Highly Defective MK II Hybrid Rocket

Team 43 Project Technical Report for the 2019 IREC

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This document presents Virginia Techs 10,000 ft SRAD Hybid Motor Catergory Launch Vehicle: Highly Defective MK II and subsequent subsystems. The launch vehicle and a majority of the subsytems follow a new design philosophy for the team: modularity. Highly Defective MK II offers a completely student made airframe, active drag system, and a robust payload.

I. Introduction

Rocketry at Virginia Tech is an organization dedicated to exploring and testing the limits of high-powered amateur rocketry and sharing that experience with the rest of Virginia Tech. As a club, we help students get their high power rocketry certification from Tripoli Rocket Association or the National Association of Rocketry with the support of New River Valley Rocketry, the local rocket club. As a design team, we compete in competitions across the country - continually pushing the limits of what we know to go higher, faster, and farther. This is our fourth year as a team.

This year, our team is composed of 50+ undergraduate students of all ages, from freshmen to seniors, across several STEM majors - from aerospace and mechanical engineers, to biomedical engineers and students of physics and engineering mechanics. Each member brings a variety of experiences to the table; we have students with several years of experience with high-power rocket competitions and with official high power certifications. Regardless of previous experience, Rocketry at Virginia Tech seeks to foster an environment in which members can explore their passion for high powered rocketry. That blend of experience and passion lends itself well to the nature of our work; we have a wide pool of experience, knowledge, and interests from which we can draw ideas from in a fun and constructive environment.

This year, Rocketry at Virginia Tech will return to compete in the 2019 Spaceport America Cup. The team aims to compete in the 10,000 foot student-researched-and-developed (SRAD) category. The competition requires launching a sounding rocket carrying a CubeSat style scientific payload to 10,000 feet, all while requiring major components to be student developed. To be competitive, Rocketry at Virginia Tech is striving to develop its most advanced rocket yet, featuring an entirely in-house-constructed composite airframe, fins, boat tail, and nose cone.

Furthermore, this year's rocket will feature a hybrid rocket motor developed and manufactured by the team which works in conjunction with an on-board health monitoring system to keep aerodynamic and motor firing stresses within nominal limits. This health monitoring system will also utilize machine learning algorithms for anomaly detection. Additional technical features include further improvements to our active drag system.

2019 will mark the first year Rocketry at Virginia Tech designs its competition rocket with modularity as a primary focus. We aim to build a rocket with completely interchangeable components, allowing for easy repair or upgrades to the rocket, as well as the ability to have multiple flight configurations to meet varying mission requirements.

This will be our most ambitious year yet, and the team is eager to meet the challenge!

II. System Architecture Overview

A. Propulsion Subsytem

1. Propulsion Subteam Overview

The propulsion subteam consists of eight undergraduate students, ranging from freshman to senior students. Majors represented are aerospace, chemical, and mechanical engineering. For the 2019 Spaceport America Cup, Rocketry at

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Virginia Tech endeavors to compete in the 10,000 feet SRAD hybrid/liquid category. Consequently, the propulsion subteam aims to produce Rocketry at Virginia Tech's first ever student researched and developed motor. Furthermore, in the process the subteam strives to reinforce safe operating procedures, adhere to the team's modular design philosophy, improve upon technical analysis and validation relative to prior years, and allow for ease of motor assembly and integration into the launch vehicle.These objectives are summarized in Appendix H6.

The remainder of the propulsion section will focus on the development and design rationale of the propulsion systems and subsystems. Design rationales given will frequently reference relevant requirements at the system, subsystem, and assembly level. The summary of these requirements can be found in Appendix H7. These requirements are derived from ESRA rules and regulations, as well as from the aforementioned objectives for the propulsion subteam.

For the 2019 Spaceport America Cup, Rocketry at Virginia Tech has developed an SRAD hybrid rocket motor. The motor utilizes 23 lbm at 70 degrees F of nitrous oxide for use as an oxidizer and 3.4 lbm of paraffin wax for fuel. The quantity of oxidizer and fuel required were determined using a hybrid rocket motor performance simulation MATLAB program developed by Rocketry at Virginia Tech. The program is fitted such that inputting the same quantity of nitrous oxide and the same fuel source, the outputted impulse estimate is on par with what SRAD hybrid motors from the 2018 Spaceport America Cup, as well as other commercial hybrids, were able to achieve. A detailed analysis of the motor type choice justification and sizing procedure are provided in appendices H1 and H2.

The resulting simulated impulse is about 4,200 lbf*sec, with a peak thrust of just under 1,200 lbf, and a burn time of 4.5 seconds, of which about 2.25 seconds is during the liquid portion of the nitrous oxide emptying the oxidizer tank. Furthermore, the peak combustion chamber pressure was simulated to be 400 PSIG, and it was determined that a total injector orifice area of 0.001748 m^2 with a nozzle throat diameter of 0.045051 meters were optimal. The results of the simulation is displayed below in Figures 1-4.

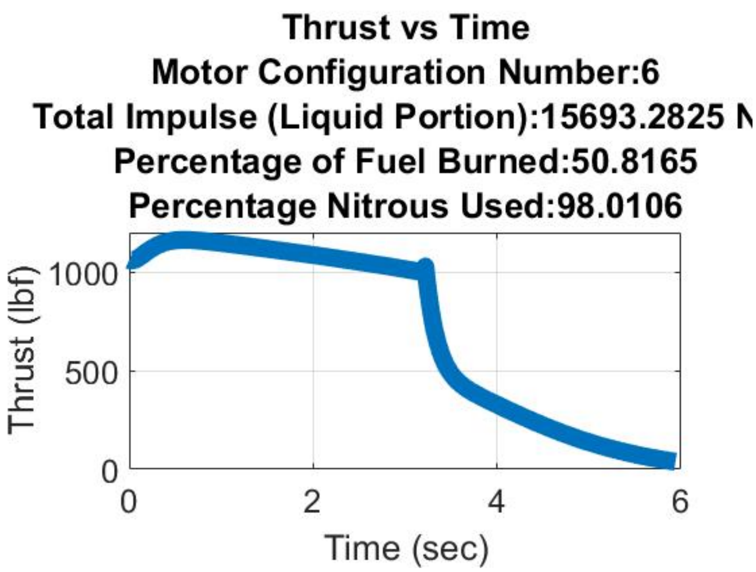

Fig. 1 Thrust vs Time Plot

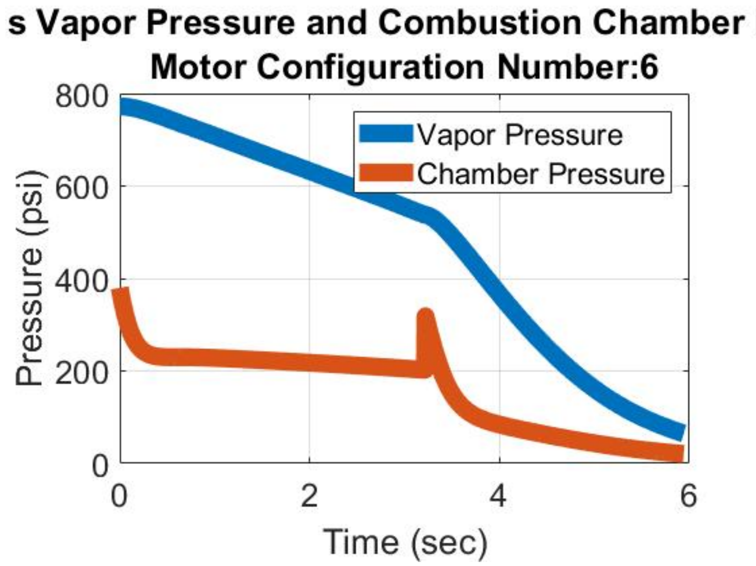

Fig. 2 Vapor Pressure and chamber pressure

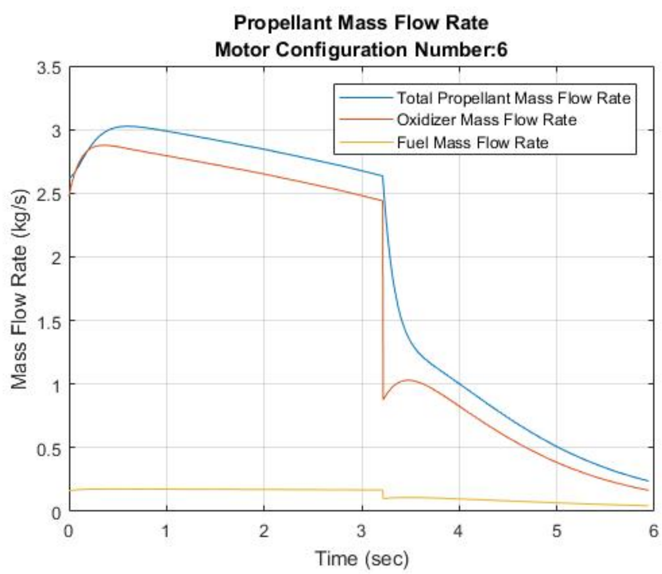

Fig. 3 Propellant Mass Flow Rate

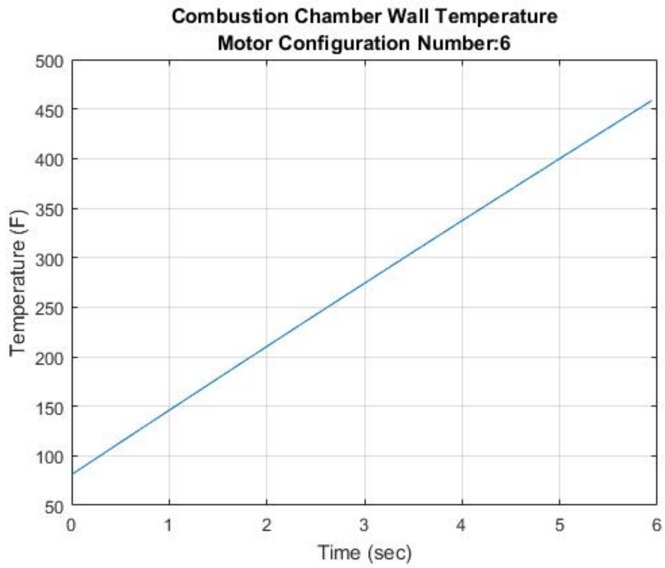

Fig. 4 Combustion Chamber Wall Temperature

The resulting apogee estimate using the simulated thrust curve in figure inputted into OpenRocket is 7,376 feet, therefore fulfilling the propulsion system requirements 3.1 and 3.2 presented in appendix H7.

2. Oxidizer Tank Subsystem

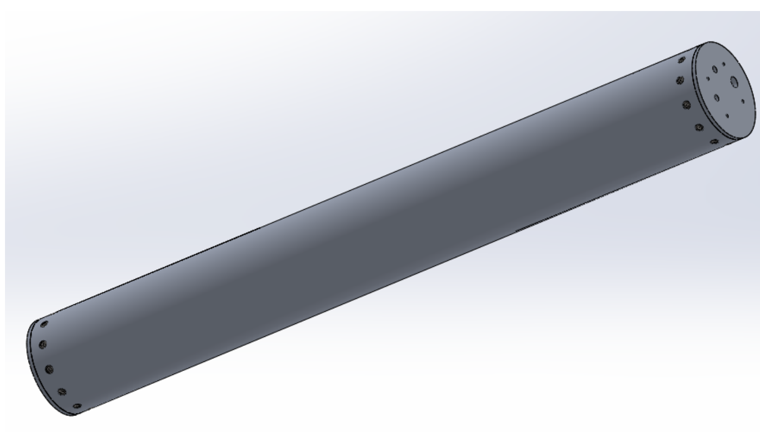

Fig. 5 Oxidizer Tank Overview

The oxidizer tank is constructed of 6061 T6 aluminum; a compatible material with nitrous oxide, therefore satisfying the oxidizer tank subsystem requirement 1 and the propulsion system level requirement 2.1.3. The oxidizer tank is 6 inches in outer diameter and 5.5 inches inner diameter. The length of the tank is 48 inches, allowing for a maximum nitrous oxide capacity of 26 lbm at 70 degrees F with an ullage space of 15%, hence meeting propulsion system requirement 3.1. The 6 inch outer diameter is the maximum diameter that can be used that is a commonly stocked tube size and can be fit inside the booster of the launch vehicle. As a result, the oxidizer subsystem requirement 2.1 is fulfilled. The inner diameter of 5.5 inches was chosen as it is a commonly stocked tubing inner diameter and yields adequate an adequate safety factor for hoop stress of 3.18 assuming a max operating pressure of 800 PSIG. Appendix H3 details the hoop stress calculations.

The oxidizer tank is sealed at either end using bolted on closures. The bolted connection goes radially around the oxidizer tank and into the closures. The tank has holes bored to the diameter of the head of the bolt, and the closures are tapped as well as countersunk such that the head of each bolt sits tangent to the surface of the tank. O-ring seals prevent nitrous oxide from leaking. This arrangement is displayed below in gure 28.

The fasteners used in this connection are 5/16-18 alloy steel socket head screws. These screws conform to ASTM A574 standards. These standards state that the minimum allowable single shear load these fasteners are rated to is 6,800 lbf. There are a total of 12 of these fasteners; assuming equal load distribution the factor of safety for shear loading on the fasteners is 4.23 for the max intended operating pressure of 800 PSIG and 10gs of acceleration on launch. Appendix H4 provides further explanation of this calculation.

The bolt holes on the tank are spaced as recommended by AISC standards, specically section J3.3. This recommended spacing is three times the diameter of the bolts from hole to hole, with a distance from the edge of the material of at least the diameter of the bolt. In Rocketry at Virginia Tech's design, the distance from the edge of the material is 0.64 inches, and a spacing of about three times the diameter of the bolts from hole to hole is utilized. Using AISC section J3.3 analysis methods, detailed in appendix H5, the bearing strength on each bolt hole is found to be 6,040 lbf. Assuming equal loading on each hole, the resulting safety factor is found to be 3.15.

FEA analysis was also conducted with the combined loading of the 800 psig of internal pressure and an axial load of 19,256 lbf applied to each of the 12 bolt holes. A xed constraint was applied on one end of the tank. The safety factor result plots are displayed below in gures 6 and 7. The FEA results show high safety factors greater than 2 on the entirety of the tank with exception to a localized area on the bolt holes were the safety factor is listed as 1.38. After discussion with Virginia Tech College of Engineering Faculty with expertise in FEA, it was determined that the low safety factor of the bolts was due to localized yielding of the material and due to how the loads were applied in the FEA model. Faculty advised that the true bearing strength at each hole was more accurately determined by using industry standards such as AISC J3.3. Consequently, using methods described by AISC J3.3 a safety factor of 4.23 was calculated and was determined to be more trustworthy. As a result, the safety factors for axial loading and hoop stress satisfy oxidizer tank subsystem requirements 3.1, 3.2, and 4.1 are satised.

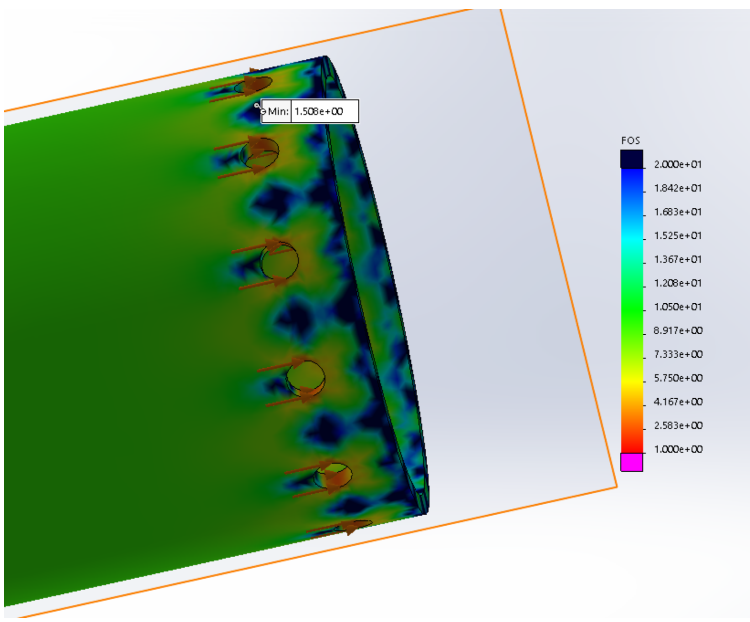

Fig. 6 FEA Results on Bolt Holes in Bearing Stress at 800 Psi

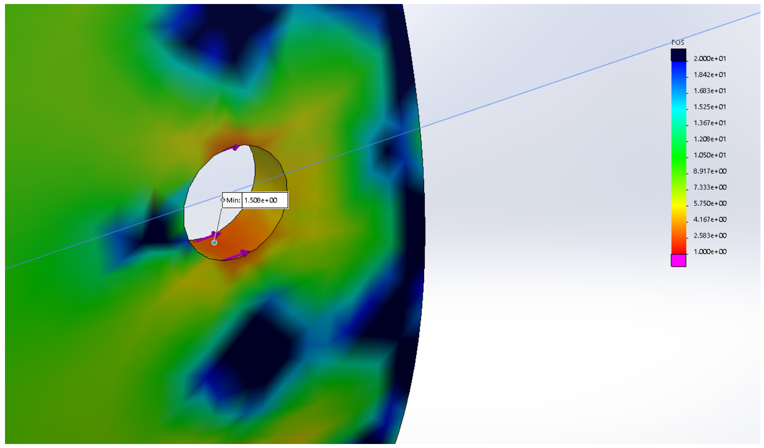

Fig. 7 Localized Stress Concentration at Bolt Hole inner Surface

The o-ring seal is designed using Parker's o-ring design calculator to size the o-rings and the o-ring seats on the closure. The o-ring size code is 2-252. The o-ring compression is designed to be 19 percent. All Parker recommendations are followed. The o-rings to be utilized is a 90 durometer Viton o-ring. Viton is a nitrous oxide compatible material. Two o-rings are utilized for redundancy. The closure is designed to limit the gap clearance between the closure and tank to be no more than 0.013 inches. This is the maximum allowable gap clearance in order to prevent o-ring extrusion, while accounting for the expansion of the tank under load. This gap clearance value for a 90 durometer o-ring is the upper limit for 800 psi of pressure. Figure 8 below depicts gap clearance values required to prevent extrusion at different pressures and o-ring durometers.

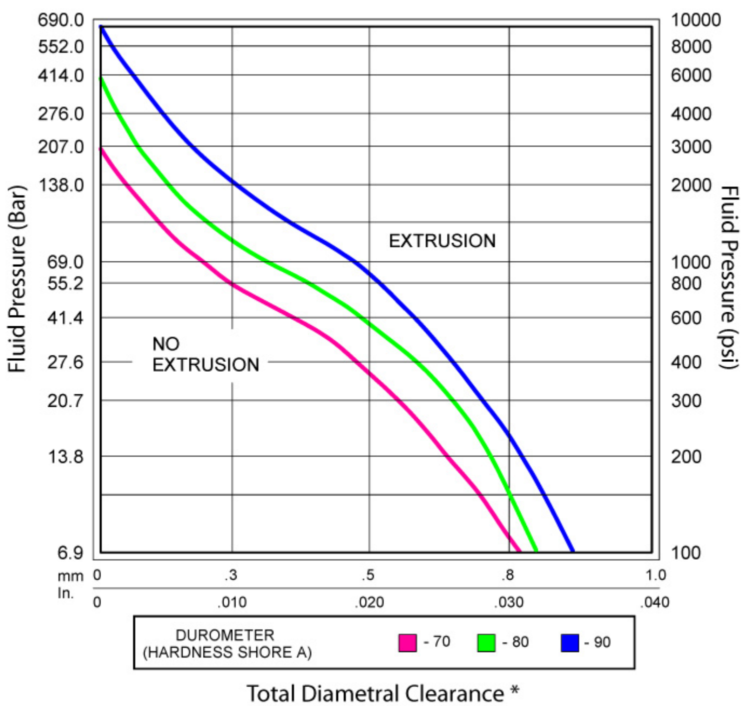

Fig. 8 O-Ring Extrusion Graph

This bolted closure design was also chosen over other options, such as welded ends, because it allows for easier oxidizer tank sizing changes. As long as the oxidizer tank has an inner diameter of 5.5 inches the size can be changes by ordering new tube stock and adding bolt holes. This process is relatively easy and cheap relative to designing and manufacturing an entirely new tank with welded ends. As a result, the bolted connection closures adheres to the team design philosophy of modularity.

Hydrostatic testing of the oxidizer tank was successfully completed to 1.5 times the max intended operating pressure for double the intended use time of the tank. No leaks or deformation was noted at the tested proof pressure of 1,200 PSIG.

3. Valving

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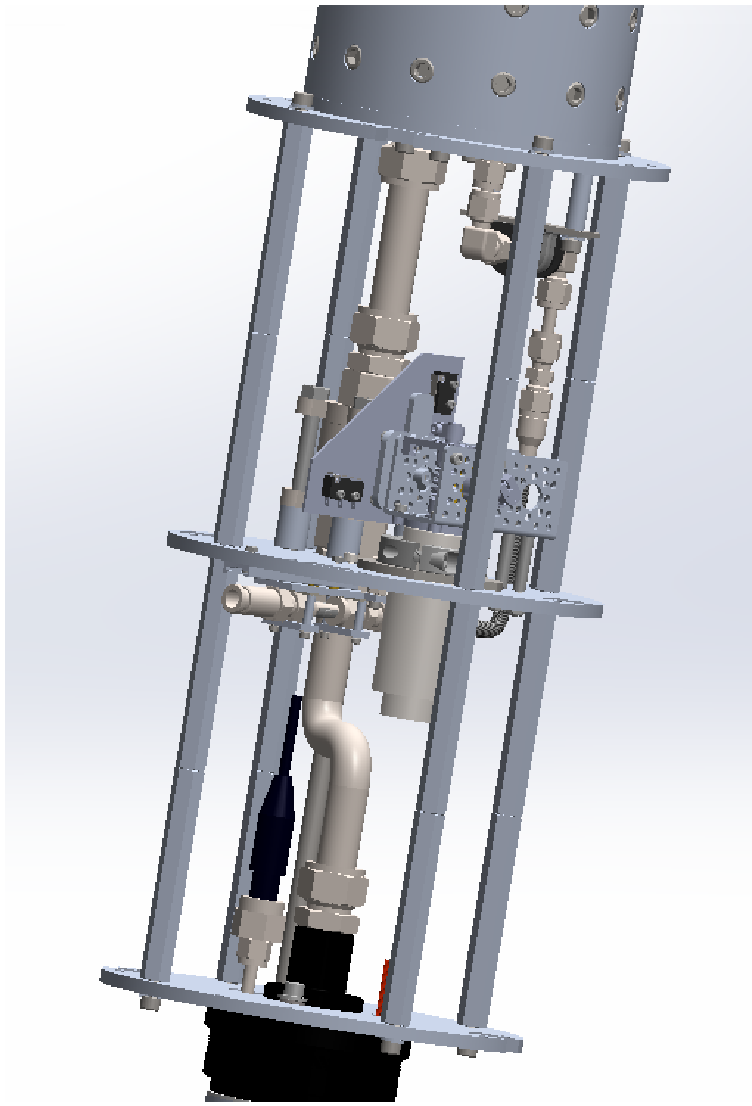


Fig. 9 Valving Assembly

The valving subsystem utilizes 316 stainless steel for all pressure lines carrying nitrous oxide, and all seals and lubricants on valving is uorocarbon based. These materials are nitrous oxide compatible, therefore meeting the propulsion system requirement 2.3.1. Furthermore, all components in the valving system carrying nitrous oxide or controlling it's ow are commercial and rated to a minimum of 1,000 psi, therefore meeting the valving subsystem requirement 5.

The main oxidizer feedline is a 3/4" 316 stainless steel tube rated for 1,500 psi. This size feedline has an area much greater than the injector area. This allows for adequate ow rates and minimizes pressure losses. Hence, valving subsystem requirement 1.1 is fullled.

In order to fulll the subsystem requirement 1.2, a motorized 3/4" ball valve is mounted in the middle of the oxidizer feedline. This valve is commercial and uses PTFE seals. The motorized actuation system uses a commercial DC motor and is axed to the valve using an SRAD coupling. The commercial DC motor includes a built in encoder, as well as a planetary gear set. Furthermore, this motor is mounted to a commercial worm gear drive with a 27:1 gear reduction. The worm gear drive amplies torque, and prevents the valve from being accidentally opened as the worm gear drive will not move unless actuated from the motor. It was measured that 350 oz*in of torque were required to open the ball valve. The motor with the worm drive reduction produces 2,600 oz*in at stall. The motor is estimated to fully open the valve in 0.35 seconds based on the no load revolutions per minute of the motor with the gear reductions. The motors encoder can be utilized to detect the position of the valve. In addition, two micro switches are mounted on a plate xed to the valve. Each switch is triggered either when the valve is fully closed or fully open. This assembly is displayed below in Figure 10.

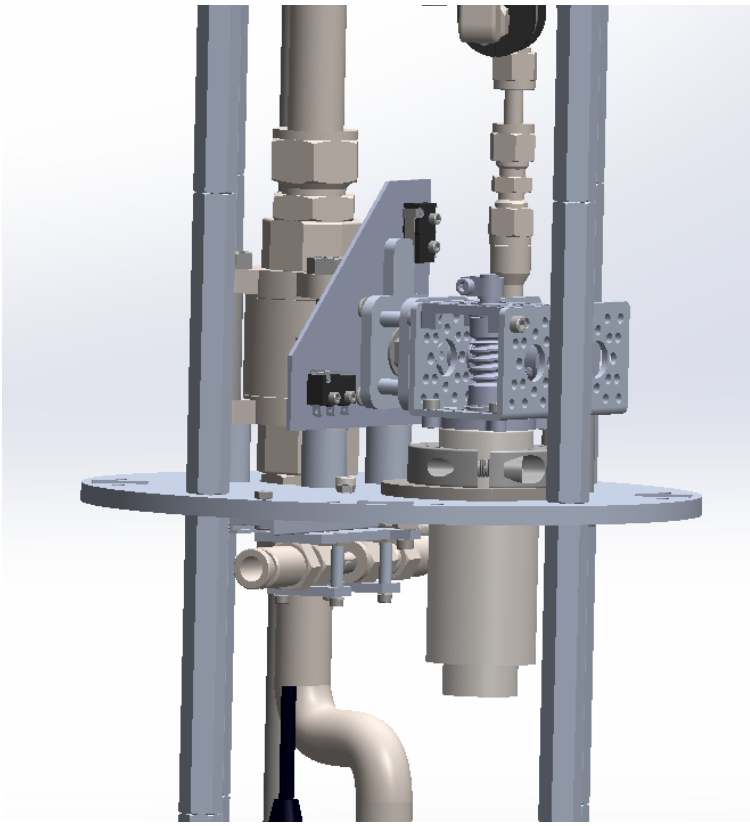


Fig. 10 Main Oxidizer Valve Assembly

In order to maximize oxidizer and fuel mixing and fulll valving subsystem requirement 1.3 an injector plate assembly is mounted to the aft end of the combustion chamber forward closure. This injector plate assembly includes an injector plate mount as well as the injector plate itself. The plate mount is constructed of 6061 T6 aluminum and has a machined section in which the injector plate sits in. PTFE gasket seals are utilized to provide seals on the injector plate and mount, as well as on where the plate mount sits against the forward closure. In addition, an o-ring seal is utilized on the edge of the injector plate mount in order to seal combustion chamber gases away from the threading on the forward closure. The injector plate mount is xed to the forward closure by four 1 4-20 socket head screws with a rubber sealing washer to prevent combustion chamber gases from entering the screw holes. The injector plate itself is a two inch diameter 1/4 inch thick aluminum disk in which injector orice geometry can be machined. This size disk can easily be acquired without any added required machining aside from the desired injector orice geometry. This assembly allows for multiple injector plate designs to easily be swapped into the injector plate mount, which is consistent with the team design philosophy focused about modularity. Figures 11, and 12 depict the injector plate assembly mounted to the combustion chamber forward closure

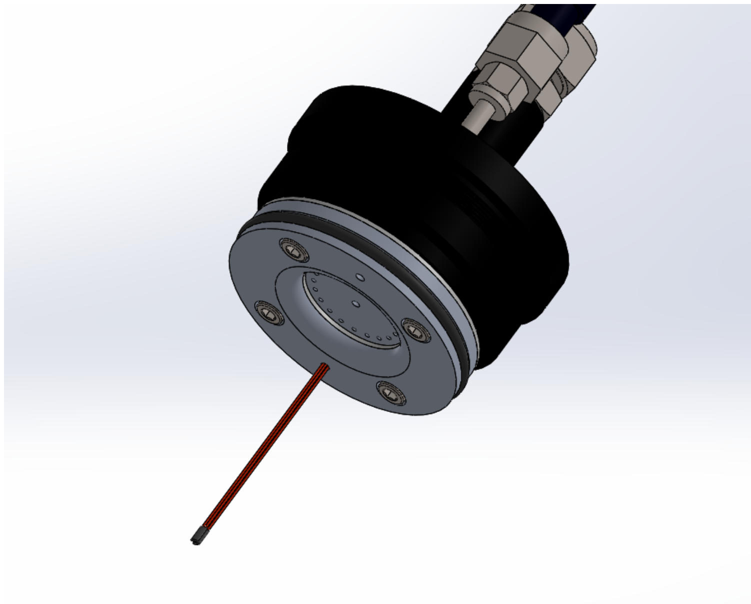


Fig. 11 Injector Plate Assembly Isometric View

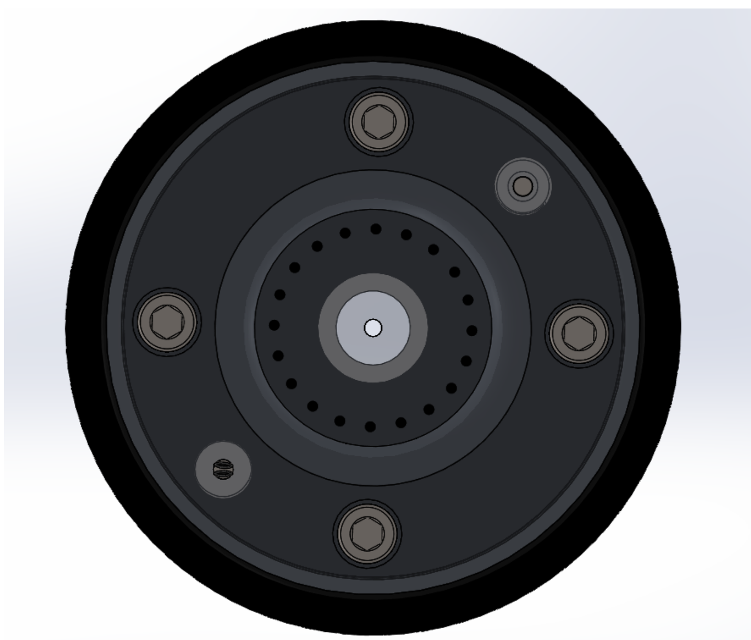


Fig. 12 Injector Plate Assembly Front View

The valving assembly also includes a ll line assembly designed to allow for remote oxidizer loading, thereby meeting the valving subsystem requirement 6. The ll line assembly includes a Swagelok 1/4” male quick connect stem mounted to a movable sled installed on an aluminum bulkhead. The sled uses a bushing and track in order to allow for the quick connect stem to be moved inward and outward radially. A tension spring is mounted to the sled to keep it pulled inward towards the center of the bulkhead when the external fill line is not attached. This translational motion allows for the quick connect stem to be pulled out past the body of the launch vehicle, making it easier to attach an external oxidizer lling line. To allow for this translational movement, a exible 1/4” 316 braided stainless steel line is used to connect the quick connect stem to the oxidizer tank. The external oxidizer ll line uses an accompanying female swagelok quick connect fitting. Both quick connects are shut off stems which act as check valves when disconnected.

Both the main oxidizer feedline and the oxidizer ll line assemblies are mounted between the combustion chamber and oxidizer tank. In order to prevent damage, aluminum bulkheads and standos are utilized to transfer the loading around the valving assemblies. The aluminum bulkheads also provide mounting points for the valving components, oxidizer tank, and combustion chamber assembly. The aft bulkhead which mounts to the combustion chamber forward closure, is constructed of 7075 alloy for the superior yield strength of 62,000 psi as opposed to 35,000 psi of the 6061 alloy used in the other bulkheads. This increased strength is required as this bulkhead has heavy loading acting on it from the thrust of the motor. Each bulkhead is 1 4” thick, and the standos are 4 inches long and 1 2” wide. Two standos are xed together to allow for 8 inches of spacing between each bulkhead. The threading is 1 4 - 20. The standoffs help transfer load to the other two bulkheads forward of the aft bulkhead. The center bulkhead is attached to the outer diameter of the launch vehicle, thus allowing for load to be transferred into the carbon fiber skin. FEA using an 1,200 lbf axial load applied on this assembly with the 7075 alloy on the bulkhead xed to the forward closure of the combustion chamber demonstrated adequate safety factors. As a result, subsystem requirements 4, 8.1, 8.2, and 9 are satised. The FEA safety factor result plot is displayed in Figure 13 below.

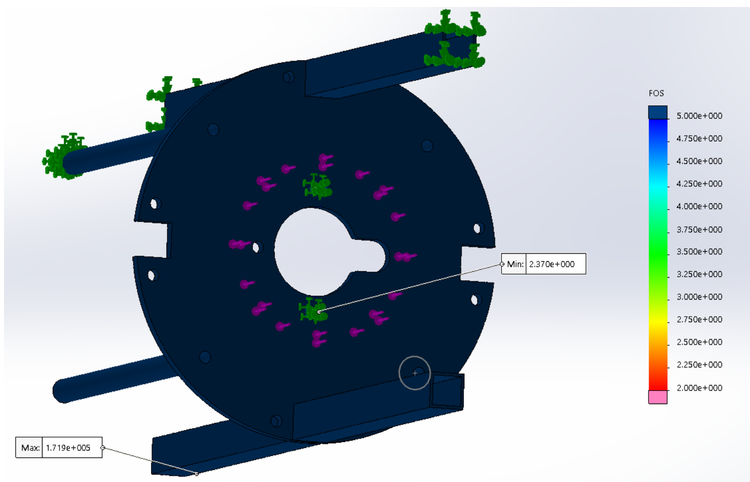


Fig. 13 FEA Results for Thrust Bulkhead at 1200lbf

The valving system also includes components mounted on the forward end of the oxidizer tank. These components include a pressure transducer to measure the nitrous oxide vapor pressure, an additional ¼” motorized ball valve for oxidizer ooading and pressure relief, and a 1500 psi burst disc in case of severe oxidizer tank over-pressurization. In addition, a 6061 T6 aluminum bulkhead is mounted to the top of the oxidizer tank in which components can be mounted to, and to allow to mounting to the internal structure of the launch vehicle. Furthermore, to determine when the oxidizer tank is lled, a 316 stainless steel oat switch rated to 1,000 psi is inserted into the oxidizer tank such that 15% ullage space is left to accommodate the expansion of the nitrous oxide in desert heat. As a result valving subsystem requirement 7 is fullled. Figures and display the valving components mounted on the top of the oxidizer tank.

4. Combustion Chamber Subsystem

The combustion chamber subsystem is a Aerotech RMS 98/15360 casing with the standard aft closure and a modied forward closure. The forward closure is modied to allow for mounting to the valving assembly, as well as include a pressure transducer mount and a mount for a Conax compression tting. The compression tting is a four element tting rated to well above 1,000 psi. This tting allows for the installation of up two electric matches into the chamber. These electric matches are for ignition of the motor. Two electric matches are used for redundancy. The to mount tot the valving subsystem two male-female standos are installed to hold the forward closure against the aft bulkhead of the valving subsystem, and the top of the forward closure is tapped with 1 2” NPT threads to allow the main oxidizer feedline to it. The modied forward closure is displayed in gure 14.

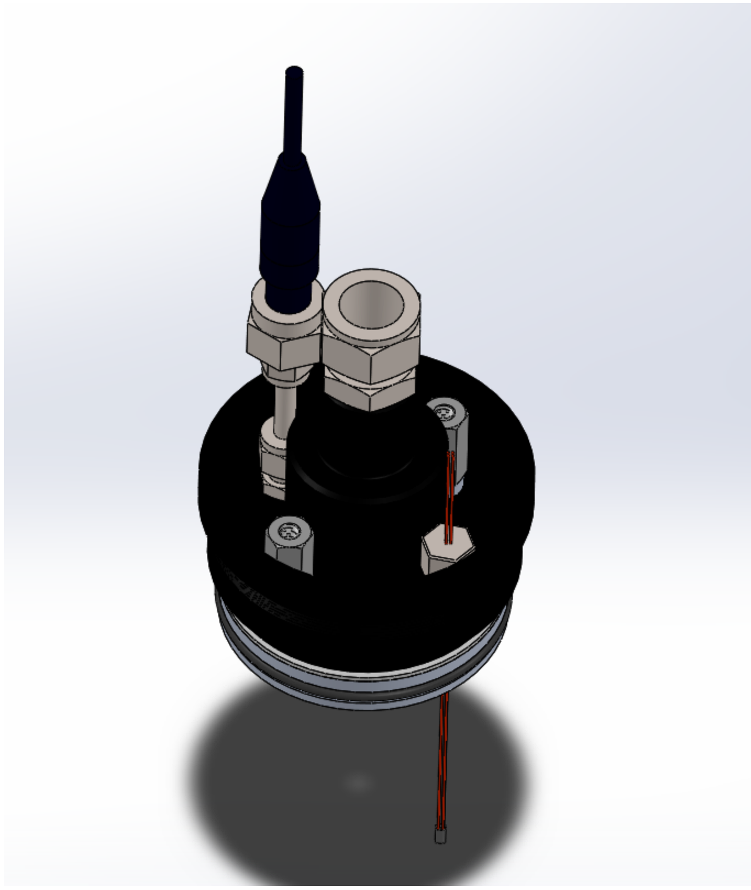


Fig. 14 Modified Forward Closure

The commercial Aerotech casing can accommodate more than enough fuel, and allows for use of commercial liners and nozzles, thus minimizing machining expenses and improving modularity of the propulsion system. Furthermore, hoop stress analysis conducted of the chamber at a peak chamber pressure of 500 PSIG and at peak temperature of 450 degrees F yields an adequate safety factor of 3. This safety factor is calculated using the reduced yield stress of the 6061 aluminum alloy used in the chamber's construction. Appendix details this stress analysis.

5. Ground Support System

The ground support subsystem includes a housing for the nitrous oxide cylinder used for filling that keeps the nitrous oxide cool, as well as the feed system and remote disconnect for the fill line into the rocket.

The cylinder housing uses a strainer mounted over the cylinder filled with dry ice to keep the nitrous cool. A set of small computer fans push the cold vaporizing dry ice gasses down over the cylinder. The housing also is wrapped in insulation to help keep the nitrous oxide cool for the duration of operations.

The fill and disconnect system uses a ¼" 316 stainless steel flexible line hooked up to the fill cylinder to carry nitrous oxide to the fill port on the launch vehicle. The flow of nitrous oxide is controlled by a large nitrous oxide solenoid. The fill line interfaces with the motors onboard fill line via a female ¼" Swagelok quick connect which hooks to the male end onboard the launch vehicle. The female quick connect is mounted onto a linear actuator using a 3D printed collar and plate that is then is used to disconnect the fill system from the rocket. The actuator provides enough force to decouple the quick connects and is mounted to the ground support stand using a hinge such that, once released, the fill system falls clear of the launch vehicle.

6. Propulsion System Control and Instrumentation

The motor utilizes a BeagleBone Black to drive all on board electronics, as well as poll and log data from a variety of sensors. Sensor data polled include the combustion chamber, nitrous oxide vapor pressure, and temperature on two points of the combustion chamber. The pressures are read from two pressure transducers, one mounted on the forward closure of the combustion chamber and the other on the forward end of the nitrous oxide tank. The temperature is measure via two k-type thermocouples taped to the combustion chamber wall, one near the forward end, the other over the nozzle. The motor controller is commanded over RF from a ground station host computer.

The ground support subsystem controls the linear actuator and solenoid valve via an Arduino Mega microcontroller. The computer fans run continuously off their own power supply. There is an RF link between the motor controller aboard the launch vehicle and the ground support subsystem provided via two Xbee modules operating at 915 Mhz. This RF link is to ensure each system is able to coordinate actions in a predetermined sequence. For instance, ignition cannot occur if the motor controller has not received confirmation from the ground support subsystem of that the fill line has been disconnected. This RF link also allows commands to be forwarded to the ground support subsystem.

Commands are sent to both the motor controller and the ground support subsystem from a host computer. To accomplish this, two LoRa radios are used operating at 434 Mhz. Both LoRa radios are operated via an Arduino Mega, one installed on a host computer linked to a GUI which is operated by Rocketry at Virginia Tech Personnel, the other is installed onboard the launch vehicle. The Arduino Mega with the LoRa radio onboard the launch vehicle operates as a hub which forwards all messages, commands, or data to either the motor controller BeagleBone, the ground support subsystem Arduino, or back to the host computer. This system forms the Rocketry at Virginia Tech Launch Control System . Further information is provided in appendix and the following section.

7. Launch Control System

With the goal of flying a hybrid motor at competition this year the team needed to write software that could communicate safely over radio signals during the filling and launch stage of the rocket. We have designed the overall system to communicated using two arduinos connected with the RFM95 radios.

Brief Overview: If you refer to figure 15 below you can see an outline of the way we have set up out system. The Ground System (GS) is one of the communicating arduinos. The GS is connected with a host computer GUI that is shown in figure 16. The GUI allows a user to send/receive commands, view live plots of sensors, and unlock the rocket for launch. The HUB is the arduino that receives and relays packets from the different nodes of the system. The HUB just acts as an intermediary between the GS and the other three nodes, it will do no computation and relay any packets that it receives to the other computers. COMP 1-3 are various computers that will be controlling different aspects of the rocket. If these three computers need to relay something back to the GS, then they will send the command to the HUB and the HUB will take care of routing the commands. While creating this system there are multiple safety requirements that ESRA have outlined for us to meet.

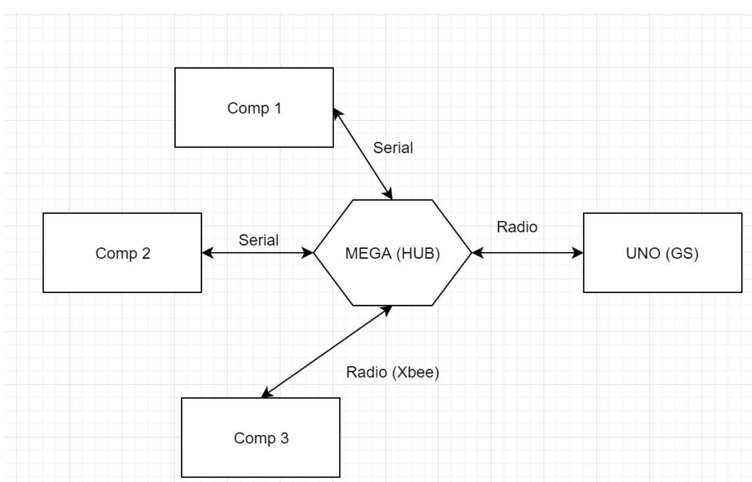


Fig. 15 Modified Forward Closure

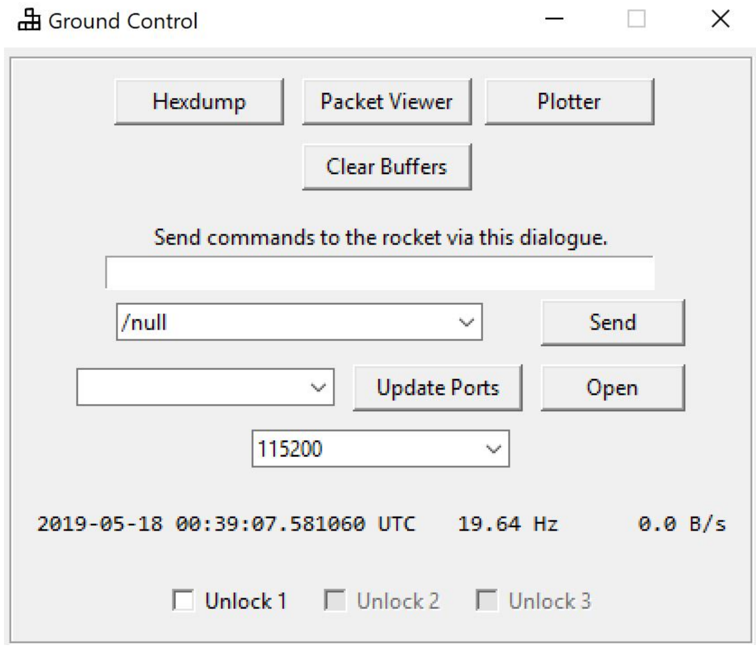


Fig. 16 Modified Forward Closure

At least two deliberate actions are required to fire the system: The very first action required to start the system is to flip a switch to enable the match circuit. After this there are three cascading checkboxes on the GUI that need to be checked in a certain order for the rocket to fire. On the initial boot up only one of these checkboxes can be interacted with, after it is checked the next box will become intractable and so on. If any checkbox is unchecked during the launch initiation sequence then the whole system will abort and go back to the initial boot up stage.

Power Interrupt: While personnel are at the pad the circuit to arm the ematch is under their control alone. The GS will not be able to trigger this part of the system. After the switch has been activate all personnel should immediately leave the pad.

Uncontrolled RF space: Since the rocket and the GS will be communicating in uncontrolled RF space we need to verify the packets received are sent by us. In order to do this we use sync bytes at the front of every packet if a packet does not start with a pre chosen set set of bytes then we know not to process it. To test this we had the arduino send random packets as well as legitimate packets to make sure that the software of the begal bones and the GUI of the ground system will be able to filter extraneous packets out. (This was to simulate the possibility of catching other teams packets while trying to fire in the desert)

B. Aero-Structures Subsytems

1. Aero-Structures Overview

The Aero-Structures subteam consists of twelve undergraduate students, ranging from freshman to seniors. A variety of majors make up the subteam including mechanical and aerospace engineering and engineering science and mechanics. For the 2019 Spaceport America Cup (SAC), the subteams objective is to safely design and fabricate a launch vehicle that will incorporate the various subsystems the team is working on and deliver them to a target apogee of 10,000 ft. AGL. In addition, the team will adhere to the requirements set by the Experimental Sounding Rocket Association for the 2019 SAC and internal team derived requirements. These requirements can be found in appendix F.

The remainder of this section will detail the preliminary design the team has developed thus far. In addition, it will detail the rationale behind selecting the designs of the internal structure and airframe.

2. Launch Vehicle Overview

The launch vehicle as seen in Figure 17, will adhere to all requirements as mentioned in the previous section. The launch vehicle will be approximately 16.75 ft long and will have a wet mass of 181 lbs.

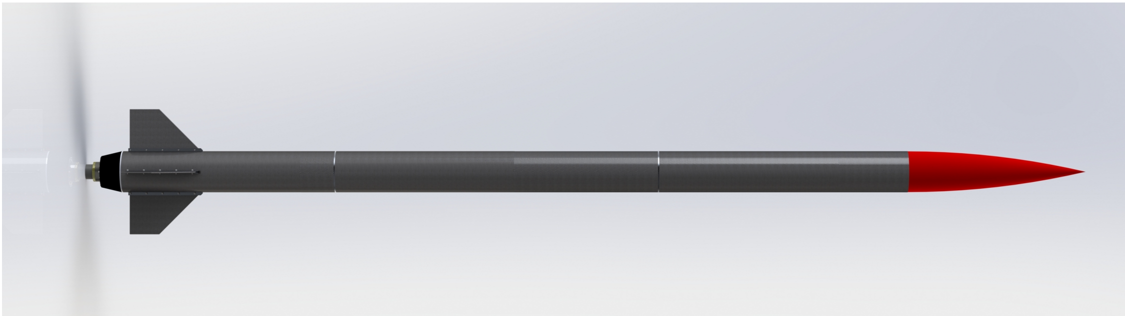


Fig. 17 Overview of the launch vehicle.

Figure 18 shows the full internal structure assembly with all subsystems installed.

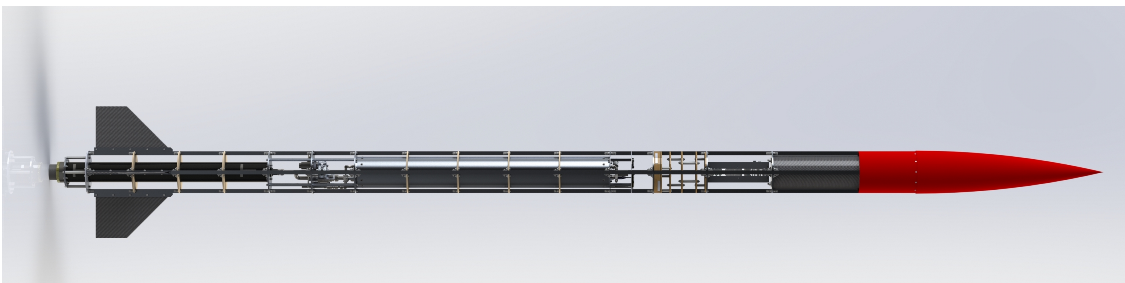


Fig. 18 Full internal structure assembly with subsystems.

3. Dimensional Justification and Design Philosophy

For the 2019 SAC, the Aero-Structures subteam decided to to stray from a traditional monocoque structure to a semi-monocoque structure. This was done for a variety of reasons. First and foremost, the team added a new requirement: to have a fully modular vehicle structure. In other words, the structure should be able to be fully disassembled down to individual components such that any component can be replaced and the entire system can be reassembled. In previous years, the vehicle assembly has required the presence of individual subsystems such as the electronics bay or the payload in order to fully construct the vehicle. Furthermore, previous monocoque designs had a majority of structural components permanently bonded to the vehicle. Thus in the event that a specific component failed, it could not be easily replaced. For example, Whipsnake, the competition vehicle for the 2018 Spaceport America Cup experienced a water landing during its first test launch. This resulted in significant structural damage as well as electronics damage. Because a majority of these components were permanently bonded to the vehicle, they could not easily be replaced and required hours of modification to the internal structure.

With the adaptation of a semi - monocoque structure, all subsystems, including the hybrid motor, can be mounted directly to a set of stringers. This is different from previous years, as these systems would typically be housed in coupling hardware and would be required to be flight ready. This allows complete independence in assembly from any individual subsystems. For example, the launch vehicle can be assemble without the payload and still be flight ready. In addition everything is fastened to these stringers instead of a traditional epoxy system. Thus, the entire stringer assembly can be taken apart to replace any individual component even a single bolt. The design of the semi-monocoque structure is detailed further below in section:internal structure.

The team began with designing the launch vehicle structure around the hybrid motor due to its complexity and the requirement to safely transfer the loading from the motor to the reset of the structure. Furthermore, the structure surrounding the propulsion system, needed to be removable such that the entire system could serviced after flights. This further reinforced the philosophy of modular design throughout the vehicle. The propulsion subteam developed initial parameters for the hybrid motor such as oxidizer tank volumes, wall thicknesses, combustion chamber lengths etc. These parameters were essentially constant and would directly impact the hybrid motor length and the overall vehicle length depending on the inner diameter of the vehicle.

Some other parameters that were important while determining the dimensions of the vehicle, were the internal usable diameter. With the use of stringers, throughout the body, the usable inner diameter would actually be less than the actual inner diameter of the vehicle. This would force multiple subsystems to create additional stacks on top of each other when originally they would be able to place everything on a single bulkhead. Most notably, this would impact the parachutes since they would need to packed into a smaller diameter which in turn would required a longer vehicle.

The team took all these parameters and plotted them at different vehicle inner diameters to observe the impact on vehicle characteristics. The results can be seen in Figure **??**. Only 6 inch through 8 inch inner diameters were considered due to the availability of commercial components and drag characteristics of larger diameter vehicles. In Figure **??**c, it can be seen that the large the inner diameter, the shorter the length of motor. Similarly, Figure **??**b shows the a similar result: the larger the inner diameter, the shorter the overall vehicle length. Figure **??**d shows the relationship between the inner diameter of the vehicle, and the usable inner diameter. Quite obviously, the larger the inner diameter, the more usable space there is inside the vehicle. Thus, as the the inner diameter increases, the length of the parachute bay decreases as seen in Figure **??**a.

After weighing the results, the team decided that keeping the rocket shorter was beneficial for multiple reasons. For one, a longer vehicle would have significant bending loads at any joints or separation points in the vehicle compared to a shorter vehicle. If only a 6 inch and 7 inch inner diameter are considered, the overall vehicle length would be approximately 22 ft. and 29 ft. respectively. A vehicle at this length would be impractical to transport or assemble reliably with the equipment that the team owns. In essence, a larger inner diameter vehicle seems to provide the most stable launch vehicle with the most usable inner diameter. Thus an 8 inch inner diameter vehicle was selected for the 2019 SAC.

4. Internal Structure Overview

As previously mentioned, in order to incorporate modularity, the team decided to adopt a semi-monocoque structure as opposed to a monocoque structure which were used in previous year. An overview of this structure can be seen in Figure 20.

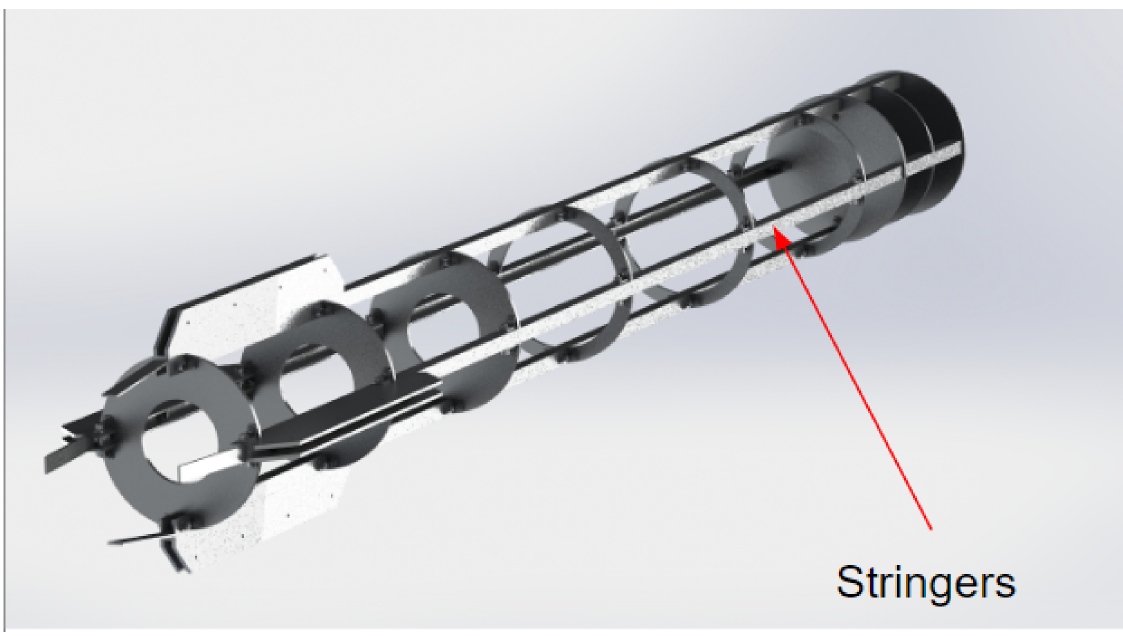


Fig. 20 Internal structure overview.

The series of stringers will run axially throughout the entire launch vehicle. The material for the stringers will be 6061 Aluminum Square tubes. They will be 0.75" x 0.75" and will be purchased commercially. Furthermore, the beams will help stabilize the outer skin after the application of the compressive aerodynamic loads. In certain instances the stringers will break and connect to bulkheads that match the outer diameter of the vehicle. This can be seen in Figure 21.

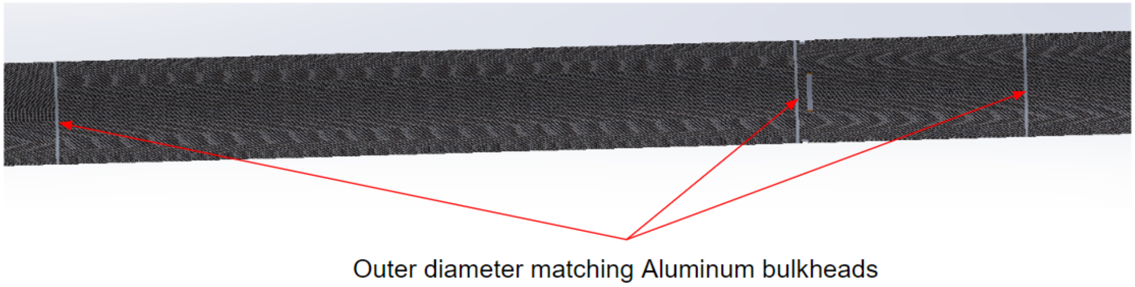


Fig. 21 Example of stringer break points at bulkheads that match the outer diameter of the vehicle.

The instances of bulkheads matching the outer diameter of the vehicle are used to transfer the loading from the internal structure to the air frame. The contact area between the aluminum bulkheads and the carbon fiber body tubes allows this load to transfer. Furthermore, this also acts as a fail safe against buckling of the internal structure. If the stringers were to buckle, all the loading would then be transferred through the carbon fiber airframe instead. Our requirements state that the carbon fiber airframe should be able to withstand all forces experienced during flight.

The fully assembled internal structure can be seen in Figure 22.

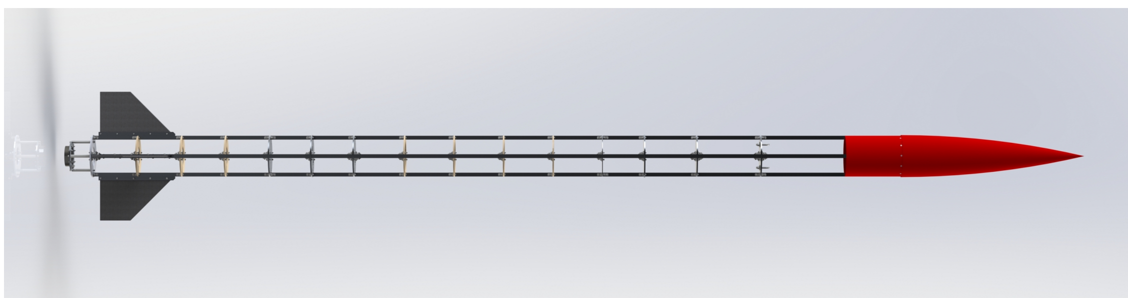


Fig. 22 Fully assembled internal structure without body tubes.

5. Interfaces

Every interface between the stringers will implement a standardized bracket and bolt structure. This can be seen in Figure 23. The bracket system and hardware will both be commercially bought. The bracket material will be steel and will be fastened using 8-32 machine screws.

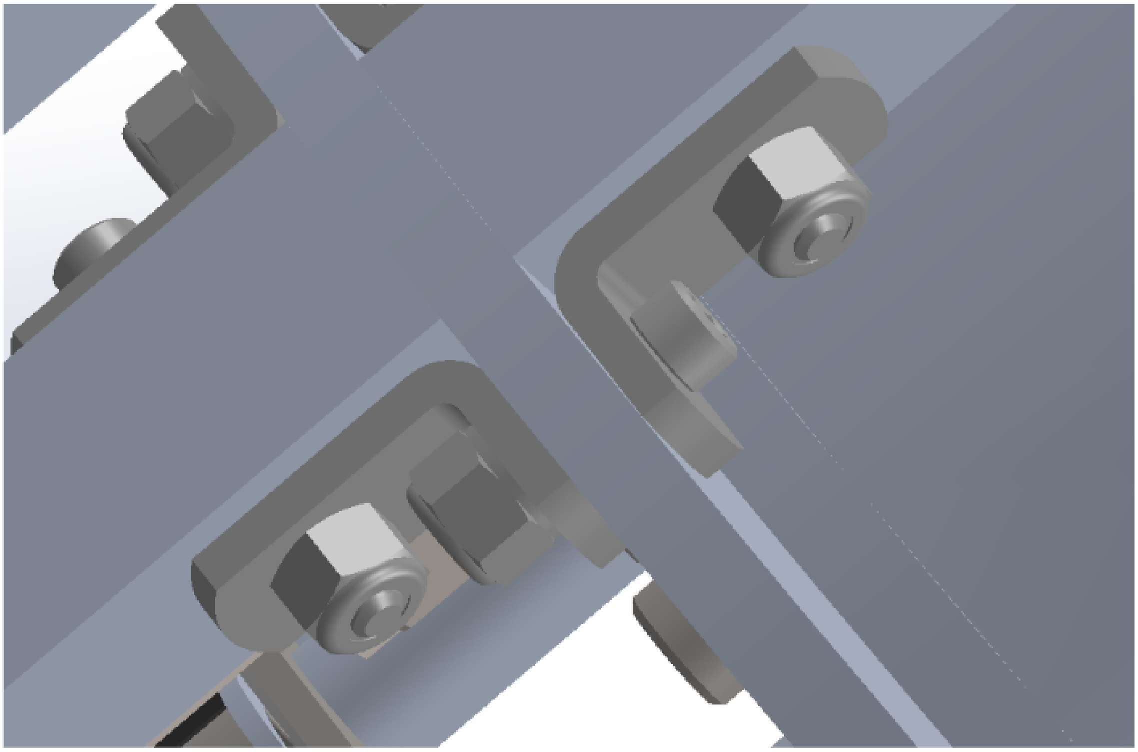


Fig. 23 Example of interface between stringers and bulkheads.

6. Loading Analysis

To understand the loading subjected on the rocket airframe during flight, a linear loading amount of 8.5 lbm per foot under 5 g's acceleration was used. This was a simplified assumption, as it is hard to characterize the exact weight of components at this stage in the design phase. As one moves down the rocket airframe, the loading compounds until the region separating the oxidizer tank and combustion chamber. The region between the combustion chamber and oxidizer tank therefore experiences the highest compressive loading during flight with a value of approximately 550 lbf. Immediately following the region between the combustion chamber and the oxidizer tank the thrust produced by the combustion chamber exceeds the inertial compressive loading. The aerodynamic forces on the outer skin and tensile inertial forces of the remaining airframe components are small relative to that of the previous section. Figure 24 below represents the expected axial loading along the length of the rocket. The horizontal axis is the distance from the tip of the rocket in inches. The vertical axis is the compressive loading in lbf.

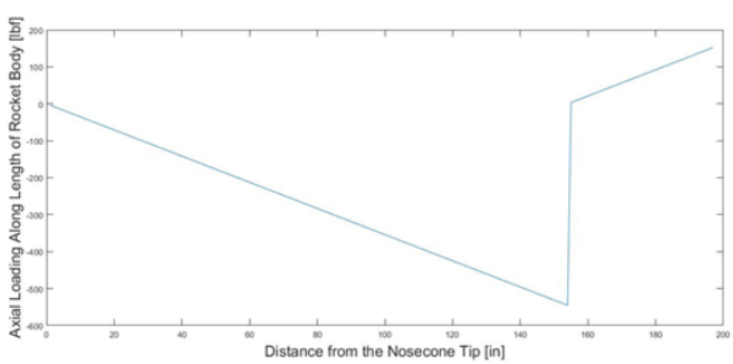


Fig. 24 Axial loading as a function of length.

The simplistic loading scenario above serves as an initial approximation for validating the structural integrity of the rocket airframe. A few shortcomings of the above analysis should be noted. First, the linear loading is only an approximation of the true mass distribution. Subsequent work needs to include the distribution of individual components. Secondly, aerodynamic forces are not included. The ongoing development of computational fluid dynamic models for flight will aide in developing realistic pressure distributions along the rocket.

7. Buckling Analysis of Stringers

The 6061 aluminum struts will serve as stringers to support the carbon fiber composite outer skin under compressive loading and aerodynamic moments. Furthermore the, the bulkheads placed through the length of the air frame will decrease the buckling length and thus increase the critical buckling load of the struts. An Euler beam buckling analysis was performed on these aluminum struts to ensure that the inertial compressive forces during flight would not exceed the critical buckling load of the struts. Equation 1 below is the Euler beam buckling formula. An n value of 4 was used to model a fixed-fixed beam; characteristic of the loading scenario.

$$F_{critical} = \frac{nEI\pi^2}{L^2} \tag{1}$$

A strut was taken as independent of the outer skin and fixed at both ends with aluminum bulkheads. For a 1 ft section of aluminum strut, it was found that the critical buckling load of the section is approximately 37400 lbf. In the simplistic scenario in which the outer skin of the rocket supports no loading during flight, each aluminum strut will support approximately 137.5 lbf. This is found by simply dividing the maximum compressive force by the number of struts. The critical buckling load of the strut far exceeds that expected compressive loading per strut, which is 137.5 lbf.

8. Von-Mises Stress in Bulkhead

FEMAP was utilized for a simple finite element analysis. This program was chosen for its ease of use and the familiarity that the existing team has with the software. A conservative loading was used for the analysis consisting of 1100 lbf acting longitudinally along the rocket airframe in static conditions. 1100 lbf force is far above the expected maximum thrust produced by rocket and is meant to account for initial shocks produced by the rocket upon takeoff. Figure 25 below shows the Von Mises stress distribution across the area of the 7705-aluminum bulkhead.

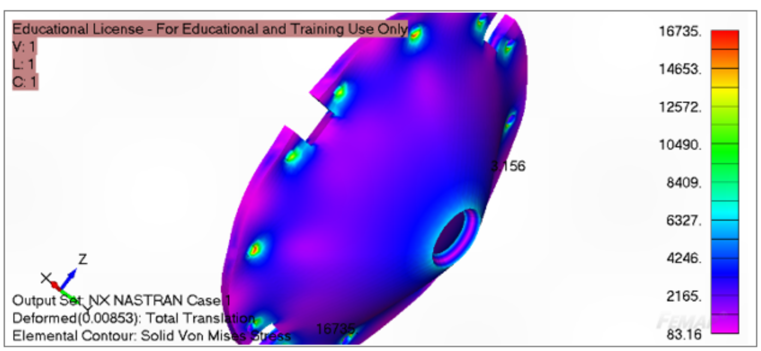


Fig. 25 Von Mises stress distribution across the area of aluminum bulkhead (psi)

From the above analysis, one notices local areas of stress concentration near the points of attachment of the bulkhead. These are to be expected but should be noted. During assembly, care should be taken in fastening the bulkhead to ensure even load distribution between each of these attachment points. It is found that that maximum Von Mises stress achieved under the conservative loading conditions is about 17000 psi. The tensile yield stress of 7705 aluminum is 73000 psi, giving a factor of safety of about 4.3.

9. Analysis of Bracket Interface

Stainless steel brackets will be used to attach the previously mentioned aluminum struts to the aluminum bulkheads along the length of the rocket airframe. For each bulkhead plate, there will be 8 brackets attached; 2 for each of the 4 struts. The attachment brackets are stainless steel which has an ultimate strength of 73200 psi. A finite element analysis was used to validate the structural integrity of the attachment brackets during flight. For the scenario, a loading of 137.5 lbf was used acting normally to the face of one side of the bracket. 137.5 lbf is 1/8 of the maximum 1100 lbf expected upon initial takeoff of the rocket as this load will be distributed across 8 brackets. Figure 26 below shows the stress concentrations across the bracket.

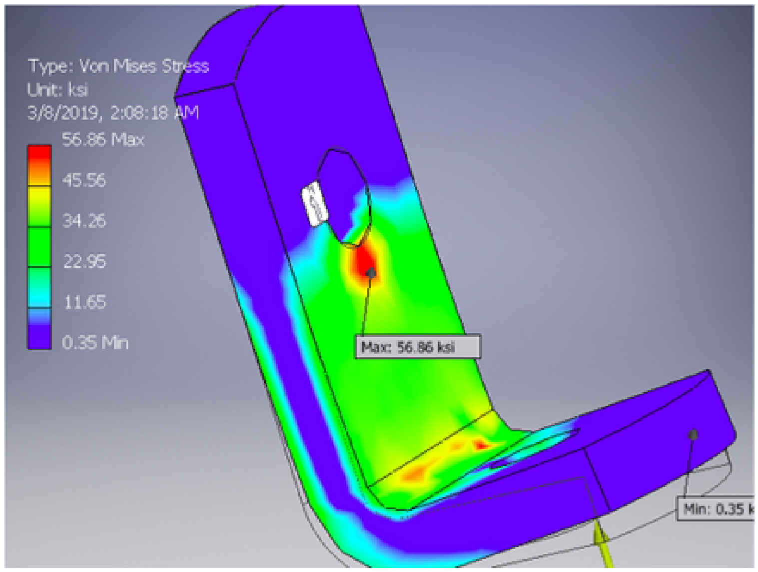


Fig. 26 Von Mises stress distribution for bulkhead attachment brackets

Clearly, one can see local areas of stress concentrations around the joint of the bracket and around the point of connection of the bolt. From the above analysis it was found that the maximum Von Mises experienced during flight is about 56,900 psi. This is below the ultimate strength of stainless steel by a factor of safety of about 1.3. While this is not a high margin, it should be noted that the loading values used in the above simulation are conservative in that the expected loading per bracket will be much less during much of the flight.

10. Airframe Overview

The airframe will consist of the structural components on the exterior of the vehicle. More specifically, the airframe will consist of the nose cone, body tubes, and the fins. The design rationale and proposed manufacturing processes for these components are outlined below. In order to satisfy the requirement 4 which states that the team should design and build the launch vehicle in house, the team has decided to manufacture all airframe components in house.

11. Body Tube Material Selection & Manufacturing

The team has considered a range of material choices for the body tubes. As with most aerospace applications, materials with high strength to weight ratios are desired. However, this dramatically reduces the types of materials that are feasible to use in this application. The most common materials that fit this are criteria are composite materials. In previous years, the team has considered the application of three common composites: kevlar, carbon fiber, and fiberglass. In previous years, the team has developed a standard layup wet procedure for manufacturing sandwich core laminates which use carbon fiber. Within the previous year, a significant amount of carbon fiber cloth has been donated to the team. Due to the relatively high cost of composite materials, the team has decided to use donated carbon fiber cloth as the primary material.

The team will also implement a sandwich core laminate. Sandwich core laminates are commonly used in aerospace applications such as missiles. The use of a sandwich core has the following benefits: higher strength ratio, high bending stiffness to weight ratio, high resistance to mechanical fatigue, and good damping characteristics. Furthermore, the use of a sandwich core requires less carbon fiber cloth usage which dramatically reduces costs. The final laminate can be seen in Figure 27.

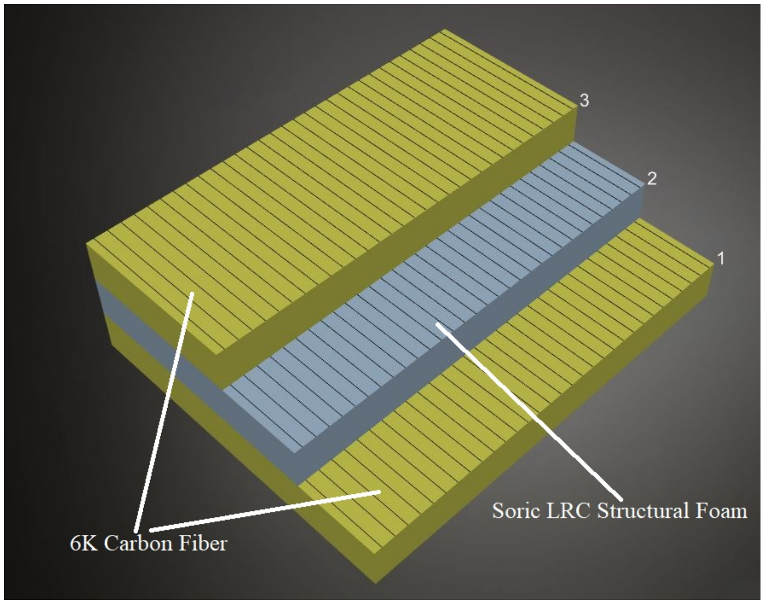


Fig. 27 3D model of final laminate stacking sequence

Composite materials have been widely analyzed for aerospace applications, most notably missiles. According to one source, carbon fiber with an epoxy matrix outperforms most other materials such as basalt and fiberglass. When analyzing these composites in compression, the carbon fiber - epoxy laminate had a maximum compressive strength of about 130,000 psi while the fiberglass laminate had a maximum strength of 45,000 psi (Radwan et al., 2016). Although these values are not representative of the setup the team will use, it does indicate that carbon fiber offers superior performance in compression over other composites. In previous years, tests of similar laminates have resulted in peak strengths of about 3200 lbf. This year, the team has reinforced the body tubes with an additional ply of carbon fiber resulting in a peak compressive strength near 25,000 lbf.

A wet layup process will be used to manufacture the body tubes of the vehicle, which will have an inner diameter of 8 inches. This inner diameter was chosen because it is large enough to hold the payload, while also being a common size for interfacing with COTS rocket components as mentioned previously. It also makes finding an appropriately sized mandrel easier. An aluminum mandrel will be used as the mold for the layups. After the carbon fiber cloth and the structural foam are saturated with epoxy resin, they will be wrapped around the aluminum mandrel and cut to length. Next, a Mylar sheet will be placed over the wrapped cloth to ensure a uniform surface finish. Finally, heat shrink tape will be utilized to remove excess epoxy from the laminate. After the body tubes finish curing in an oven, dry ice will be used to rapidly cool the aluminum mandrel. Since aluminum has a much higher coefficient of thermal expansion than the carbon composite, the mandrel will contract when cooled to create a small amount of separation. This process allows for easy removal of the parts from the mold. The final manufactured laminate can be seen in Figure 28.

Fig. 28 Example of final body tube laminate.

12. Fin Design & Manufacturing

The team has successfully flown plywood and balsa wood core fin with carbon fiber or fiberglass overwrap, this year we will be flying nomex core carbon fiber fins. Once again this is because these materials are readily available and provide high strength to weight ratios. The manufacturing process we have developed has led to a lightweight and strong fin with a smooth finish straight from casting with no vacuum bagging required. The method is highly repeatable and reliable for creation of large casts for multiple fins.

In order to ensure that the fins can withstand the aerodynamic loads of flight, some basic simulations were performed in an open source software known as FinSim. The team determined the loading requires to use during static loading and produce a factor of safety of at least two. The simulations below are for the largest possible fin dimensions (as manufactured) to get an estimate of the pressure distribution across the fin. The fins are manufactured with a 16" chord, 12" span and 7.5" tip chord Figure 29,this is the size oversize fin we are manufacturing to be cut down to the appropriate size for flight and chosen caliper. We are testing with a maximum velocity of mach 0.6 and it should be noted these simulations are not to predict failure of the fins, just the loads present. Specifically, we are looking at the 112 lbf load and 60 ft-lb moment Figure 29 due to a 2 degree angle of attack as this is the primary concern for the structural design of the fins, chord moment generated from lift. The fins will be designed to handle a higher load than this depending on the maximum off-course angle of attack we choose but the relationship is linear in FinSim so we can extrapolate the force easily as a force-per-degree angle of attack. The team will verify whether the chosen maximum loads are handled adequately by the fins by static distributed load testing.

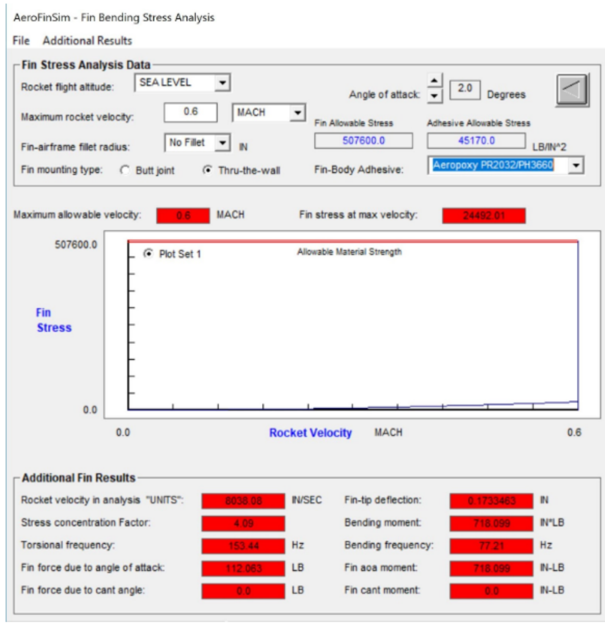


Fig. 29 FinSim Analysis.

The number of layers of carbon fiber is chosen for each side of the fin. Then 12x24" layers are wetted and laid on a high-density fiber board covered with mylar as a method of release, the mylar gives a great finish and comes off clean with no mold release needed. Then a 12" x 24" piece of rectilinear nomex core is placed on the wetted carbon with the long side of the rectangles extending in a direction optimal for counteracting applied fin moments during flight and made sure it is wicking up the appropriate amount of epoxy for adhesion before being covered with another piece of high-density fiberboard to sandwich the cast together. The cast is then placed in an oven for roughly 2 hours to cure and the result is a smooth cast of one side of the fin (Figure 30). This method allows us to control each side of the carbon as they are cast individually and gravity does all the work of keeping epoxy from soaking too far in the honeycomb while maintaining a glass finish (Figure 31). After, the layers for the other side are wetted and placed on another 12" x 24" sheet of mylar on high density fiberboard where the previous cast is then placed nomex side down on the carbon fiber before being cured in the oven. The result is a light, hollow, nomex core fin with a glass finish on both sides. Structures will continue to investigate the structural integrity and consistency of the fin casts. Additionally, the epoxy used is recorded after each cast to ensure consistency of mass and to be compared to final fin mass. It should be noted these are the fins with square edges before aerodynamic leading and trailing edges have been attached (Figure 33), those are currently under development.

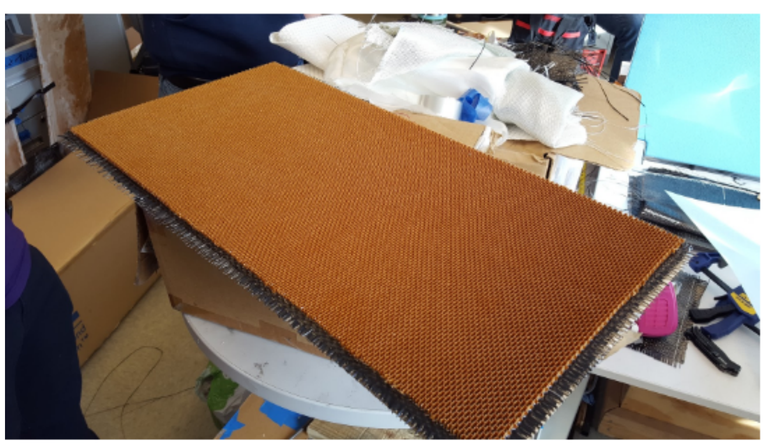


Fig. 30 Nomex side up after casting one side.

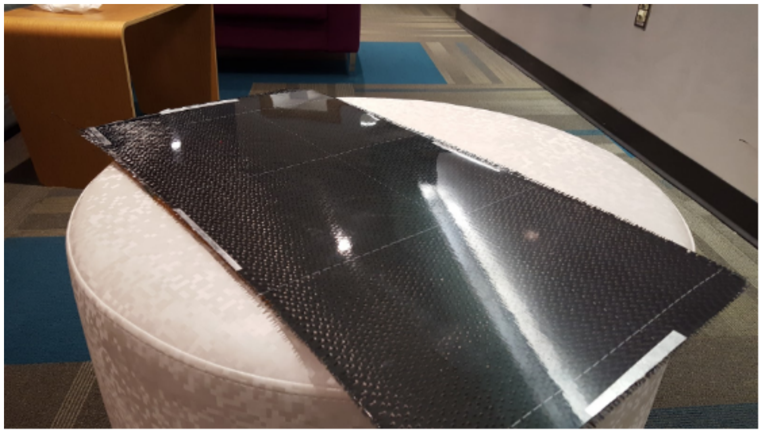


Fig. 31 Smooth finish of one side

Fig. 32 Close-up of the epoxy wicking up to the nomex for full adhesion

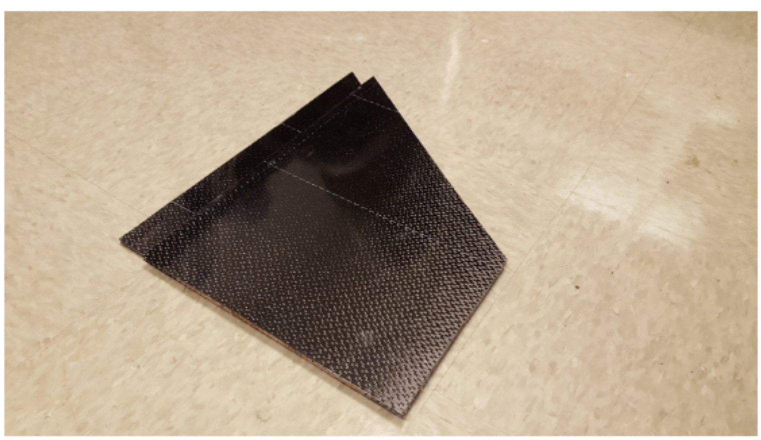


Fig. 33 Fins cut out

The fins produced below are 16” chord and 12” span, these are drastically oversized fin plate casts to allow us room for design changes in placement of mass or adjustments in calipers by adjustment of the center of pressure. If we need to increase the fins for any reason, we will have the material to do so and if we need much smaller fins we can cut them down. This will be time saving when it comes to the final design and mass placements.

The fins will be tested by manually distributed loads (sand bags etc.) to determine the failure point of the fins to make sure we maintain a consistent safety factor of two as compared to FinSim predicted aerodynamic forces.

13. Nose Cone Design & Manufacturing

The nose cone design mainly focused on determining geometry with the least skin friction drag. At an estimated maximum velocity of our rocket, Mach 0.66, it was found that the pressure drag is minimal on the nose cone regardless of geometry. Similarly, in our analysis the form drag caused by the nose cone varied very little from geometry to geometry. Of the two designs that were considered – Von Karman and a Tangent Ogive – the Tangent Ogive with a fineness ratio of 3:1 was chosen. This specific fineness ratio and geometry produced the least skin friction and pressure drag in our OpenRocket analysis, seen in Figure 34. To manufacturing simple, the nose cone picked was modeled after an available commercial nosecone.

	Diameter	Mach	Fineness Ratio	Length	Nose cone Cd	Total Cd	Stability	Apogee
	in	-	-	in	-	-	cal	ft
Ogive	8.25	0.6	3	24.75	0.03	0.55	1.67	8279
			4	33	0.04	0.55	1.63	8198
			5	41.25	0.04	0.56	1.59	8117
		0.67	3	24.75	0.03	0.56	1.73	8279
			4	33	0.04	0.57	1.69	8198
			5	41.25	0.04	0.57	1.65	8117
Von Karman	8.25	0.6	3	24.75	0.03	0.55	1.69	8296
			4	33	0.03	0.55	1.65	8214
			5	41.25	0.04	0.56	1.62	8134
		0.67	3	24.75	0.03	0.56	1.75	8296
			4	33	0.03	0.57	1.71	8215
			5	41.25	0.04	0.57	1.68	8134

Fig. 34 Nose cone shape optimization analysis

To design a Tangent Ogive nose cone, basic dimensions of the rocket must be established. The rocket will have an estimated 8" ID and 8.25" OD. With a fineness ratio of 3:1, the nose cone is 24" long with a 6" coupler section. To manufacture the profile geometry of the nose cone is calculated and used to create a CAD model of the nose cone. The model is split into sections to be 3D printed and assembled to create a nose cone shell which is then be reinforced by 5 layers of fiberglass.

The Tangent Ogive function is created in MATLAB to create a vector of multiple sets of points to create the geometry for the nose cone. These points are imported into Inventor to create a spline from which a side profile of the nose cone is made. These plots can be seen in Figure 35. This geometry is revolved and shelled with a wall thickness of 0.25" to create the basic internal structure of the nose cone. This model is then split into several printable sections, keeping in mind the printing bed of the 3D printer being used.

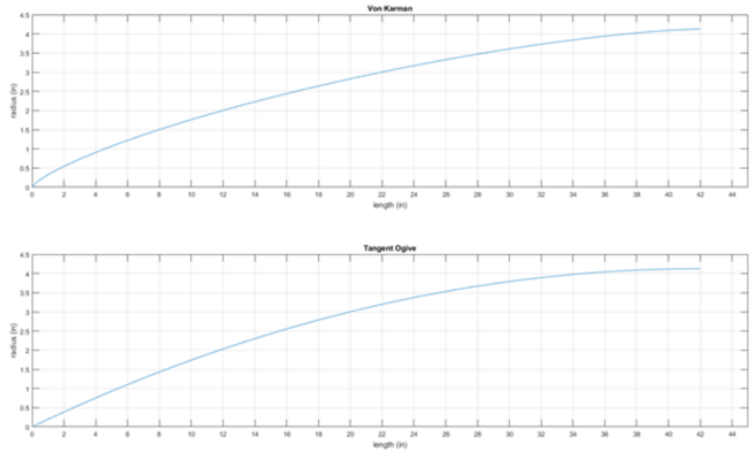


Fig. 35 Nose cone profiles plotted in MATLAB

These parts are then printed at a 0.3 mm layer height, with 15% infill. Though this is the internal structure of the nose cone, the fiberglass will reinforce the nose cone to withstand the forces the rocket will experience. A wet hand layup will be done on the 3D printed shell, while alternating the location of the fiberglass layer seams to avoid areas of weakness. The 3D printed shell can be seen in Figure 36. Each fiberglass layer will be the transpose of the nose cone surface area onto a 2D plane.

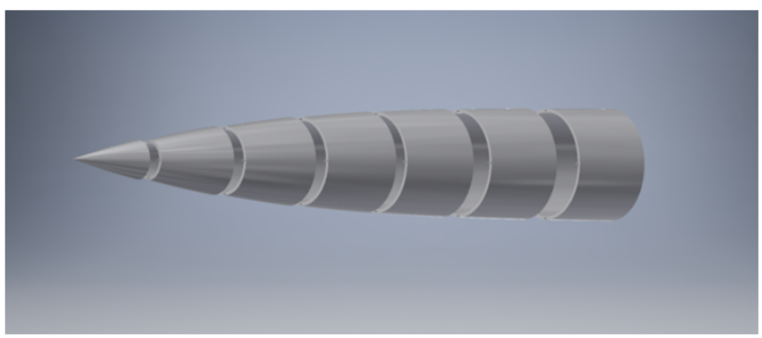


Fig. 36 Nose cone 3d printed shell

14. Stability Analysis

The launch vehicle was primarily modeled in Open Rocket to determine the resulting aerodynamic properties. The Center of Gravity was estimated using a SolidWorks CAD model which had the most detailed mass properties for each subsystem. Furthermore, the Center of Pressure was determined by inputting the geometry of the Launch Vehicle into OpenRocket. The result are as follows:

CG = 115.7 inches as measured from the tip of the nose and CP = 132 inches as measured from the tip of the nose.

These parameters can be used to calculate the static stability margin:

$$CP - CG/8.25 = 1.91 \tag{2}$$

The static stability margin complies with requirements in Appendix F.

C. Avionics and Recovery Subsytems

Recovery and Tracking

1. Tracking Systems

A successful recovery rocket worth 35% of the points possible. And the proximity of the apogee of the rocket to 10,000 ft, which can only be confirmed upon the recovery of the rocket, worth 35%. So, to guarantee a successful recovery, the team set up two criteria. First, the range of the tracking system should be at least 1.5 mi. This requirement is based on the last year's test launch and launch at the competition. At the test launch last year, the rocket landed about 0.2 mi away from the launch site. At the competition, the rocket landed about 0.8 mi away from the launch site despite strong wind. So, as a safety factor, the team set up an internal criteria of 1.5 mi range. Second, the accuracy of the tracking system should be within 10 m. 10 m accuracy will allow the recovery team to spot the rocket in plain sight. Furthermore, SA Cup Design, Test, and Evaluation Guide states that the team should implant two separate tracking systems for redundancy. And, it also states that at least one should be COTS. So, the team has developed one tracking system and chosen one COTS tracking device independent from the first one.

The first tracking system uses the telemetry system that the team has developed. The telemetry system uses Adafruit Ultimate GPS, Adafruit RFM96W LoRa Radio Transceiver, and Great Scott Gadgets ANT500 antenna all connected to BeagleBone Black. One setup will be sat inside the body tube with the antenna mounted externally through a pressure hole. The exact same setup will be held by the team at the basecamp to communicate with the onboard setup. Ultimate GPS provides 3 m accuracy, and RFM96W provide 1.5 mi range. These meet the team's criteria. The telemetry system will be further explained later. For the second tracking system, the team is using Seagull Rocket System by Eagle Tree Systems. Seagull Rocket System is a telemetry/logging system used for high powered rocketry and is composed of two parts. It includes transmitter for the rocket and a dashboard for the ground-base. Onboard transmitter will be mounted inside the nose cone which is made out of fiber glass allowing radio signal to be transmitted through it. Seagull Rocket System allows the team to track GPS position, altitude, and the rocket's trajectory in real time. Even if the transmission stops when the rocket lands, the last known position will be available on the ground-base dashboard. It is used for more intense rocketry having range of 32000 ft (6.6 mi).

2. Tracking Systems Testing

Throughout testing, the team determined the GPS was highly accurate, as the GPS coordinates shown on Google Earth were within a few feet of being exactly where the GPS was physically. The team encountered a problem with the GPS not getting a fix, but determined the problem was because the GPS was in a car or in a building (The metal roof would block the GPS's signal). To avoid the problem, the team placed the GPS on the dashboard of the car under the windshield, allowing the GPS to obtain a consistent fix. The team then ran a test where the GPS was on the dash of the test vehicle and the dashboard and receiver were at the base site with the rest of the team. The dashboard integrated with Google Maps accurately mapped the GPS's position in Google Maps in real time. The car drove around in a circle with an increasing radius having the monitoring system at the origin. Reliable connection was confirmed in the given range, and the operation time was about 4 6 hours.

During launch, the team determined it would be best for the Seagull Rocket System to start recording data once the rocket has reached 10000 ft AGL. The alternative option is to start recording when powered on. Due to memory limitations of recording times of 45 minutes maximum, the team would not be able to rely on this method, as the rocket may sit on the pad for more than 45 minutes or sit in the desert for several hours at competition.

3. Altimeters

Section 2.5 of SAC Rules and Requirements requires the team states that the team need to carry a COTS barometric pressure altimeter with on-board data storage. And, the altitude data from that altimeter will be used for the official scoring. So, for the official altitude logging, the team decided to use StratoLogger CF since it's a barometric pressure altimeter and provides internal storage. The team chose StratoLogger CF based on its reliability from our past launches. The team has been using StratoLogger for the past three years, and it has performed well without failing. For the redundancy, the team uses three StartoLogger CFs each with separate battery. Also, these altimeters will be used for the parachute deployment. The altimeters send electrical pulses to leads connected to them when certain flight events occur, such as apogee or when a specific programmed altitude is reached during descent. These electrical signals are used to ignite electrical matches (E-match) that set off parachute ejection charges. When these altimeters are armed, they sound a unique beep sequence that is used to audibly confirm ejection charge continuity. Each of the altimeters have their own main and drogue E-match, 9-volt battery, and rotary arming switch to maximize the redundancy in the recovery system.

4. Static Port Sizing

The altimeters must measure the launch vehicle's external static air pressure to accurately calculate the altitude barometrically. Static pressure ports must be added to the airframe to allow the atmosphere to flow through. To determine the size of the ports, the team used the equation provided by the manual of the altimeter the team use [1]. Using the diameter of 8 in and the electronics bay length of 9 in, the equation conclude 4 holes with the diameter of 0.4608 in. 3 holes will be evenly made around the body tube where the electronics bay will be located and one hole will be used where an antenna for the telemetry system will be exerted out externally.

5. Parachutes and Descent Rates

Section 3.1.1 and 3.1.2 of SAC Design, Test, Evaluation state required ranges for the descent rate for each deployment event. For the initial deployment when the drogue chute comes out, the descent rate should be between 75 and 150 ft/s. For the main deployment when the main chute comes out, the descent rate should be less than 30 ft/s. To select parachutes that will bring the descent rates to the required ranges, the team used the parachute descent calculator provided the manufacture [2]. The drogue is a 36 in Iris parachute which will bring the descent rate down to 81.8 ft/s, and the main-chute is a 168 in Iris parachute which will bring the descent rate below 17.5 ft/s. The iris parachutes were chosen because the iris parachutes, having the apex (crown) of the parachute pulled down inside canopy, create Cd of 2.2 or more allowing them to create more drag compared to regular parachutes with the same dimensions thus reducing the packing volume. The fully deployed parachute can be seen in 37.

Fig. 37 Iris Parachute fully inflated.

6. Shock Cord and Knots

Both parachutes are connected to 1/2 inch diameter Kevlar shock cords. These cords were given sufficient length to prevent the launch vehicle sections from colliding into each other during descent. The ends of the shock cords are tied to a stainless steel quick link using a Palomar knot. This knot was used because it is easy to tie and has an approximate tensile strength loss of only 5% [3].The main chute is attached part way along its shock cord using an alpine butterfly loop. This knot was used for similar reasons, having a tensile strength loss of 20% [4].Throughout the deployment of both parachutes, all rocket components remain attached via the shock cords. The shock cord is tubular kevlar rated up to 7200 pounds, giving it the necessary strength to withstand the shock load during parachute deployment.

7. Ejection Charge and Shear Pin Sizing

An online calculator [5], which is certified by IEEE, will be used to determine the size of the black powder charges and number of shear pins required following the team's faculty advisor's recommendation. The calculator's formula is CDDL = m (Where C is the psi value, D is the airframe diameter, and L is the length of the tube). The calculator also displays the number of shear pins needed to hold the sections of the rocket together at the parachute bay openings until the charges detonate. A psi value of 10 to calculate the initial amount of black powder needed for each charge. This amount of black powder would be used for the first ground test. More about testing of the ejection charges is explained in the next section.

8. Ejection Charge and Shear Pin Testing

A series of tests will be performed to test the accuracy of the calculator prior to the launches. The exact amount of black powder and shear pins will be implanted, then they will be tested if they are sufficient for separating the body tubes. Then, depending on the result, 10 percent of the total amount of black powder will be added or deducted. The same steps will be repeated until the right amount of black powder is found. Also, the same test, the team will test if the number of shear pins are sufficient to hold the tubes together. Traditionally, the number of shear pins from the calculator was lower than the experimental value. So, the team expect to have about 10 percent increase in the number of the shear pins after the tests.

Telemetry

For the secondary tracking and motor control, the team is developing a telemetry system which uses COTS transceivers and microcontrollers (BeagleBone Black). One Adafruit RFM 96W LoRa Radio Transceiver (frequency of 433 MHz) will be attached to one BeagleBone Black inside the rocket. A Great Scott Gadgets ANT 500 antenna attached to the transceiver will come out externally through an 1 in diameter hole. The set inside the rocket will communicate with the same set on the ground-base where team can communicate and monitor. The set on the ground-base will be attached to a laptop. This transceiver has about 1.5 mi range depending on the environment provided. The range is provided by the manufacturer. This setup was chosen to meet section 10.3 SAC Design, Test, Evaluation which sets the required operational range. For the launch control, it states that the maximum operational range should not be less than 2,000 ft and that 3,000 ft range is preferred. After testing, it was confirmed that the system can deliver the signal without a drop in the data rate until about 1,800 ft, and the desired operational rage that surpasses the recommend operational range was also achieved.

Electronics Bay

In previous years, the electronics bays were mounted in a fixed location inside the rocket; therefore, accessing and working on the electronics bay while the rocket was assembled challenging. So, the team developed an electronics bay that allows each plate to get separated individual from the bay. As shown in Figure38, each plate is connected by threaded rods and bolts or screws and standoffs. Since each plates are held together by threaded rods that are independent from other plates, each plate can be separated individually by removing the bolts. To hold the plates in place, spacers were used in between the plates with threaded rods going through. Also, to keep the bottom and top plates flat, standoffs with threading inside were used for the top and bottom plates instead of the spacers, and negative spaces were made on the top and bottom plates so screwheads don't stick out externally. Since the load of the rocket is carried by the aluminum tubes that run through the rocket, there is only minimal load applied to the bay. So, having short screws and spacers placed in alternating locations is structurally safe.

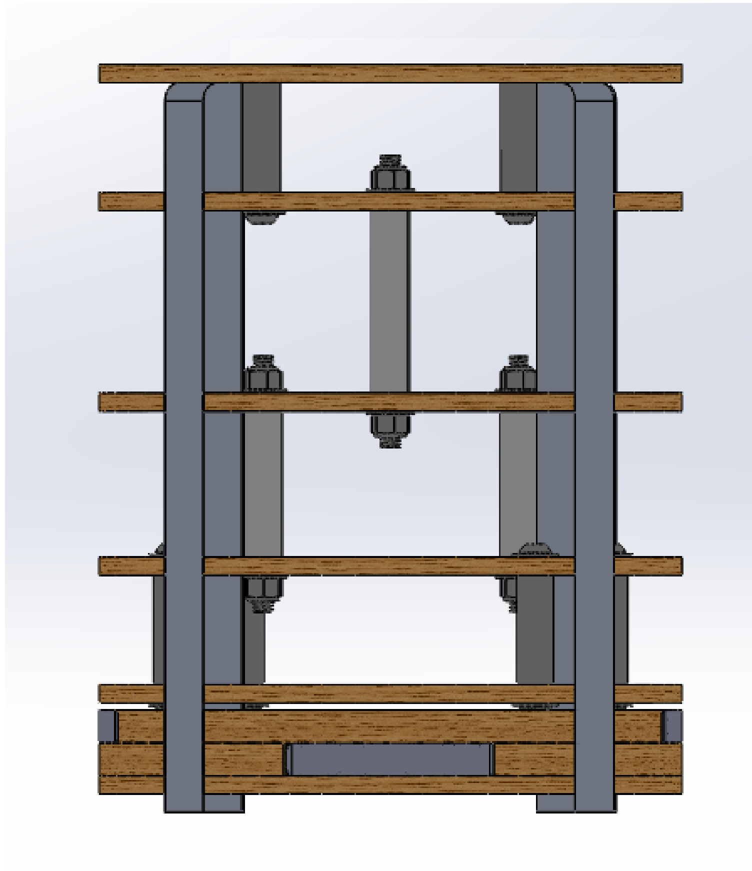


Fig. 38 Electronics Bay Assembly

Active Drag System (ADS)

9. Introduction and Motivation

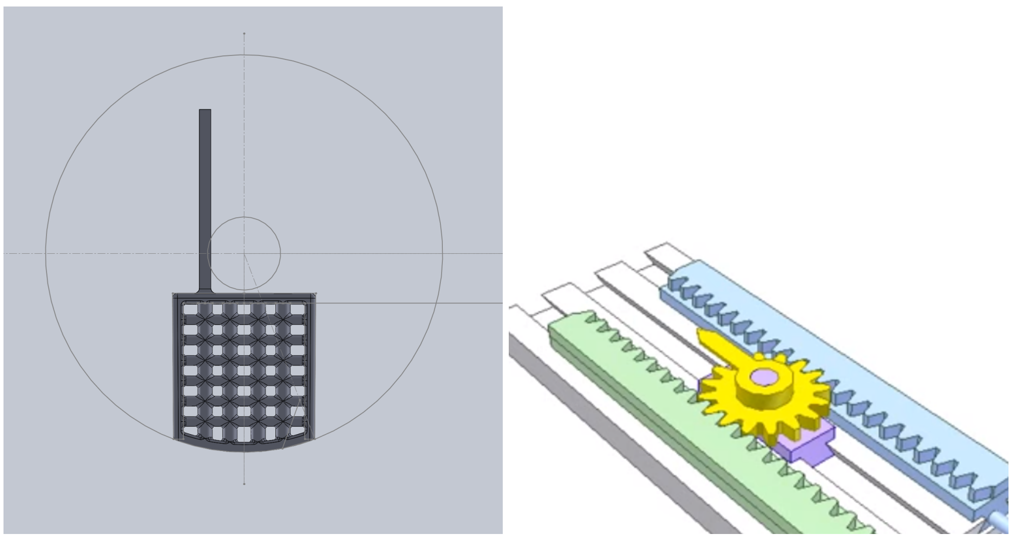
Thirty five percent of the scoring at the Spaceport America Cup is based upon the accuracy of the rocket's predicted apogee. Weather conditions and differences in the manufacturing of rocket motors, factors outside of the control of the rocket designers, cause trajectory analyses to be somewhat inadequate in accurately predicting a rocket's apogee. Therefore, finding an active method to control the rocket's apogee in flight was desired. The two main ways to accomplish this are throttling the rocket motor thrust or implementing an Active Drag System (ADS). Propulsion team is developing a hybrid motor for this purpose, and Avionics team has been implementing ADS for the past two competition. After motor burnout the ADS predicts apogee using a combination of temperature, acceleration, and air pressure data. The system then deploys air brakes, slowing the rocket down, and decreasing its apogee. The ADS continues to predict the trajectory and actuate the air brakes accordingly till apogee is reached.

10. ADS Concept Generation and Selection

The ADS design from the 2018 rocket had a drawback. Air brakes for 2018 ADS were thin solid plates and were actuated outward in a rotational motion. While they generated more drag, being solid plates, they made the rocket unstable possibly making the rocket to roll. Also, it made the rocket to roll more by actuating in a rotational motion. The team drew this conclusion from a flight footage video that shows the rocket start being unstable as the ADS brakes come out and IMU data that shows how the acceleration data fluctuate greatly after the burnout. So, the team designed a new system where the air brakes have grid design (not a solid plate) and actuate outward in a lateral motion.

11. Hardware and Mechanical Design

To solve the instability issue, the team designed a fin (airbrake) that looks like a grid shown in Figure 39a. Each grid line of the fin has round edges. This design let the airbrakes to create drags with less vortexes thus making the rocket more stable. However, having a grid design causes the break to have less surface area. So, it creates less drag than a solid plate. However, this year's rocket is taller and have bigger diameter; therefore, this year's airbrakes got bigger and thicker compensating the drag losses. To actuate the airbrakes linearly, the team is implanting a gear and rack system shown in Figure 39b. Each rackwill have one airbrakes attached to an opposite end. Two layers of this system will be stacked crossing each other sharing one tall center gear. As the servo motor rotates the center gear, total four airbrakes will be actuate uniformly. The rotation will stop before the racks are exposed. For the previous years, the team used acrylic plates for the airbrakes. With serious of testing and launches, the team concluded that an inch thick acrylic plates are strong enough to withstand the drag force during the flights. However, due to the complex design, the team decided to 3D print the airbrakes. According to MatWeb, both PETG and ABS (material available for 3D printing) exceed material strength of common acrylic. But, due to the nature of 3D printing where it reduce the density of the 3D printed material, simple data of material strength wouldn't be sufficient to decide the validity of the material. To test if the material is sufficient to withstand the drag, the team will do stress analysis. The team will use drag from a calculation made by assuming the airbrakes are thin plates. This assumption will work as a safety factor itself since thin plates generate significantly more drag than the grid.



(a) ADS Grid Fin **(b) ADS Gear System**

12. ADS Electronics

The ADS electrical system consisted of five main components: a servo, a flight computer, sensors, power regulators, and a battery. The servo is a RJX FS-0521HV which, at 8.4 V, can output 295.9 oz-in. This servo also has quick actuation speed that allows the ADS to fully deploy in only 0.05 seconds. The flight computer consists of one BeagleBone Black. A microSD card reader shield is attached to the BeagleBone Black for data logging during the rocket's flight. For its low cost and ease of implementation all sensors used are located on a Pololu AltIMU-10 v5. This circuit board has gyros, accelerometers, and an altimeter The power to the BeagleBone Black and servo are regulated by two LM2596 Buck Converters. One regulator has been adjusted to output 8.4 V for the servo. The other was set to output 5 V for the BeagleBone. The BeagleBonehas an onboard regulator that supplies 5 V to the AltIMU. The ADS is powered by a 2200 mAh 11.1 V LiPo battery. With this battery, the ADS has a standby time of at least 4 hours. The battery is also connected to a HobbyKing LiPo Checker. The checker will output a loud buzz when the battery voltage lowers beyond a safe value. All of the ADS electronics are bolted to shelves and stacked above the ADS petals.

13. Algorithm Development

The development of the algorithm was based off of the mathematical model that the team made and was written in python. The team used the numpy and scipy libraries to emulate ode45 from MATLAB. The algorithm first detects the launch of the rocket. Once the launch has been detected, the algorithm will write the data from the IMU to a file so that it can be used later for data analysis. Once the algorithm detects burnout of the rocket motor, calculations for the apogee prediction are done. The first case of this prediction is special, since the ADS flaps will only actuate to 70 percent of the calculated required flap deployment to attain an apogee of 10,000 ft, to avoid overcorrection initially. From there the algorithm will keep predicting the apogee, using a constantly updating state vector, until the rocket either reaches apogee, or the prediction comes within 100 feet of the target. After the algorithm senses the rocket has reached apogee, the ADS is fully retracted and data collection is shut off.

D. Payload Subsystems

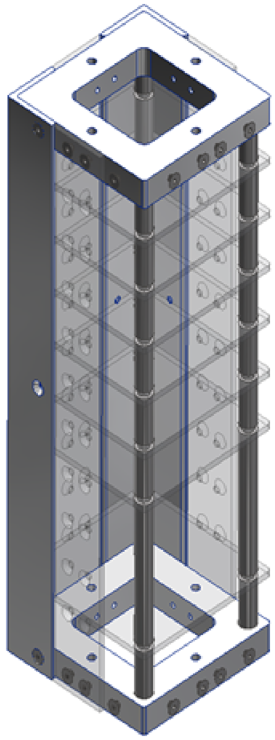
The 2018-2019 payload subteam consists of 7 members and is focusing on two primary objectives. The first is to design a new modular CubeSat structure that is customizable and reusable year to year. The second goal is to compete in the Space Dynamics Laboratory CubeSat challenge with a vibration isolation study to determine effective ways to mitigate the transmission of vibrations on a sounding rocket. This will be conducted by using passive vibration isolation materials and the development of an active control system, as a stretch project, to target low frequency vibrations. The CAD and constructed model can be seen in Figure 40 and the system requirements for the payload can be found in appendix G.

(a) CubeSat experiment CAD model

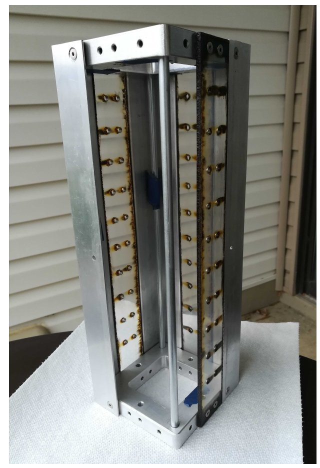
(b) Assembled CubeSat

Fig. 40

The CubeSat is designed to be a reusable modular structure with easy accessibility to on board components within a 3U (10x10x30 cm) CubeSat form factor. The CubeSat is designed around movable platforms for supporting experimental hardware on the CubeSat. The platforms are internally adjustable to meet our modularity design requirement and can be changed to different heights to support a wide variety of layouts within the CubeSat. The platforms are installed on a system of 4 quarter inch steel rods with aluminum piping as spacers. The 4 rods then mate with a top and bottom frame to constrain the rods to the CubeSat form factor. The primary load bearing components of the structure are 4 1.5 inch aluminum angle irons that are attached to both frames. Additionally, 4 acrylic side panels are placed to give viewing access into the payload and for mounting additional switches, hardware, and small electronics. The CAD rendition and the final structure and be seen in Figure 41.



(a) CubeSat structure CAD model



(b) manufactured CubeSat structure

Fig. 41

All components are rated to withstand 675N, 16 g's of acceleration of the CubeSat mass. Load Calculations can be found in Appendix G. The CubeSat integrates with the rocket through 8 10-32 bolts that are inserted through each end frame and secured to the adjacent bulkheads.

Excessive shock and vibrations are a known cause of failure for avionic and payload specific electronics and hardware. The vibrations experienced on-board rockets can originate during the launch process, at separation events, from aerodynamic forces, and motor vibrations. The bulk of the high amplitude vibrations experienced on board a rocket occurs during the launch process and comes from the motor. These vibrations can break soldering joints, pcb boards, electrical wires, fasteners, and can introduce noise into sensitive electronics and imaging equipment. This in turn can limit the on-board hardware and electronics used as they may not be suitable for the high vibration environment [6]. Our Payload is focusing on analyzing the lower frequency vibrations,1-2000 Hz, as explored in a 2013 California Polytechnic State University Thesis paper by Steve Furger focusing on Analysis and Mitigation of the Cubesat Dynamic Environment [7]. We have selected to employ two forms of vibration isolation: Passive and Active.

The first method tested is using passive vibration isolation materials. Passive vibration isolation materials are unique as they have a low internal spring rate and internal hysteresis dampening. When massed and sized appropriately, the low natural frequency and internal hysteresis dampening reduces force transmissibility of higher frequencies. This drives the desire to select materials with the lowest spring rates and highest damping possible. After extensive research online, the passive materials are selected as Sorbothane, neoprene, and memory foam. The can be seen in Figure42.

Fig. 42 Sorbothane (left), Neoprene (middle), Memory Foam (right)

Sorbothane is selected as it has a low spring rate and high internal damping. Neoprene does not perform as well as Sorbothane but is easily commercially available to all consumers making it ideal for amateur rocketry hobbyist. Memory foam is selected based on a hypothesis that through its flexible and soft characteristics it shows some promise for isolating small masses. Sizing the materials was proven difficult due to the small masses each material is supporting. Both elastomers are cut into a 1.5 inch diameter annular ring with a wall thickness of 0.25 inches to decrease stiffness. A copper mass is applied to both material to make the supporting mass equal to 30 grams. These decision lower the natural frequency of both materials as seen in Table 1.

Dynamic Modulus frequencies	Sorbothane Natural Frequency, Hz	Neoprene Natural Frequency, Hz
15	130.64	255.22
30	150.85	294.17
50	168.65	329.49

Table 1 Sized and massed passive elastomers material's natural frequency

The foam is left as a 1.5 inch cylinder with a height of 0.5 inches. Each material is mounted to its own standalone form and mounted to a polycarbonate platform as shown in Figure 43.

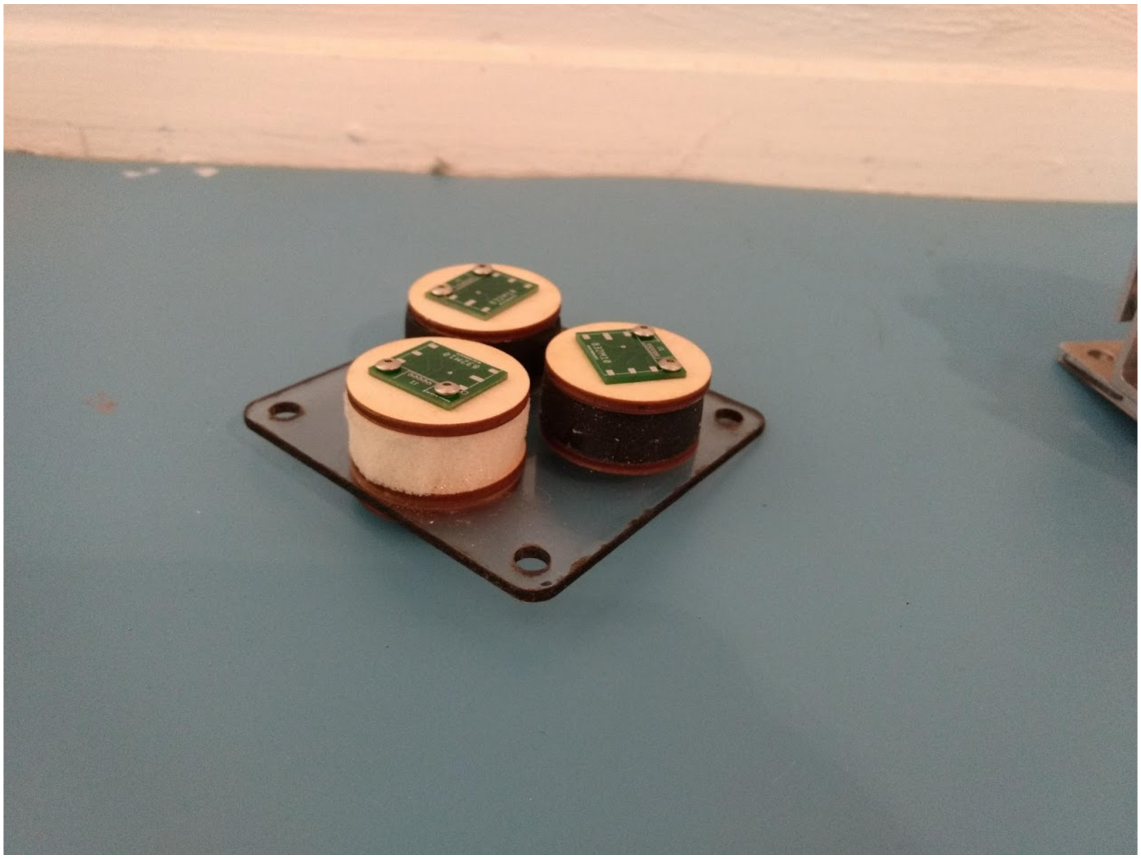


Fig. 43 Modular testing method for passive materials and passive material platform

A high performance 3 axis accelerometer is mounted to each material. All of the samples are secured to the platform in a radial pattern where the x direction of the accelerometer points radially and the y direction of the accelerometer points tangential to the circular pattern. This configuration is done to improve the symmetry and similarity of experienced vibrations.

The second method explored is an active vibration isolation system. This system functions by actively exerting a displacement and force with equal magnitude and in opposite direction of an experienced vibration such that net experienced force of the object is near zero. The forcing device selected is a 35.4 Newton voice coil actuator with a 1.64 cm stroke. The experiment will be performed as a single degree of freedom on in a direction, red, normal to the center axis of the rocket, black, as shown in Figure44.

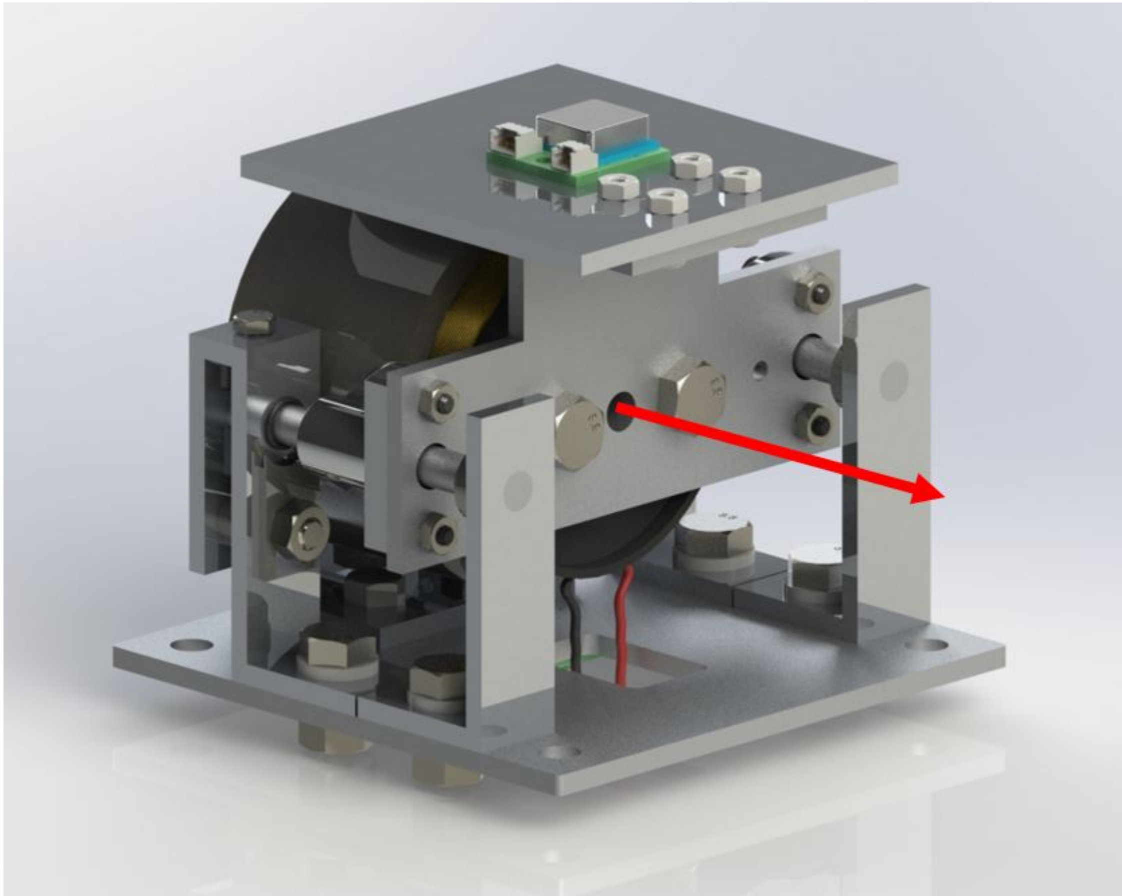


Fig. 44 Active vibration isolation hardware and mechanical system

The control system for the active system is fully analog to reduce latency commonly associated with a microcontroller. The tunable analog system can be seen in Figure 45.

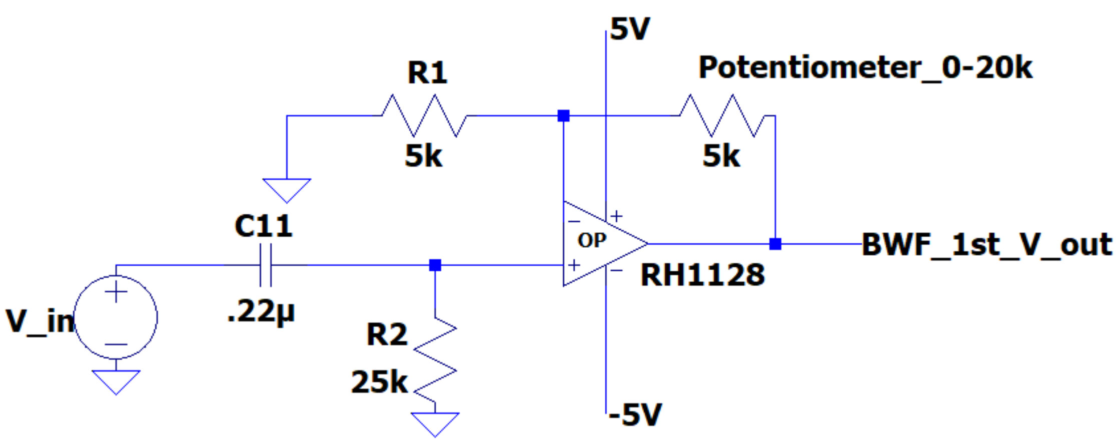


Fig. 45 Active vibration isolation signal conditioning circuit and commercial amplifier

A vibration sensed by our accelerometer on the active platform is passed through a first order ButterWorth high pass filter with a cutoff frequency of 10 Hz. A first order high pass filter is selected due to the minimal phase change resulting from the filter and is used to remove the changing acceleration resulting from gravity and the rotation of the launch vehicle by negating the max expected roll rate frequency of 5 Hz. The filtered signal is pass through a tunable inverting op amp where the gain can manually be adjusted using a 20k ohm potentiometer. A secondary amplification stage is then used provide the necessary voltage and current for the voice coil actuator. The both audio channels of the MAX9744 20 watt commercial audio amplifier are used in parallel to provide 6.03 Amps of current to the voice coil actuator with minimal latency and phase change. Additionally, the amplifier gives the ability to adjust the gain by 20 db. Testing is still underway to determine the exact gain and phase change needed to be applied to the voice coil to have adequate vibration isolation. While very early in its development, we plan to continue the development and accuracy of the active vibration isolation system of the next sequential years. With the use of the voice coil actuator, we were concerned about the potential magnetic field disrupting the onboard measurement devices. To mitigate this, magnetic shielding materials have been placed in the payload to protect the data acquisition systems. The magnetic shielding was testing and the results can be found in Payload appendix G.

The accelerometers used are 6 TE-Connectivity 832M1 +/- 25g accelerometers rated to 6000 Hz. A custom PCB, Payload Figure 7, is designed and manufacture to convert the accelerometer output to be 0-1.8V corresponding to +/-6g (x/y axis) and +/- 8g (z axis).

The acceleration range selected is based on historical IMU data and the voltage scaling is preformed such that the data can be recorded by a Beagle Bone Black. The Beagle Bone Black is used as the Data Acquisition (DAQ) System (as it has a 12 bit Analog to Digital Converter and is capable or recording up to 200K samples per second. Using the python library libpruio, our beagle bone black is capable of recording 7 analog pins at a rate of 5000 Hz, well above the Nyquist frequency of 4000 Hz. The payload is powered by a 6000mAh 7.2 volt li-po battery for both beagle bones and a 9000mAh 10.8 volt li-po battery for the active vibration isolation system. These batteries are capable of supplying the DAQ system for 3 hours and the active system for 1.5 hours. However, the system powers on when a launch command is received by the telemetry system to further improve battery life.

All of the data is collected as a CSV and is processed in matLab to compare the vibration transmissibility between the tested methods and their respective controls. The target benchmark of all methods is a minimum 6dB transmissibility loss. The collected results will be tabulated and compared against each other to asses the performance and to create a recommendation for vibration mitigation in sounding rockets.

III. Concept of Operations

An overview of the High Defective MK. II CONOPS is shown in Figure 46. Pre-Launch Phase begins with the attachment of the fill umbilical to nitrous fill port. After filling is complete, the umbilical is detached and the launch vehicle is placed on the launch rail. Active drag system and tracking/recovery electronics armed, and flight data collection begins. This phase ends with motor ignition.

The motor burn phase is defined by IMU detection of 3Gs in the vertical direction, which is approximately half the maximum expected loading that the vehicle should experience before clearing the launch rail.

The coast phase is defined by IMU detection of an acceleration sign change in the axis along the centerline of the launch vehicle, corresponding with the transition from acceleration from motor burn to that of drag in the opposite direction. For redundancy, this transition may also be defined by the passing of 7 seconds since the beginning of the Motor Burn Phase. Actuation of active drag system begins for apogee correction to 10,000 ft. AGL.

Decent phase one is defined by the moment at which the launch vehicle no longer has a vertical velocity, detected by the on-board barometer altimeters (StratoLoggers). For redundancy, this transition may also be defined by the passing of 7 seconds since the beginning of the Motor Burn Phase. During this phase, all three StratoLogger altimeters fire separate e-matches to ignite a black powder charge to deploy the drogue parachute, and the active Drag flaps fully retract. At 5,000 ft. AGL, both payload bay StratoLogger altimeters ignite a black powder charge to deploy the payload probe, which transmits atmospheric data to the payload bay, where it is recorded

Decent phase two is defined by StratoLogger detection of 700 ft. AGL, at which point all three StratoLogger altimeters fire separate e-matches to ignite a black powder charge to deploy the main parachute.

Recovery begins when the vehicle touches down, and electronics are used to position and retrieve the launch vehicle and probe.

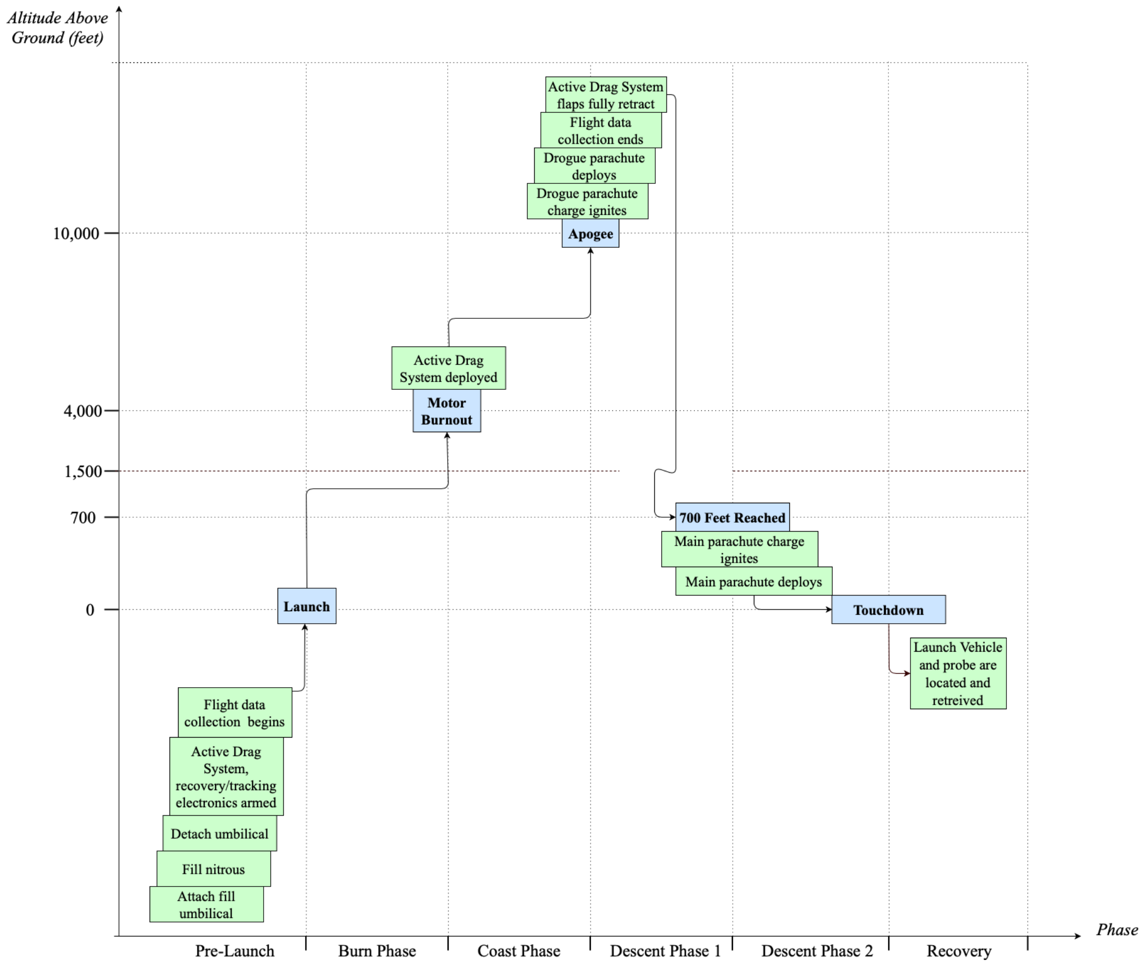


Fig. 46 Concept of Operations.

The motor burn phase is defined by IMU detection of 3Gs in the vertical direction, which is approximately half the maximum expected loading that the vehicle should experience before clearing the launch rail.

1) Pre-Launch Phase:
    1) Transition defined by the completion of the launch vehicle assembly checklist
    2) Attach fill umbilical
    3) Fill
    4) Detach umbilical
    5) Launch vehicle placed on launch rail
    6) Active drag system and tracking/recovery electronics armed
    7) Flight data collection begins
    8) Motor ignition
2) Motor Burn Phase:
    1) Transition defined by IMU detection of 3Gs in the vertical direction, which is approximately half the maximum expected loading that the vehicle should experience before clearing the launch rail
3) Coast phase:
    1) Transition defined by IMU detection of an acceleration sign change in the axis along the centerline of the launch vehicle, corresponding with the transition from acceleration from motor burn to that of drag in the opposite direction
    2) For redundancy, this transition may also be defined by the passing of 7 seconds since the beginning of the Motor Burn Phase
    3) Actuation of active drag system begins for apogee correction to 10,000 ft. AGL

4) Descent Phase 1:
    1) Transition defined by the moment at which the launch vehicle no longer has a vertical velocity, detected by the on-board barometer altimeters (StratoLoggers)
    2) or redundancy, this transition may also be defined by the passing of 7 seconds since the beginning of the Motor Burn Phase
    3) All three StratoLogger altimeters fire separate e-matches to ignite a black powder charge to deploy the drogue parachute
    4) Active Drag flaps fully retract
    5) At 5,000 ft. AGL, both payload bay StratoLogger altimeters ignite a black powder charge to deploy the payload probe
5) Descent Phase 2:
    (Transition is defined by StratoLogger detection of 700 ft. AGLAll three StratoLogger altimeters fire separate e-matches to ignite a black powder charge to deploy the main parachute
6) Recovery:
    1) Transition defined by when the launch vehicle touches down
    2) Recovery electronics are used to position and retrieve the launch vehicle and probe

IV. Conclusion and Lessons Learned

The 2019 Spaceport America Cup launch vehicle represents the most challenging and ambitious undertakings Rocketry at Virginia Tech has ever attempted. This years launch vehicle is 30% wider and longer, with over double the mass and installed impulse of any other launch vehicle the team has ever created. In addition, the inclusion of an internal structure, SRAD hybrid motor, and SRAD launch control system represents major technical challenges that up until this year the team had no prior background in. In terms of technical growth, the team learned it is imperative to begin and complete design as early as possible, preferably before the end of the Fall semester. In addition, it was learned that there are not only limits to the scope of what an undergraduate design team can do in a single academic year. As the project scope grows more pressure is put on maintaining timelines. This pressure can result in technical analysis, build quality, or timelines not being maintained. This year the team put a large emphasis on early technical analysis and coordination with Virginia Tech faculty. Consequently, more time was spent early in the year writing design reports for faculty and conducting design reviews. While the team is proud of this, it has made sticking to a timeline very difficult, and in many cases not feasible. To remedy this, the team was expanded to larger than any prior year, but it was soon realized this creates additional organizational challenges. It was found exceedingly difficult to bring 40 plus members up to speed on subsystems and prerequisite knowledge, especially while under a constant time crunch to hit timeline objectives. As a result the team leadership decided to institute the following changes for the 2020 Spaceport America Cup:

- Take advantage of the modular rocket design and reuse it for the 2020 year. Decide what to subsystems to redesign and make those changes, but do not redesign without reason. Implementing thsi will free up immense amount of time that can be directed towards developing additional flight systems, improving technichal analysis, or improving already existing subsystems, as well as helping to make the following changes listed below possible.
- Use the beginning of the Fall semester to conduct workshops for team members. These workshops will be designed to build engineering skills. Workshop ideas include how to Use Solidworks, OpenRocket, where to source materials, ect. These workshops will also better allow for knowledge to be transferred down to new team members.
- Have occasional one on ones between team leads and team members to give better ability for each to receive feedback. This will also help build more effective team members and future team leads.
- Do not allow Co-Captains to also be subteam leads. It was found this puts too much strain on the co-captains and results in poorer organization.
- Have team members conduct presentations for the team every week or every other week in order to demonstrate what that individual has been working on, as well as further engage the other team members. It is also aimed that this will foster pride in what each team member contributes.

It is Rocketry at Virginia Tech's belief that making these changes in operational procedures and changes in how the team goes about analyzing a technical problem will improve team dynamic, leadership, and technical competencies.

V. Appendix A1: Third Progress Report

Spaceport America Cup

Intercollegiate Rocket Engineering Competition
Entry Form & Progress Update

Color Key

SRAD = Student Researched and Designed

IMPOTANT CHANGE EFFECTIVE IMMEDIATELY FOR SA CUP 2019 EVENT

All inputs are mandatory for all submitions of this document. We undestand some data may change over time, this is completely acceptabl

Feel free to add additional commanets where needed, and be sure to fill out the last page. Treat the last page as a "cover letter" for your project.

Date Submitted: 5/20/2019

Country: United States

Team ID: 43 * You will receive your Team ID after you submit your 1st project entry form.

State or Province: Virginia

State or Province is for US and Canada

Team Information

Rocket/Project Name:	Highly Defective MK II
Student Organization Name	Rocketry at Virginia Tech
College or University Name:	Virginia Polytechnic Institue and State University
Preferred Informal Name:	Virginia Tech
Organization Type:	Club/Group
Project Start Date	9/1/2018
Category:	10k – SRAD – Hybrid/Liquid & Other

*Projects are not limited on how many years they

Member	Name	Email	Phone
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Alt. Student Lead	Johnny Jaffee	jjaffee@vt.edu	203-240-720
Faculty Advisor	Kevin Shinpaugh	kashin@vt.edu	540-231-124
Alt. Faculty Adviser	Eric Paterson	egp@vt.edu	540-231-231

For Mailing Awards:

Payable To:	Johnny Jaffee
Address Line 1:	29 Stony Hill Road, Ridgefield, CT. 06877
Address Line 2:	
Address Line 3:	
Address Line 4:	
Address Line 5:	

Demographic Data

This is all members working with your project including those not attending the event. This will help ESRA and Spaceport America promote the event and get more sponsorships and grants to help the teams and improve the event.

Number of team members

High School	0	Male	46
Undergrad	48	Female	3
Masters	1	Veterans	0
PhD	0	NAR or Tripoli	2

Just a reminder the you are not required to have a NAR, Tripoli member on your team. If your

STEM Outreach Events

Demonstrated the basic concepts of a rocket a activity regarding an air powered rocket with the children for the Virginia Tech Child Developme for Learning and Research.

Fig. 47

Rocket Information

Overall rocket parameters:

	Measurement	Additional Comments (Optional)
Airframe Length (inches):	201	
Airframe Diameter (inches):	8.25	*The base diameter of the vehicle is 8 in. The additional 0.2 incorporates the wall thickness of the airframe.*
Fin-span (inches):	24.125	
Vehicle weight (pounds):	145	
Propellent weight (pounds):	36	
Payload weight (pounds):	8.8	
Liftoff weight (pounds):	181	
Number of stages:	1	
Strap-on Booster Cluster:	No	
Propulsion Type:	Hybrid	
Propulsion Manufacturer:	Student-built	
Kinetic Energy Dart:	No	

Propulsion Systems: (Stage: Manufacturer, Motor, Letter Class, Total Impulse)

Please use a new line for each motor/engine. !!See examples below and DELETE EXAMPLES WHEN FINISHED!!
1st Stage: SRAD Hybrid , 12 pounds of paraffin wax fuel and 24 pounds of nitrous oxide, N class, 18,800 Ns

Total Impulse of all Motors: 18800 (Ns)

Predicted Flight Data and Analysis

The following stats should be calculated using rocket trajectory software or by hand.
Pro Tip: Reference the Barrowman Equations, know what they are, and know how to use them.

	Measurement	Additional Comments (Optic
Launch Rail:	ESRA Provide Rail	
Rail Length (feet):	17	
Liftoff Thrust-Weight Ratio:	6.63	peak thrust of ~1200lbf
Launch Rail Departure Velocity (feet/second):	61	
Minimum Static Margin During Boost:	2.2	*Between rail departure and bu
Maximum Acceleration (G):	8.1	
Maximum Velocity (feet/second):	643	
Target Apogee (feet AGL):	10K	
Predicted Apogee (feet AGL):	7500	

Payload Information

Fig. 48

system, very low frequencies (1-50 Hz) will be the primary focus of the control system. The control system employs a bu
filter to remove extermely low frequencies associated with the change in pitch and roll of the launch vehichle, an inverti
with tunable gains, and a secondary audio amplifier to delieve the required power to the voice coil actuator. If time and
allow, we will focus on tuning the system to operate at higher frequencies. The accelerometer data is collected using 2 b
blacks capable of sampling at a rate of 50 Hz. All of the acclerometer signals are conditioned using a custom PCB that s
clips the acceleromter output to be +/- 6 or 8 g and to be compatible with the beagle bone black ADC. The first round
collection will be performed during test launch with several passive materials to asses there performance and to have th
test additional materials at SAC. At SAC both the active and passive vibration isolation methods will be used. From this w
record a 6dB reduction in the vibration magnitude across a large range of frequencies between 1-2000 Hz. The intended
hod for isolating sensitive electronics from the harmful vibrations experienced at events and during launch.

Recovery Information

The drogue chute comes out at the apogee. The main chute comes out at 1000ft. Both parachutes use barometric pre
trigger the event. Our rocket will have one separatio point. However, to make sure that each parachute comes out at th
altitudes, both parachutes will be wrapped by the straps that detects the altitude using an altimeter and get release
preprogrammed altitudes. Despite the separation points, separated pieces of our rocket will be connected with shock c
will stay together.
Our rocket will have two onboard tracking systems. The first system will be rocketed inside the nosecone of the rocket;
commercial radio tracking device. The second system will be located on the lower part of the rocket (inside the electr
located right above the booster section); it will be a GPS device that will transmit the GPS data to the ground base usin
telemetry system that our team made.

Fig. 49

Date	Type	Description	Status	Comments
3/18/19	Ground	Tracking system range test	Successful	ystem had the range that was well ov
4/27/19	Ground	lemetry system (launch control system) range te	Successful	o for the recommended range. 2600
5/24/19	Ground	Cold Flow testing of motor	TBD	
5/26/19	In-Flight	Test Flight	TBD	using commercial solid
3/16/19	Ground	compression test of CF tubes	Successful	fail at 26,000 lbf
4/7/19	Ground	Hydrostatic test of SRAD oxidizer tank	Successful	tested to 1.5x max intended operati
3/15/19	Ground	packing test for ejection charges	Successful	
3/29/19	Ground	test of chute ejection	Major Issue	resulted in design change to single
6/2/19	Ground	full scale static fire of motor	TBD	
5/24/19	Ground	RF interference test for Launch control system	TBD	

Fig. 50

Any other pertinent information:

Rocketry at VT is into it's testing intensive phases. To accomplish this, many team mbers are in Blacksburg, Va for the su
work on completeing testign goals before leaving for competition.

Fig. 51

VI. Appendix A2: Project Test Report

A. Recovery Systems Testing

1. Redundancy

The recovery system is a dual deployment system, in which black powder ejection charges are utilized to eject the parachutes. Both parachutes are stored in the same bay, consequently to prevent the main chute from deploying at apogee two Jolly Logic chute release devices are used. These devices use an elastic band wrapped around the main chute to keep it closed until a preset altitude is reached. Once the desired altitude is reached, the chute release ejects the elastic band, thus freeing the main chute and allowing it to deploy. These releases use built in altimeters to determine when to release. For redundancy, two of these chute releases are used and connected in series. Consequently, only one chute release needs to eject the elastic band to cause the main chute to deploy.

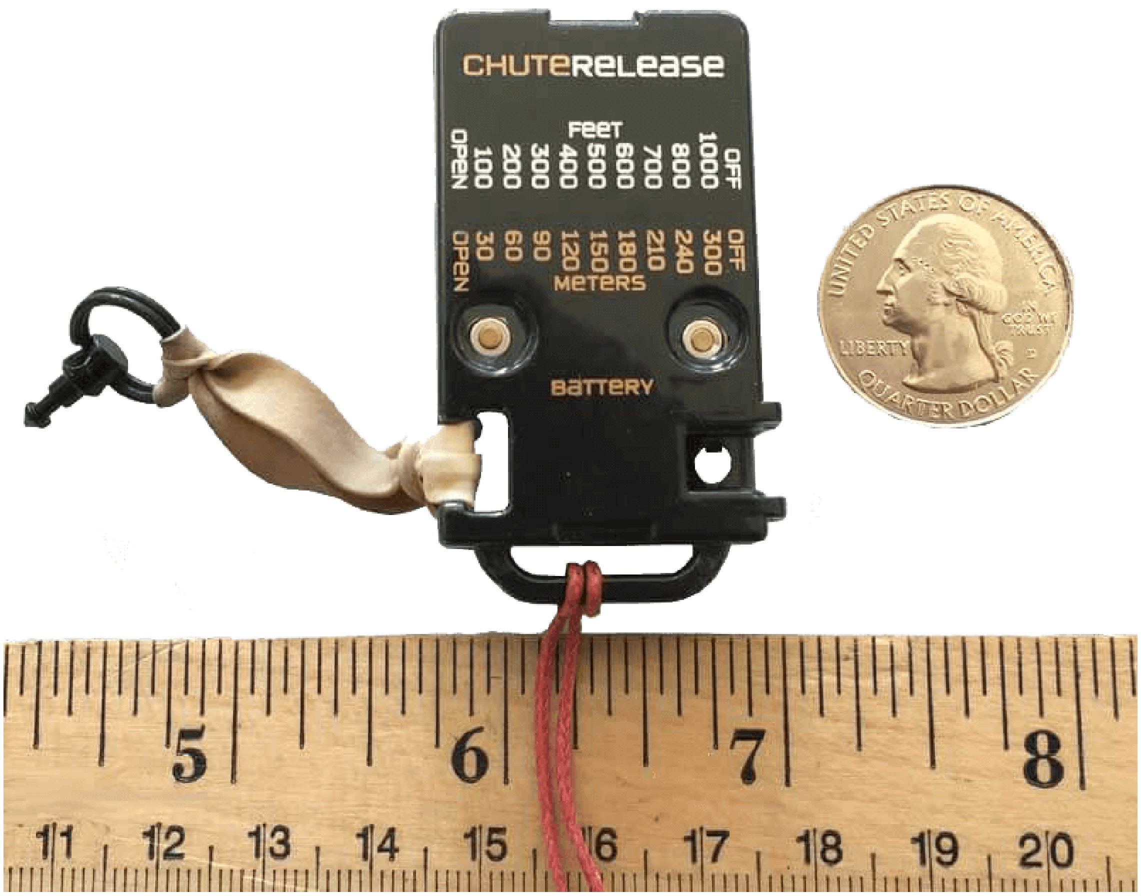

Fig. 52 Jolly Logic Chute Release System.

The black powder ejection charge is ignited via a Stratologger altimeter. For redundancy, two Stratologger altimeters and black powder charges are used. Each altimeter uses an independent power supply, and is connected to its own black powder charge. The altimeters are set at staggered altitudes to avoid firing both charges at the same time and having an overly energetic chute ejection.

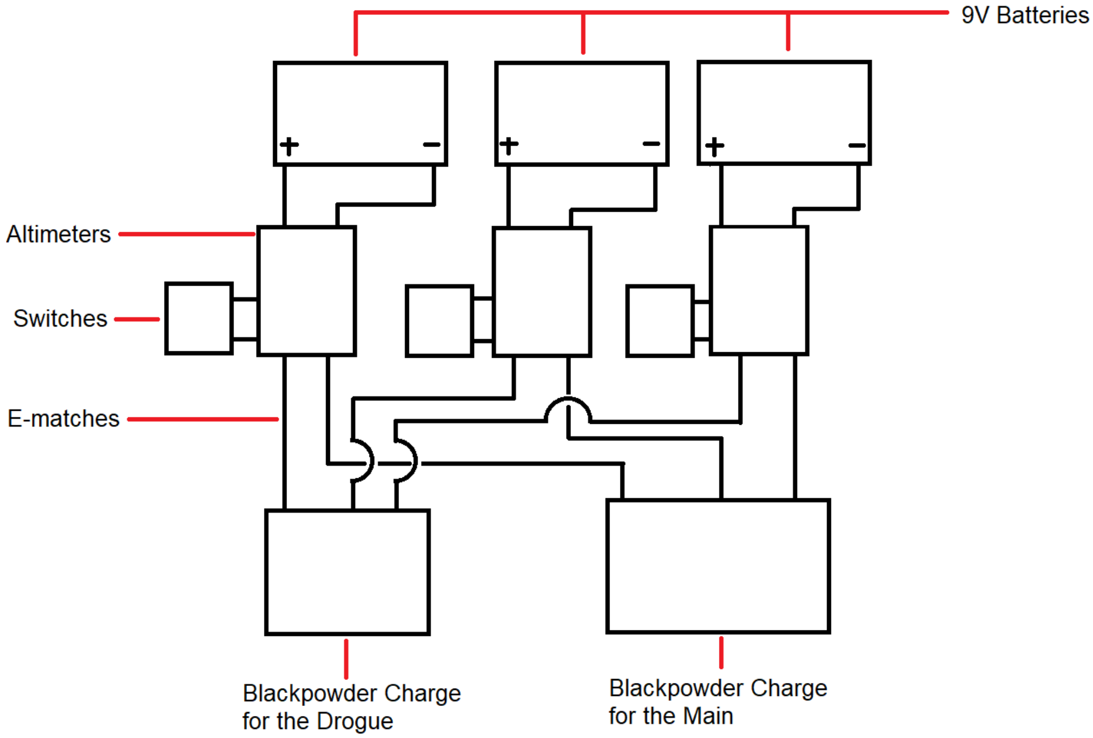

Fig. 53 Recovery System Diagram.

2. Initial Deployment Testing

Initially Rocketry at Virginia Tech had planned to utilize two separate chute bays; one for drogue and one for main. The design for this system required that for the main chute to deploy the launch vehicle would need to separate at in about the middle of the rockets long axis. Consequently, the internal structure was required to slide through cutouts in an aluminum bulkhead. Initial testing showed that this form of separation had a tendency to bind up. As a result, the deployment was modified to the aforementioned single bay method of deployment, where the bay is located at the forward end of the nosecone such that only the nosecone needs to separate from the launch vehicle.

3. Ejection Charge Packing Testing

The team conducted a test designed to verify that the method of packing ejection charges resulted in reliable ignition with the e-match. To test this, the charges were packed as they would be for flight, including the insulation packed around the charge to help shield the chutes from hot gasses, and using the e-matches to be used in flight. Following this, the e-matches were set off using a remote power supply, with all personnel at a safe distance away. The test yielded that the method of packing ejection charges did result in successful ignition of the charge.

4. Jolly Logic Chute Release Testing

The Jolly Logic Chute releases were tested to ensure that they would release with the elastic band in tension while wrapped around the main chute, and they were tested to ensure they were firmly affixed to the main chute. To accomplish this, the chute releases were install in series, as they would be for flight, and the main chute was weighted and dropped with slack in a rope. When the slack in the rope was used up, the shock created would simulate the ejection of the chutes. This testing showed that the chute releases were probably affixed and did not shake loose. Furthermore, after this was conducted, the chute releases were triggered to release by using the Jolly Logic provided procedure for triggering the release on the ground. Both devices ejected the elastic band, demonstrating that they operated nominally under tension.

5. COTS Telemetry System Testing

The team tested the commercial Seagull telemetry system to ensure it operated properly. This test was conducted by having the a group of team members carrying the telemetry transmitter distance themselves from the ground receiving computer, while another telemetry system operator verified that a stable telemetry link, in the form of GPS data being received, was still attained. The transmitter was continuously moved further and further from the receiver until the link was lost. These initial tests failed, as no link was able to be received regardless of the range. As a result the unit had to be sent back to the manufacturer, in which it was confirmed that the unit was defective. A new unit was delivered under warranty, and the test was reconducted. The second unit proved to operate nominally, with a stable link being maintained at a range of several miles.

6. Recovery System Planned Tests

Tests that are slated to be completed before the team leaves for the 2019 Spaceport America Cup include:

Ground Test Demonstration - Eject chutes on the ground from the launch vehicle with black powder ejection charge to verify proper ejection charge size and correct number of shear pins.

B. Propulsion System Testing

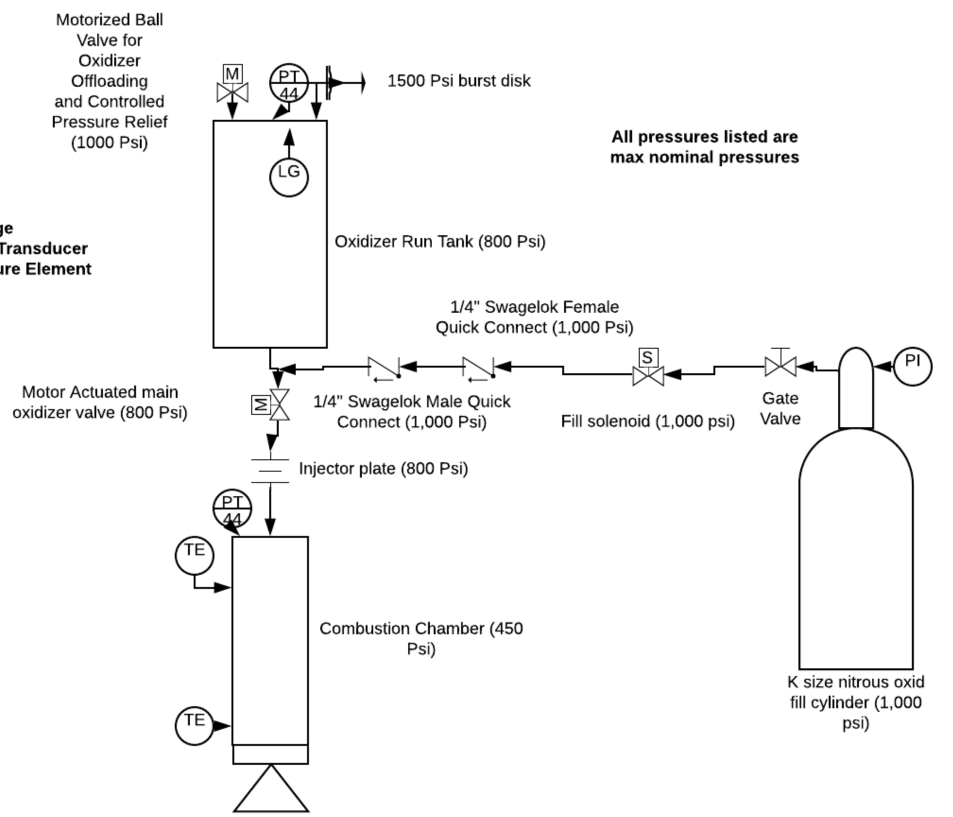

Fig. 54 Propulsion System Overview

1. SRAD Oxidizer Tank Test

The oxidizer tank was hydrostatic pressure tested to 1,200 psi and left at that pressure for about two times the max intended operating time. The design pressure is 800 psi, so 1,200 psi is 1.5 times that. The oxidizer tank however will have a valve designed to release pressure at 950 psi to avoid supercritical nitrous oxide. This test resulted in no noticeable deformation or leaks of the tank, thus verifying that the tank operates nominally.

2. Main Oxidizer Valve Actuation Testing

The main oxidizer valve actuation system was assembled and tested to ensure that the motorized ball valve could produce enough torque to actuate the valve. This test result was successful.

3. Launch Control System Range Testing

The launch control system was tested to operate reliably at a line of sight range of 2,600 feet.

4. Fill Line Remote Disconnect Testing

The ground support subsystem's remote fill line disconnect was tested by hooking the female Swagelok quick connect to the male end and actuating the linear actuator to verify if the system can disconnect the fill line. The test results were that the linear actuator used has ample amounts of force able to disconnect the fill line, however, that the ground support frame needs to be stiffened and secured to the ground to prevent it from bending or falling over.

C. Propulsion System Planned Tests

The following tests are planned to occur before the team leaves for the 2019 Spaceport America Cup.

1. Combustion Chamber Hydrostatic Test

Even the the combustion chamber is a commercial motor casing, the team plans to test it with the modified forward closure. The chamber will be hydrostatic tested to 700 psi, which will be just over 1.5 times the max intended operating pressure of 450 psi.

2. Cold Flow Tests

The motor will be tested with only oxidizer loaded. This test is designed to verify simulated oxidizer flow rates, and ensure all systems operate nominally under lesser stress than full scale static firing.

3. Static Fire Testing

A full scale static fire test is planned to verify the motor operates as intended, and produces the required impulse.

Launch Control System RF Interference Testing The launch control system will be tested with additional LoRa radios operating at the same frequency constantly transmitting random data. This test will be conducted at a line of sight range of greater than 2,000 feet and is designed to ensure the launch control system does not become adversely affected by nearby RF systems.

D. Additional Tests

The following tests are planned to occur prior to the team leaving for the 2019 Spaceport America Cup:

1. Simulated Launch Procedure

This test will be a full scale simulated launch, without using any propellants or ejection charges. It it designed to build familiarity with the launch vehicle and it's various subsystems among the team as well as identify any flaws in launch procedures.

2. Flight Demonstration

A full scale flight demonstration will be conducted to test the launch vehicles subsystems in flight. This test will occur using a commercial N-class motor due to limitations of the nearby launch site. Furthermore, the use of a commercial motor for the first flight demonstration will also help serve as a demonstrator on the modularity of the launch vehicle as a whole.

VII. Appendix B: Hazard Analysis

Hazard	Causes	Level of Risk and Rationale	Mitigation	Risk of Injury After Mitigation
Premature ejection charge ignition during handling causing burns, or explosive trauma, potentially severe.	Failure to ensure non-live circuit. Accidental contact with ignition source.	Moderate; Student packed charges, hence significant handling increasing chances of accidental ignition.	All ejection charges shall be placed only after ensuring firing circuit is not armed. Leads to ejection charges will be wrapped with non-conductive tape to avoid accidental contact with ignition sources All personnel will statically ground themselves to avoid the potential for static discharge as an ignition source.	Low
Premature SRAD Motor ignition during handling, resulting in severe burns or explosive trauma.	Failure to ensure non-live circuit. Accidental contact with ignition source.	Low; Student handling is minimal.	Motor shall be placed only after ensuring firing circuit is not armed. All personnel will statically ground themselves to avoid the potential for static discharge as an ignition source. E-match will only be placed when ready to launch.	Low
Two part epoxy used improperly causing skin irritation, dizziness from vapor inhalation.	Lack of personal protective equipment.	Moderate; Due to the composite airframe, handling epoxy is very common.	Gloves, masks, and safety glasses shall always be used while working with epoxy.	Low
Improper use of power tools such as drill, band saw, and dremel causing potentially severe injury in the form of lacerations, debris inhalation, or eye damage/irritation.	Lack of personal protective equipment. Poor understanding of power tool operational procedures.	High: Working with power tools is required often, and can be difficult to do safely with lack of experience.	Gloves, masks, and safety glasses shall always be used when operating power tools . All personnel of Rocketry At Virginia Tech must complete an online power tool safety course offered by Virginia Tech Environmental Health and Safety Department.	Low

Fig. 55 Hazard Analysis

Inhalation of composite particulate matter during post lay-up processing. The resulting particulate matter is a respiratory irritant, with the potential to cause long term respiratory problems.	Lack of personal protective equipment.	High; The composite airframe requires large amounts of post processing, which produces very fine dust. This dust can easily be inhaled unless great care is exercised.	Masks must be worn at all times when working with composites.	Low
Improper usage of chemicals, including Frekote mold release, enamel paint, spar urethane, and acetone, causing respiratory irritation, skin irritation, eye irritation, potential long term organ toxicity effects. Possible fire.	Lack of personal protective equipment. Usage in poorly ventilated spaces. Failure to remove ignition sources.	High; Composite lay-ups and post processing requires the use of a variety of potentially hazardous chemicals, which leads to unanticipated exposure unless proper measures are taken.	Gloves, respiratory masks, and safety goggles should be worn at all times. Use with any of the listed chemicals must occur in a well ventilated area. Any sources of ignition must be avoided.	Low
Premature ejection charge ignition of the payload during handling causing burns, or explosive trauma, potentially severe.	Failure to ensure non-live circuit. Accidental contact with ignition source.	Moderate; Student packed charges, hence significant handling increasing chances of accidental ignition.	All ejection charges shall be placed only after ensuring firing circuit is not armed. Leads to ejection charges will be wrapped with non-conductive tape to avoid accidental contact with ignition sources . All personnel will statically ground themselves to avoid the potential for static discharge as an ignition source. All personnel will avoid being in the direct path of the ejection barrel when handling the rocket	Low
Inhalation of Gas	Tank leakage or spillage. Opened release valve. Damaged feed lines.	Medium: Drowsiness, dizziness, central nervous system depression, unconsciousness fatigue, nausea, vomiting, and headaches are all potential hazards if mishandled.	Inspect all plumbing during assembly and filling Some form of fill procedure? Proper equipment used? Minimal exposure of personnel?	Low

Fig. 56 Hazard Analysis

			Masks? Gasses will be handled in well ventilated area.	
Skin Exposure to Gas	Contact with source of tank leakage or spillage. Contact with gas from opened release valve. Contact with gas from damaged feed valve	Low: Frostbite and irritation may occur.	Wear proper safety equipment when handling. Check all plumbing before handling. Ordered procedure of handling to minimize total contact with gas.	Low

Fig. 57 Hazard Analysis

VIII. Appendix C: Risk Assessment

Rocketry at Virginia Tech Risk and Mitigation Analysis Matrix IREC 2019					
Flight-Stage	**Incident**	**Possible Source**	**Level of Risk and Rationale**	**Mitigation and Prevention**	**Risk of Injury after Mitigation**
Ground Setup	Parachute recovery system ord payload deploys in pre-launch operations causing injury	Accidental ignition command is set to ejection charges	Low; Firing circuit is controlled via commercial altimeters built for rocketry use.	Continuity of firing circuit is denied until ready for launch via a switch.	Low
	Rocket falls from launch rail during pre-launch operations causing injury to personnel.	Launch lug failure. Chaotic pre-launch operations cause mistakes.	Medium; Excitement of launch and limited launch lug testing.	Launch lug integrity validated through testing and FEA. Pre-designated and trained launch operations team used to mount rocket on the rail.	Low
	An explosion of rocket motor during launch creating debris that may injury nearby personnel.	Cracks in propellant grain. Gaps between the propellant section or nozzle Chunks of propellant breaking off and clogging the nozzle Motor case failure Motor closure failure	Low; The motor is commercial and has been used multiple times in amateur rocketry applications.	The motor assembly is conducted by trained personnel following all manufacturer recommendations.	Low
	The motor does not ignite when commanded to but does when personnel approach to troubleshoot.	Delayed firing signal	Low; motor and E-match are commercial and have substantial testing.	A failed motor ignition will lead to a 15-minute wait prior to approach. When approaching, the firing circuit will be unarmed.	Low
	Exothermic Decomposition	Local, thermal ignition.	High:	Keep the nitrous tank in a consistent	Medium

Fig. 58

Ignition		Rapid depressurization of Nitrous vapor. Homogeneous ignition, a large quantity of nitrous oxide must be exposed to relatively large quantities of heat at once.	Explosion resulting in severe injury or death, damage to surrounding property, and severe burns.	temperature environment, removed from any source that would induce an unpredictable change in temperature. Utilize a cooling system on the ground support system to slow temperature changes	
	Graphite-Insert Fragmentation	Collection of moisture prior to motor ignition.	High: Ejection of sharp, high-temperature fragments at high velocities can result in severe injury or death, and damage to surrounding property.	Keep tank and fuel in a dry environment, well away from any humidity and water. Transported in a dry container, insulated from humidity.	Medium
	Ignition of Surroundings	Firing of the motor within proximity to other flammable materials.	High: Explosion resulting in severe injury or death, and damage to surrounding property.	Keep away from other flammable sources, safety officer ensures adherence to rule.	Medium
Powered Ascent	Rocket deviates from a nominal trajectory and hits personnel.	Poor rocket stability Launch lugs unable to withstand forces from launch.	Medium; Limited testing of launch lugs and rocket stability from actual flight tests.	CFD and OpenRocket simulations to verify rocket stability. FEA analysis to ensure launch lug integrity. All personnel at minimum 2000 feet from the rocket and behind a barrier.	Low
	Failure of the airframe during flight causing break up of rocket.	Failure to design for stresses of flight.	Medium; Student-built airframe with limited testing.	FEA analysis on the airframe to ensure adequate safety factor.	Low

Fig. 59

				Compression testing of airframe prior to flight to validate FEA simulations.	
	Over Pressurization of Pressure Vessel	Hydraulic-overpressurization, expansion of gas via heating of the tank, and an increase in vapor pressure.	High: Explosion resulting in severe injury or death and damage to surrounding property.	Monitoring of tank pressure and temperature throughout all stages of its life cycle Pressure vessel tested to 1.5x expected maximum pressure.	High: Even with mitigation, pressure vessels are exceedingly dangerous and must be treated with the utmost respect and caution.
Powered Ascent	Over Pressurization of Combustion Chamber	Unexpected, high oxidizer flow rates. Unexpected, high fuel regression rates. Blockages and erosion in the exhaust nozzle. Weakened wall structure from thermal stress. Fuel grain fissures and cracks.	High: Explosion resulting in severe injury or death, damage to surrounding property, and severe burns.	Having controls on oxidizer valve. Cleaning exhaust nozzle before use. Testing thermal stress of combustion wall. Testing fuel grain arrangements for points of failure, package fuel grain with cushioning to reduce vibration and impact shock.	Medium
Un-controlled Descent	Main parachute deploys at or near apogee causing the rocket to drift excessively.	Erroneous firing signal to main chute ejection charge. Drogue chute deployment causes premature shear pin failure.	Medium; Firing circuit controlled by highly tested commercial altimeter. However, limited flight testing can make shear pun number estimates difficult.	Test flights, and on the ground ejection testing as well as shock analysis to determine the required number of shear pins, with safety factor applied.	Low
Controlled Descent	Parachute recovery fails or doesn't fully deploy, leading to rocket or	Parachute rips or tears. Parachute ejection fails.	Medium: Testing recovery systems is limited to full-	Commercial parachute designed explicitly for high power rocketry use is utilized and inspected	Low

Fig. 60

	payload contacting personnel at high speeds.	Shock cord fails.	scale flight tests.	for defects prior to launch. Parachute ejection charges are tested numerous times on ground prior to use. Parachute ejection charge has a redundant backup, with each charge connected to independent altimeter with fresh batteries. Commercial kevlar shock cord designed for high power rocketry use is utilized and inspected for defects prior to flight.	

Fig. 61

IX. Appendix D: Assembly, Preflight, and Launch Checklist

A. Propulsion Launch Day Readiness Procedures

1) Prepare vehicle for launch by cleaning or performing any necessary maintenance of rocket components
2) Setup remote communication systems and test functionality of rocket control and data transmission
3) Follow procedures outlined by other subteams to assemble the interior frame of the launch vehicle
4) Prior to full assembly conduct pre-flight propulsion, avionics, and payload electronic tests to ensure all critical hardware is working as expected
5) Conduct Radio transmission test to ensure there has been communication established with the launch vehicle
6) Ensure that e-matches are not active and require the necessary number of distinct actions to become live
7) Ensure that the manual release valve and main injection valve can be remotely operated and are functioning properly
8) Check that emergency pressure release disk is seated properly
9) Load fuel-grain into the combustion chamber
10) Ensure all power systems on the interior of the rocket are active and in an on state
11) Finish assembly of the launch vehicle and continue to ground support
12) Setup ground support stand and launch stand
13) Ensure that the launch stand is properly anchored and can readily accommodate launch vehicle
14) Ensure ground support stand is properly weighted/anchored
15) Test functionality of remote disconnect system and its remote operation as well as remote fuel loading activation
16) Test functionality of nitrous cooling systems
17) Turn on nitrous cooling systems
18) Load nitrous oxide into ground support stand and connect to the oxidizer fill line on the ground support stand
19) Setup launch vehicle on the launch pad
20) Ensure the rocket is properly secured on the launch pad
21) Perform final inspection of safety equipment, including the manual release system, and ground support systems.

Ensure that e-matches are not armed and that the oxidizer fill system is not active

22) Fully extend linear actuator and ensure that the exterior female oxidizer fill valve is in line with the interior male oxidizer fill valve
23) Use provided fuel loading tool labeled INSERT NAME to pull the male end of oxidizer fill valve out of the interior of the launch vehicle
24) Carefully couple the male and female ends of the oxidizer fill connect valves. Ensure that the female valve is encased by the black release collar
25) Possible step to test the release system and recouple fill line
26) Remove all personnel and non-launch essential equipment from a 3000-foot radius of the rocket
27) Commence oxidizer fill sequence remotely, ensure no personal enter launch safety radius of the launch vehicle
28) Fill rocket to approximately 85% oxidizer capacity denoted full by fill float
29) Remotely close oxidizer fill line and manual release vent
30) Remotely decouple oxidizer fill line
31) Perform final venting tests if necessary*
32) Announce loudly that launch vehicle is considered potentially dangerous
33) Perform final continuity test
34) Ensure onboard electronics are functional
35) Start data recording and transmission
36) Initiate filling ECT and bring e-matches into armed state
37) Loudly announce that launch vehicle is live
38) Loudly begin the countdown to launch
39) Initiate Launch
40) If the launch is successful begin recovery sequence
41) If the launch is unsuccessful follow substeps below
    1) Abort e-match firing command
    2) Reduce e-matches from their armed state into a locked state
    3) Ensure that no personnel entire 3000ft launch safety radius
    4) Inspect range/launch pad for any visual clues behind the launch issue
    5) If necessary begin emergency manual venting procedure
    6) Leave vents open
    7) Do not approach rocket until 15 minutes have passed since oxidizer has finished venting or until the range is declared safe
    8) Perform complication analysis and resolution procedures

B. Avionics Readiness Procedures

1. Day before

1) Charge the batteries that are needed for the operation. The rechargeable batteries include one lipo battery for the servo and one power bank for the beagleboards.
2) Make sure three 9 V batteries are prepared.

2. Day of Procedures

1) Place and constrain three 9 V batteries on a plate with the number 2 (Plate 2) with the zipties, then connect each battery to a corresponding
2) Turn the swichesfor the altimeters, and check if they make a beep sound. Then, turn the switches off.
3) Check if attached rechargeable batteries are held in places securely.
4) Mount the servo onto Plate 5 by using the screws and the nuts preinstalled on the plate.
5) Place a pair of airbeaks on Plate 6. Then place another pair and Plate 5 perpendicular to the other. Make sure gear fits and can rotate all the airbreaks uniformly.
6) Insert 1.5 in standoffs to the holes on Plate 5 and 6, and insert screws to both ends of the stands offs.
7) Place 2 in spacers and washers between Plate 5 and 4. Insert threaded rods and nuts.
8) Place 2 in spacers and washers between Plate 4 and 3. Insert threaded rods and nuts.
9) Place 2.25 in spaces and washers between Plate 3 and 2. Insert threaded rods and nuts.

10) Connect the ematches to the altimeters on Plate 2 and wire it out to Plate 1.
11) Place 1 in standoffs and whaswers between Plate 2 and 1. Insert screws.
12) Connect wires that need to go across different plates.
13) Put locktide on the threaded rods.
14) Using a flat head driver, turn on the switches for the altimeters. Check if all of them make a beep sound.
15) Make a black powder package by wrapping black powder, insulation, and ematches with paper.
16) Connect the parachutes to the shock cords and U bolts, then pack the parachutes.
17) Mount the tracking device inside the nosecone.
18) Check the connectivity of the tracking device and telemetry system (launch control system).
    ]

C. Payload Readiness Procedures

1) Obtain both the Payload and the Payload section of the launch vehicle
2) Ensure the side strips of the payload are not secured to the payload. The wiring can remain connected
3) Place 4 10-32 bolts into the bottom framing piece of the payload section.
4) Sparingly apply blue loc-tite to exposed threads
5) Place the Payload between the two bulk heads and ensure all 4 screws of the bottom section align and fall through the 4 holes on the adjacent bulk head
6) Sparingly apply blue loc-tite to exposed threads
7) Lightly tighten all 4 bolts with 4 nuts
8) Align the payload top frame bolt holes with the top bulkhead
9) Insert all 4 bolts through the payload framing and out through the bulk head
10) Lightly tighten all 4 bolts with 4 nuts
11) Secure side strips to payload with the 6-32 screws
12) Using the custom payload wrenches, slip the wrench through the side strips and hold the head of the bolt in place. Tighten all 8 nuts with a wrench or socket, being sure to not over tighten and damage the payload wrenches
13) Connect the white marked wire harness with the 2 position male circular connector to the connector above the switch on the voice coil platform
14) Connect the red marked wire harness with the 2 position male circular connector to the connector below the switch on the beagle bone platform
15) Check to make sure NO lights turn on inside of the payload, if they do switch both switches counter clockwise to the off position
16) On pad, using a long flat head screw driver, rotate both switches clockwise to the on position.
17) Visually inspected through the switch hole to see if red + blue indicator lights are on

D. Pre-Flight and Assembly Checklist

- Day Prior To Launch
  - Power
  - ☐ Charge internal propulsion batteries
  - ☐ Obtain and pack two fresh 9-volt batteries for avionics
  - ☐ Ensure charge on payload SRAD battery and backup battery
  - ☐ Charge launch control station and backup control station
  - Supplies
  - ☐ Pack extra 9-volt batteries
  - ☐ Pack shear pins
  - ☐ Pack Binoculars (x2)
  - ☐ Pack Data storage systems
  - ☐ Micro SD
  - ☐ Pack weather station
  - ☐ Pack launch control computer and backup control station
  - ☐ Pack RF antenna
  - ☐ Pack micro USB/Arduino connection cables
  - ☐ Pack Allen Wrench Set and Allen keys

☐ Pack long Phillips screwdriver and screwdriver set
– Payload
☐ Ensure full charge on all power systems
☐ Assemble any necessary components of the payload system
☐ Perform a test of payload functionality and data acquisition
☐ Ensure all payload settings are correct
– Recovery Systems
☐ Pack main chute charges
☐ Pack main chute
☐ Pack drogue chute charges
☐ Pack drogue chute
☐ Ensure deployment settings are correct
– Propulsion
☐ Pack ground support stand
☐ Pack fuel grain and backup fuel grain
☐ Ensure nitrous oxide is properly housed and ready for launch day

- Morning of Launch
  ☐ Confirm completion of the day prior to launch checklist, fulfill any uncompleted steps
  ☐ Prepare Nitrous oxide tank for transport
  ☐ Have Equipment laid out in the hall before packing
  ☐ Ensure all power systems are charged and functional
  ☐ Check payload battery voltage
  ☐ Tidy any necessary areas before leaving
  ☐ Ensure no fire violations are present
  ☐ Pack water cases
  ☐ Pack food
  ☐ Ensure all team members have had a meal before leaving
  ☐ Ensure all team members are hydrated or have access to water
  ☐ Pack trailer securely
  ☐ Double check all necessary equipment is packed
  ☐ Head to launch facility
- At the Launch Facility
  – Workspace
  ☐ Setup on-site workspace
  ☐ Assemble tent and fold out table
  ☐ Layout necessary tools and equipment in an organized fashion
  – Motor
  ☐ Ensure personnel are wearing proper PPE
  ☐ Assemble motor
  ☐ Ensure oxidizer tank closures are secure
  ☐ Ensure rupture disk is seated correctly
  ☐ Ensure emergency vent line is properly affixed
  ☐ Ensure oxidizer fill line is properly affixed
  ☐ Ensure injection line is properly affixed
  ☐ Connect any necessary power cabling to all motor control hardware
  ☐ Ensure motor control hardware is powered
  ☐ Seat injection plate into the correct position atop the combustion chamber
  ☐ Tightly secure top closure of combustion chamber
  ☐ Connect injection line to the combustion chamber
  ☐ Run any continuity tests on motor control hardware
  ☐ Ensure remote functionality of oxidizer injection system and emergency vent system
  ☐ Load and secure fuel grain into the combustion chamber
  ☐ Secure combustion chamber with motor retainer
  ☐ Affix nozzle to the end of the combustion chamber

- ☐ Secure bottom of the launch vehicle with the boat tail
- – Charges
- ☐ Ensure personnel are wearing proper PPE before starting
- ☐ Ensure the main chute and drogue chute are properly secured
- ☐ Ensure shear pins and charges are in place
- ☐ Run any necessary continuity tests
- – Payload
- ☐ Ensure data storage systems are in place
- ☐ Ensure data can be properly transmitted
- ☐ Load payload into payload bay and secure in place
- ☐ Activate power systems and payload
- – Avionics
- ☐ Load hardware into the avionics bay
- ☐ Activate all avionics systems
- ☐ Ensure hardware is functional
- ☐ Ensure hardware is secured in launch vehicle
- ☐ Activate avionics systems
- – Structures
- ☐ Ensure all systems are secure in rocket
- ☐ Ensure all internal systems are on
- ☐ Attach fins onto internal structure rails
- ☐ Slide carbon-fiber skin over launch vehicle structure
- ☐ Double check that the first layer of skin's access port is aligned with the internal nitrous oxide fill valve
- ☐ Secure carbon-fiber tubing
- ☐ Affix Nose Cone onto the top of the launch vehicle
- – Software
- ☐ Establish communication connection with rocket
- ☐ Ensure data collection systems are functional
- ☐ Ensure the launch vehicle is in a locked state

- On Pad
  - ☐ Ensure no team members ever stand directly in the line of payload barrel
  - ☐ Perform any final safety checks and ensure the rocket is securely assembled
  - ☐ Secure launch vehicle onto launch rail
  - ☐ Use the fill tool to pull out the male end of the nitrous oxide fill valves and couple the nitrous oxide fill systems
  - ☐ Power on any ground support coolant or filling systems
  - ☐ Remove any personnel from a 3000ft radius of the launch vehicle
  - ☐ Begin filling procedures remotely
  - ☐ Decouple fill line
  - ☐ Arming:
    - ☐ Arm payload systems
    - ☐ Arm GPS tracking system
    - ☐ Arm Altimeter measurement system
  - ☐ Check Local Weather
    - ☐ Check METAR/AWOS
    - ☐ Record Temperature
    - ☐ Record Pressure
    - ☐ Record Wind Speed

X. Appendix E: Engineering Drawings

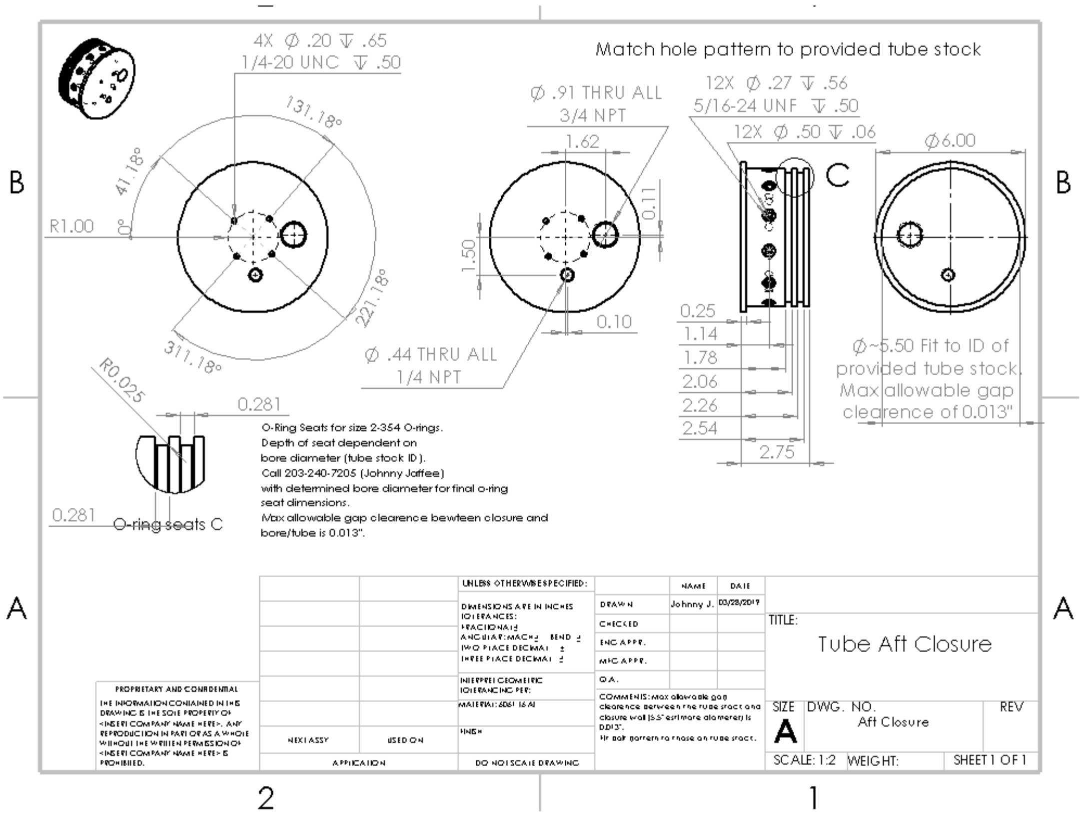

Fig. 62 Aft Oxidizer Tank Closure

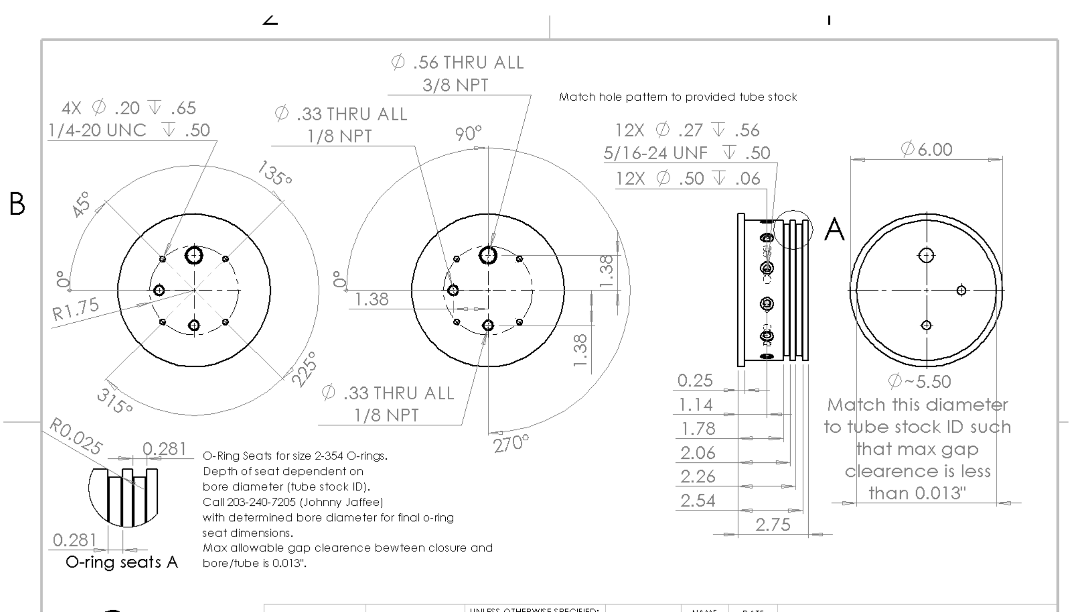

Fig. 63 Forward Oxidizer Tank Closure

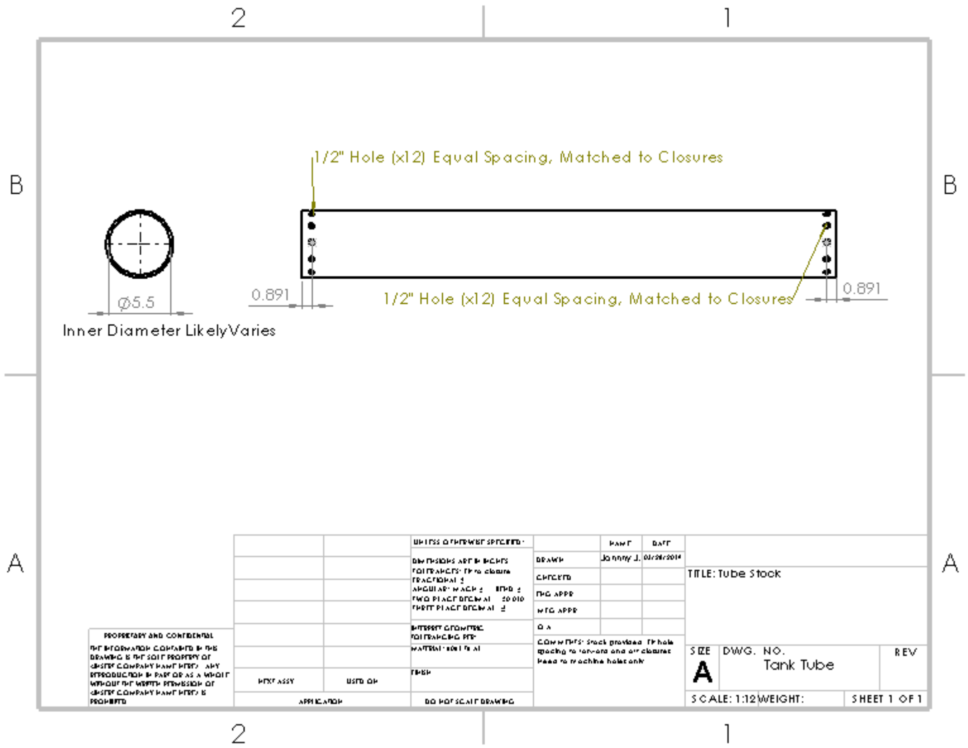

Fig. 64 Oxidizer Tank Drawing

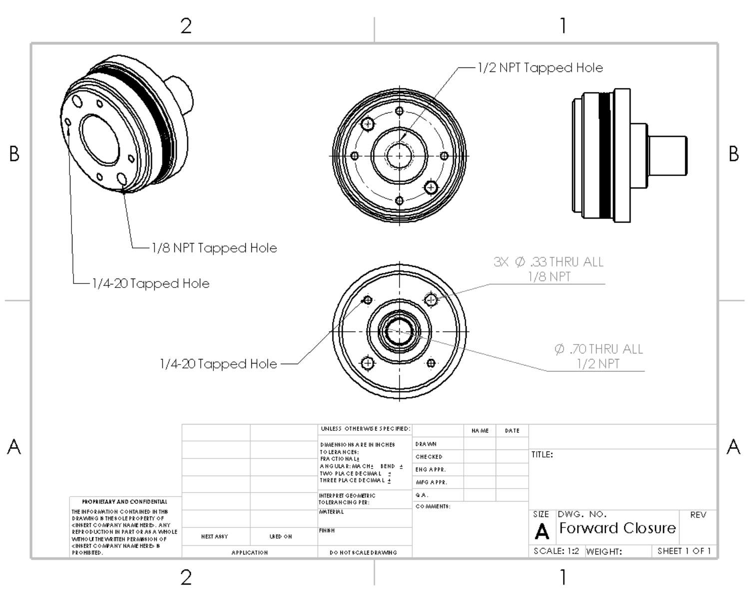

Fig. 65 Forward Closure Assembly

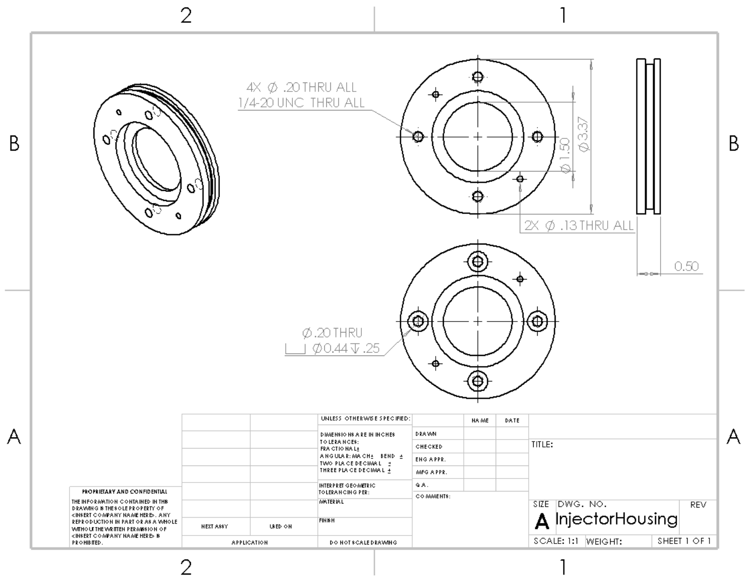

Fig. 66 Injector Housing

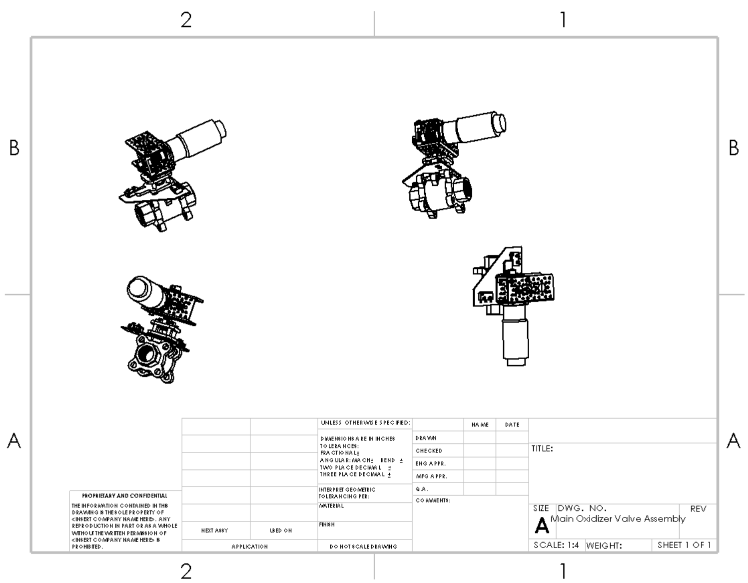

Fig. 67 Full Oxidizer Valve Assembly

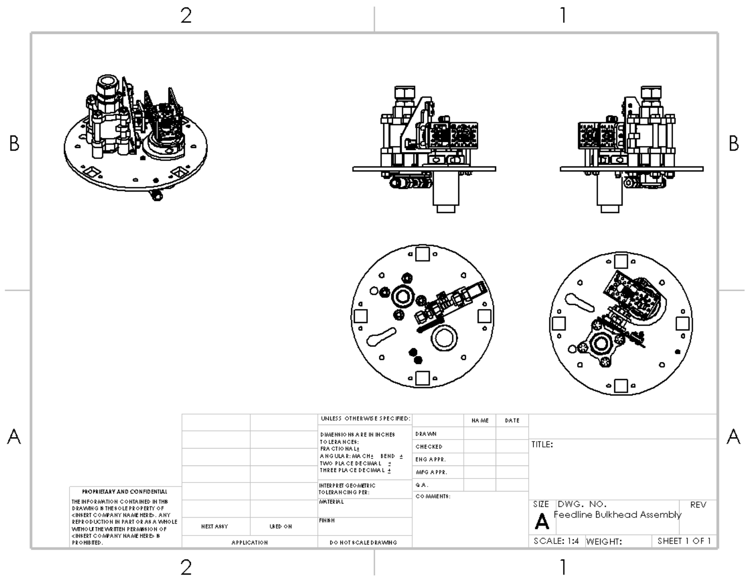

Fig. 68 Center Feedline Plate Assembly

XI. Appendix F: System Requirements

The following Appendix will derive the requirements for the propulsion, launch vehicle, payload, and avionics systems.

A. Propulsion Requirements

1) The propulsion system shall adhere to Spaceport America Cup Rules and Specification for the 10K SRAD Hybrid/Liquid Category
    1) The propulsion system shall be student researched and developed, as defined by ESRA, "...designed by students – regardless of whether fabrication is performed by students directly, or by a third party working to student supplied specifications – and can include student designed modifications of COTS systems."
    2) The propulsion system shall not exceed 9,200 $lbf * s$ of impulse.
    3) The propulsion system shall be a liquid or hybrid implementation.
    4) The propulsion system shall have the capability of accelerating the launch vehicle to a minimum velocity of 50 ft/s upon leaving the launch rail, or the required velocity to achieve a stability margin of at least 1.5 calipers
    5) The propulsion system shall have pressure relief valves on all pressure vessels set to open at no greater than 1.5x the max intended operating pressure.
    6) The propulsion system complete a static fire ground test or flight test prior to the competition date.
    7) The propulsion system shall utilize non-toxic propellants.
    8) The propulsion system feature remote onloading and offloading for all liquid propellants utilized.
    9) The propulsion system shall have all SRAD pressure vessels designed to a minimum safety factor of 2 if metallic, or 3 if composite.
    10) The propulsion system shall NOT be armed until all personnel are at a minimum of 50 feet away. ESRA states, "A propulsion system is considered armed if only one action (eg an ignition signal) must occur for the propellant(s) to ignite."
2) The propulsion system shall be designed to withstand the loading associated with it's usage and flight.
    1) The oxidizer tank shall be designed to withstand the loading associated with it's usage and flight with a minimum safety factor of 2.
        1) The oxidizer tank shall be designed to a safety factor of 2 in hoop stress.
        2) The oxidizer tank shall be designed to a safety factor of 2 in axial stress.
        3) The oxidizer tank shall be constructed from material compatible with the oxidizer.

   2) The combustion chamber shall be designed to withstand the loading associated with it's usage and flight with a minimum safety factor of 2.
      1) The combustion chamber shall be designed to a safety factor of 2 in hoop stress at maximum operating temperature.
      2) The combustion chamber shall be designed to a safety factor of two in axial stress at maximum temperature.
   3) The valving assembly shall be designed to withstand the loading associated with its usage, flight, and thrust.
      1) The valving assembly shall be constructed from material compatible with the oxidizer.
3) The propulsion system shall carry sufficient propellant to achieve an apogee of 10,000 feet AGL.
   1) The propulsion system shall carry sufficient oxidizer to achieve an apogee of 10,000 feet AGL.
   2) The propulsion system shall carry sufficient fuel to achieve an apogee of 10,000 feet AGL.
4) Safety shall be prioritized first and foremost
   1) Propulsion system shall be subjected to design reviews with Virginia Tech College of Engineering faculty.
   2) Propulsion system shall be tested in a controlled location.
   3) Propulsion system testing shall be coordinated and approved by the Virginia Tech Aerospace Engineering department.
   4) Propulsion system shall undergo "mock" testing without any propellant present to verify systems operate as designed and team personnel are well versed in the systems operation.
5) The propulsion system shall integrate into the booster section of the launch vehicle.

1. Oxidizer Tank Subsystem Requirements

1) The oxidizer tank shall be constructed from an aluminum alloy
2) The oxidizer tank shall be designed to integrate into the booster section of the launch vehicle
   1) The oxidizer tank shall have an outer diameter no greater than 6 inches.
   2) The oxidizer tank shall include mounting locations to the launch vehicles internal structure.
3) The oxidizer tank shall be designed to accommodate an internal pressure of 1,000 PSIG with a minimum safety factor of 2.
   1) The oxidizer tank shall be designed to a minimum safety factor of 2 under hoop stress associated with 1,000 PSIG of internal pressure.
   2) The oxidizer tank shall be designed to a minimum safety factor of 2 under axial loading associated with 1,000 PSIG of internal pressure.
4) The oxidizer tank shall be designed to withstand a minimum of 10g's in the axial direction.
   1) The oxidizer tank shall be designed to withstand an 250lbf of axial loading in additional to the axial loading created by an internal pressure of 1000 PSIG while maintaining a minimum safety factor of 3.
5) The oxidizer tank shall accommodate mounting locations for all valving and instrumentation components.
   1) The oxidizer tank shall include mounting locations for the oxidizer feedline.
   2) The oxidizer tank shall include mounting locations for the oxidizer fill line.
   3) The oxidizer tank shall include mounting locations for a pressure transducer.
   4) The oxidizer tank shall include mounting locations for the oxidizer offloading line.
   5) The oxidizer tank shall include mounting locations for the oxidizer pressure relief valve.
   6) The oxidizer tank shall include mounting locations for tank fill level sensors.

2. Valving Subsystem Requirements

1) Valving shall transfer oxidizer from the oxidizer tank to the combustion chamber
   1) The oxidizer feedline shall be large enough to allow for sufficient flow rates to achieve the thrust necessary to fulfill propulsion system level requirement 1.4.
   2) Oxidizer transfer from oxidizer tank to combustion chamber shall be controllable remotely.
   3) Oxidizer transfer to the combustion chamber shall be designed to maximize oxidizer and fuel mixing.
2) Valving shall be designed for remote oxidizer offloading on command.
3) Valving shall be designed to release oxidizer when pressure in the oxidizer tank reaches 1.5x the max intended operating pressure.
4) Valving shall be designed to withstand 1,500 lbf of axial load minimum safety factor of 2.

5) Valving shall be designed to withstand 1,000 PSIG of internal pressure.
6) Valving shall be designed for remote filling of the oxidizer tank
    1) Fill line shall detach from the launch vehicle remotely when the oxidizer tank is full.
7) Valving shall be designed to detect a full oxidizer tank.
8) Valving shall integrate with the rest of the propulsion system.
    1) Valving shall include mounting points for the propulsion systems combustion chamber.
    2) Valving shall include mounting points for the propulsion systems oxidizer tank.
9) Valving shall integrate with booster section of launch vehicle.

3. Combustion Chamber Subsystem Requirements

1) The combustion chamber shall be designed to accommodate 500 PSIG if internal pressure at max temperature with a minimum safety factor of 2.
    1) The combustion chamber shall be designed to limit max temperature to 400 degrees F.
2) The combustion chamber shall integrate with the propulsion system valving.
3) The combustion chamber shall integrate with the booster section of the launch vehicle.
4) The combustion chamber shall accommodate a minimum of 3.2 lbm of fuel.
5) The combustion chamber shall accommodate a nozzle.

4. Launch Control System Requirements

1) The launch control system shall adhere to Spaceport America Cup rules and specifications
    1) The launch control system shall have minimum operational range of 2,000 feet.
    2) The launch control system shall be at least single fault tolerant via implementing a removable safety interlock in series with the launch switch.
    3) The launch control system shall implement momentary type ignition switches.
    4) The launch control system shall require a minimum of two deliberate actions to cause ignition.

5. Motor Controller Subsystem Requirements

1) The motor controller shall control all electronics on the motor.

6. Ground Support Subsystem Requirements

1) The ground support subsystem shall provide remote disconnect of the oxidizer fill line.
2) The ground support subsystem shall keep the oxidizer fill tank cool.
3) The ground support subsystem shall integrate with the avionics telemetry system to provide remote control of the launch control system.

B. Launch Vehicle Requirements

1) The launch vehicle should not pose any threat to the safety of the team or bystanders
2) The launch vehicle shall maintain stability for the duration of the ascent. Stable is defined as maintaining a static margin of at least 1.5 to 2 body calibers, regardless of CG movement due to depleting consumables and shifting center of pressure (CP) location due to wave drag effects. Falling below 1.5 cals is considered a loss of stability.
3) The launch vehicle should avoid becoming over-stable during the ascent. The rocket is considered over-stable with a static margin significantly over 2 cals.
4) The launch vehicle should obtain a rail exit velocity of 100 ft/s off of the launch rail.
5) The launch vehicle shall launch at an angle of 84° ± 1° and a launch azimuth defined by competition officials.
    1) Team provided launch rails should comply with the nominal launch elevation specified in section 8.1 (It should not permit launch above 85 degrees and below 70 degrees if adjustable).
6) The airframe for the launch vehicle should be fully student researched, designed, and developed
7) The internal structure of the launch vehicle should be modular in design.
    1) Individual subsystems (e.g. Payload, Avionics) should not be required to assemble launch vehicle
    2) Individual components on board launch vehicle should be able to be replaced in case of damage or redesign

8) The launch vehicle shall be built to sufficiently withstand the operating forces acting during flight
    1) The skin of the launch vehicle shall be able to withstand all operational forces
    2) No PVC, PML, Quantum Tube, or Stainless Steel shall be used in load bearing capacity
    3) All eye bolts are closed eye and steel (not stainless). Any bolt used will be steel.
    4) All body tube couplers shall extend no less than one body caliber on either side of the joint
    5) All launch lugs must be mechanically attached to the body of the rocket to prevent rips and tears during flight
    6) The aft-most launch lug must be able to hold the entire mass of the rocket while on the launch pad

C. Avionics and Recovery Requirements

Terminology:
- SAC DTEG: Spaceport America Cup Design, Test, and Evaluation Guide
- SAC RRD: Spaceport America Cup Rules and Requirements Document
- SAC RSOP: Spaceport America Cup Range Standard Operating Procedures

1) All the flight electronics shall have a standby time of at least 2 hours.
2) The recovery system will enable the team to successfully retrieve the rocket.
    1) The tracking system should be able to provide the location of the rocket reliably and accurately.
        1) The tracking system shall have a range over 1.5 mi.
        2) The tracking system shall have an accuracy of 10 m or under.
    2) The tracking system shall be composed of two tracking subsystems for a redundancy.
        1) At least one redundant COTS tracking electronics subsystem (SAC DTEG 3.3.1).
        2) Redundant tracking device should be dissimilar to the other device (SAC DTEG 3.3.2).
    3) The tracking system shall be tested for its range and reliability prior to a launch.
        1) The team shall include a figure and supporting text that describes the test results of the dual redundancy of tracking system electronics in the appendix on the Project Technical Report.
    4) The team shall follow "dual-even" recovery operations concept (CONOPS) including the initial deployment event (a drogue parachute) and a main deployment event (a main parachute) (SAC DTEG 3.1).
    5) The drogue parachute and the main parachute should be highly dissimilar from each other visually (SAC DTEG 3.1.5).
    6) The drogue parachute shall bring the descent rate down to between 75 and 150 ft/s, and the main parachute shall bring the descent rate down to between 75 and 150 ft/s (SAC DTEG 3.11 and 3.12).
3) The telemetry system will enable the team to communicate with the rocket reliably.
    1) The telemetry system for the motor control shall have the minimum maximum operational range of 2,000 ft (SAC DTEG 10.3).
    2) The telemetry system shall be tested for its reliability and range prior to the launch.
4) The electronics bay will be modular while hosting the parts securely.
    1) Each section of the electronics bay should be easily accessible.
    2) The electronics bay shall slide along the structural square tubes of the rocket.
        1) The electronics bay shall have flat top and bottom surfaces.
        2) The electronics bay shall have multiple notches that match the characteristics of the structural square tubes.
5) The general avionics system should be able to make the in-flight data will be available.
    1) A COTS barometric pressure altimeter with on-board data storage shall be carried inside the rocket (SAC RRD 2.5)
        1) Redundant barometric pressure altimeters shall be implanted.
    2) On-board flight computer shall collect altitude, acceleration, and gyroscopic data.
6) The Active Drag System (ADS) will be able to control the apogee of the rocket.
    1) The ADS shall actuate the airbrakes to create drag.
        1) The airbrakes shall be actuated linearly.
    2) The ADS shall bring the apogee of rocket to 10,000 ft with a percent error of 5 percent if the rocket is exceeding 10,000 ft apogee.

D. Payload Requirements

1. Payload System Level Requirements

1) The Payload will adhere to the Spaceport America Cup rules and specifications.
    1) The Payload shall weigh no less than 8.8 lbs, no more that 25% “boiler plate" mass.
    2) The Payload shall be fully independent and easily removable of the launch vehicle.
    3) The Payload shall not contain significant quantities of lead or any hazardous materials.
    4) The Payload should conform to the 3U CubeSat form factor.
2) The Payload will contain a scientific experiment to create a possible solution for mitigating vibrations in a sounding rocket.
    1) The Payload shall be entered into the Space Dynamics Laboratory Cubesat Challenge at IREC
        1) The payload shall meet the requirements in Section 1
        2) The Payload shall weigh heavy emphasis on the SDL Payload Challenge Judging Criteria
3) The Payload structure will be designed for year to year re-usability

2. Payload Subsystem Requirements

1) The Payload structure will be designed for year to year re-usability
    1) The Payload will be precision manufactured to exact dimensions of the 3U standard
    2) The Payload shall withstand 16g’s of acceleration in all directions
        1) The Payload should have a 16g rating factor of safety greater than 2
        2) The payload fasteners should be pre-loaded to withstand vibrations under max loading
    3) The Payload should provide modularity for year to year competitions
        1) The Payload should have removable and relocatable platforms for hardware/electronics mounting
        2) The Payload should include vertical mounting capabilities for electronics and optics
        3) The Payload should have the ability to be precisely massed in accordance to system level requirement 1.1
    4) The Payload should be designed for maintainability and ease of access
        1) The on-board payload hardware will be accessible from all four sides of the CubeSat
        2) The number of unique components should be minimized
        3) All structural points should use the same fasteners
    5) The Payload shall be integrated into the 2 adjacent bulk heads on the launch vehicle
        1) The integration points shall be capable of withstanding 16g’s of acceleration in all directions
2) The payload will contain an on-board vibration isolation scientific experiment
    1) The Payload experiment will addess vibration isolation methods for on board electronics and avionics on sounding rockets
        1) The Payload should address vibration frequencies between 1-2000 Hz
        2) The Payload should target a transmission decrease equal to 6 dB average loss across the target frequency spectrum
    2) The Payload shall employ multiple vibration isolation methods for the frequency range in requirement 2.1.1
        1) The payload will us passive materials for vibration isolation
            1) A minimum of 3 passive materials should be investigated
        2) The Payload should create an active vibration isolation method
            1) The active vibration isolation hardware should be capable of isolating vibrations at a minimum frequency of 300 Hz
            2) The mechanical hardware shall be limited to a 10 cm x 10 cm platform
            3) The active system shall not operate at a frequency corresponding to any of the CubeSat’s natural frequencies
            4) The active system should be able to function in a high acceleration environment
        3) The accelerometers used for the isolation methods should be capable of recording a minimum of 8g acceleration at a minim frequency of 4000 Hz
        4) The acceleromters used to record launch vehicle vibrations should be capable of recording a minimum of 8g acceleration at a minim frequency of 2000Hz

        1) The Accelerometers shall use a maximum voltage of 5V
    5) The Data Acquisition System should be capable of recording, processing, and logging the accelerometer data a frequency greater than or equal to 400 Hz
        1) The Data Acquisition System should be able to fit within a 10cm x 10cm platform
        2) The Data Acquisition System should function off of a maximum voltage of 5V
        3) The Data Acquisition System should be 10 bit or higher
    6) The Payload should be capable of having an “idle" mode until launch is detected to conserve battery
    7) The Payload system shall be capable of running under max curent draw for 30 minutes
        1) The data logging equipment should include an additional 1hr of redundancy from the power supply
    8) The vibration isolation experiment should not result in the payload exceeding the minim payload weight by more than 10%
    9) A conclusion and recommended implementation procedure shall be created after the competition launch

3) The Payload Subteam will prioritize the Space Dynamics Laboratory judging criteria in the Payload’s design
    1) The Payload shall have scientific or technical objective
        1) The payload should select a scientific experiment relevant to the aerospace industry with an emphasis on topic relevant to sounding rockets
            1) The Payload experiment will address vibration isolation methods for on board electronics and avionics on sounding rockets
    2) The payload shall be well constructed
        1) A reusable Payload will be developed for year to year usage
            1) The Payload will be precision manufactured to exact dimensions of the 3U standard
        2) The Payload should provide modularity for year to year competitions
        3) The Payload should be designed for maintainability and ease of access
    3) The Payload subteam shall demonstrate overall professionalism
        1) The Subteam shall be well versed on all aspects of the payload
        2) The Subteam will prepare an exhaustive technical paper
            1) The paper shall contain information regarding the experiment and findings
        3) The Subteam shall conduct themselves according to the Hokie Handbook during launches, podium sessions, and interactions with judges and the public
        4) At least one Subteam member should attend the 2018-2019 SAC as an expert on this year’s payload
    4) The Payload shall not interfere launch operation and require minimal maintenance once integrated with the launch vehicle
        1) The Payload shall integrate with the two adjacent bulkheads in the Payload bay
            1) The integration should be capable of being performed in parallel with other tasks during launch vehicle assembly
        2) The Payload should be installed and removed in under 10 minutes
        3) The Payload should not require any modifications after integration and until the mission incomplete
        4) The Payload shall have adequate battery life for withstanding 1 hour on pad under the TBD max power consumption
    5) The Payload shall carry our the scientific or technical objective as intended with adequate data collection
        1) The vibration isolation Payload shall be flown in its operation state during rocket test flight
            1) The Payload should be adequately tested and validation on ground prior to flight for its accelerometer and data logging capabilities
            2) The data collected should be assessed to determine adherence to prior requirements
            3) The active isolation system should be tested for it’s functional ability to address frequencies with in it’s target range
        2) The Payload shall have sufficient data logging electronics
        3) The Payload experiment should be reassessed, refined, and improved after first test launch to further test more effective isolation methods and/or materials
        4) A conclusion and recommend implementation procedure shall be created after the competition launch

XII. Appendix G: Payload Specifications

A. Calculations

1. Fastener Safety Factor

The following calculations assume 1 fastener is supporting the entire mass of the payload and is under 16g acceleration (151.955 lbf)

$$\tau_y = 0.58 * S_y \tag{3}$$

where S_y equals 50.8 KPsi, the yield strength of Steel Alloy (the material of both), resulting in τ_y to equal 29.464 KPsi. The shear stress experienced by the fasteners is equal to,

$$\tau = \frac{F}{\pi * \frac{d^2}{4}} \tag{4}$$

Where F is the loading calculated above and d is the diameter of the bolt. The 10-32 screws have a diameter of 0.19 in and the 6-32 screws have a diameter of 0.138 in.

This results in the 10-32 screws having a factor of safety of 5.50 and the 6-32 screws having a factor of safety of 2.90

2. Fastener Torque Specifications

The torque specification is calculated to prevent the bolts from backing out during flight. We choose to take a conservative approach and are assuming the force driving the bolt out is equal to 151.955 lbf. The torque equation used can be seen bellow,

$$Torque = k * d * F * \% \tag{5}$$

Where, k is the constant for conditions of the bolt (assume lowest value of 0.2), D is the diameter of the bolt, F is the load, and % corresponds to the lubrication constant.

This results in a torque spec of 5.774 lbf/in and 4.194 lbf/in for the 10-32 and 6-32 respectively.

3. Load Bearing Members Safety Factor

The primary load bearing member for the payload are 1/8 inch aluminum angle irons spanning the length of the CubeSat. Using the same loading as the prior cases, but applied in bending, the safety factor of bending stress is computed below. Computing the area of inertia, I_{XY},

$$I_{xy} = \frac{t(5L^2 - 5Lt + t^2)(L^2 - LT + t^2}{12(2L - t} \tag{6}$$

where t is the thickness of the angle iron, 0.125 in, and L is the length of the CubeSat, 11.811 in.Next the area, A, and centroid, X_c is computed using

$$A = (2L - t) \tag{7}$$

$$X_c = \frac{\frac{t}{2}L^2 + Lt - t^2}{A} \tag{8}$$

The distance from the neutral axis can then been computed by,

$$D_{Neutral} = L - X_c \tag{9}$$

where L is the length of the angle.

The bending moment, Mx, is then computed by,

$$M_x = F * L \tag{10}$$

which is found to be 74.67 in-lbf. With the moment know, the bending stress, σ_b, is found by

$$\sigma_b = \frac{M_x * D_{Neutral}}{I_{XY}} \tag{11}$$

which if found to be 18.14 KPsi. Te yield strength of aluminum used is 40 Ksi, resulting in a safety factor of 2.2

B. Payload Electronics: Electronic and Component Data Sheets

The following are hyperlinks for Component Data Sheets:

- BeagleBone Black
- 1DOF High Performing Accelerometer
- 3 DOF High Performing Accelerometer
- ADXL345 Acceleromter
- High Fidelity Operational Amplifier
- DC/DC Converter
- Diode
- Voice Coil Actuator Specifications:

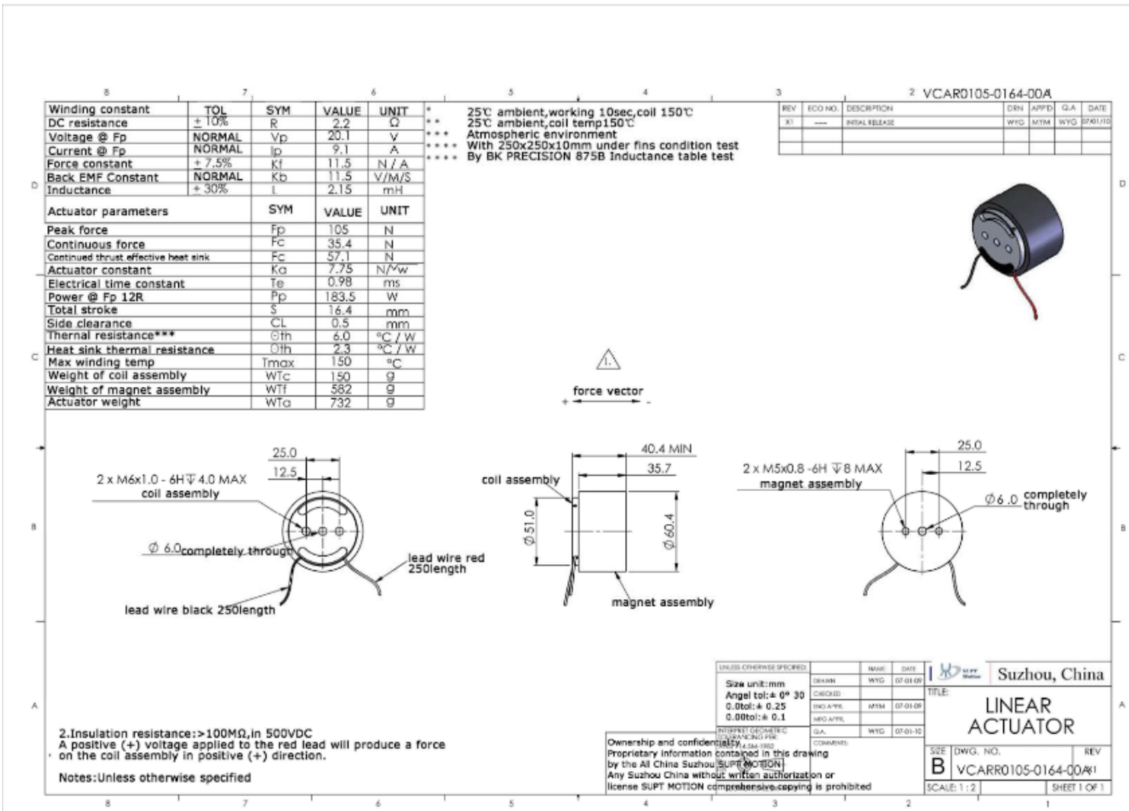

Fig. 69 Voice Coil Actuator Specifications

XIII. Appendix H1: Propulsion Motor Type Selection

The propulsion subteam began design by determining to pursue a hybrid motor implementation for the launch vehicle propulsion system. The hybrid motor was chosen over a liquid implementation as it requires less complex mechanical systems. Liquid motors require additional feedline and flow control components due to the presence of a liquid fuel source (figure). This liquid fuel also requires a pressure vessel to accommodate it. Due to Rocketry at Virginia Tech attempting its first SRAD motor, it was decided that these added complexities may be impractical or overwhelming. Perhaps in the future, once experience is built handling a liquid oxidizer, a liquid propulsion system could be implemented. Instead, a hybrid motor was chosen to be developed for the 2019 Spaceport America Cup as it is believed to be within the teams capability, and it meets the system requirements 1.1 and 1.3.

A. Oxidizer Selection

Following the decision to develop a hybrid rocket motor, choices for oxidizer were considered. Research conducted by the propulsion subteam found the three common oxidizers were utilized in hybrid rocket motors: liquid oxygen, high concentration hydrogen peroxide, and nitrous oxide (insert source). Liquid oxygen usage in rocketry applications requires storage as a cryogenic liquified gas, and requires specialized equipment and handling procedures. In addition, either a mechanized pump or external pressure source would be required to drive the liquid oxygen into the motors combustion chamber. source This added complexity and required specialized components associated with the a cryogenic liquified gas resulted in the propulsion subteam ruling out it's use for an oxidizer. High concentration hydrogen peroxide was also found to require a mechanized pump or an external pressure source to drive the oxidizer into the motors combustion chamber. Furthermore, hydrogen peroxide is often used as a monopropellant due to its tendency to decompose source. This possible decomposition poses increased safety risks. These increased safety risks coupled with the need for an external pressurization system or mechanized pump resulted in ruling out hydrogen peroxide as an oxidizer. As a result, nitrous oxide was deemed the ideal oxidizer, as it was self-pressurizing due to high vapor pressures, and required no cryogenic specific components due to nitrous oxide transitioning to a liquid at relatively low pressures and high temperatures relative oxygen. The self-pressurizing characteristic greatly reduces complexity as no pumps or additional pressure sources are required for driving oxidizer flow. This leads to a motor layout similar to that shown in figure . Nitrous oxide is also considered non-toxic by ESRA, therefore meeting requirement 1.7. Nitrous oxide can however under the right conditions exothermically decompose (see appendix (insert sections of safety report)). This risk was carefully analyzed, and the team determined that such risks could be mitigated through subsystem design features and establishing proper handling procedures. Nitrous oxide safety is more extensively analyzed in appendix , and this analysis forms many of the requirements listed in appendices -. With these risks analyzed and requirements developed to mitigate them (summarized in appendix), the propulsion team felt comfortable moving forward with nitrous oxide as the chosen oxidizer.

B. Fuel Selection

The chosen fuel is paraffin wax. Paraffin wax was chosen to be the ideal fuel source mainly due to the high regression rates it exhibits relative to other fuel sources. Testing conducted by Stanford University and NASA Ames demonstrated substantially higher regression rates utilizing paraffin wax as fuel as opposed to more traditionally utilized HTPB. Testing results form these studies are displayed in figure .

This high regression rate is vital in meeting requirement 1.4 as the high regression rate increases the overall mass flow rate and therefore the thrust of the motor. This thrust increase is highly favorable as it is instrumental in achieving the required off the rail velocity for stable flight. Calculating thrust required to meet required off the rail velocities and additional fuel grain design considerations and choices will be discussed in greater detail in proceeding sections. One more reason paraffin was chosen as the main fuel source was due to its non-toxicity, therefore meeting requirement 1.7.

XIV. Appendix H2: Propulsion System Sizing

Following the oxidizer selection of nitrous oxide and fuel selection of paraffin wax, the team worked to determine optimal quantities of both to be stored on board the propulsion system. The team began this investigation by tabulating data from both commercial hybrid motors and student researched and designed hybrid motors used in the 2018 Spaceport America Cup. This investigation was in effort to understand possible total impulse values based on quantities of oxidizer and fuel. The results of this research are displayed below in Figure 70 and Figure 71.

Motor	Oxidizer	Fuel	Mass of Oxidizer (lbm)	Mass of Fuel (lbm)	Total Impulse (lbf*s)	Comments
Contrail K234BG 54mm	Nitrous Oxide	HTPB with additives, single port	2.5 (@ 70 degrees F)	1.0	372.6	Tested this motor at a prior internship. Only about 30-40% of the fuel would be used per burn.
Contrail L1428SF-P 75mm	Nitrous Oxide	HTPB with additives, single port	5.5 (@ 70 degrees F)	2.6	1064.9	
Contrail M2800BG 98mm	Nitrous Oxide	HTPB with additives, single port	9.0 (@ 70 degrees F)	4.9	1363.5	
Contrail O6300BS 152mm	Nitrous Oxide	HTPB with additives, multi port	25.5 (@ 70 degrees F)	11.7	6120.0	
Rattworks M900 64mm	Nitrous Oxide	Polypropylene, single port	6.3 (@ 70 degrees F)	1.1	1450.2	
Hypertek M1001 5478/98-RG-M 98mm	Nitrous Oxide	Thermoplastic, single port	9.3 (@ 70 degrees F)	2.3	2212.9	
University of aterloo Kimet Engine	Nitrous Oxide	90% HTPB, 10% powdered aluminum by mass. Pseudo-finocyl port geometry.	33.3 (@ 70 degrees F)	6.7	4959.0	Achieved about 60% efficiency relative to recorded Isp vs ideal Isp. 70% of fuel grain burned. Predicted to lift 143 lb launch weight rocket to 12,480 feet. Actual apogee was 13,412 feet. This team one first place in the 10k SRAD Hybrid/Liquid Category in the 2018 Spaceport America Cup.
University of Washington *Boundless* Hybrid Rocket Motor	Nitrous Oxide	89% Paraffin wax, 4% stearic acid, 2% vybar, 5% aluminum powder, by mass. Single port.	36	12	9200.0	Specific impulse of 191 seconds. Designed to hoist launch vehicle with a lift off weight of 178 lbs to 30,000 feet. Achieved an impulse of 10,500 lbf*s on a static fire test.

Fig. 70

ULA Prometheus Hybrid Motor	Nitrous Oxide	Paraffin HTPB mixture. Single port.	10.9	1.7	1761.3	Modified Contrail M-1575
Oronos Polytechnique Montreal Prometheus Hybrid Motor	Nitrous Oxide	Paraffin with Vybar additive and black dye, Single Port	18.0 (@ 70 degrees F)	6.2	3712.5	
University of Calgary Atlantis Hybrid Motor	Nitrous Oxide	Paraffin with 10% tar. Single port.	48.5	7.9	9000	Predicted apogee of 30,180 fete with 140 lbm launch weight. Did not launch in 2018 Spaceport America Cup.3

Table 1: Comparison of Impulse and Propellant Quantities

Fig. 71

This data was used to better understand the range of possible impulses that can be expected from varying quantities of fuel and oxidizer. Furthermore, Rocketry at Virginia Tech in past years has developed a MATLAB program that generates thrust curves, chamber pressures, and nitrous oxide vapor pressure of a hybrid rocket motor utilizing inputs of nitrous oxide mass, fuel mass, fuel port diameter, initial oxidizer temperature, and fuel type. For the 2019 spaceport America Cup, this program was updated to include modeling of flight trajectories, heat transfer to combustion chamber walls, oxidizer to fuel mass flow rate, max launch vehicle velocity, nitrous oxide flow velocity through feed lines, and percentages of nitrous oxide and fuel used during the burn. In addition, characteristic velocity efficiency and a exit velocity coefficient are considered in generating thrust curve and chamber pressure data. The characteristic velocity (c*) efficiency is computed via equation 12.

$$C^*_{efficiency} = c^*_{actual}/c^*_{ideal} \tag{12}$$

Where: $c*_{actual}$is the recorded c* in testing, $c*_{ideal}$ is the idealized simulated c*
Characteristic velocity can be computed alone via equation 13:

$$C^* = P_c * A_t/\dot{m} \tag{13}$$

Where: P_c = combustion chamber pressure, A_t = area of nozzle throat, $\dot{m}$ = mass flow rate
The exit velocity coefficient is computed via equation 14:

$$ExitVelocityCoefficient = V_{actual}/V_{ideal} \tag{14}$$

Where: V_{actual} = the actual recorded/calculated exit velocity of the nozzle, V_{ideal} = ideal/theoretical nozzle velocity

The velocity coefficient and c* efficiency values allow for corrections that account for inefficiencies of the motor's combustion and nozzle expansion. This significantly scales thrust curve outputs down and allows for more realistic estimates. The c* efficiency is of particular importance, as hybrid rocket motors often exhibit low combustion chamber efficiencies due to not all the available regressed fuel mixing and combusting with the oxidizer in the combustion chamber. C* efficiency provides a method of quantifying this incomplete mixing.

Before applying c* efficiencies or velocity coefficients, the program calculates ideal values for chamber pressure. A key component in finding ideal chamber pressure is mass flow rate. The mass flow rate is broken up into two varieties; the nitrous oxide mass flow rate and fuel mass flow rate. Nitrous oxide mass flow rate is found based on an algorithm introduced in a technical paper published by the United Kingdom based organization Aspire Space. The technical paper, titled Modelling the Nitrous Run Tank Emptying, presents a model for describing a nitrous oxide tank emptying at discrete time steps. The model introduces a formula for finding mass flow rate nitrous oxide based upon a modified formula for mass flow rates of liquids based on a pressure differential. This formula is given by equation 15 below:

$$\dot{m} = sqrt2 * rho * dP/D_{loss} \tag{15}$$

Where Rho = current density of nitrous oxide, dP = pressure drop across injector, And Dloss is given by equation 16:

$$D_{loss} = K/((N * A_{inj})^2) \tag{16}$$

Where: K = empirical loss coefficient, N = number of injector orifices, A_{inj} = area of each injector orifice

It is important to note that the loss coefficient K must be empirically derived, and if not the same as flow coefficients given in commercial flow components. This is due to nitrous oxide being a two-phase fluid as it pases through an injector orifice. As it passes, the nitrous oxide goes through a phase transition from a liquid to a gas. This process causes stadard loss coefficients for single phase flows to be invalid, and thus the loss coefficient used in this formula must be empirically derived. When this formula is applied over a discrete time step, the total mass lost from the nitrous oxide tank can be found via numerical integration. Once the mass lost over the time step is found, the physical properties of the nitrous oxide, such as density and vapor pressure, can be found by updating the value of the current temperature of the fluid. The change in temperature can be found by solving for the heat lost via evaporative cooling. This process allows for the mass flow rate to be found at each discrete time step, and for the physical properties of the nitrous oxide to be updated from time step to time step.

With the nitrous oxide mass flow rate calculated, the mass flow rate of the fuel is found. The formula for mass flow rate is given by equation 17:

$$\dot{m}_F = \dot{r} * rho_F * BA \tag{17}$$

Where: rho_F is the density of the fuel, BA is the surface area of the port, is the regression rate given by equation 18, and is introduced in Sutton's Rocket Propulsion Elements:

$$\dot{R} = a * G^n \tag{18}$$

Where A = empirical constant, G = oxidizer mass flux, N = empirical constant

This regression rate formula assumes any variance in regression rate down the length of the port to be negligible. For the purposes of Rocketry at Virginia Tech, this is assumed to be a reasonable assumption. With nitrous oxide and fuel mass flow rate calculated, the total mass flow rate is known. With mass flow rate known, and assuming adiabatic combustion, adiabatic flame temperature is calculated using a constant volume process. Assuming isentropic flow through a conical converging diverging nozzle, and using isentropic relations, combustion chamber pressure and nozzle exit velocities are calculated.These represent the ideal values.The chamber pressure is then scaled down by being multiplied by the inputted c* efficiency and the exit velocity is scaled down by being multiplied by the inputted velocity coefficient. Thrust is then calculated using these scaled down values. The resulting thrust curve is considered to adequately reflect hybrid motor inefficiencies.

With a thrust curve calculated, the program calculates trajectory using Euler's method and inputted values for rocket mass, propellant mass, drag coefficient, and starting altitude. The model is one dimensional, and is designed mainly to be a quick estimation for apogee based on different motor designs. For more accurate estimations other software, such as OpenRocket, is utilized by the Aerostructures subteam.

The aforementioned model and collected data in table 1 was used by the propulsion subteam for determining the optimal nitrous oxide and paraffin mass. The model was first calibrated by adjusting c* efficiency and thrust coefficient values until the returned results for impulse matched the tabulated data in table 1. Regression rate coefficients used are 0.12 for a and 0.5 for n. These values were calculated from testing done at Stanford University insert source. A loss coefficient value of 2.0 was utilized, as this was recommended point by Aspire Space in the aforementioned technical paper. It was found a c* efficiency of 0.65 and a velocity coefficient of 0.92 were required to for the model outputs to match the collected data. Consequently, those values were utilized when using the developed MATLAB model.

XV. Appendix H3: Propulsion Hoop Stress Calculations

In order to calculate hoop stress, an internal pressure of 800 PSIG was utilized, as at this pressure nitrous oxide is past it's supercritical point, which for both performance and safety reasons the team aims to avoid reaching through venting and nitrous oxide fill bottle cooling. As a result, this value for pressure was considered the upper limit for pressure in the oxidizer tank. The hoop stress was calculated with the cylinder hoop stress formula, listed as equation 19 below:

$$HoopStress = P * d/(2 * t) \quad (19)$$

Where: P is the internal pressure, D is the inner diameter of the tank, T is the thickness of the tank wall,

The resulting hoop stress was found to be 8,800 psi. The yield stress of the aluminum was listed as 35,000 psi from the supplier, and as a result there is a safety factor of 3.97 for hoop stress.

XVI. Appendix H4: Propulsion Fastener Selection

Based on the design internal pressure of 800 PSIG, and the oxidizer tank inner diameter of 5.5 inches, it was found that the axial loading due to the internal pressure is 19,006 lbf. In addition, factoring in the acceleration of the launch vehicle at an upper value of 10g's, an additional load of 250 lbf from the mass of the nitrous oxide is considered, giving a total axial load of 19,256 lbf. The fasteners used in this connection are 5/16-18 alloy steel socket head screws. These screws conform to ASTM A574 standards. These standards state that the minimum allowable single shear load these fasteners are rated to is 6,800 lbf. There are a total of 12 of these fasteners; assuming equal load distribution each bolt supports 1600 lbf of loading resulting in the factor of safety for shear loading on the fasteners of 4.23.

XVII. Appendix H5: Propulsion

The bolt holes on the tank are spaced as recommended by AISC standards, specically section J3.3. This spacing is three times the diameter of the bolts from hole to hole, with a distance from the edge of the material of at least the diameter of the bolt. In Rocketry at Virginia Tech's design, the distance from the edge of the material is 0.64 inches. AISC section J3.3 states that the bearing strength of each bolt hole in lbf in the material of the bolted connection can be calculated by equation 20

$$R_n = 0.75 * 1.2 * L_C * t * F_u \quad (20)$$

where, L_c is the distance from the bolt hole to edge of the tank, t is the thickness of the tank, F_u is the tensile strength of the tank material

Using this equation the strength of each hole in the oxidizer tank is found to be 6,040 lbf. Assuming equal loading in each hole (1600 lbf per bolt hole), the safety factor for bearing stress on the holes drilled into the oxidizer tank is calculated to be 3.15.

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