the time constraint for this project does not allow time to learn the gas dynamics needed to design a bell shaped nozzle.

The final decision was to use a conical nozzle, based on the facts that it is the most efficient design at a low altitude, and it can be easily designed. Once this decision was reached, the next step was to design the nozzle itself.

3.3.2 Nozzle Design

To produce a successful experimental nozzle, the design must be performed analytically first. For the analytical nozzle design, the nozzle was assumed at ideal conditions. Ideal conditions entail the following: Pressure at the exit of the nozzle, P_e, is equal to the ambient pressure, P_a, secondly that the Mach number at the throat, M_t is equal to 1, thirdly that the gases expanding through the nozzle behave as ideal gases, and finally that the flow through the nozzle is 1-dimensional.

The most important parameter of a nozzle design is the throat diameter, D_t. The throat is the intersection of the converging and diverging portions of a nozzle. The diameter of the throat determines how fast or slow the exhaust gases exit the rocket. Making the throat diameter too small can cause over pressurization of the chamber, which can lead to a motor case rupture causing a catastrophic failure of the rocket. On the other hand, opening the throat too wide will cause the motor to be under pressurized, which will bring about a decrease in thrust and overall efficiency of the rocket. So it is critical that the throat diameter be the proper size to ensure that the rocket fires properly. The throat diameter was numerically calculated to be 1.15” in diameter.

Once we have the physical dimensions of the nozzle, a material must be selected to manufacture the nozzle. A desirable material for the nozzle must have good thermal conductivity and a high melting point. The material should also be relatively light since the overall weight of the rocket is critical. The material should also be easily machined. After considering several materials, graphite was selected because it has a high melting temperature, it is light, it is easy to manufacture, and it is relatively inexpensive. After selecting the material, the nozzle was machined by students on a lathe.

In addition to a numerical result for the nozzle geometry, an analytic approach was used to verify the nozzle. Gambit software package were used to create the geometry and mesh of the nozzle. Fluent, a CFD (computational fluid dynamics) post-processor software package, was used to analyze the model made in Gambit. Fluent was chosen due to its wide use in industry as well as its capabilities of solving for unsteady compressible flows. A useful option within Fluent is the capability to select a Spartan-Almaratus turbulent flow model. This turbulent flow model was created for aerospace applications in particular. The energy equation mode was also selected in order to enable an ideal-gas assumption for the propellant gas.

After convergence was reached the results were analyzed for pressure, velocity, and thrust through the nozzle. Contours for pressure, velocity, and mach were analyzed with the contour plot of velocity shown in Fig. 2 as well as a plot of velocity along the line of symmetry. These results prove the validity of the model as well as the numerical results.

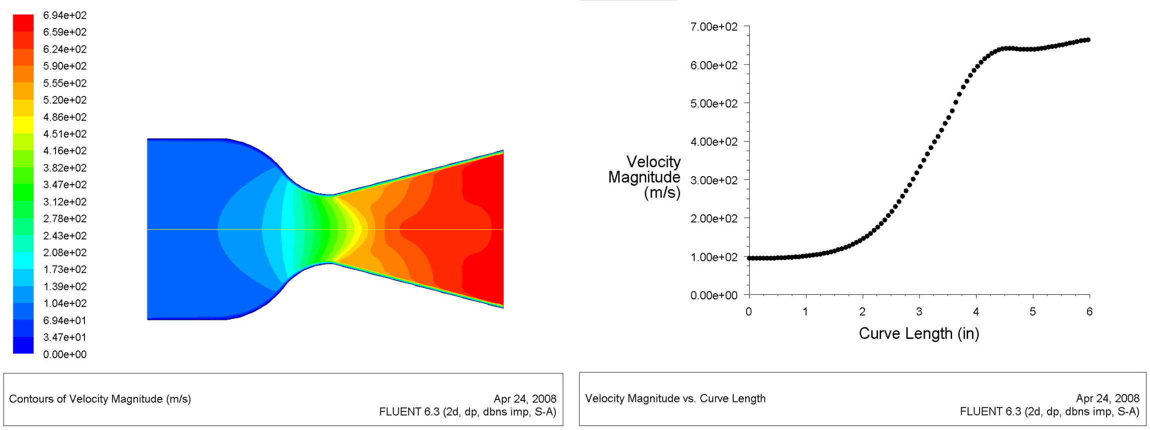


Fig. 2. Contour plot of velocity through the nozzle. (Note: velocity at nozzle walls is 0)

From the exit velocity the exit Mach number was found and the average thrust was found to be 760 lbs. This is lower than the numerical approach. This difference can be attributed to poor gas properties. Gas properties for a composite based ammonium perchlorate propellant exhaust are hard to find. If better gas properties are found, the error between the two approaches should approach zero. Overall, the analytic model validated our numerical results and the values of pressure and velocity through the nozzle.

4. AIRFRAME

4.1 Material Selection

There are a wide range of materials available for the construction of the airframe. Aluminum, fiberglass, carbon fiber, and phenolic tubing have all been used in amateur rocketry. G10 fiberglass was chosen for a number of reasons. It is lightweight, strong, easily available, cost effective, and has no government restrictions. The FAA restricts any all metallic airframes from being launch by amateurs. Carbon fiber was widely researched, but was found to be undesirable due to its high cost and limited availability.

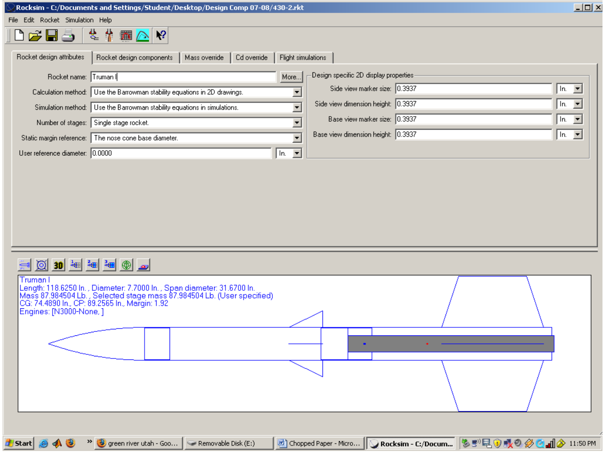
4.2 Aerodynamics

4.2.1 Profile

The aerodynamic profile of the rocket was chosen to be a 7.5” diameter body tube with an ogive nosecone with a total length of the rocket being 11 ft. This profile was chosen because the body tubes and nosecone were donated to the team and our limited budget forced us to make this profile work as well as preliminary research and discussion with the local rocketry club. Using RocSim it was found that a rocket of this size could be successfully launched to an altitude of 10,000 ft using an ‘N’ motor.

4.2.2 Fin Size and Placement

The rocket was assembled with its entire payload and assemblies along with a ‘dummy’ motor, the exact size and weight of the actual motor, and the center of gravity, *Cg*, was found. (The final ‘wet’ weight of the rocket is 88 lbs.) This data was then entered into Rocksim. Using Rocksim different fin arrays shapes, and placements were varied until a stability margin of 2 or greater was achieved as seen in Fig. 3. This value corresponds to the center of pressure, *Cp*, being located two or more body tube diameters behind the *Cg*. It was found that an array of four triangular fins along with four trapezoidal fins near the aft allowed for a stability margin of 2.05.



Rocket name: Truman I
Calculation method: Use the Barrowman stability equations in 2D drawings.
Simulation method: Use the Barrowman stability equations in simulations.
Number of stages: Single stage rocket.
Static margin reference: The nose cone base diameter.
User reference diameter: 0.0000 In.
Side view marker size: 0.3937 In.
Side view dimension height: 0.3937 In.
Base view marker size: 0.3937 In.
Base view dimension height: 0.3937 In.

Truman I
Length: 118.6250 In. , Diameter: 7.7000 In. , Span diameter: 31.6700 In.
Mass 87.984504 Lb. , Selected stage mass 87.984504 Lb. (User specified)
CG: 74.4890 In., CP: 89.2565 In., Margin: 1.92
Engines: [N3000-None,]

Fig. 3. Screenshot of Rocksim software package.

4.3 Construction

The outer body tube of the rocket is made of G10 fiberglass. G10 is common on amateur rocketry due to its lightweight and strong nature. Also, metal airframes had to be excluded due to government regulations. It is measured by its inner diameter of 7.5 in and has a wall thickness of 0.125 in. A four foot section makes up the lower body tube (housing the motor), connected to an upper four foot section by a coupler made of the same fiberglass tubing of a slightly smaller diameter, so as to fit, bolted, between the two.

Centering rings are placed between the body tube and the motor in the lower section of the rocket. These centering rings hold a dual purpose. They help distribute the load of thrust during takeoff and motor retention during recovery. They also prevent the rocket shaking within the rocket. These centering rings are made of simple ¾-inch plywood to reduce cost and maintain strength with a low weight design. The have a 7.5 in outer diameter and a 4 inch inner diameter. There are three centering rings placed throughout the lower body tube. One bulkhead, a solid wooden piece of the same outer diameter as the centering rings is placed flush above the top of the motor, contacting the motor bulkhead. These wooden rings are held in place via fiberglass cloth coated in epoxy. The cloths drape the inside of body tube, over the corners where the centering rings meet the tube. With this design, the rocket maintains less drag than bolting while ensuring a strong bond through the fiberglass.

The motor's retention upon upward thrust is handled by a load distributed to both the bottom centering ring and the contact of the motor bulkhead to the aforementioned wooden bulkhead. The motor bulkhead will be hand screwed into the wooden bulkhead via a hatch cut in the side of the rocket just before launch. On the bottom centering ring a snap ring will fit into the aluminum motor, which will sit simultaneously flush with the bottom centering ring while the motor bulkhead sits flush with top wooden bulkhead. As both thrust bearing materials are the same ¾-inch plywood with no preload, the thrust load should be similar for both. Upon recovery, the weight load will similarly apply in the opposite direction for both the bottom centering ring and the top bulkhead. When the parachute deploys during recovery, the weight of the motor casing will distribute load against the bottom centering ring and the bolt retaining the motor at the top bulkhead.

For redundancy, in case either the plywood or the epoxied fiberglass fails, the centering rings are rigidly connected to each other via ¼ inch steel allthread. Each centering ring and bulkhead is bolted and epoxied top and bottom to the allthread. This will insure that as some members encounter plastic deformation before failure, load will be distributed to the two centering rings, placed between the bottom centering ring and the top bulkhead, as well as the opposing outer centering ring or bulkhead. This improves the safety factor assembly in epoxying the fiberglass error as well as manufacturing defects in the cheap plywood.

5. RECOVERY

5.1 Parachute

With the target weight of the rocket being 80-90 lbs. and using an online parachute calculator it was found that a 10' diameter parachute would successfully recover our rocket with a descent rate of 26.84 ft/s [12]. Different materials were researched in the construction of our parachute but it was found that ripstop nylon was the most desirable for its lightweight and strength.

5.2 Separation

The separation of our rocket to eject the parachute was decided to be at the joint of the two body tubes. The separation method of blowing the nosecone off was researched; however was found to be less desirable due to the location of our payload in the nosecone. The separation is achieved by using a 4 gram black powder charge contained in a used 'D' size Estes motor ignited by an E-match. This amount of black powder was chosen with the help of online calculators. To ensure premature separation does not occur at burnout two nylon shear pins are placed through the airframe. The separation was tested on the ground to ensure the charge was sufficient to shear the pins and eject the upper body tube.

To deploy the Parachute and retain both halves of the rocket a shock cord is connected between the coupler bulkhead and a bulkhead placed midway in the upper body tube. The shock cord is connected using U-bolts and D-links. The parachute is then attached to the shock cord at about its midpoint. When the separation occurs the shock cord is extended and pulls the parachute out of the upper body tube deploying the chute. To ensure the charge does not harm the parachute a Kevlar welder's cloth is wrapped around the chute and then placed in the body tube. When deployed the cloth separates leaving the exposed chute to 'catch' air.

5.3 Altimeter & Timer-

A Perfect Flite Mini Alt W/D was chosen because it was decided to be the best altimeter out of the three possible altimeters. This altimeter was chosen because it is a precision sensor up to 25,000 ft, it has the capability to record 5 minutes of flight data at 20 samples per second which can later be downloaded to a PC/Mac, and the data is recoverable even if a power failure were to occur. A timer was installed in the electronics bay as a backup in case the altimeter fails or the separation charge it is connected to fails. The timer is connected to a separate separation charge for redundancy. The timer is set to 27 seconds which was found by adding a little over a second to the approximate time to apogee found using Rocksim and Matlab. The approximate time to apogee is 25.75 seconds.

6. PAYLOAD

6.1 GPS

A GPS system was installed in our payload to allow the tracking of the rocket for recovery. Different GPS systems were researched including a HAM radio system, a falconry beacon, and a Garmin Astro unit. In order to use a HAM radio system a user must have a permit which none of the members possessed so this option was ruled out. A falconry beacon uses a small transmitter placed inside the rocket while the user has an antennae and device which beeps when the user is headed in the right direction. This option does not display a map however showing the location of the rocket. The Garmin Astro unit was designed for hunters to place on their hunting dogs to track their dogs while out in the field. This unit was found to be ideal because the collar unit could be installed in the rocket while the handheld unit displayed the exact location of the collar unit. However the unit is very expensive and was outside of our budget, but thankfully the unit was donated to the team by Garmin.

6.2 Accelerometer

The other section of our payload consists of a 'homemade' accelerometer as well as an off the shelf accelerometer. The purpose is to compare the data of the two devices to see how accurate the 'homemade' unit is. The 'homemade' accelerometer consists of an 8 lb mass connected to a spring. The mass is then connected to a string potentiometer. The string potentiometer measures the displacement of the mass by outputting a voltage as the mass moves. The output of the potentiometer is linear as seen in Fig. 4. This displacement can then be applied to the system to solve for the acceleration of the mass through the use of the following

$$F = k \times x = m \times a \qquad (3)$$

Where k is the spring stiffness, x is the deflection of the mass, m is the mass, and a is the acceleration of the mass, while F is force which is simply F=kx for a spring without damping. In this system damping is being neglected, but a close approximation is expected. From the measured acceleration, its derivative can be found to produce velocity and altitude.

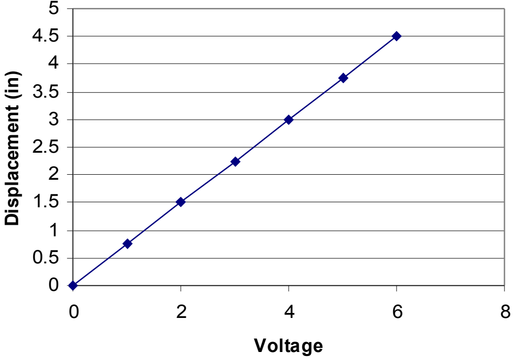


Fig. 4. Plot of Voltage vs. Displacement for linear string potentiometer.

6.3 Eagle Tree Telemetry System

A telemetry system will be utilized in the payload section as well. The maker of the telemetry system is Eagle Tree. They are well known for the RC plane systems. Some of the telemetry system components were salvaged from a previous design competition the University of Missouri AIAA branch participated in last year. This telemetry system will send and receive real time data; however it is only good up to 1 mile. However the transmitting unit has internal memory which can be recovered with the use of a USB cable. This telemetry system will record the voltage from the string potentiometer, the temperature of the nose cone, as well as the acceleration of the rocket using an Eagle Tree accelerometer which is our off the shelf unit as described in section 5.2.

7. PERFORMANCE

7.1 Rocket Engine

The rocket motor was statically test fired to ensure the construction of the motor was sound. The test fire proceeded with no mishaps and exceeded the team's expectations. The motor produced an average thrust of about 870 lbs of force with a maximum peak thrust of about 1300 lbs with a burn time of 4.16 seconds as seen in Fig. 5. This equated to a specific impulse of 3,619 $lbf \cdot s$ which was right above the target specific impulse. The maximum motor pressure was also about 700 psi which is well below the maximum allowable internal pressure. The thrust profile was inputted into Rocksim as well as a Matlab Simulink model. Both programs produced an acceptable altitude with Rocksim and the Simulink model showing an apogee altitude of around 11,000 ft. Figure 6 shows the altitude plot created in Matlab.

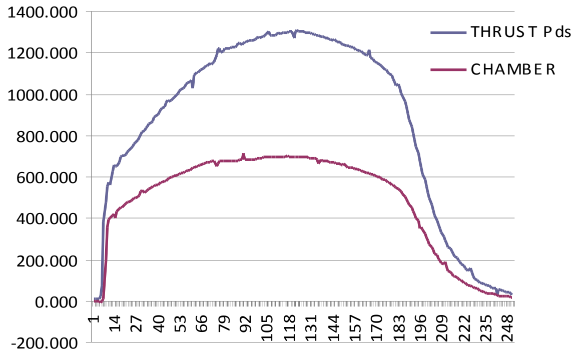


Fig. 5. Perfomance of experimental 'N' size rocket motor.

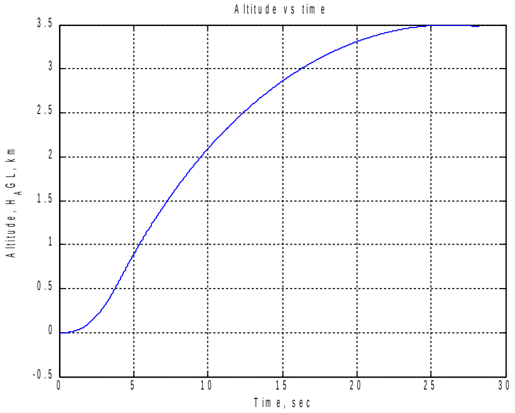


Fig. 6. Matlab Altitude plot versus time. Peak alt. of 3.5 km (11,500 ft).

8. SAFETY

When working with high explosives and solid rocket propellant safety is a major concern for not only those on the rocket team, but those in the surrounding area that could be affected by an unplanned explosion. To avoid such catastrophes many safety precautions were implemented during the construction of the propulsion system. Other safety concerns were prevalent during the construction that is not associated with the high explosives component of the project. These other safety concerns were not overshadowed and will be discussed here as well.

Mixing the propellant was one of the more dangerous and hazardous operations. To ensure that the propellant mixing and casting proceeded successfully safety, procedures were set in place and an exact mixing procedure was followed. Also all mixing of the propellant was done under the supervision of one low explosive manufacturer permit holder and one low explosives users permit (LEUP) holder. When mixing the propellant the worker was grounded by wearing a conductive strap attached to metal (similar to grounding wrist straps worn by computer technicians) to prevent a spark from igniting the mixture. Respirators were also worn to ensure none of the chemical particles were inhaled. The chemicals were mixed in an industrial Hobart bread mixer. The chemicals were added to the mixer and the operator then retreated to a safe distance of 200 ft. before turning the mixer on. This prevented any injury in case of ignition while in the mixer. After all the chemicals were mixed together the propellant was stable to handle and was then packed. The propellant was then allowed to cure and then stored in an approved explosive container on the premises of the LEOP holder.

When working with high explosives one mistake may be the last so it was important to ensure every safety step was followed. The team was updated or refreshed of the procedure about to be carried out and the safety concerns associated with that task prior to the task. This allowed the group to be on the same page, address any topics that may have been overlooked, and most importantly to hold each accountable. Emergency plans were also set in place of any emergency. A cell phone was always ensured to be readily available to call emergency crews in case of an accident. This plan has proved to be successful thus far and should be continued without slack.

9. CONCLUSION

The team has designed a rocket capable of lifting a 10 lb payload to at least 10,000 ft. Even though the rocket has not been test fired the team is confident that the rocket will perform as expected. The project allowed the team to practice knowledge learned while attending the University of Missouri. Teamwork was also tested and taught everyone how to remain flexible and patient to achieve the common goal. The lessons learned while building and testing can be applied to future engineering positions within the aerospace industry.

10. REFERENCES

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