Solid Rocket Motor Burning Rate: Experimental Proposal

ENME712: Measurement and Instrumentation Techniques for Thermal and Fluid Processes

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5/17/24

Effort Summary

Colin Scott completed:

- **Background and Motivation**
  - Conducted background research into solid rocket motors and literature surrounding previous experimentation and design optimization.
- **Objective**
  - Outlined objectives of proposal and attainable outcomes of our proposed experiment
- **Selection of Experimental Approach (split)**
  - Discussed proposed experiment procedure and addressed materials, hardware and instruments required.
- **Parameter Map**
  - Created a non-dimensional space by which to represent our experiment parameters and created a map to illustrate..
- **Design of Experimental Test Matrix**
  - Designed a two-level factorial test by selecting variables and ranges to test for a relationship and effect and created a test matrix.
- **Document Formatting**

Rafael Meza completed:

- **Executive Summary**
  - Synthesized a summary of the entire research effort to create the proposal.
- **Selection of Experimental Approach (split)**
  - Discussed proposed experiment procedure and addressed materials, hardware and instruments required.
- **Hardware Design**
  - Designed all components of the rocket body and test apparatus hardware assembly in SolidWorks.
  - Sourced commercial off-the-shelf components from McMaster Carr.
  - Worked with Protolabs manufacturing to generate quotes for machinable parts and ensure machinability of components.
  - Researched and designed the three nozzle designs used for the model rocket.
- **Calibration, Data Reduction Procedure & Uncertainty Estimates**
  - Answered the questions about likely sources of systematic error.
- **Budget and Research**
  - Compiled and reported budgeting expenses for the proposed projects.
  - Created the Gantt chart to help extrapolate costs of the test campaign over a yearly cycle.
- **Document Formatting and References**

Executive Summary

The following proposal is based on the problem statement detailed regarding "Solid Rocket Motor Burning Rate". The objective of this proposal is to produce a testing apparatus for a solid state propellant model rocket body. The research gap to be addressed is the previously understudied area of solid state propellant erosion, within the combustion chamber of the rocket and how a variety of factors affect the erosion rate. The erosion rate is directly correlated to the burning rate of the propellant and thus the thrust produced by the model rocket. In this proposal Colin Scott and Rafael Meza will outline, design, and budget a hardware test assembly capable of measuring the thrust produced, the local wall shear stress, internal chamber pressure and temperature.

The test apparatus designed consists of three sections. The design of the model rocket itself, the hardware support structure needed to run tests in a stationary assembly, and the sensor arrangement consisting of multiple distinct measurements to help describe combustion behavior. The design of the model rocket is proposed in a simple fashion. Mainly focused on material selection for body components. The pressure build up within the combustion chamber will cause a dynamic change in hoop stress and shear stress along the local rocket body wall. The rocket body itself is made up of three components including the end cap which helps funnel thrust toward the open end. At the open end of the rocket body a set of three interchangeable rocket nozzles were designed. The motivation for designing a series of nozzles is that the throat area where the flow is compressed affects the pressure increase within the chamber. Depending on the initial burning rate, the pressure build up in the combustion chamber over time can impact the rate of change of the burning rate as the full combustion of the propellant continues. The team looks to study the effects of pressure within the chamber on accelerating or extinguishing the chemical reaction driving the propellant burn. By adding sensors to measure the thrust generated and the flow exit velocity, the team will study the efficiency of the rocket with different throat areas at the nozzle. Adding a port for a piezoelectric pressure sensor will allow the team to measure the instantaneous combustion chamber pressure. Finally, using electronic strain gauges mounted on the exterior surface of the rocket body, measurements will be collected to quantify stress changes in the local wall as the combustion reaction ensues.

Controlling the solid propellant core geometry and material is an important consideration for this project. The team will use an aluminum based composite, solid propellant, specifically with an ammonium perchlorate oxidizer. Creating a cylindrical propellant core with a straight bore through the axial center of the core. NASA has explored alternative propellant core geometries in the past such as a conical bore. The standard circular bore to be used in this project provides a uniform initial surface area which eliminates the complexities of analyzing change of surface area shape as the bore burns in a conical core. The team expects the bore to expand but remain cylindrical throughout combustion.

The test campaign should produce verifiable results characterizing the behavior of propellant burn rate and the erosion of the cylindrical core as combustion happens. Once ignited the solid state rocket will burn completely. By measuring the time from ignition to finish, the team will be able to derive a propellant core erosion rate as we observe changes in thrust and exit velocity. With the knowledge that the propellant bore surface area expands as the reaction completes.

TECHNICAL PROPOSAL

Background and Motivation

The concept of a solid rocket propellant has been around for centuries. Primitive solid rocket propellant designs date back to 14th century China where “fire arrows”, tubes filled with gunpowder and an open nozzled end for gas to escape upon ignition, were used to fight the Mongols. Several other proto solid rocket propellant designs emerged throughout history, leading up to the 1942 invention of the modern U.S. composite solid rocket motor which is the foundation of solid rocket motor design we utilize today. We will be considering this modernized design for our proposal for experimentation.

Solid rocket motors operate by utilizing a solid fuel source for propulsion. In a solid rocket motor, the fuel and oxidizer are mixed together into a solid propellant. The solid propellant is then packed into a solid cylinder. The solid fuel propellant has a cavity that runs through the center of the cylinder which acts as the combustion chamber for the rocket. The solid propellant is then ignited at the area of solid fuel exposed to the combustion chamber, called the flame front. At the end of the cylinder is a nozzle where the exhaust is choked at the throat of the nozzle. The mass flow rate of the exhaust is determined by the cross-sectional of the throat. The exit velocity and exit pressure are determined by the area ratio of the throat to nozzle exit (1). The general design described above is illustrated by the figure below with the resulting thrust equation, given a free stream pressure of p_o:

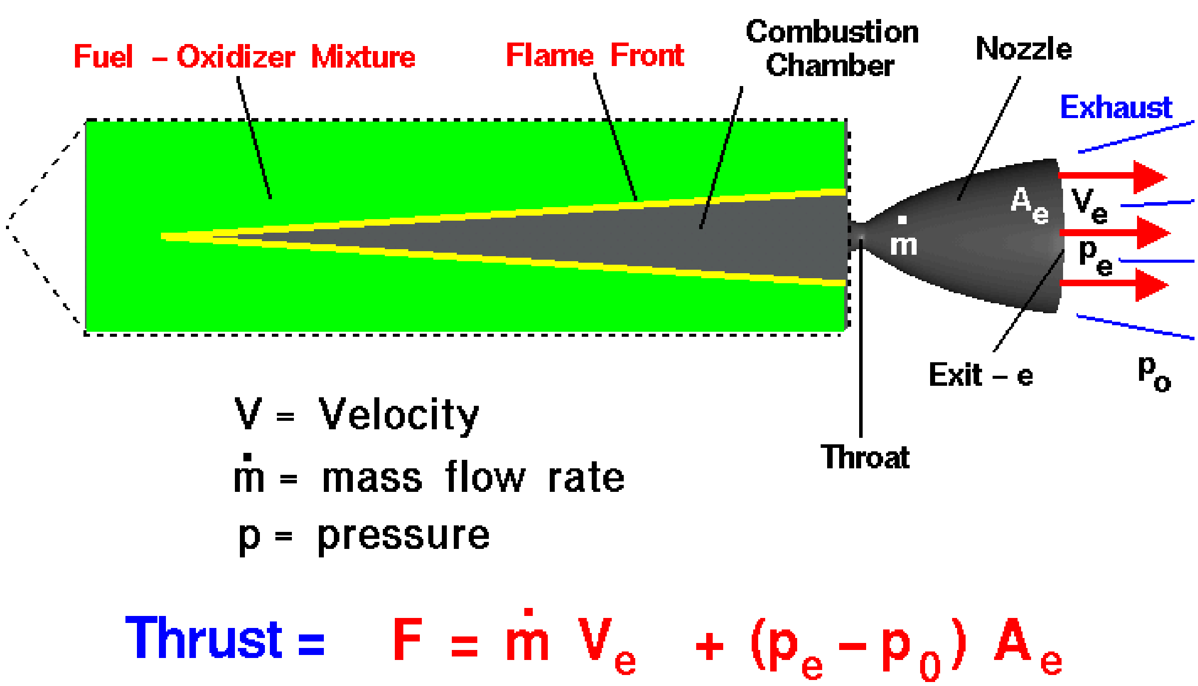

Figure 1: *Solid Rocket Motor Schematic with Thrust Equation [1]*

The design of modern solid motor rockets varies widely and is dependent upon specific mission requirements. Since the invention of their modern incarnation in 1942, solid rocket motors have seen extensive research into the optimization of their design. This research into design optimization has included research into solid propellant selection, propellant material grain geometry for optimal combustion, throat and nozzle design, and general casing design to name a few areas of research. We have found that not much research and investigation has been dedicated to the relationship between the compressible exhaust flow inside the combustion chamber and the local reaction rate and erosion of the solid propellant. We aim in this proposal to lay out an experiment that tests for the local erosion rate of the solid propellant and other physical parameters pertinent to solid fuel erosion in the solid rocket motor. These will all be fleshed out in the next section.

Objective

In the section above, we have already discussed a brief background on solid rocket motors as well as our motivation for the experimentation and investigation of solid rocket motors. In this section we are going to outline the objectives of both this proposal and the proposed experiment.

In this technical proposal, we aim to specify an experiment and its methodologies for completing the objectives later laid out in this section. This experiment specification includes facility and material requirements for our test procedure as well as required instrumentation. We aim to provide a parametric map illustrating where we plan to operate in non-dimensional space. We aim to provide a detailed hardware design for the testing apparatus with all instrumentation and a schematic. We aim to provide a 2-level factorial design test matrix for testing our objectives. Finally, we aim to discuss calibration methods for our testing procedure along with a data reduction procedure and uncertainty estimates.

In the previous section we discussed the basics of solid rocket motors, previous areas of solid rocket motor research and our desired direction for solid rocket motor investigation. In this section we would like to clearly define the objectives of our experiment. Solid rocket motor casings are made in a wide variety of sizes based upon the mission requirements and design constraints of a given mission. For our experiment we plan to create a small casing (size specified later in proposal) model to simulate solid rocket motor operation with the objective to establish a relationship between the erosion rate of solid fuel propellant to other physical parameters of solid rocket motor operation, including pressure and temperature within the combustion chamber. From this simulation with our proposed experiment, our intended measurable outcomes are the local erosion rate of the solid material, the local wall shear stress of the cylinder, and the temperature and pressure within the combustion chamber. The model will include a nozzle carbon steel pipe to act as the rocket casing, a solid propellant with a borehole for a combustion chamber and all required instrumentation. The hardware details, particular instrumentation and model scale will be specified later in this proposal.

Considering our specified parameter map as well as our experimental test matrix, we are also setting out to determine a relationship between initial flame front area, initial solid propellant mass, and throat area with the erosion rate of the solid propellant and hopefully with thrust and exhaust velocity.

Selection of Experimental Approach

Given the fundamental physical principles of solid rocket motors and the objectives for our experiment, we would like to elaborate upon our selected approach and methods for experimental testing. In this section we are going to discuss the facility requirements and hardware overview of our model, the required instrumentation for each measurable outcome, and the overall process of conducting the experiment. We will not get into hardware and instrumentation details as that will be discussed later in the proposal.

Before discussing our required testing faculties and instrumentation we would like to propose our experiment procedure. Prior to rocket simulation, we intend to weigh the modeled rocket casing to determine the weight of the rocket without propellant. Upon insertion of the solid propellant, we then intend to weigh the rocket, casing and propellant, to determine the amount of propellant that will be expended during the simulation. We intend to begin our solid rocket motor simulation by igniting our model rocket in an anchored position and beginning a timer upon ignition. We intend to let the rocket fully combust the solid propellant. During the combustion reaction, we intend to take continuous measurements of pressure and temperature within the combustion chamber, strain measurement of the casing, strain induced within the anchor to deduce thrust, and exhaust flow from the nozzle. Upon completion of the simulation, we intend to stop the timer to determine the exact combustion duration. Weight of propellant and duration of propellant combustion is essential to determining thrust and impulse.

In terms of solid propellant erosion and combustion rate, we will attain an average value at the end of the experiment by taking the known original mass/weight of the propellant and dividing by the time of the experiment. However, we are more interested in obtaining a transient solid propellant erosion rate to see how the propellant erodes over the course of the experiment and see how it relates to the other physical parameters being tested and measured. Namely, we want to see how the erosion rate of the propellant is affected by the pressure and temperature within the combustion chamber and how the thrust is affected by the changing erosion rate as well as how casing shear stress is affected by erosion rate.

We will need a testing facility capable of accommodating our desired testing procedure outlined above. We will need to adequately anchor the modeled rocket to prevent motion of the rocket and adequately measure the thrust/impulse. The impulse produced is dependent on the mass of solid propellant expended and the thrust is the impulse integrated over the time span of the rocket combustion. We will investigate the specifications of our hardware and materials later in the proposal, but assuming a cylinder size (rocket casing) of 2" diameter by 12" length, a composite solid propellant density of 1.8 g/cm^3, a combustion time span of 240 seconds, and giving a large safety factor for variations to these parameters for experimentation, we want an anchor for the casing that can withstand at least 1 kN of thrust. As we are igniting a combustion chamber that will be exhausting at high temperature, pressure and velocity, we will need a testing space that can properly accommodate that level of exhaustion.

Finally we will need proper instrumentation to conduct our experiment. We will need a scale for determining the weights of the casing and the solid propellant. We intend to plant a pressure sensor in the chamber to measure pressure and a temperature sensor to measure temperature within the chamber. We also intend to include a strain gauge on the outer wall of the chamber to determine the shear stress of the casing devoid from direct temperature influence. To measure the thrust we intend to measure the stress in the rocket anchor. To measure the flow at the outlet of the throat we intend to place a velocity flow meter and calculate the volumetric flow rate by multiplying the velocity at the throat by the cross-sectional area of the throat.

Parametric Mapping

When considering the design of our experiment we would like to consider the parameters that are going to influence the operational space with which we will be conducting our experiment. Constraints on the operational space of our experiment can come from several places, including the physical properties of the hardware under test, the ambient conditions of our testing facility, and any other design constraints we set out for our experiment.

When creating a map of our parameters, we need to consider a non-dimensional space that we intend to transform our parameters to. To map the parameter space of our experiment, we would like to transform our parameters to throat diameter to casing inner diameter ratio (d_t / D) and the initial combustion chamber volume to inner rocket casing volume ratio (V_{cc} / V_{rc}). It is to be noted that the difference between the rocket casing volume and the initial combustion chamber volume is assumed to be the initial volume of rocket propellant ($V_{rp} = V_{rc} - V_{cc}$).

With a non-dimensional space defined we can begin to transform our parameters into this space to create our operational space. The parameters which we can and intend to control for our experiment include initial propellant mass, which is a function of combustion chamber bore hole diameter assuming a uniform solid propellant density and therefore initial combustion chamber volume, and throat diameter. With a propellant density of ϱ_{rp} (we are going to assume a solid propellant density of 1.9 g/cm^3), and a bore hole in the propellant of diameter d_{cc} and a casing length of L, we can relate propellant mass to initial combustion chamber volume as:

$$m_{rp} = \rho_{rp} V_{rp} = \rho_p (V_{rc} - V_{cc})$$
$$V_{cc} = V_{rc} - \frac{m_{rp}}{\rho_p} = \frac{\pi}{4} d_{cc}^2 L$$

The casing diameter and the volume of the rocket casing, our normalizing factors, are as follows:

$$D = 1.939\ inches = 4.925\ cm$$
$$V_{rc} = \frac{\pi}{4} D^2 L = \frac{\pi}{4} (1.939)^2 (12) = 35.435\ inches^3 = 228.612\ cm^3$$

Our operational space for this experiment is constrained by our hardware design. For the rocket to even work, we need a combustion chamber created by a bore hole in the solid propellant. The minimum bore hole size we want to test is a 1/4" diameter bore hole, or 0.635 cm. The maximum bore hole size we want to test is 1" diameter bore hole, or 2.54 cm, which would be over half the diameter of the solid propellant. This results in initial combustion chamber volumes and combustion chamber to rocket casing volume ratios of:

$$V_{cc} = \frac{\pi}{4} (0.635)^2 (12) = 3.8\ cm^3$$
$$V_{cc} / V_{rc} = 3.8 / 228.612 = 0.0166$$

$$V_{cc} = \frac{\pi}{4}(2.54)^2(12) = 60.8\ cm^3$$
$$V_{cc}/V_{rc} = 60.8/228.612 = 0.266$$

We can then solve for the mass of the propellant from these equations, which can be confirmed by a scale upon experimentation:

$$m_{rp} = \rho_p(V_{rc} - V_{cc}) = (1.9)(228.612 - 3.8) = 427.143\ g$$
$$m_{rp} = \rho_p(V_{rc} - V_{cc}) = (1.9)(228.612 - 60.8) = 318.843\ g$$

We also are going to constrain our operational space by the ratio of throat diameter to inner casing diameter. Typical throats range between one-tenth the diameter of the casing to three-tenths the diameter of the casing. This can be represented as:

$$d_t/D = 0.1$$
$$d_t/D = 0.3$$

Solving for the throat diameters, we find:

$$d_t = 0.1D = 0.1(4.925) = 0.4925\ cm$$
$$d_t = 0.1D = 0.3(4.925) = 1.4775\ cm$$

We can visualize this operational parameter space with the following parameter map:

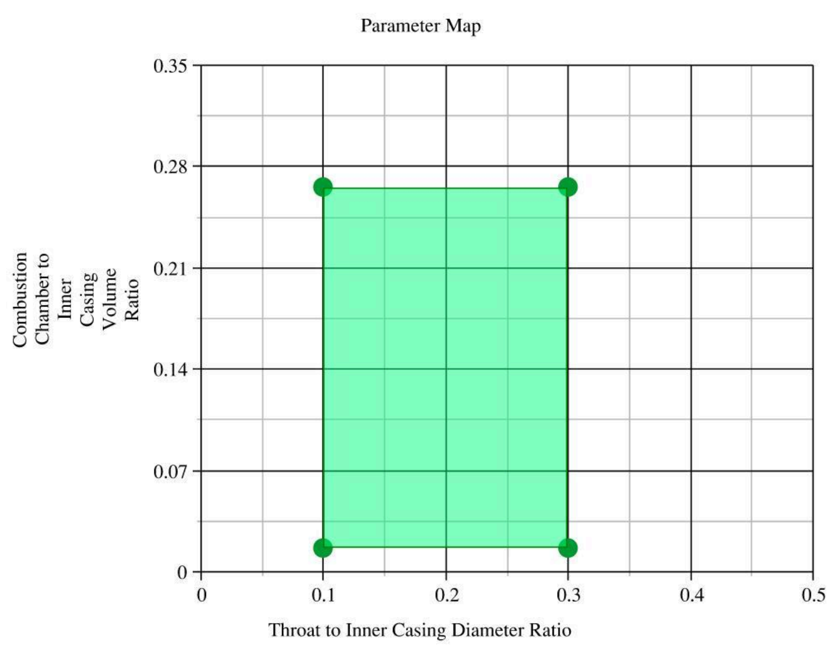

Figure 2: *Parameter Map for Operational Space*

Hardware Design

In this section we would like to specify in detail, the materials we plan to use for our rocket model, the hardware for our testing apparatus, and the instrumentation needed for our experiment. We also are going to provide a schematic detailing our testing apparatus and calling out specific components and instrument locations.

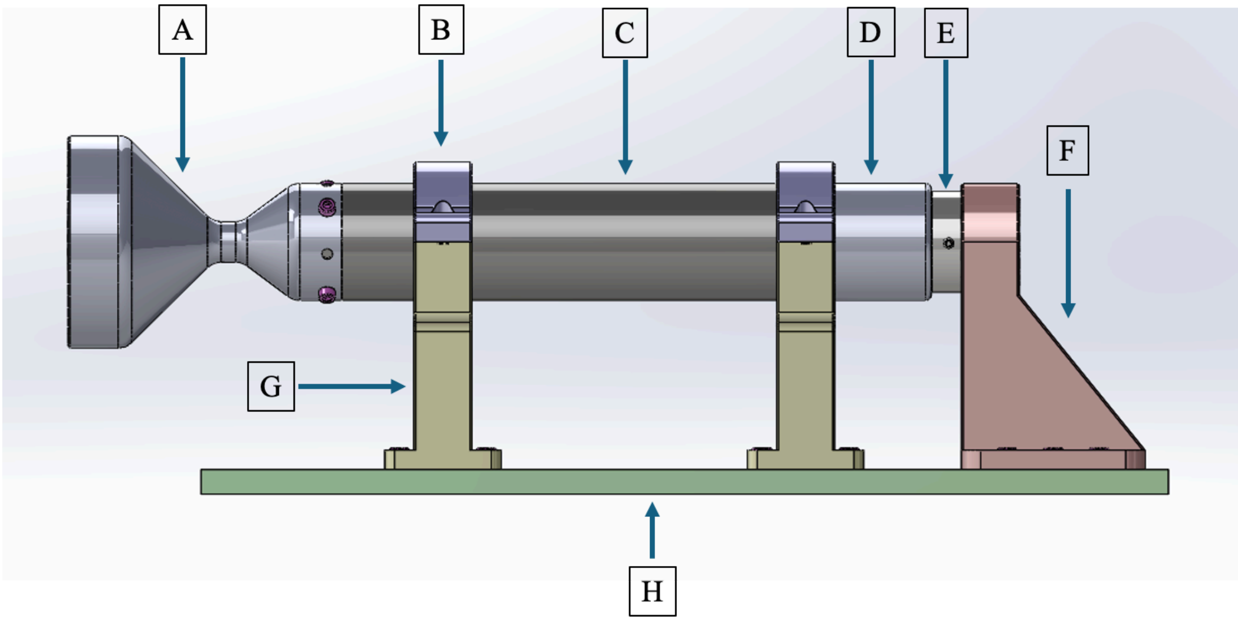

Figure 3: *Decomposition of Elements in Model Rocket Test Assembly*

For our modeled solid rocket motor, we want our casing to be made out of 316-SS stainless steel 2” Schedule 80 pipe. 2” Sch 80 pipe has an outer diameter of 2.375” and an inner diameter of 1.939”. We chose this alloy over other steel alloys or even other metal alloys because of its mechanical properties, namely its tolerance to high pressures and temperatures. In this case it is important to choose materials capable of withstanding high hoop stress fluctuations throughout the combustion process. We chose a 2” pipe to accommodate a smaller scale test for construction of apparatus in a testing facility. We chose Schedule 80 pipe for the higher pressure rating. We chose stronger materials in light of cheaper, less durable materials as we want our rocket casing to have a high level of performance throughout the many tests we intend to perform. For our fuel source, we want to use an aluminum based composite solid propellant, specifically with an ammonium perchlorate oxidizer. Casing and propellant geometry to be illustrated in schematic below.

We would next like to specify the instrumentation we intend to use in our experiment. Due to the nature of our experiment and how we want to deduce relations between parameters over continuous measurements, we want instruments with high frequency responses. For our pressure sensor we chose to incorporate a piezoelectric gauge. We intend to have a pressure tap in the casing wall in the combustion chamber leading tubing to our piezoelectric gauge. We chose to incorporate a resistance temperature sensor (RTD) to measure the temperature in the combustion chamber. In order to measure the hoop stress and local wall shear stress of the rocket body as the combustion reaction happens, a series of strain gauges will be mounted on the exterior surface of the model rocket body. As the testing phase begins the local wall should experience a dynamic stress propagating throughout the rocket body. In order to collect this data a centralized data acquisition system will need to be used to combine signals and report them through a user interface software such as Labview.

The experimental hardware setup is made of eight distinct components. Referencing **Figure 3** the components are labeled A-H. Beginning with the model rocket body components. Component A, the convergent divergent nozzle, is an interchangeable nozzle head. There are 3 individual nozzles which were designed for this experimental set up. With a combustion chamber internal diameter of 1.939" (ID), we targeted the nozzle throat diameter as a primary testable parameter. The nozzle was designed with 3 separate throat diameters, 0.1*ID, 0.2*ID and 0.3*ID. The exit diameter of the nozzle is a standard 2*ID. Refer to **Figure 1d, 1e, 1f** for the three depictions of the targeted nozzle geometries. Component C in **Figure 3** is the Combustion Chamber Body, a 316-SS stainless steel pipe. The pipe will have two interference fit features machined to create a coupling surface on the ends of the pipe at half of the wall thickness. This coupling feature allows the nozzle and chamber end cap to be attached. Component D in **Figure 3** is the end cap of the rocket body. This feature uses the interference fit coupling to slide onto the end of the chamber body, where it is then welded in place at the seam between the fitted surfaces. The combustion chamber end cap has a compatible end surface feature machined to match the face and pin design on the force sensor. This feature guarantees full contact to the force sensor face, minimizing slop between the rocket body and the sensing hardware. Component E in **Figure 3** is the Force sensor which is purchased through McMaster Carr. This force sensor was chosen bearing in mind the targeted 1kN of thrust to be produced by the propellant core during a given test. The amount of thrust is variable depending on the nozzle head chosen but the force sensor has an upper limit of 1,000 pounds of force. One kilo-Newton of force is equivalent to approximately 225 pounds of force. This choice gives the force sensor ample room to experience higher loads or to be repurposed for a larger testing apparatus. The force sensor has a measurement uncertainty of 250g. For the purpose of measuring thrust this is an acceptable tolerance.

Continuing with the components which make up the testing apparatus. Component H from **Figure 3** is the Aluminum 7075 baseplate which is used as the grounding structure for the entire set up. For the test apparatus Aluminum 7075 was chosen as the desired material because these hardware components do not need to bear dynamic loads and temperatures to the extent of the combustion chamber components. Aluminum 7075 is easier to machine than 316 Stainless Steel and cheaper for the project. While still providing a comparable ultimate tensile strength of 572 MPa (316 SS - 580 MPa) and a higher tensile strength 503 MPa (316 SS - 290 MPa) [3][4]. Component F in **Figure 3** is the Force Sensor Mounting Structure. In this case we are concerned with the cross moment load applied to the cantilevered end of the mounting structure. For the full implementation of the testing hardware and any future changes to the maximum thrust generated by the model rocket body, extensive Finite Element Modeling needs to be completed to ensure survivability. A gusset feature was added to the base of the sensor mounting structure to provide fracture resistance and distribute the axial loads through a larger fulcrum arm. Components B and G from **Figure 3** are the rocket body supports (G) and the combustion chamber clamps (B). The rocket body supports allows us to align the model rocket body to the force sensor at the end of the test apparatus. The chamber clamps are meant to restrict the degrees of freedom of the system. In our test design we are focused on allowing movement only across the axial direction of the cylindrical body of the rocket. Therefore, by adding the chamber supports and clamp we create a structure eliminating lateral and vertical mobility of the rocket body preventing pitch or yaw of the structure during combustion. The clamping features are designed to have a clearance fit around the rocket body, allowing 0.250 millimeters difference between the outer diameter of the rocket body and the internal diameter of the clamping feature, permitting the model rocket to move freely axially.

Calibration, Data Reduction Procedure & Uncertainty Estimates

The main sources of systematic error in this system will be the sensing components. This is static uncertainty the sensors will have from the original equipment manufacturer. The strain gauges found on McMaster carry an uncertainty of ±0.3% in measurement. The force sensor has a uncertainty of ±250g. The pressure sensor chosen has an uncertainty of ±1.00%. The resistance temperature sensor has an uncertainty of ±10%.

$$\Delta R = \sqrt{\left(\frac{\partial R}{\partial A}\Delta A\right)^2 + \left(\frac{\partial R}{\partial B}\Delta B\right)^2 + \left(\frac{\partial R}{\partial C}\Delta C\right)^2 + \left(\frac{\partial R}{\partial D}\Delta D\right)^2}$$

Figure 4: *Decomposition of Elements in Model Rocket Test Assembly*

Depending on the post processing of the system and the true nominal values of temperature, strain, pressure, and thrust; **Figure 4,** the propagation equation for the four element system can be applied to find the total propagation of error in the system. The quantity we are ultimately looking for is the rate of erosion of the cylindrical bore on the propellant core. This bore will erode outward, expanding its diameter as the combustion reaction occurs. By finding the pressure difference generated within the combustion chamber and relating that to the volumetric flow rate of the nozzle exit, we can begin to draw mathematical relationships between the reaction in the chamber and thrust produced. A good start location for this process will be to extract the exit properties of the nozzle as the internal pressure of the chamber is actively measured. **Figure 5** shows the compressible flow equations necessary to calculate the exit velocity at the nozzle head.

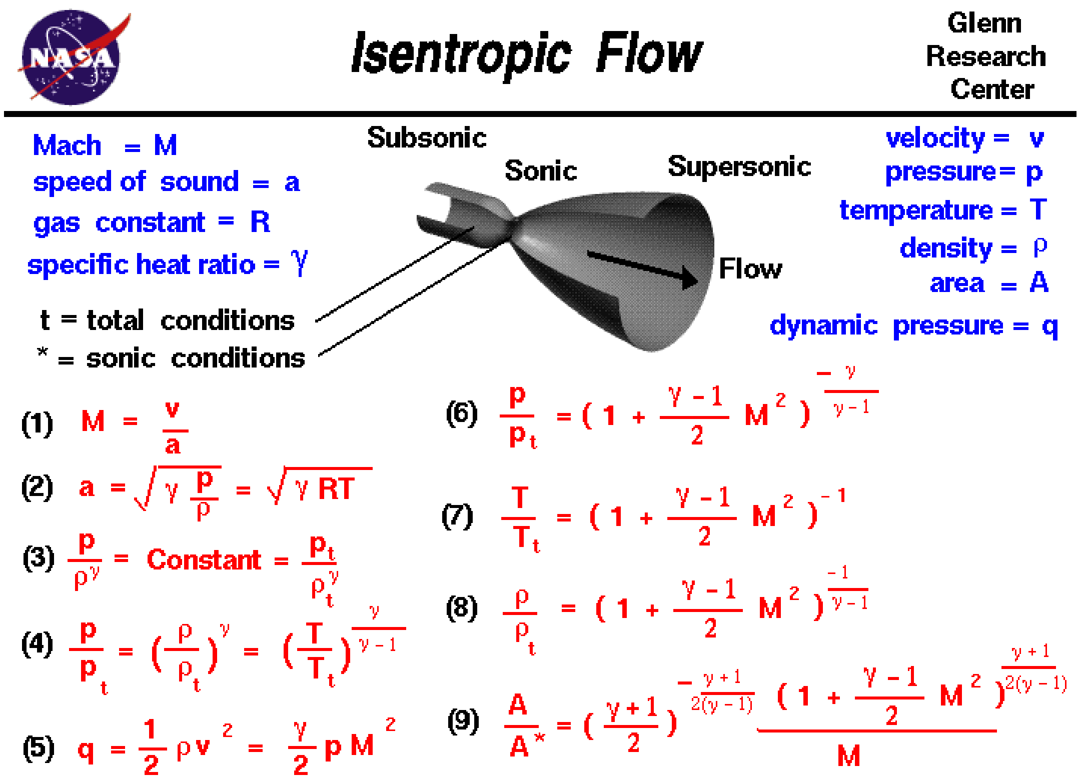

Figure 5: *Decomposition of Elements in Model Rocket Test Assembly [5]*

Calibration techniques which will be useful for this test set up include having a known signal source. All of the sensors described in this proposal will be electronic instruments which may carry a DC offset affecting the zero point. By running a known signal, for example from a function generator, the team can verify the validity of the observed result in the LabView environment.

A proper correction procedure to achieve a zero centered experiment will be possible in the verification and validation phase of the timeline described in **Figure 6.** Once the system is fully assembled the engineer will have the ability to conduct a deep analysis of the system. Beginning by working to minimize system disturbances, the ignition and combustion of a solid state propellant rocket motor is a harsh and loud process. Eliminating all disturbances or even finding their sources may be near impossible but decoupling sensors from surrounding noise can be done. For example, isolating the strain gauge from temperature influence by measuring the stress on the exterior of the wall is one step. Ensuring proper adhesion of strain gauges and tight seals on the pressure sensor all work to minimize system and sensor interaction. The diagnostic testing of the DAQ system is important to achieve zero points on measurements as described above. A proper data reduction program would consider both failures in initial assumptions about internal chamber conditions and calibration errors. Due to the high fluctuations in temperature, pressure, and thrust which may occur within the span of tens of seconds calibrating the system to reflect accuracy at its lowest and highest limits will be a challenge. This will be an interactive method conducted with the hardware assembly on-site.

Design of Experimental Test Matrix

As a part of our proposal we would like to provide a 2-level factorial experimental test matrix to help determine the most statistically significant parameter as well as develop a relationship between the physical parameters to be tested in our proposed experiment.

As a part of our two-level factorial design we want to define our measurand R as the erosion rate of the solid propellant. As a part of our two-level factorial test, we want to consider three factors for their influence on the measure. The three factors we want to consider for this test are time of experiment, initial mass of solid propellant (which is a function of bore hole in propellant), and throat area, represented as x_1, x_2, and x_3 respectively. Three-factor design parameters tabulated below:

Variable	Factor Coding	(-)	(+)	Units
x_1	Time of Experiment	120	360	s
x_2	Initial Propellant Mass	318.843	427.143	g
x_3	Throat Diameter	0.4925	1.4775	cm
R	Erosion Rate of Propellant	-	-	kg/s

Given we are considering three factors, we will need to conduct trials at the bounds of each factor, resulting in 8 trials. We will also want to conduct a trial at the center of each factors' range, resulting in 9 trials. The trials with their parameter coding can be seen below:

Trial #	Factors		
	x_1	x_2	x_3
1	120	318.843	0.4925
2	360	318.843	0.4925
3	120	427.143	0.4925
4	360	427.143	0.4925
5	120	318.843	1.4775
6	360	318.843	1.4775
7	120	427.143	1.4775
8	360	427.143	1.4775

9	240	372.993	0.985

To strengthen the accuracy of our 2-level factorial test, we want to replicate trials. We want to conduct trials one through eight two times each, and conduct the trial at the center point of our parameter space a total of four times, resulting in a total number of trials of 20 trials.

Budget & Justification

Item	Quantity	Material	Supplier	Cost	Justification
M6 Socket Head Screws	1 (Pack of 100)	18-8 Stainless Steel	McMaster	$16.90	These screws are used for mounting the force sensor mounting structure and the body supports to the baseplate.
M5 Socket Head Screws	1 (Pack of 50)	18-8 Stainless Steel	McMaster	$12.62	These screws are used for fixing the force sensor to the force sensor mounting structure.
Baseplate [H]	1	Aluminum 7075-T65 1/T6	McMaster	$175.68	A support baseplate is needed to support the entire hardware set up.
Body Supports [G]	2	Aluminum 7075-T65 1/T6	Protolabs	$1940.72	This set of supports create a base for the cylindrical chamber body which houses the propellant core. These components are grounded to the Baseplate.
Combustion Chamber Body [C]	1	316/316L Stainless Steel	McMaster	$129.34	The chamber body is used to house the solid state propellant. It is a 316-SS stainless steel 2” Schedule 80 pipe.
Combustion Chamber Cap [D]	1	316/316L Stainless Steel	Protolabs	$436.58	The chamber cap is the end cap of the propellant housing which has a compatible geometry to the force sensor face.
Chamber Clamp [B]	2	Aluminum 7075-T65 1/T6	Protolabs	$655.68	The chamber clamp is a loose fitting structure constraining the chamber body to one degree of freedom. It allows sliding in the chamber axial direction but limits lateral or vertical movement.
Force Sensor [E]	1	N/A	McMaster	$570.00	The force sensor rated for a maximum of 1000 pounds of force is well above our threshold of 1kN of thrust. Allowing for flexibility of measurement toward the upper limit of thrust.
Force Sensor Mounting Structure	1	Aluminum 7075-T65 1/T6	Protolabs	$1283.39	The force sensor mounting structure is compatible with the OEM mounting hole pattern on the force sensor. It provides a grounding point for the axial direction.

Nozzle 0.1D Throat Area (Price estimate) [A]	1	316/316L Stainless Steel	N/A	~$2000	This nozzle has an internal throat area of 0.1D. Where D is the internal diameter of the chamber body, 1.939 inches. It has an exit diameter of 2D.
Nozzle 0.2D Throat Area (Price estimate) [A]	1	316/316L Stainless Steel	N/A	~$2000	This nozzle has an internal throat area of 0.2D. Where D is the internal diameter of the chamber body, 1.939 inches. It has an exit diameter of 2D.
Nozzle 0.3D Throat Area (Price estimate) [A]	1	316/316L Stainless Steel	N/A	~$2000	This nozzle has an internal throat area of 0.3D. Where D is the internal diameter of the chamber body, 1.939 inches. It has an exit diameter of 2D.
National Instruments DAQ equipment	1	N/A	N/A	~$10,000	The National Instruments equipment is compatible with the strain gauges and the RTD , pressure sensors and pitot tubes used in this experiment. The individual components of this system need to be picked individually.
Strain gauge kit	1	N/A	McMaster	$200	A strain gauge kit including adhesives and different gauge types to measure the stress at multiple locations around the exterior of the rocket body.
Pressure Sensor	1	N/A	McMaster	$200.82	Pressure sensor with wire leads rated for 1000 psi. The purpose will be to measure the pressure within the combustion chamber.
Resistance Temperature Sensor	1	N/A	McMaster	~$500	The resistance pressure sensor is an insertion flow temperature transmitter. The kit with the appropriate fittings and rating for the combustion practice can be found on McMaster.
Machining Service	N/A	N/A	N/A	~$1000	The stainless steel pipe for the chamber body needs to have 2 sections machined on the pipe head with a lathe. To create 2 interference fit sections to couple to the Nozzle and Chamber End cap.
Welding Service	N/A	N/A	N/A	~$1000	The stainless steel pipe needs to be welded to the combustion chamber end cap at the interference fit joint.

Total Material Costs: $24,121.73

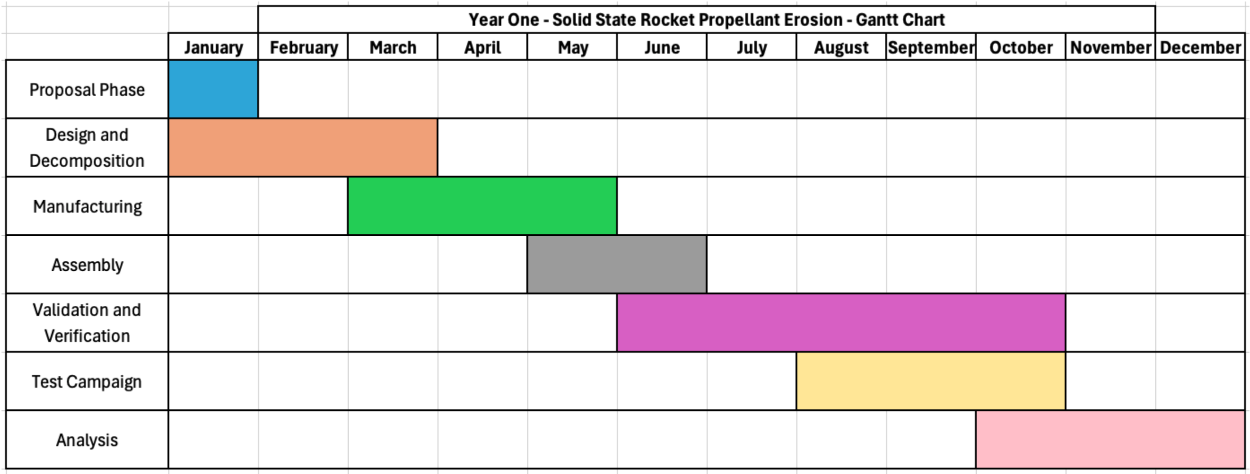

Figure 6: *Year One - Solid State Rocket propellant Erosion - Gantt Chart*

The budget for this project encompasses the entire scope of the effort over one year. The project may proceed through the testing campaign further than 12 months. At this point a new budget for continuous annual testing needs to be drawn out. The proposed budget details the costs of defining, designing, and assembling the project to then spend the remaining months of the year running validation and verification. With a small test campaign to analyze initial results. This Budget and time table is based on the common trajectory of the systems engineering V-diagram.

Figure 6 depicts the project timeline, beginning with the proposal and research phase in blue the team will begin expediting resources on January 1st 2025. The design and decomposition required to define the necessary subsystems and how they benefit the overall goal are completed within the first three months of the project. Manufacturing of components can begin once enough design has been completed and select subsystems are declared complete. The manufacturing process may be a long stretch depending on how much money is willing to be spent on speed. Once the system is assembled, the second half of the year encompasses the validation and verification stage. Setting up the Labview environment and working with the NI DAQ hardware will lead into the first test campaign where preliminary results can be collected. The final portion of the year will conclude with analysis of system performance metrics and a report of how well the system hardware achieved the preconceived goals.

For this project it will be assumed that at least two engineering salaries are needed, for Colin and Rafael. This team of engineers has the capability and knowledge to address all levels of the V-diagram running from project decomposition to systems integration and finally testing.

Total Engineer Wages (annual, two engineers): $170,000

Employment benefits (annual, two engineers): $42,500

Overhead costs, Lab expenses, Testing Expenses (annual): $85,000

Expense Type	Amount	Recurring (annually)	Justification
Employment	$170,000	Yes	The mean annual wage from the Bureau of Labor Statistics is $105,000. For two entry level engineers the wage should be $85,000.
Benefits	$42,500	Yes	The benefits are equivalent to 25% of the annual wages necessary per person.
Overhead	$85,000	Yes	The overhead costs include items that help the facilities stay in working order. Power, water, cleaning, security, heavy equipment rent, etc.
Hardware	$23,621.73	No	The hardware includes the items outlined in the budget table for the assembly of the proposed testing apparatus.

Total Operational Costs for the Entirety of the First Calendar Year: $321,621.73

References Cited

[1] "Solid Rocket Engine." *NASA*, NASA, www.grc.nasa.gov/www/k-12/airplane/srockth.html. Accessed 15 May 2024.

[2] *Design Optimization of Solid Rocket Propulsion: A Survey of Recent Advancements | Journal of Spacecraft and Rockets*, arc.aiaa.org/doi/10.2514/1.A34594. Accessed 16 May 2024.

[3] *AISI Type 316 Stainless Steel*. ASM material data sheet. (n.d.-a). https://asm.matweb.com/search/SpecificMaterial.asp?bassnum=mq316a

[4] *Aluminum 7075-T6*. ASM material data sheet. (n.d.-b). https://asm.matweb.com/search/SpecificMaterial.asp?bassnum=ma7075t6

[5] *Isentropic Flow*. https://www.grc.nasa.gov/www/k-12/airplane/isentrop.html

[6] U.S. Bureau of Labor Statistics. (2024, April 3). *Mechanical engineers*. U.S. Bureau of Labor Statistics. https://www.bls.gov/oes/current/oes172141.htm

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[1] Force Sensor: *McMaster Carr*, www.mcmaster.com/3487N18/. Accessed 17 May 2024.

[2] Combustion Chamber Body: *McMaster Carr*, www.mcmaster.com/4815T181/. Accessed 17 May 2024.

[3] Baseplate: *McMaster Carr*, www.mcmaster.com/8885K923/. Accessed 17 May 2024.

[4] M5 Socket Head Screws: *McMaster Carr*, www.mcmaster.com/91292A125/. Accessed 17 May 2024.

[5] M6 Socket Head Screws: *McMaster Carr*, www.mcmaster.com/91292A134/. Accessed 17 May 2024.

APPENDICES

Appendix 1: Hardware CAD Model & Instrumentation Quotes

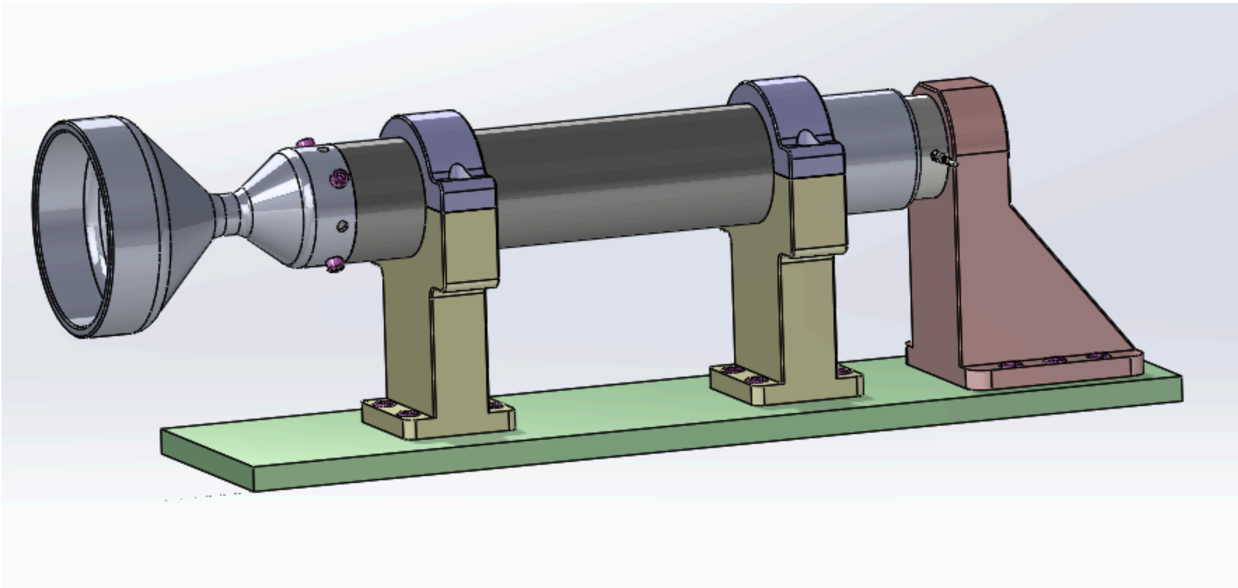


Figure 1a: *Isometric View of Model Rocket and Hardware*

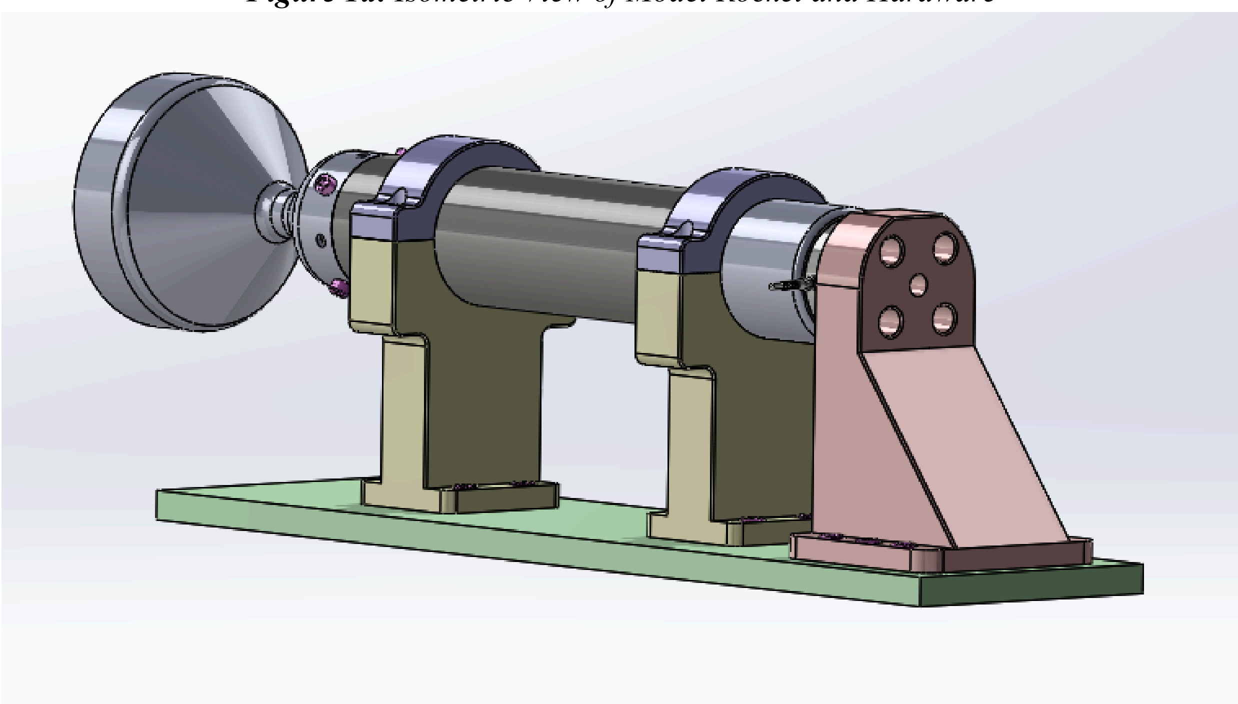


Figure 1b: *Isometric View of Model Rocket and Hardware, back side*

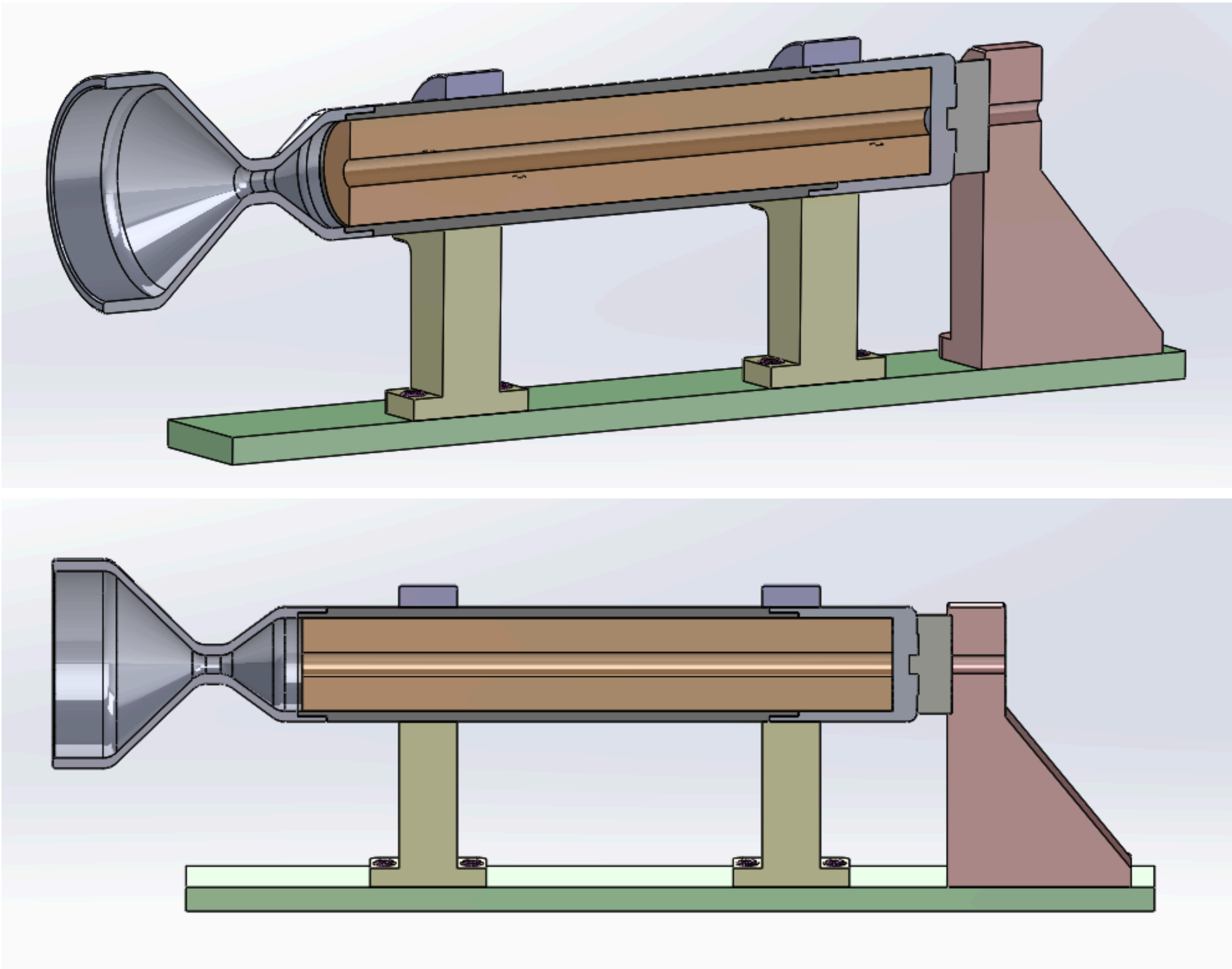


Figure 1c: *Section Cut of Model Rocket and Hardware*

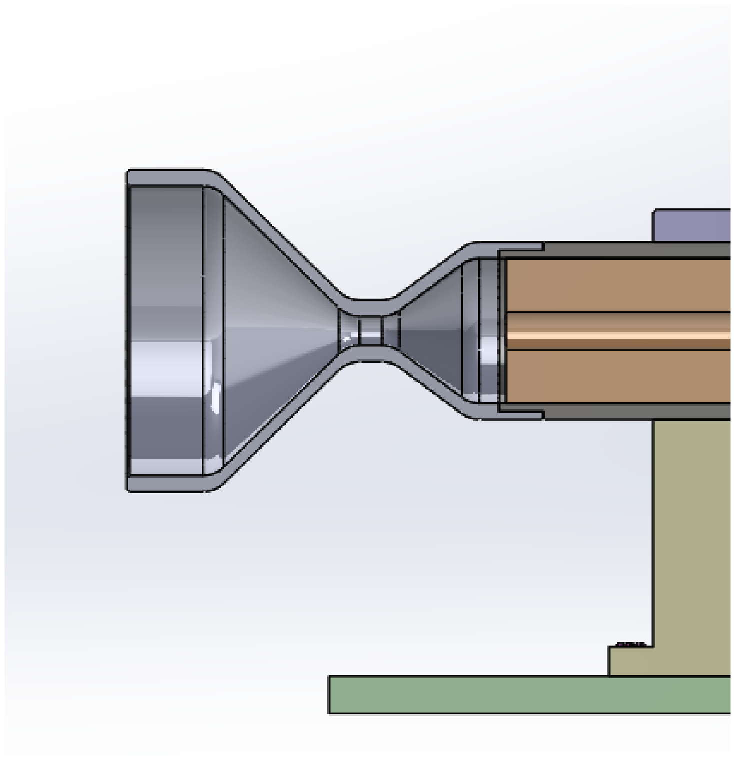


Figure 1d: *Section Cut of Model Rocket and 0.1*D Nozzle*

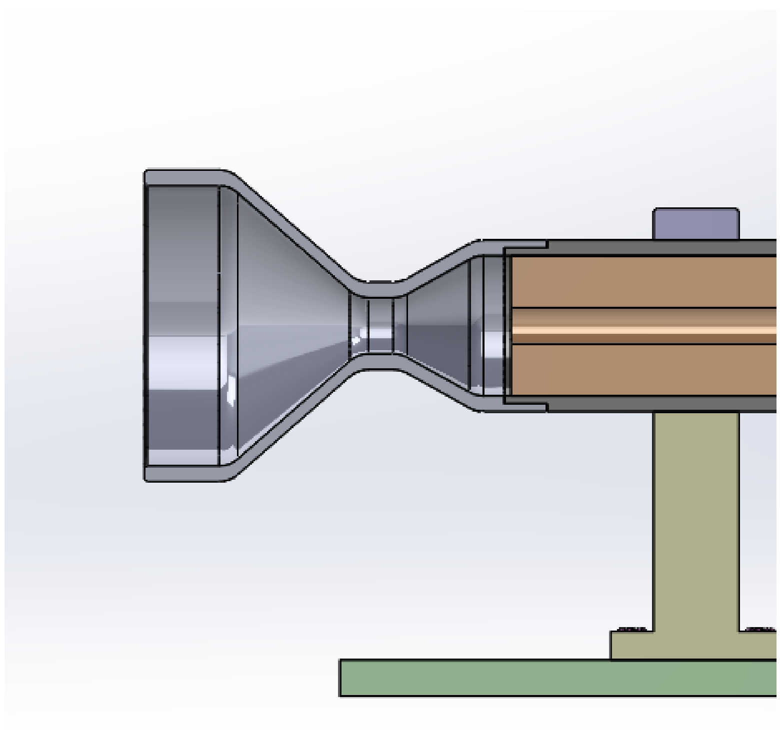


Figure 1e: *Section Cut of Model Rocket and 0.2*D Nozzle*

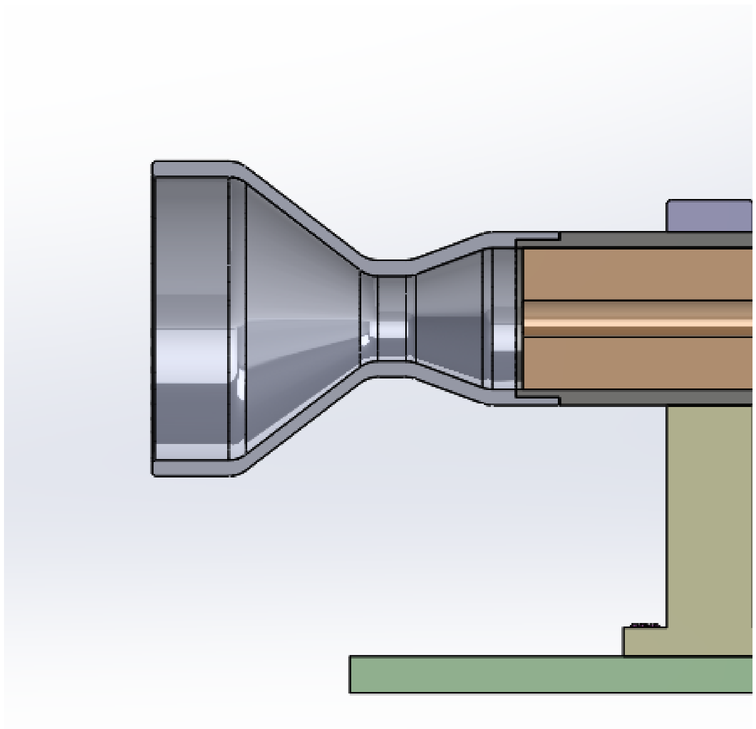
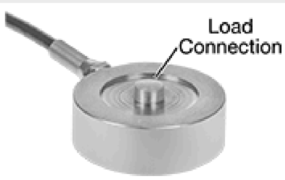


Figure 1f: *Section Cut of Model Rocket and 0.3*D Nozzle*

Round with Button



Round with Button

Capacity						Sensors	
English	Metric	Dia.	Ht.	Temp. Range, °F	Environmental Rating		Each
For Measuring Compression							
4 lbs.	2 kg	1/2"	1/4"	50° to 100°	—	3487N446	$684.29
10 lbs.	5 kg	1 1/4"	13/32"	70° to 85°	—	—	—
20 lbs.	10 kg	1/2"	1/4"	50° to 100°	—	3487N447	684.29
50 lbs.	25 kg	2 1/16"	7/8"	50° to 100°	IP66	3487N176	455.71
50 lbs.	25 kg	3"	1/2"	70° to 85°	—	—	—
100 lbs.	50 kg	1 1/4"	13/32"	70° to 85°	—	—	—
100 lbs.	50 kg	2 1/16"	7/8"	50° to 100°	IP66	3487N177	455.71
200 lbs.	100 kg	1 1/4"	13/32"	70° to 85°	—	—	—
200 lbs.	100 kg	2 1/16"	7/8"	50° to 100°	IP66	3487N178	455.71
400 lbs.	200 kg	2 1/16"	7/8"	50° to 100°	IP66	3487N179	455.71
400 lbs.	200 kg	3"	1/2"	70° to 85°	—	—	—
1,000 lbs.	500 kg	2 1/16"	7/8"	50° to 100°	IP66	3487N18	570.00
2,000 lbs.	1,000 kg	2 1/16"	7/8"	50° to 100°	IP66	3487N181	570.00

Figure 1d: *Force Sensor Specification and Quote*

18-8 Stainless Steel Socket Head Screws

Fully Threaded

Made from 18-8 stainless steel, these screws have good chemical resistance and may be mildly magnetic. Length is measured from under the head.

Black-oxide stainless steel screws have a matte-black finish.

Metric screws are also known as A2 stainless steel screws.

Coarse threads are the industry standard; choose these screws if you don't know the pitch or threads per inch.

Screws that meet **ASME B1.1, ASME B18.3**, **ISO 21269,** and **ISO 4762** (formerly DIN 912) comply with standards for dimensions.

CAD For technical drawings and 3-D models, click on a part number.

			Head							
Lg., mm	Threading	Thread Spacing	Dia., mm	Ht., mm	Drive Size, mm	Tensile Strength, psi	Specifications Met	Pkg. Qty.		Pkg.
M6 × 1 mm										
18-8 Stainless Steel										
12	Fully Threaded	Coarse	10	6	5 mm	70,000	DIN 912, ISO 4762	50	91292A134	$9.84
Black-Oxide 18-8 Stainless Steel										
12	Fully Threaded	Coarse	10	6	5 mm	70,000	DIN 912, ISO 4762	50	90348A025	16.54

Figure 1g: *Screws Specification and Quote*