Project Raziel

Preliminary Design Review

2016 - 2017

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1. Executive Summary

1.1. Team Overview

The MIT Rocket Team (hereafter, “the Team”) is a well-established student group focused on rocket-related projects. The organization’s mission is to proliferate the knowledge of sciences related to rocketry, to foster the development of skills and techniques related to the field, and to provide a hands-on, project oriented outlet for application of theory learned in the classroom setting. To do this, the team provides resources for personal high-power rockets, and competes in intercollegiate rocketry competitions.

1.2. About this Document

This design review attempts to cover several subteams, all of which are at different stages of development; therefore, this design review may not fully reflect a PDR in industry. Some of the subsystems for Project Raziel have progressed beyond the traditional PDR, and the teams are refining them for this project (e.g., the fiberglass tubes, Pyxida flight computer). Other systems are totally new, and have not been tested yet (e.g., the rover payload). Therefore, be aware that not every subsystem is at PDR level.

The Team does its best to balance time spent obtaining feedback through design reviews with time spent prototyping and flying rockets. As such, the purpose of this PDR is to obtain feedback on all of the subsystems within its individual level of maturity to improve Project Raziel, and prepare for both test and competition flights.

One goal of last year’s submission to the IREC was to build a rocket with as many student-built components as possible, even though it was not required for our category. This way, we always had a commercial, off-the-shelf (COTS) backup for any particular part of the rocket. This rocket, Therion, was 6” in diameter, and stood ~12 feet, 8 inches tall. Many of the subsystems of Project Raziel have been matured by last year’s project, in particular, Avionics and Structures, whereas Payload and Ground Support Equipment are brand new this year, while Propulsion and Recovery are somewhere in between.

This PDR utilizes a standardized risk table to delineate risks associated with the subsystem. Risks highlighted in red constitute a no-go. Yellow risks are questionable, and a decision will be made by team leadership with respect to a launch. The following table serves as a legend and example for subsequent tables.

Guest User: I'm glad that you are using a standardized risk table, but you may want to rethink the axes of the chart. See my next comment for more information

Risk	20%					
	10%					
	5%					
	1%					
	0.1%					
		1	2	3	4	5
Subsystem		Impact				
Description		Mission not compromised or minor loss of data	Damage to subsystem, substantial loss of data	Loss of subsystem, loss of critical data	Total loss of vehicle, miss major milestone due to schedule slip, loss of all data	Loss of mission, injury to team members, external parties adversely affected

Guest User: You may want to rethink this axis: Typically risk charts show impact vs a likelihood or probability of an event occurring.

In fact this seems to be what you are trying to convey in this chart by showing values as a percentage. Otherwise, what is a 20% Risk?

Also I am unsure why you chose the value range you showed here. Is it possible for a risk to have a greater than 20% likelihood?

Suggestion: Change this axis to likelihood and let it go from 0-100% in some convenient number of steps.

1.3. Goals, Competition Details

Previously, the Team competed in the Intercollegiate Rocket Engineering Competition (IREC), through the Experimental Sounding Rocket Association (ESRA) in Green River, Utah. This year, ESRA and Spaceport America have teamed up to create the Spaceport America Cup (SAC), moving the IREC to New Mexico to support more launches and higher altitudes.

The Spaceport America Cup is split up into six categories as follows, based on COTS and student researched and developed (SRAD) propulsion, propellant type, and target altitude.

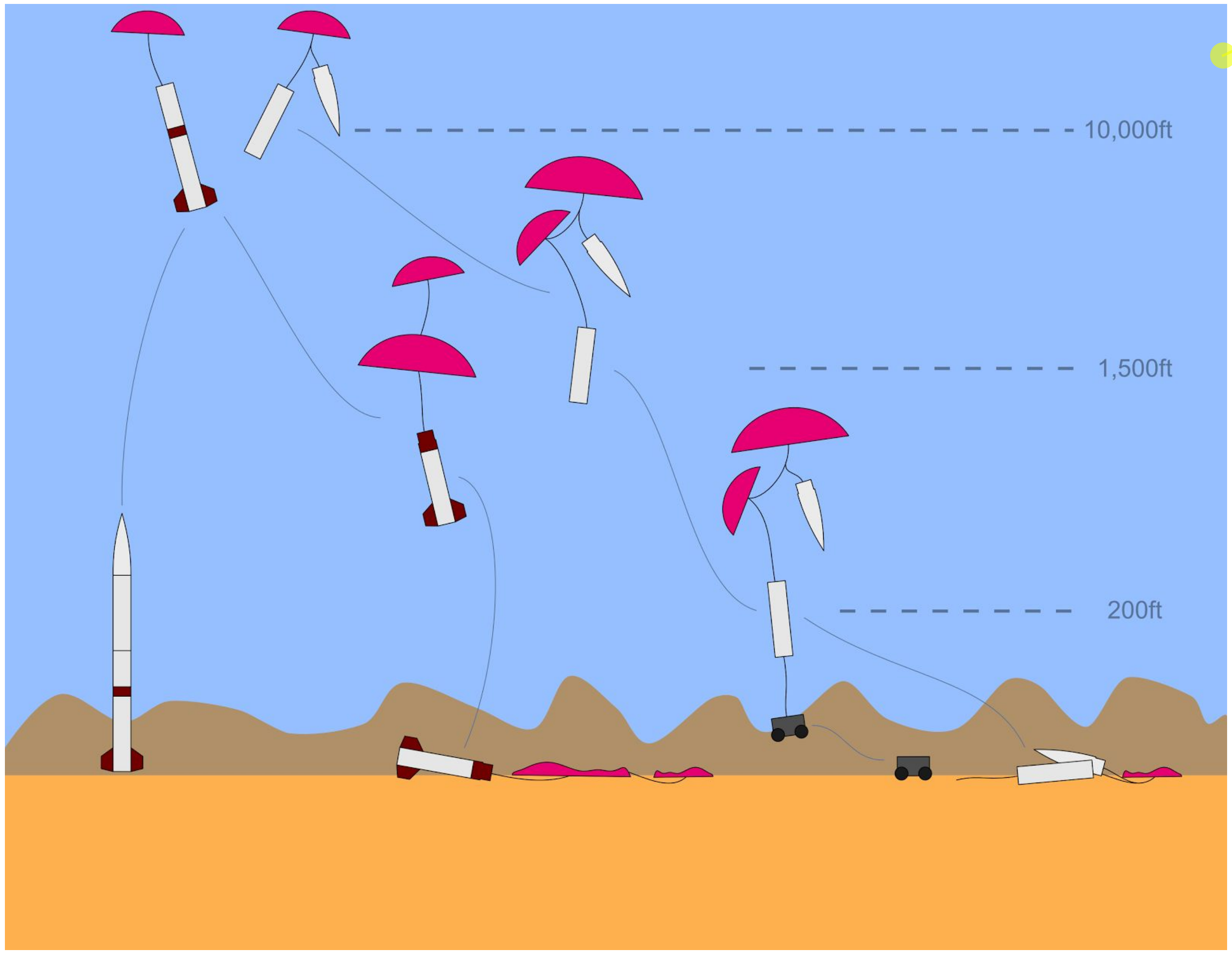
Target Altitude (AGL)	10,000 feet	30,000 feet
COTS Solid or Hybrid		
SRAD Solid	Team's entry	
SRAD Hybrid or Liquid		

The SRAD solid propellant category closely aligns with the team's goals from Project Therion moving forward, because the team wanted to compete using custom propulsion and an entirely student-built rocket this year.

At a high level, the SAC requires the delivery of at least 8.8 lbs of payload to the specified target altitude. The competition is scored as follows:

Item	# Points	Description
Entry form, progress updates	100	The on-time delivery of the entry form and 3 progress updates each provide 25 points towards the final score. They are awarded points are on a pass-fail basis.
Technical Report	200	Points are awarded for quality of technical writing (40), completeness of the report (40), and rigorous analysis of necessary trade spaces and constraints (120).
Design Implementation	200	Competency of design & construction (100) points are awarded for robust systems and quality of hardware. SRAD component implementation receives the other half of the points for this category (100). The most key areas for SRAD are all airframe sections and parachutes.
Rocket Performance	500	The rocket performance is scored based on accuracy to the target altitude (350) and the successful recovery of the rocket in re-flyable condition (minus consumables). (150)
Total	1000	
Infractions	-20	Teams are penalized 20 points per infraction for unsafe or unsportsmanlike conduct as deemed by the judges. This includes failure to use appropriate PPE, failure to use checklists, violating traffic rules, and the like.

1.4. Vehicle Concept of Operations (CONOPS)



Guest User: I like the graphic, but no figure numbers. You might as well get used to adding figure numbers and captions now, since you'll be expected to use them throughout industry.

The mission concept of operations proceeds as follows:

1. *Vehicle* launches from launch rail, and ascends to 10,000ft
2. *Vehicle* ejects *payload segment*, and deploys the *vehicle drogue*
3. *Payload segment* deploys the payload drogue
4. *Vehicle* and *payload segment* descend to 1,500ft under drogue
5. *Vehicle* releases *vehicle main chute*
6. Payload segment releases *payload main chute*
7. *Vehicle* lands on the ground
8. *Payload segment* descends to about 200ft
9. *Payload segment* deploys *rover* to *landing position*
10. Upon or shortly above ground contact, *rover* separates from *payload segment*
11. *Payload segment* lands on the ground

1.5. Team Organization

Per the Team Constitution, the Executive board is comprised of a President, Vice-President, Treasurer, Publicity Chair, Social Chair and Safety Chair. In addition, the team appointed a Launch Ops Chair, which oversees launch operations at test flights and at competition.

To complete this project, 6 subteams have been formed, each with its own leader: Structures, Recovery, Avionics, Propulsion, Ground Support Equipment (GSE), and Payload. Each team is tasked with certain deliverables essential to the fulfillment of the goals of the team, and the requirements of the competition. These deliverables and requirements, as well as their relation to system-level requirements are detailed in later sections.

General team members are free to work for whichever subteam they desire, and may work for multiple subteams. Currently, we have approximately 50 regularly attending members on the Team.

2. Systems Overview

2.1. Requirements and Goals

The Team has outlined goals based on the the previous two years' performance at IREC. ESRA has posted technical requirements for the Spaceport America Cup online. Of these requirements, we have checked which ones are most relevant to Project Raziel, and listed them in Combined Requirements Table (Appendix A). Each subteam is responsible for complying with their designated requirements, as well as the overall Team requirements. Subteam requirements are documented in their respective subteam sections.

The following are the Team's goals for this year's competition:

	Baseline	Target	Reach
1	Members are safe	Members learn something	Members become rocket engineers
2	50% of systems are student-designed	80% of systems are student-designed	90% of systems are student designed
3	Rocket has partially-successful flight test	Rocket has fully-successful flight test	Rocket has two fully-successful flight tests
4	Rocket integration shall take less than 30min	Any system of the rocket accessible within 5min	
5	Flight apogee between 9,000 and 11,000ft	Flight apogee between 9,500 and 10,500ft	Flight apogee between 9,800 and 10,200ft
6	Structures can withstand all flight loads	Structures can withstand 110˚F, direct sunlight for 2 hours	
7	Flight data stored on Pyxida	Flight data transmitted to ground station	Flight data presented in intuitive format
8	Rocket recovered in reflyable condition	Rocket recovered using non-pyrotechnic devices	
9	Rover lands safely	Rover transmits back data from sensors	Rover transmits back live video

Guest User: can greater than 2hrs be the reach goal here?

2.2. Timeline

Project Raziel features test plans for several of the subteams and are detailed in each subteam's section of this PDR. At the highest level, the team has planned to complete a flight test before the end of the fall semester (December 23rd), primarily using the techniques refined from Project Therion. This flight test is planned to test the major deployment sequences of the rocket, such as the deployment of the parachutes and the deployment of a dummy rover. A successful test of these deployments gives the team a working competition rocket that can fly with a minimally complex payload at SAC, regardless of the results of subsequent development cycles.

The planned schedule is as follows:

Guest User: What are the significance of the red dates? Are they milestones? Perhaps a table of milestones/key dates would be useful here.

Task	9/3/2015	9/10	9/17	9/24	10/1	10/8	10/15	10/22	10/29	11/5	11/12	11/19	11/26	12/3	12/10
Requirements Development															
System Design															
Payload System Design															
Strand Burn Testing															
54 mm, 1 grain program															
98 mm, 1 grain program															
Flight Motor Testing															
Recovery Deployment Dev															
Recovery Test Plan															
Avionics Test Plan															
PDR															
Raziel 1 Build															
Flight Test #1															
Structures Dev. Testing															
CO2 Deployment Dev.															
Payload Prototyping															
Development Reviews															
Raziel 2 Build															
Flight Test #2															
Flight Test #3															
Preparation for SAC															
SAC															

Task	1/2/2017	1/9	1/16	1/23	1/30	2/6	2/13	2/20	2/27	3/6	3/13	3/20	3/27
Requirements Development													
System Design													
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Flight Test #2													
Raziel 3 Build													
Flight Test #3													
Build Raziel 4													
Preparation for SAC													
SAC													

Task	4/3/2017	4/10	4/17	4/24	5/1	5/8	5/15	5/22	5/29	6/5-19	6/20-24
Requirements Development											
System Design											
Payload System Design											
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Flight Test #2											
Raziel 3 Build											
Flight Test #3											
Build Raziel 4											
Preparation for SAC											
SAC											

2.3. Safety Plan

Given the dangerous nature of many of the rocket's components, including energetic, toxic, and electrical hazards, as well as general safety concerns such as the use of power tools, the Team implements strict regulations to ensure the safety of its members. These regulations are documented in a safety plan, which is reviewed by the MIT AeroAstro Facilities Committee. The team also elects the Safety Officer, who is responsible for ensuring team member compliance with the safety plan. The safety plan and Safety Officer focus on day-to-day lab activities.

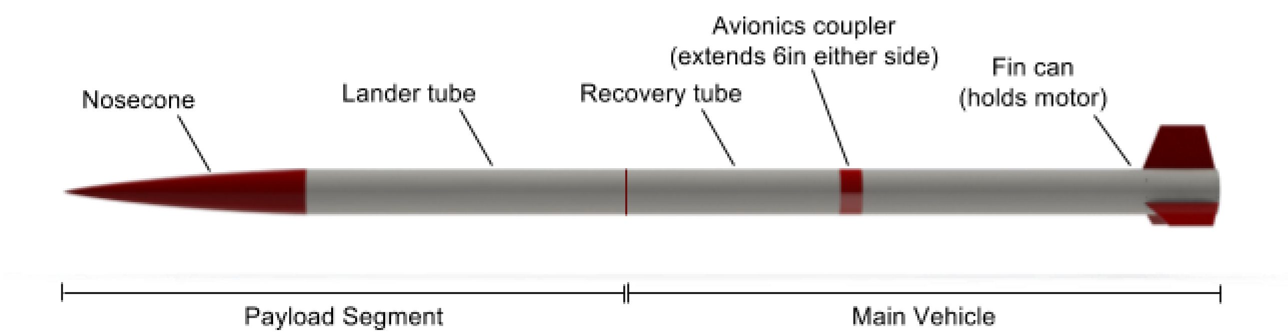
In addition to the safety practices available in the safety plan, members are required to complete safety trainings to access certain labs and tools. In order to access MIT's main aerospace lab, the Gelb Lab, members are required to complete an MIT AeroAstro online safety course. Additionally, in order to access the AeroAstro Machine Shop, members are required to complete four hours of Machine Shop training on lathe and mill.

2.4. Vehicle Overview

Project Raziel stands 13' 8" tall and has a projected weight of 26 kg without propellant. It is powered by custom ammonium perchlorate composite propellant. The deployments are controlled by a custom flight computer and a COTS flight computer in parallel, contained in the avionics bay. The recovery system is single separation, dual-deploy, and is located between the payload segment and the avionics bay. The payload segment is the uppermost section of body tube, and the nose cone, which contain the rover, a separate set of flight computers and parachutes, and the rover deployment mechanism. All of these subsystems are described in detail in the subsections that follow.

Guest User: Mixed unit system!! Either pick one, or use dual units with a standardized format.

Below: Render of Raziel



Below are the mass allocations for Raziel (without propellant). For comparison, we have also included mass allocations and usages from last year.

Subsystem	Structures	Propulsion	Avionics	Recovery	Payload
Therion Allocation (kg)	7.0	3.4	0.6	10.0	4.5
Therion Usage (kg)	6.7	3.3	0.7	5.7	9.1
Raziel Allocation (kg)	7.0	5.5	1.0	7.0	5.5

Guest User: May want to clarify that Therion was the rocket from last year for those unfamiliar with the names.

Guest User: Is this "As Flown" ?

The large discrepancy in the recovery and payload allocations for Project Therion came as a result of not flying the parafoil section, and instead replacing it with ballast, which was then added to the payload mass.

3. Structures

3.1. Overview of Requirements

Requirements as defined by the team, taken from Appendix A.

No.	Source	Description
3.0	DTEG 6.2	Structure shall withstand all loads from flight, landing, and transportation.
3.1.1	PDR 6	Rocket shall be able to withstand bending moment from aerodynamic pressure at expected maximum angle of attack.
3.1.1.1	DTEG 6.2.3	All coupling tubes shall extend at least one “tube diameter” through each adjacent tube
3.1.2	PDR 6	Structure shall be able to withstand 7,000N of thrust.
3.1.3	PDR 6	Structure shall be able to bear deployment loads.
3.1.3.1	DTEG 6.2.2	All eye bolts/nuts shall be closed-eye, forged steel
3.1.4	PDR 6	Structure shall bear the maximum landing loads due to rocket landing under main parachute.
3.2	PDR 5	Rocket shall have at least one place to adjust ballast weight for purposes of altitude tuning.
3.3	DTEG 8.2	Rocket shall use the ESRA-provided launch rails
3.3.1	DTEG 8.1	Rocket shall launch at a nominal elevation angle of 84°±1°, and as low as 70°.
3.3.2	DTEG 6.2.4	Rocket shall incorporate a minimum of two rail guides. These rail guides shall support the vehicle’s fully loaded launch weight when suspended horizontally, and the aft most rail guide must support the launch vehicle’s fully loaded launch weight while vertical.
3.3.3	DTEG 8.2-8.3	Rocket shall have a stable angle of attack upon launch rail exit, and shall remain stable for the entire ascent.

Guest User: Unfortunately I do not have time to go through all of your requirements, so I will take a look at a few and hope the rest are good.

I am very happy that you guys are thinking about them though!

Guest User: I know this comes from the competition, but be aware that NAR/Tripoli rules dont let you fly more than 15deg from vertical. So keep that in mind for test flights.

3.3.4		A person shall stand no higher than 4 feet on a ladder to access the rocket on the launch pad.
3.4		To allow for radio to be placed freely inside the rocket, the body material shall be radio transparent.
3.5	PDR 4	Structure shall be designed for accessibility. Any subsystem shall be accessible within 5 minutes of disassembly.
3.5.1		Fins shall be removable.
3.6.1	DTEG 6.1	Airframe shall be adequately vented to prevent unintentional separations due to pressure.
3.6.2	PDR 6	Structure shall tolerate temperature and heat flux from motor and environmental conditions.
3.7.1	DTEG 6.3	The team ID shall be visible from all sides of the rocket.
3.7.2	DTEG 6.3	The rocket shall be painted with the project name and academic affiliations.
3.7.3		The rocket shall be painted with all Inconel and Gold level sponsors.

3.2. Design Process

Structures designed a 164-inch (13 foot, 8 inch) rocket based on input from other subteams as to how much space would be necessary. Structures designed around constraints from the other subteams, making the rocket radio frequency (RF) transparent for avionics, and with 6” inner diameter tubes to accommodate payload’s cubesat configuration. The goal of the Structures team is to minimize weight while still designing and manufacturing a rocket that can withstand all expected loads, including shock and landing loads.

3.3. Technical Design/Analysis

3.3.1. Fabric Type

Structures chose a 9oz, 8-harness satin weave S-glass for construction of the rocket because of the high strength to weight ratio of S-glass, as seen in the comparison table below. While carbon fiber provides the greatest strength to weight ratio, it is not RF transparent. Therefore, structures elected to use fiberglass as to not interfere with avionics’ tracking devices. Structures chose a 9oz glass because it is very tightly woven, providing superior strength, and a harness satin weave to better conform to odd shapes like the tubes and nose cones.

Material	Ultimate Tensile Strength at ~22C [MPa][1]	Density [g/cc]	Ratio
Carbon Fiber, Generic	4150	1.78	2331
S-Glass Fiber, Generic	4585	2.48	1848
E-Glass Fiber, Generic	3790	2.54	1492
E-Glass Fiberglass/Epoxy Composite	490	1.9	258
Aluminum	310	2.7	115

3.3.2. Tube Layup Process

Rocket Body Tube Custom Layup Jig

The tubes for the rocket body will each be made in-house using a custom jig, pictured above, with a 5' long, 6" outer diameter (OD) rotating aluminum mandrel held horizontal by two wooden supports. The mandrel can be rotated by hand or with the use of two variable-speed stepper motors on each side. The aluminum mandrel is covered with a sheet of Mylar to make the removal of the finished tube easier. The jig can also be rearranged to fit a 2' long, 5.85" OD mandrel for coupler layups.

[1] All material data in Structures section from similar materials defined in Matweb database

Each main body tube for the first test flight of Raziel will be made with 10 layers of an S-glass fiberglass and epoxy composite. The team chose to use 10 layers of fabric initially because this procedure was tested last year. However, the tubes will also be tested after the first flight test to determine the optimal number of plies. To make an individual tube, the fiberglass fabric and epoxy will be simultaneously added to the mandrel: after every layer of fiberglass is added, enough epoxy to bond the fabric layers will be added to the glass with putty knifes. This will ensure the highest possible composite strength after the fiberglass and epoxy are combined. The current epoxy to fiberglass ratio is 3:4 which is very close to the optimal ratio[2] of 1:1. The fiberglass is added in straight wraps, such that the fibers are in 0 and 90 degree directions, which was determined optimal as opposed to a quasi-isotropic (-45/+45-degree alternating with 0/90) pattern based on an FEA analysis. Because the glass is only strong in the direction of the fibers, having the fibers in the direction of the vertical tensile forces and in the direction of hoop stress is the optimal design. The tube cures with single film of perforated plastic covering the exterior of the tube, which will ensure a smoother finish of the tube. Each tube is given at least 12 hours to cure in a well-ventilated area.

Each layup is made 0.5 to 1 inch longer than desired on each side because the edges have poor fiber structure. These rough ends are cut off so that the end product of every tube is uniform throughout. After finishing the layup, a second layer of epoxy is applied to fill in gaps and allow for sanding the tube without affecting the fibers. The tube is cut to size and the exterior is sanded with grit starting from 60 and incrementally increasing to 800. Imperfections are filled in with epoxy, Bondo, or gel coat depending on the severity of the imperfection. Primer, base coat, and paint are sprayed onto the tubes after sanding to create an ideal surface finish. This leaves a fully finished tube ready for implementation.

3.3.3. Future Layup Process

Future layups will be vacuum bagged and cured in an oven. The vacuum bagging serves two purposes: it squeezes out excess epoxy to remove unnecessary weight, and it compresses the fiberglass layers. This increases the strength of the layup by increasing the friction between layers. To implement this, the vacuum bag will be placed between the mandrel and the layup, and over the top of the layup. A hole will be drilled in the mandrel in order to insert the vacuum pump through the center of the tube to the outside of the tube. A few layers of wet fiberglass and bleeder fabric will be placed over the vacuum bag to keep the bag stiff and flat at that point. A layer of perforated plastic and a layer of bleeder fabric will be placed over the layup to create a smooth surface finish and allow excess epoxy to bleed out into the bleeder fabric.

In choosing the material for the mandrel, Structures decided on aluminum because it will expand the most without melting in the oven. An aluminum mandrel has desired properties for the team's procedure such as a maximum service temperature 1080°F and a change in circumference in the oven of .0434". Aluminum tubing is easily found with a 6" outer diameter. Aluminum better fits the team's needs than PVC, which lacks the thermal properties we need

[2] "Composites Part B: Engineering," *Composites Part B: Engineering*, vol. 79, Sep. 2015, pp. 132–137.

and differs in outer diameter size. Aluminum also has preferable thermal properties to steel, while also having a lower cost.

The structures team is building a composite oven in which to cure future tubes. The oven has a layer of concrete board to avoid flammability, a layer of fiberglass insulation to contain the heat, and a wooden frame. A modified space heater with a ducting system is placed on top of the oven to provide even heating to the oven. It will be controlled in order to provide the appropriate cure and post-cure cycles.

The oven allows us to use an epoxy system with a higher glass transition temperature as well as make a higher quality tube. The new epoxy system, 3000/3120 from Fiber Glast, has a maximum service temperature of over 250°F. The layup is cured at room temperature, and then uses the following post cure cycle[3]:

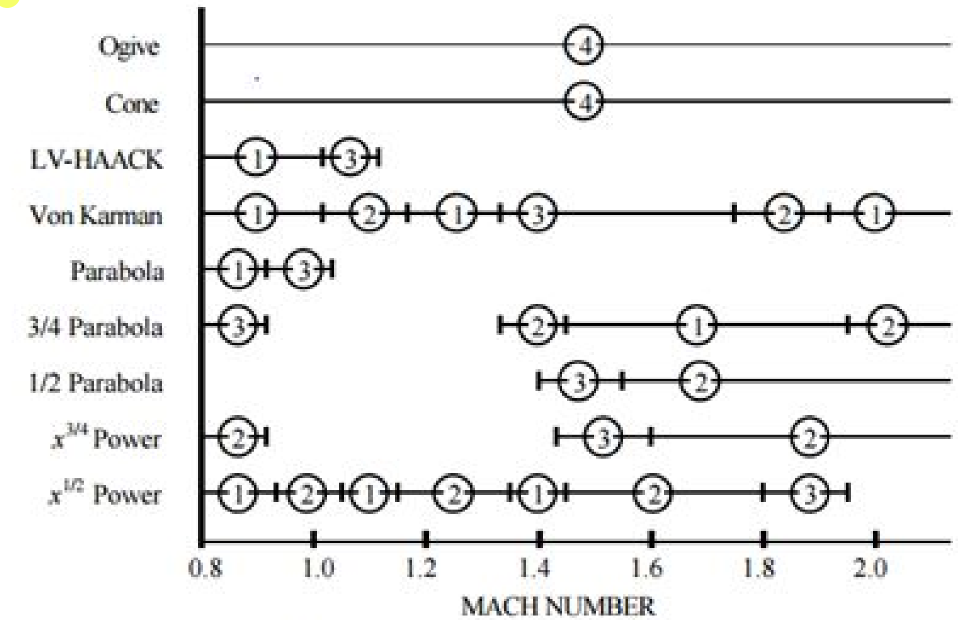
1. Ramp up temperature at a rate of no more than 2-5°F per minute.
2. Hold at 150°F, 250°F, and 300°F for 3 hours each.
3. Ramp down temperature to 100°F at a rate of no more than 2-5°F per minute. Do not shut down the oven and leave to cool.

3.3.4. Nose Cone Shape

Since the rocket will be traveling just over Mach 1, the Structures team chose a Von Kármán nose cone. The Von Karman shape is calculated to have the theoretical minimum wave drag, and has been proven to have the least drag of nose cone shapes in the team's flight regime. Based on the results show in the graphs below, the Von Karman shape has the least drag just under Mach 1, as well as low drag over Mach 1 (for the Mach 1 to Mach 1.4 range).

[3] "System 3000 High Temp Epoxy Kit," *Fiber Glast Development Corporation* Available: http://www.fibreglast.com/product/high-temp-epoxy-resin-3000/epoxy_resins.

Guest User: Figure label and caption would be nice.



Comparison of drag characteristics of various nose shapes in the transonic-to-low Mach regions. Rankings are: superior (1), good (2), fair (3), inferior (4). [4]

[4] Crowell, G. A., "The Descriptive Geometry of Nose Cones," 1996.

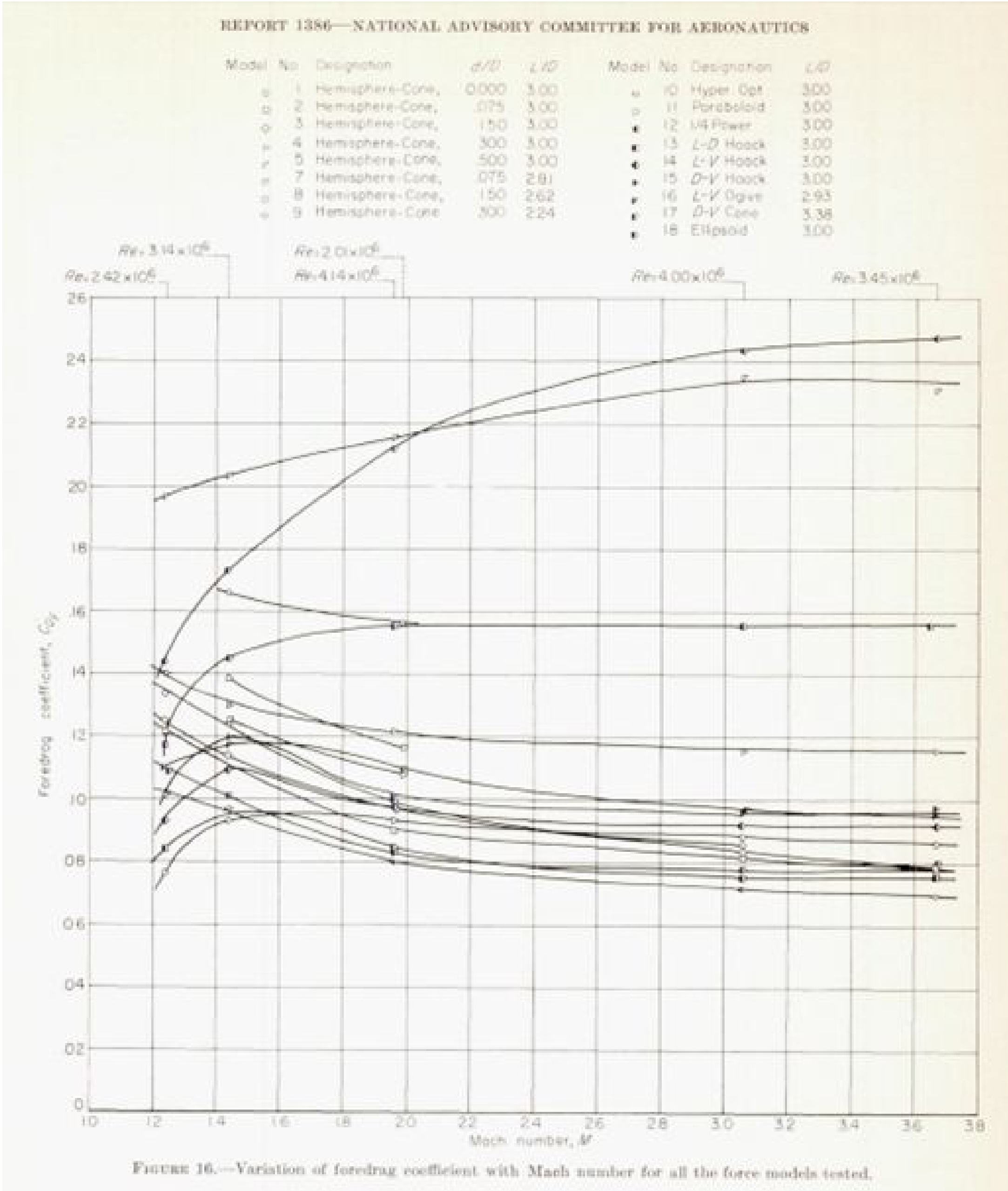


FIGURE 16.—Variation of foredrag coefficient with Mach number for all the force models tested.

[5]

Other design considerations include the fineness ratio of the nose cone and blunting the tip of the nose cone. We will be using a nose cone with a fineness ratio (ratio of length to base diameter) of 5.5, since increasing the fineness ratio past 5.5 does not yield as significant a benefit as increasing fineness ratio up to 5.5, as shown in the graph below[6][7].

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[5] Perkins, E. W., Jorgenson, L. H., and Sommer, S. C., *Investigation of the Drag of Various Axially Symmetric Nose Shapes of Fineness Ratio 3 for Mach Numbers from 1.24 to 7.4*.

[6] Crowell, G. A., “The Descriptive Geometry of Nose Cones,” 1996.

[7] Alvero, V., and Toft, H. O., “Deciding Which Nose Cone Shape To Use For A High-Altitude Rocket,” *Peak of Flight*, Oct. 2014.

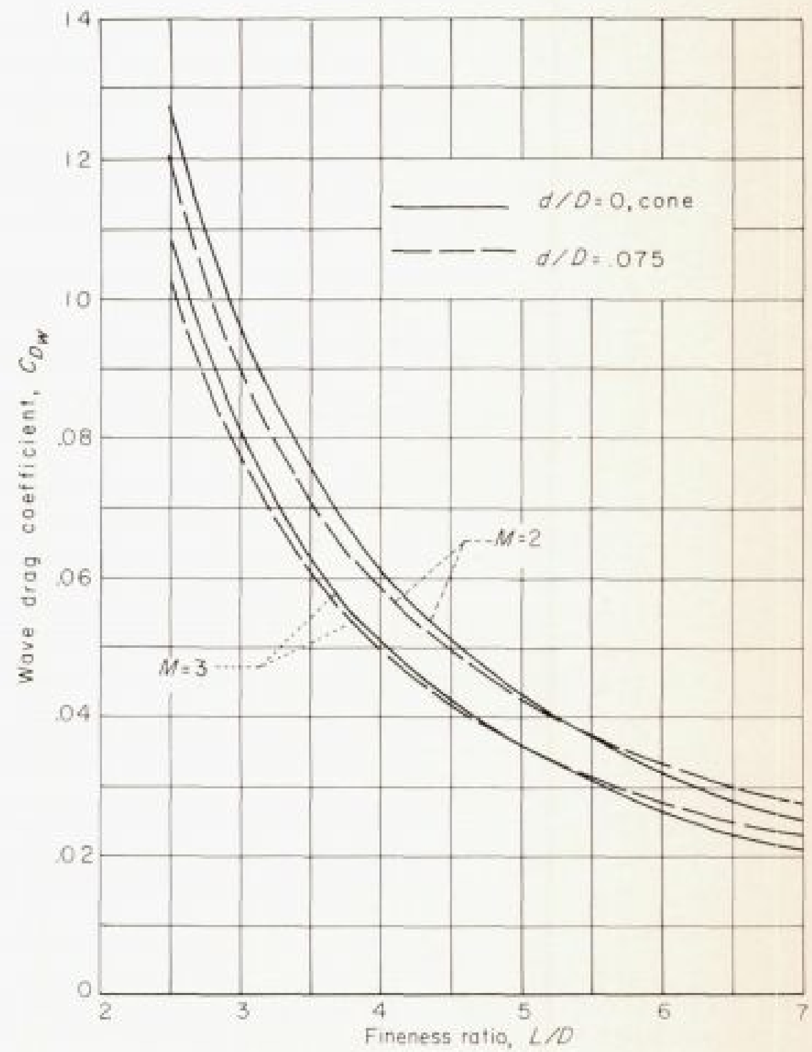

FIGURE 9.—Comparison of the estimated wave drag variation with fineness ratio for a sharp-pointed cone and a 7.5-percent hemispherically blunted cone at Mach numbers 2 and 3.

[8]

For lower fineness ratio nose cones, up to a fineness ratio of about 5.25, blunting the nose cone tip can decrease the drag by increasing the drag at the tip, over a small area, while decreasing the drag along the sides. Since there is not a significant difference in wave drag above a fineness ratio of 4.5, we will not make the nose cone more complicated by blunting the tip. We may consider slightly blunting the nose cone (approximately 0.5" diameter hemisphere for a 6" diameter base) if we choose to make a shorter nose cone in the future to save weight[9][10].

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[8] Perkins, E. W., Jorgenson, L. H., and Sommer, S. C., *Investigation of the Drag of Various Axially Symmetric Nose Shapes of Fineness Ratio 3 for Mach Numbers from 1.24 to 7.4*.

[9] Perkins, E. W., Jorgenson, L. H., and Sommer, S. C., *Investigation of the Drag of Various Axially Symmetric Nose Shapes of Fineness Ratio 3 for Mach Numbers from 1.24 to 7.4*.

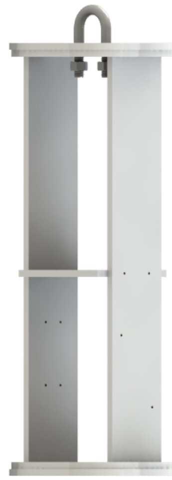
3.3.5. Nose Cone Layup Process

The team used computer-aided design to create a model of the nose cone. This design was cut out of plywood on a ShopBot to create the parting board, and the commercial fiberglass nose cone was fit into the parting board. The nose cone was then taped into the parting board, allowing half to protrude above the parting board. The holes between the parting board and cone were sealed with clay to ensure that layers of epoxy and gel coat did not leak through the cracks during the layup. After sealing the cracks, a layer of wax was applied to the parting board and nose cone to keep the epoxy from sticking to the wood. A thin layer of gel coat was then applied over the waxed cone and allowed to cure overnight to create an easily sanded surface and improve the mold's surface finish. After curing, the surface finish was sanded to prepare the surface for a layup. Five plies of fiberglass were applied to the top half of the waxed cone and parting board and allowed to cure to create one half of the mold. This process was repeated on the other half of the nose cone: the team waxed the nose cone and first mold, applied a thin coat of gel coat to the nose cone, and made another fiberglass mold for the other half. Since this procedure created small errors in the mold layup, the team will route an MDF mold to better control the nose cone shape in the future. The mold will be sanded, sealed, and waxed to prepare it for a layup. The layup process with the MDF mold will be the same as the fiberglass mold.

After creating the mold, the team could do the first nose cone layup. Each side of the mold was covered with gel coat and allowed to cure. Then the team laid up a five-ply nose cone inside each half of the mold. The team decided upon five layers to create a more durable nose cone than the earlier test nose cones. The team also plans on conducting tests to determine the optimal number of plies for the nosecone. The two halves of the nose cone were then bolted together, and another two layers of fiberglass were added to the seams inside of the cone to seal the halves together. Once dry, the bolts were removed and the nose cone was sanded.

[10] Seiff, A., and Sandahl, C. A., *The Effect of Nose Shape on the Drag of Bodies of Revolution at Zero Angle of Attack*.

3.3.6. Avionics Bay Sled Design



Avionics Bay Sled

The avionics bay consists of two sets of stepped bulkheads designed to seal the avionics bay from recovery gasses, two avionics sleds connecting the bulkheads, and a shelf as a safety measure to separate the CO2 canister from the avionics equipment. The inner bulkheads have two sockets, one for each avionics sled. The sockets will increase the security of the sleds within the bay and decrease any movement of delicate components which might otherwise occur due to vibrations of the rocket. Two bolts go through this assembly and secure the sleds in place. The sleds are placed far enough apart to accommodate a CO2 canister necessary for the recovery system, and an access hole was included in the recovery/avionics bulkhead for this canister. The avionics bay will be constructed of 0.25" polycarbonate because it is light, strong, nonconductive, and easy to machine. More research can be done into decreasing the weight of the avionics bay structure without significantly decreasing its structural integrity.

3.3.7. Bulkhead Design

The non-avionics bulkheads will be designed similarly to last year's design – a ½" thick plywood bulkhead with two layers of fiberglass on either side as reinforcement. Avionics bulkheads will be made from two pieces of 0.25" polycarbonate. Recovery hardware will include 3/8-16 U-bolts, with a maximum strength of 1075 lbs, which is expected to have a shock load of at most 1000 lbs. The flexural yield strength is:

$$\sigma = \frac{3F(L - L_i)}{2bd^2}$$

The maximum force is 1000lbs, the length of the support (L) is 6", ID of the U-bolt (Li) is 1", the width (b) is also 6", and the thickness (d) is 0.5".

$$\sigma = \frac{3 * 1000 * (6 - 1)}{2 * 6 * 0.5^2} = 5000\,psi$$

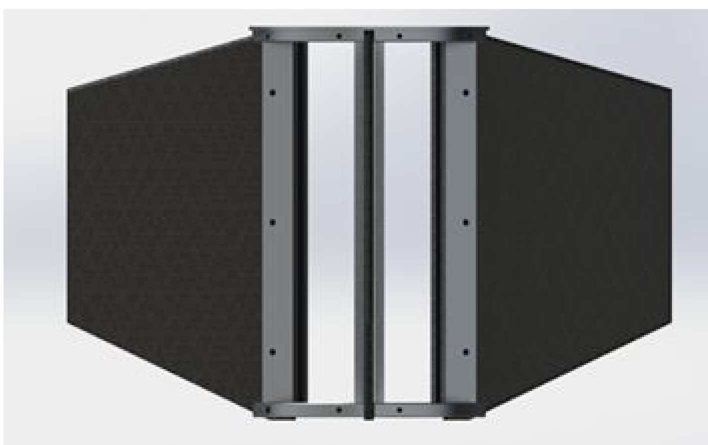
Polycarbonate has a flexural yield strength of 10400-15000 psi, and plywood has a modulus of rupture of 7000-10000 psi. The expected flexural strength required is 5000 psi, thus the polycarbonate and plywood without fiberglass reinforcement will be sufficient to handle deployment loads. The team will perform tests to confirm that the bulkheads are sufficiently attached to the tubes to not shear out of the body tubes.

The avionics bulkheads will have a step to fit over the avionics bay coupler like a lid. ¼-20 screws will screw into the avionics bay to keep the bulkheads in place. The payload/recovery bulkhead will sit between the top of the payload/avionics coupler and small tabs under the payload bay, so that the recovery system can easily eject the bulkhead when the drogue parachute is ejected.

3.3.8. Tube Attachment Mechanism

Tubes will be attached with three 8-32 screws between the tube and the coupler, placed 120 degrees apart. To ensure only one orientation of the tubes, one hole will be placed 15 degrees off of the desired location. The 8-32's will screw into a nut plate, which is attached to the tube with rivets to create a better mechanical attachment than epoxy. The shear strength of the nut plates in a generic fiberglass composite is 9000 ksi (about 60 gPa), and the area of each of the six 0.125" rivets fastening the nut plates is 0.0122 sq in. The shear strength of the tubes is 219,600 lbs, which significantly exceeds expected recovery loads.

3.3.9. Fin Can Design

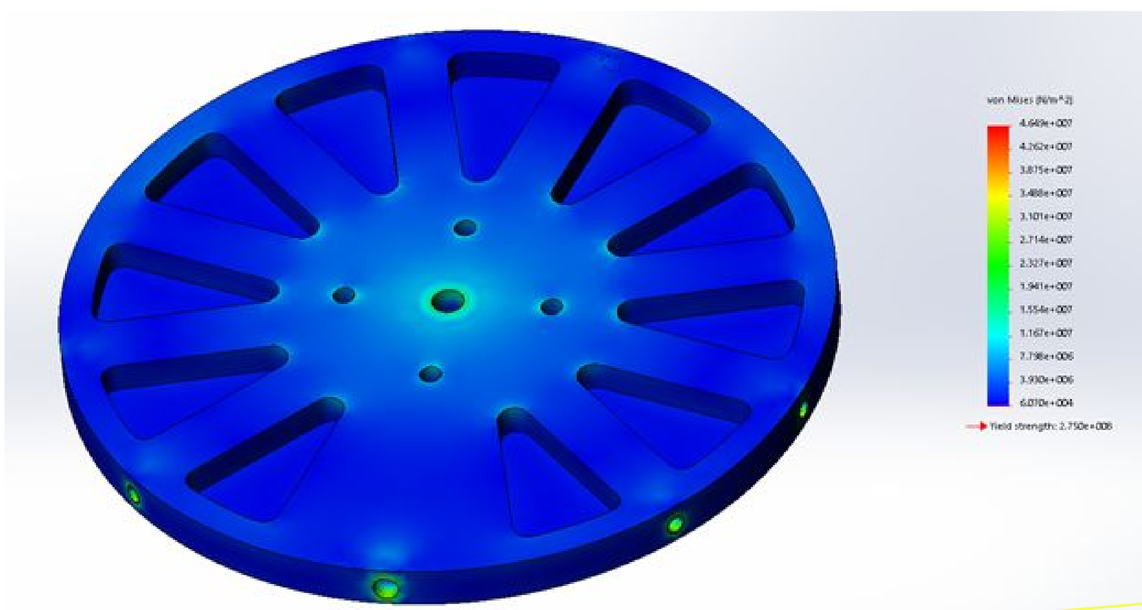


Fin Can Design

This year's fin can design was based off last year's design, which gives the team confidence that the fin can architecture will be strong enough to handle all of the predicted loads. The lower section of the fin can consists of a 0.4" thick ring of 6061-T6 aluminum to serve as a thrust plate, which transfers the thrust load from the motor to the rocket's airframe. There is a 0.375" thick aluminum centering ring 10" above the thrust plate, with fin supports connecting the two rings. This assembly will be welded together to eliminate the need for a motor tube, and to facilitate the fin can assembly. The fin supports will allow the team to change fins without creating a new fin can, since the fins are the most delicate part of the structure. Interchangeable fins also affords us the option to fulfill the “re-flyable” requirement of the competition by replacing any fins that may break on landing. The centering ring above the fin supports will hold a rail button, and thus needs to be 3/8” thick to hold the ¼-20 bolt.

Fin Can Structure

The motor retention plate is 36” above the thrust plate. A threaded rod will bolt into the motor casing and both a nut and washer will rest on the motor retention plate. Stress analysis in SolidWorks, pictured below, shows that the current design of the motor retention plate will be able to support more than the predicted loads due to the motor and casing. The maximum Von Mises stress is about 2.714e+07 N/m^2, compared to the yield stress of 2.750e+08 N/m^2, for an axial weight of 100lbs. The plate is 3/8” 6061-T6 aluminum, which is thick enough to place the second rail button 16” above the lower rail button.



Motor Retention Plate SolidWorks Stress Analysis

Guest User: Unfortunately this figure is not clear enough to read the colorbar on the right.

Both the motor retention plate and the centering ring will have ¼-20 rail buttons for a 15-15 rail. In order to meet specifications, the rail buttons must be able to hold the weight of the rocket. The aluminum has a lower shear strength than the 18-8 stainless steel screws used to attach the rail buttons to the rocket, so the length of engagement required to cause the screw to snap instead of strip can be calculated as follows:

$$Tensile\ strength\ of\ screw\ (lbs) = tensile\ stress\ area * screw\ tensile\ stress$$

$$= 0.0318 * 70000$$

$$= 2226\ lbs$$

$$Internal\ thread\ shear\ strength/in$$
$$= internal\ thread\ shear\ area/in * \ internal\ thread\ shear\ strength$$

$$= 0.539 * 30000$$

$$= 16170\ PSI/in$$

$$Length\ of\ engagement\ needed\ to\ avoid\ internal\ thread\ stripping$$
$$= screw\ tensile\ strength\ /\ internal\ thread\ shear\ strength\ per\ inch$$

$$= \frac{2226}{16170}$$

$$= 0.1377\ in$$

Since the screws will be tapped to 0.5 in, we know that the screw will fail under tensile loading instead of shear loading.

To determine if the rocket can be held by two rail buttons, the maximum force that a ¼-20 in aluminum can handle was calculated.

D_{s_min} = minimum major diameter of external threads = 0.2419"

E_{n_max} = maximum pitch diameter of internal threads = 0.2175"

$N = thread \frac{pitch}{in} = 20\ TPI$

L_e = length of thread engagement = 0.5"

Shear strength of 6061 − T6 Al = 30,000 PSI

ATS = cross − sectional area through which shear occurs

$$= \pi * n * L_e * D_{s\_min}\left[\frac{1}{2*n} + 0.57735\left(D_{s\_min} - E_{n\_max}\right)\right]$$

$$= \pi * 20 * 0.5 * 0.2419\left[\frac{1}{2*20} + 0.57735(0.2419 - 0.2175)\right]$$

$$= 0.297\ in^2$$

F = Shear strength $*$ ATS

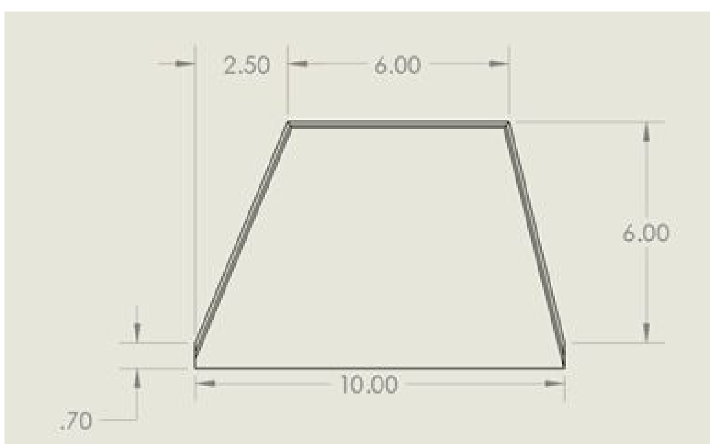
$$= 30000 * 0.297$$

$$= 8910\ lbs\ per\ rail\ button$$

Since the wet mass of the rocket is near 100 lbs, two rail buttons will provide plenty of support for the rocket.

3.3.10. Fin Design

Multiple fin designs were tested in OpenRocket with a model of this year's rocket to determine which one would best keep the rocket stable but not overstable during ascent. The fins are trapezoidal so that the tips do not extend past the bottom of the rocket and break on landing. The height of the fins is also minimized to decrease chances of fins breaking upon landing, since the forces on the fin tip will have a smaller moment arm.



Fin Design

The fins will be a sandwich panel construction of two-ply carbon fiber over foam. The foam fins will decrease the moment of inertia, and the carbon fiber is far enough from avionics as to not cause significant problems with the radio antenna. To attach the fins to the supports in the aluminum fin can structure, the team will be epoxying three carbon fiber tubes into the foam fin. These tubes will have an ID of 0.176” to accommodate 8-32 screws through the fins. One main concern is that the shear stress on the foam due to drag could cause the fins to rip out of the rocket at max Q. The shear strength of the DOW STYROFOAM Panel Core 30 Extruded Polystyrene Foam Insulation that we are using for the fins is 35 PSI or 241 KPa. The calculations for the drag of each fin is as follows:

$$v_\infty = maximum\ velocity\ (\max Q) = 328\,\frac{m}{s}$$

$$L = length = 10in = 0.254m$$

$$\nu = fluid\ kinematic\ viscosity\ at\ 100F\ = 1.79 * 10^{-4}\frac{ft}{s} = 5.45592 * 10^{-5}\frac{m}{s}$$

$$Re = Reynold's\ number = v_\infty L/\nu$$

$$= 1.527 * 10^6$$

$$\rho = density\ of\ air\ at\ sea\ level = 1.225\ kg/m^3$$

The range for transition to turbulent is a Reynold’s number between 500,000 and 3,000,000. To overestimate the drag, turbulent flow calculations were used.

$$C_D = \frac{0.031}{Re^{\frac{1}{7}}} = 0.004$$

$$A_{fin} = \frac{chord_{root}+chord_{tip}}{2} * height * 2\ sides = 92\ in^2 = 0.059m^2$$

$$A_{force} = 3\ screws * \frac{1}{2}Surface\ Area = 3\left(\frac{1}{2}\right)(\pi D) = 0.207in^2 = 0.000135m^2$$

$$F_D = \left(\frac{1}{2}\right)\rho v_\infty^2 C_D A_{fin} = 15.52\ N$$

$$Shear\ stress = \frac{F_D}{A_{force}} = 114.962\ kPa$$

Since the shear stress is 114.962 kPa, compared to the 241 kPa of the foam, this affords a safety factor of just over 2. The carbon fiber overwrap will increase the shear strength of the fins, affording a greater safety factor.

3.4. Key Technical Issues/Risk

The diagram below is a stoplight diagram of risks. Green risks are deemed “acceptable,” whereas red risks must be mitigated prior to flight. The table outlines the risks, probability, and impact on project. To minimize the risks, the structures team created a risk reduction plan for each risk.

Risk	20%					
	10%					
	5%		1		5	3
	1%			7	8	
	0.1%			2		4, 6
		1	2	3	4	5
Structures		Impact				

	Risk	Risk Reduction Plan
1.	High Crosswinds on Launch Site	Wait for a less windy day to launch
2.	Rocket unstable or overstable	Adjust CP (center of pressure) and CG (center of gravity) to keep rocket stable. Perform stability calculations in OpenRocket.
3.	Separation Failure	Test that all couplers fit without sticking under flight-accurate transverse loads (with safety margin) and that shear pins sized properly for charge size. These tests will reduce the risk to 0.1%.
4.	Motor Mount Failure	Test all mounts in a test fire first to be sure they can handle the full motor sizing.
5.	Delamination or Other Damage to Fiberglass	Test fiberglass under in-flight loads, and handle tubes with caution to avoid damaging the structure before flight
6.	Fins Detach during Launch	Load test fins to ensure they can handle launch loads, run CFD (Computational Fluid Dynamics) simulations to determine expected shock loads
7.	Rocket over Mass Budget	Make sure that subteams are constantly updating their sections in the BOM (Bill of Materials), and design to reduce weight

8.	Recovery loads pull out bulkheads	Previously tested bulkhead under similar conditions; run calculations based on recovery loads

3.5. Interfaces

Structures interfaces with other subteams are listed in the table below:

Interface:	**Interfacing Subteam:**	**Description**
Bulkhead location and design	Payload, Recovery	The payload and recovery team will coordinate with the structures team to ensure optimal placement of bulkheads within the rocket.
Material of rocket tubes	Avionics	Structures and avionics coordinated to make a fiberglass rocket to ensure that the avionics antennas will get signal.
Rocket Length	Payload, Recovery, Avionics, Propulsion	The structures team will decide the rocket length based in part on the space needs of other teams.
Rocket Diameter	Propulsion, Payload	The rocket diameter is larger than necessary for the propulsion team to determine if the current motor is the appropriate size for the team's goal. The rocket diameter fits the cubesat payload requirement.
Mass Budget	Payload, Recovery, Avionics, Propulsion	The structures team will estimate the specific mass budget based on each team's needs and last year's approximate masses. The total dry mass will be needed for propulsion team to determine if the motor is appropriately sized while being able to provide the desired velocity off the rail and enough impulse to reach the target altitude.
Thrust profile	Propulsion	The thrust-time curve of the motor is needed from the propulsion team to design a structure able to withstand the body force produced by the motor.
Avionics Bay	Avionics	Avionics and structures are coordinating to design an avionics bay to ensure that it properly houses avionics and will be structurally sound.

3.6. Going Forward Plan

Structures will be finishing the first test rocket, primarily based on last year's design, for a test launch in late November or early December. At that point, structures will switch focus to research and development of improved composite tubes. Structures will spend the rest of the semester building the oven and creating test tubes. During the January term (IAP), the structures team will test the new tube process to determine if it can handle launch loads, and if the more expensive S-glass provides a significant increase in strength for its cost. Based on the results of the tests, structures will build multiple rockets in the spring term for flight tests. These flight tests will validate the earlier research that the structure can withstand flight loads. This schedule allows for the possibility of multiple losses of vehicle, with Raziel 4 as the competition rocket.

	December	January	February	March	April	May
Composites Oven						
Test Plans						
Structural Testing						
Build Raziel 2						
Test Raziel 2						
Build Raziel 3						
Test Raziel 3						
Build Raziel 4						

4. Recovery

4.1. Overview of Requirements

The Recovery Subteam began the design process by outlining internal and SAC requirements (Table 4.1.1) to ensure that the design met both sets of requirements. These requirements can also be found in Appendix A.

Table 4.1.1: Internal and SAC requirements

Internal Requirement	SAC Requirement	Description
4.0	RRD 2.8.1.4	Rocket shall not be “excessively” damaged during recovery (i.e. could be launched again safely with consumables replaced).
4.1	DTEG 3.1	Rocket shall follow a dual-event CONOPS.
4.1.1	DTEG 3.1.1.1	The initial deployment event shall occur at apogee and significantly lower the descent velocity.
4.1.2	DTEG 3.1.1.2	The main deployment event shall occur at an altitude no higher than 1500 feet AGL, and decrease the descent velocity to less than 30 feet/second.
4.1.3	DTEG 3.1.3	Drogue and main parachutes shall have dramatic color differences.
4.2	R2T	Recovery should use non-pyrotechnic methods to initiate and complete all deployment events.
4.2.1	DTEG 3.1.2	Recovery shall protect cords, parachutes and other vital components from any hot gases.
4.3	RRD 2.5	All independent sections of the rocket shall carry a radio beacon or similar transmitter aboard.
4.4	DTEG 3.6.1	The recovery system shall undergo a functional ground test prior to SAC, replicating flight conditions for sensors.
4.4.1	P3B	Recovery shall have a successful flight test prior to competition.

4.5	DTEG 4.1	Any energetics used by recovery shall be “safed” until the rocket is in the launch position.
4.5.1	DTEG 4.2.2 & 4.2.4.1	Any non-COTS pressure vessels shall be designed to withstand twice their rated pressure. They shall be tested to 1.5 the maximum operating pressure, and contain a relief device set to open at no greater than the proof pressure.

4.2. Design Process

4.2.1. Conception of CO2 Recovery System

Previous years’ projects have seen the use of black powder to separate sections of the rocket and reveal the parachutes. This year it became desirable to use non-pyrotechnic mechanisms for Raziel’s recovery. This decision was based primarily on the team’s desire to reach higher altitudes in upcoming years. At approximately 20,000ft, simple black powder-initiated systems become significantly less reliable due to the decreased atmospheric pressure and temperature at which the reaction takes place.[11] Thus, it is practical for the team to set a precedence in non-pyrotechnic recovery prior to its necessity.

Guest User: Black Powder and Surgical tube ejection charges can also work above 20k ft, the team may want to look into them as an alternative.

We chose to pursue a CO_2 system, noting both its popularity within the rocketry community and its functional similarity to black powder systems. In Table 4.2.1 below we have compiled a list of advantages and disadvantages between CO_2 and pyrotechnic recovery in order to highlight the complexity of this decision.

Table 4.2.1: Pro-Con List Between Pyrotechnic and CO_2 Recovery

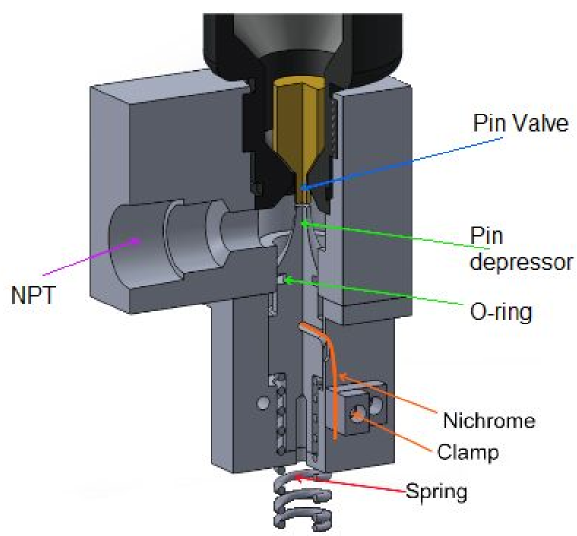
Purely Pyrotechnic Recovery System		**CO_2 System**	
Pros	**Cons**	**Pros**	**Cons**
Team has institutional knowledge—black powder is a hobby standard	Decreased effectiveness at high altitudes	Effective at high altitudes	Lack of institutional knowledge—requires new skills

[11] “CD3 Jupiter Kit (28g, 38g),” Apogee Components, https://www.apogeerockets.com/index.php?main_page=product_info&cPath=198_133&products_id=266&zenid=0c56ac81655bfef8824cf201d813a811, [November 11, 2016].

Low cost	Large room for human error—charges can be packaged incorrectly	Less room for human error, especially when using commercially filled canisters	Expensive—commercial systems cost several hundred dollars and sourcing individual components is likewise costly
Mass efficient and energy dense	Leaves residue within the rocket that can potentially damage sensitive avionics and/or payloads	Increased ability to test multiple times and yield consistent results	Adds extra weight and occupies extra space

Initially, the Recovery Subteam planned to design and fabricate a CO_2 system exclusively in-house. On a high level, CO_2 systems operate in the same way as their pyrotechnic counterparts: gas is released into a chamber adjoining two parts of the rocket and the resulting pressure breaks the shear pins connecting the tubes.

To determine the amount of CO_2 required to pressurize the recovery bay and eject the payload and parachutes, we needed to know how much force is required to break a shear pin. The team performed a test (further described in Section 4.6) and found that approximately 33.1 pounds would necessary per shear pin used. Calculations were also performed to determine how much CO_2 would be required to apply this force on the recovery-payload bulkhead. A custom valve was designed with these constraints in mind.



Following a setback requiring a redesign of the custom valve, the Recovery Subteam decided to purchase a custom off the shelf (COTS) CO_2 system, the Rouse-Tech CD3 system. Switching to this system re-introduced pyrotechnics into the Recovery CONOPS because a small black powder charge is used to drive a piston into a pressurized CO_2 canister. Fortunately, this charge is contained within a small, sealed chamber, making the mechanism viable even at high altitudes. The COTS system will be used as-purchased for flight tests in 2016, and custom revisions will be made in 2017 (further details on revision plans can be found in Section 4.6). The CD3 system will be discussed in further detail within the Technical Design/Analysis section (Section 4.3) to follow.

4.2.2. Recovery-payload interface

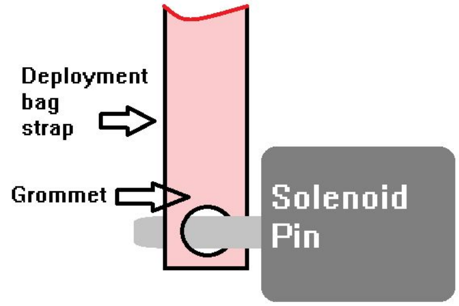
During the design process, Recovery was also faced with whether to develop a single or dual separation system. We ultimately chose to separate the rocket from a single opening between the recovery and payload tubes, releasing the drogue and main chutes from the same chamber.

Because the pneumatic force from the CO_2 would likely be insufficient to push the drogue chute out by itself, various options for drogue ejection were discussed. We initially planned to connect a deployment bag to the recovery-payload bulkhead. Upon ejection of the payload tube at apogee, the drogue would be pulled out of this deployment bag. Unfortunately, this plan was complicated by the necessity that the payload tube remain unconstrained by a bulkhead (to allow for easy rover deployment). As a result, the team decided to use a pressure bulkhead that is permanently fixed only to the drogue chute and not to either tube or the coupler. The force on the bulkhead from the CO_2 will eject the payload tube and, as the bulkhead travels away from the rocket, will also draw out the drogue chute. This system will be explained in further detail within Section 4.3.

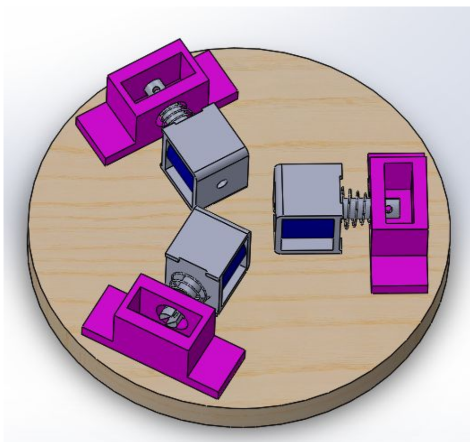
4.2.3. Main parachute retention system

Single separation presented another challenge to Raziel's recovery system: we had to design a mechanism to prevent the main chute from being prematurely drawn out of the tube by the drogue. Various methods to prevent premature inflation were thus discussed. The primary method investigated to date has been to hold the main parachute's deployment bag using a set of solenoid pins.

The solenoid pins would go through the grommets of the deployment bag, keeping the bag closed and the main parachute uninflated. At the desired altitude, Raziel's flight computer would apply a current to the solenoid pins, retracting them and thus allowing the deployment bag to open and deploy the main parachute.



We developed a preliminary CAD model and prototype using three solenoid pins to hold down the deployment bag. The decision to use three pins was largely arbitrary and an attempt to strike a qualitative balance between fully constraining the deployment bag while also limiting the number of possible hardware failures.




Preliminary testing of this system revealed that the strength of the magnetic field within the solenoids was not strong enough to overcome binding of the pins caused by applying an upwards force. We machined a new pin with a smaller tolerance in the hopes that a more snug fit between the pin and the solenoid would counteract this binding force. Unfortunately, these efforts were unsuccessful, prompting the Recovery Subteam to begin pursuing a nichrome wire pin mechanism (described in further detail in Section 4.3).

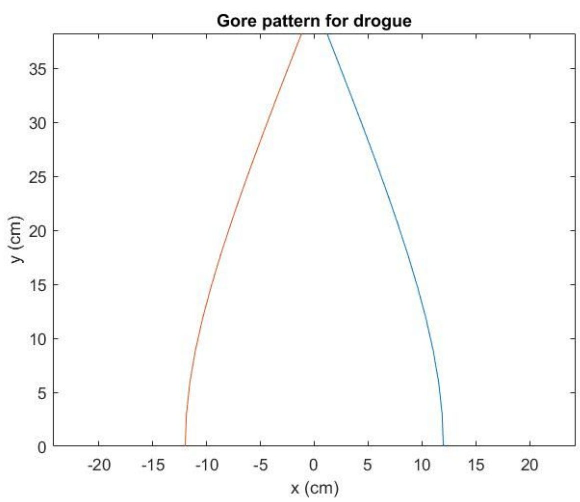
4.3. Technical Design/Analysis

Project Raziel will use a dual event parachute recovery system. This means that Raziel will carry two parachutes: a 2 foot drogue deployed at an altitude of 10,000 feet that constraints the rocket's descent velocity to around 95 feet per second and a 9.5 foot main parachute deployed at 1,500 feet that slows the rocket to the final descent velocity of 20 feet per second. Since these parachutes will be housed in the recovery bay, a commercial CO_2 system will be used to pressurize the bay and separate the rocket along the recovery-payload interface, deploying the drogue parachute. The main parachute will be held down by a steel pin secured with a nichrome wire until the rocket falls to 1,500 feet. Each of these systems is described separately in the following subsections.

4.3.1. Parachutes

Project Raziel will use a semi-ellipsoidal design for both the drogue and main parachutes to achieve desirable drag values, limit fabric usage, and conserve packing volume[12]. The sizes of the parachutes were chosen to bring the rocket's descent rates within limits recommended by the SAC guidelines. The 2 foot drogue will be constructed from 8 panels of equal size while the 9.5 foot main parachute will be composed of 16 equal panels. These values are based on institutional knowledge and present a balance between a limited number of components and a manageable component size.

To determine panel shape, the Team designed and executed a Matlab script to plot the shape of a panel given certain parameters (major radius, ratio between radii, and number of panels). By upscaling the resulting Matlab plot to a full scale curve on paper, the panels may be cut from fabric and sewn together to create the desired parachute.



4.3.2. Separation mechanism

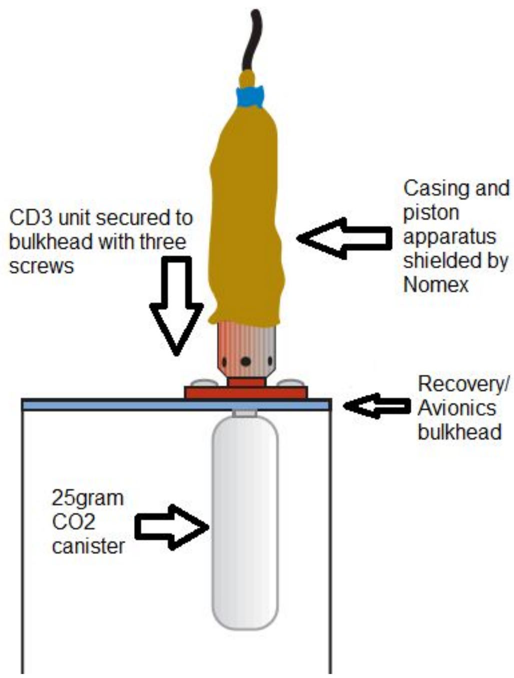
Raziel will use CO_2 pressurization to separate the recovery and payload bays and therefore release the drogue parachute. For the 2016 flight tests, the recovery team will be employing the [Rouse-Tech CD3 High Power Rocketry CO_2 Ejection Kit][13]. This kit uses CO_2 canisters and a pyrotechnic charge to puncture a compressed gas canister with a piston. The resulting pressure in the recovery bay will break apart three nylon shear pins holding together the recovery and payload bays. To determine the size of CO_2 cartridge required to pressurize the 6" diameter, 36"

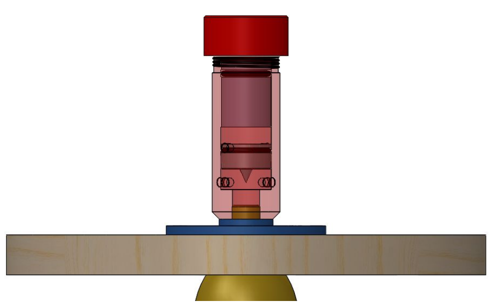
[12] Nakka, R., "Parachute Design and Construction," Richard Nakka's Experimental Rocketry Web Site, http://nakka-rocketry.net/paracon.html, January 8, 2011.

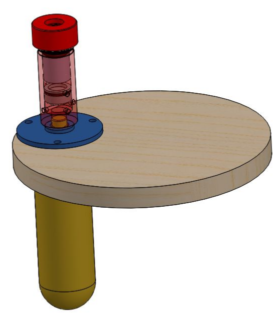
[13] ttps://www.apogeerockets.com/index.php?main_page=product_info&cPath=198_133&products_id=267

height recovery bay, the team followed the manufacturer's recommendations[14] and selected a single 25g canister. Ground testing the rocket will allow the recovery team to verify and, if necessary, modify the amount of gas required.

Featured below are the manufacturer's diagram (with modified annotations)[15] as well as two Team CAD renderings of the CD3 system:







Additionally, the ignition of the pyrotechnic charge will be controlled by the Pyxida flight computer in the avionics bay, so there will be a wired connection between the charge and the avionics bay.

[14] Rockdale, J., "Sizing Guide," CD3 Instruction Manual, https://www.apogeerockets.com/downloads/PDFs/CD3_Manual2009.pdf, [November 11, 2016].
[15] Rouse-Tech, CD3 Instruction Manual, https://www.apogeerockets.com/downloads/PDFs/CD3_Manual2009.pdf, [November 11, 2016].

4.3.3. Release mechanism – Nichrome pin release

The main parachute must be restrained until the rocket falls to 1,500 feet. To accomplish this, the grommets on the deployment bag will be attached to a steel pin on a nichrome wire release mechanism**.** A section of the pin will be surrounded by a spring that will be held under compression by a nichrome wire. After an electrical current flows through the nichrome, the wire will deform and release the pin, thus freeing the deployment bag and the main parachute with it.

The pin and the nichrome wire will be housed in a teflon structure that will be bolted to the bottom bulkhead in the recovery bay. This structure and the pin must withstand the loads exerted by the drogue during descent.

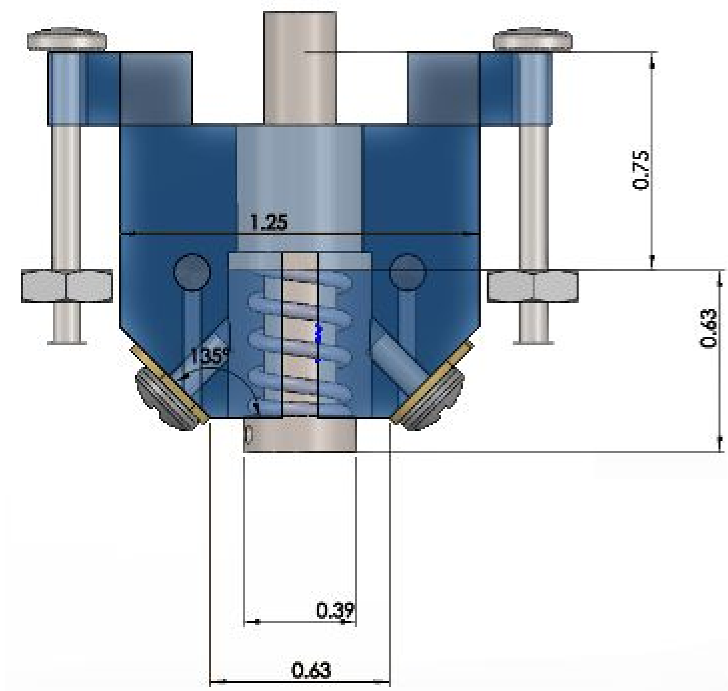


Figure 4.3.3.1: Nichrome wire pin release mechanism. The nichrome wire will be connected to the conductive plates under the screws and will hold down the center pin by compressing the spring. Modifications will be made to the casing of the system to attach it to a bulkhead and to the length of the pin to extend it through the deployment bag grommets. Units are in inches.

4.4. Key Technical Issues/Risk

Throughout the design process, the Team identified a list of risks in the Recovery System that may disturb or, in the worst case, preclude project success. Each of these risks is discussed below.

1. Parachute integrity: Since the parachutes will be built by the Team, improper design or manufacturing may compromise their structural integrity. Though a flight test will ultimately validate the integrity of the parachutes, a ground test and an inflation test will be conducted beforehand to identify visible deformations on the main parachute and verify that the drogue can withstand deployment loads.

2. Pin jamming: The steel pin may be unable to slide after the nichrome wire releases due to the perpendicular stresses from the deployment bag, which is being pulled by the drogue. This risk can be mitigated by using a spring of enough stiffness to push the pin. Following the fabrication of a prototype, we can better asses the risks associated with this failure. Until then, this failure has an unknown probability and high consequence.

3. Pyrotechnic charge malfunction: Improper handling and preparation of the commercial CO_2 ejection system may cause the piston to fail, thus inhibiting parachute deployment. Because the CD3 Recovery System is a commercial product and has a history of successful use in amateur rocketry, any risk of malfunction is minimized if the commercial system is not physically modified or used beyond its intended design.

4. Improper Pressurization: Containing the pressure within the recovery bay is essential to a successful recovery. An anomaly in the pressurization of the recovery bay may result in one of the following failure modes:

> **4a. Drogue deployment failure:** If there is low pressure in the recovery bay, there may be insufficient force on the shear pins that hold together the recovery and payload tubes, so these sections may not separate.
>
> **4b. Avionics bay pressurization**: Since the recovery bay is adjacent to the avionics bay, an unsealed interface between the two sections could result in undesired pressurization of the avionics bay. This would affect altitude calculation and, in the worst case, may cause premature deployment of the main parachute. The team will preemptively seal wire connections with epoxy resin, but any other leaks will be assessed with a ground test. If the problem is serious, the flight computer can also be programmed to ignore changes in temperature for around three seconds before and after apogee, allowing for the pressure to equalize with outside air.

5. Payload rail failure: If the payload tube's rails are not properly secured, the pressure bulkhead (described in Section 4.5.2) may not be able to apply sufficient force to separate the two tube sections. The rocket would descend in free-fall without a drogue or main parachute. Although this failure is currently high risk, ground testing can verify the design and fabrication methods.

	20%					5

Risk	10%				1	2, 3, 4a
	5%		4b			
	1%					
	0.1%					
		1	2	3	4	5
		Impact				
		Mission not compromised or minor loss of data	Damage to subsystem, substantial loss of data	Loss of subsystem, loss of critical data	Total loss of vehicle, miss major milestone due to schedule slip, loss of all data	Loss of mission, injury to team members, external parties adversely affected

Figure 4.4.1: Risk Table for the recovery subsystem. Each risk is labeled with the number used in the Key Technical Issue/Risk subsection. The Team expects that the risk percentage will be reduced with ground and flight testing of the system, but the impact estimates should be representative of the final values.

Guest User: Figure labels appeared!!

4.5. Interfaces

4.5.1. Summary of interfaces

The following chart summarizes the interfaces between the Payload, Avionics, Structures, and Recovery Subteams. Recovery does not significantly interact with any other subteams.

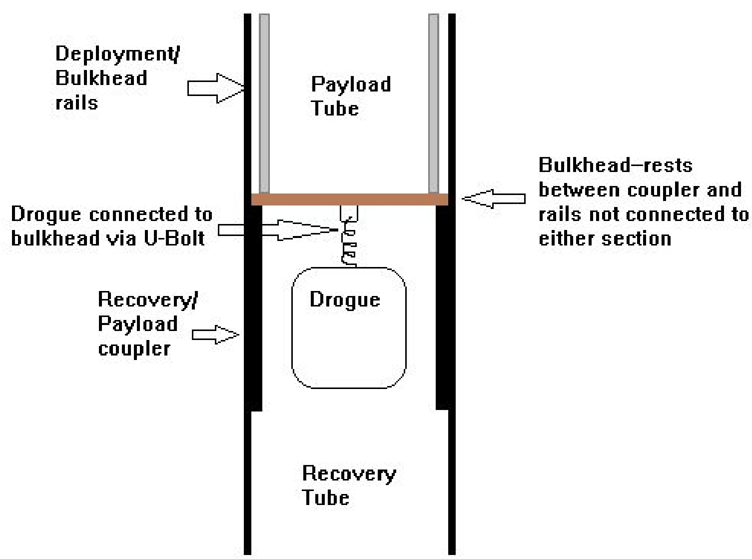
Table 4.5.1.1: List of Interfaces Between Recovery and Other Subteams

Output	**Source**	**Recipient**	**Description**
Event Initiation Times	Recovery	Avionics	Recovery will provide Avionics with the appropriate initiation times so that Avionics can program the flight computers accordingly.
Event Initiation	Avionics	Recovery	Avionics flight computers will trigger initiation events at the appropriate time during the rocket's descent.
Specific Mass Budget	Recovery	Propulsion	Specific mass budget must be maintained to ensure proper propulsive power.
Specific Volume	Recovery	Structures	Specific packing volume must be maintained by recovery to ensure that the recovery system will fit in the rocket.

Sealed Bulkhead	Payload	Recovery	A bulkhead is required between recovery and payload to ensure proper chamber pressurization for CO_2 ejection.
Removable Bulkhead	Recovery	Payload	The bulkhead between payload and recovery must not be rigidly connected in order for the rover to deploy through the payload tube.
CO_2 Canister Access	Recovery	Avionics, Structures	Avionics will provide Recovery with easy access to CO_2 canisters through the avionics bay.

4.5.2. Interface with Payload

As is referenced in Section 4.2.2 and elements 5 and 6 of Table 4.5.1.1, the recovery chamber must be appropriately pressurized while still leaving the payload tube ultimately unconstrained by a bulkhead. In order to fulfill the requirements of both the payload and recovery systems, we will be using a “pressure” bulkhead between the two sections. This bulkhead is connected to the main rocket via the drogue chute. In its integrated form, the bulkhead rests on top of the recovery/payload coupler and below a set of rails used for rover deployment.



When the CO_2 is released into the recovery tube at apogee, the increased pressure will push upwards on the bulkhead. The bulkhead will in turn transfer this force to payload’s rails and thus the entire payload tube. This force will break the shear pins connecting payload to recovery. As the payload tube and bulkhead are ejected away from the main rocket, the bulkhead will pull the drogue out of the recovery tube. Because the bulkhead is not rigidly connected to the payload tube, the rover can be easily deployed.

4.5.3. Interface with Avionics

As is referenced by element 7 of the Table 4.5.1.1, the team will access the CO_2 canisters from the avionics bay. This necessitates that a) the recovery-avionics bulkhead have an appropriately sized opening for canisters to screw into the piston casing and b) there be sufficient space within the avionics bay to easily access and replace canisters.

4.6. Test Plan

4.6.1. Shear pin test

The Recovery Subteam began the year by designing an experiment to measure the amount of force required to break a nylon shear pin. This test, completed in mid-September, provided information critical to designing a custom CO_2 system. Although the Team ultimately followed the recommendations of the CD3 CO_2 system manufacturer in choosing canister size, the shear pin test offered valuable information for future, custom recovery systems.

The test was designed based off of a similar test performed by the University of Alabama “Rocket Girls” Team. Two pieces of G10 fiberglass were screwed together with a 4-40 shear pin. A measured force was then applied to one end using a fish scale.

The Team found that the average load required to break a 4-40 shear pin was 33.10 pounds. Although there were several sources of error associated with this experiment,[16] the results align with institutional knowledge and the Team is confident that previously assumed information has been verified.

[16] Sources of error include (but are not limited to): not accounting for thermodynamic changes over the course of flight, screwing as opposed to pushing in the shear pins, and a limited data set.

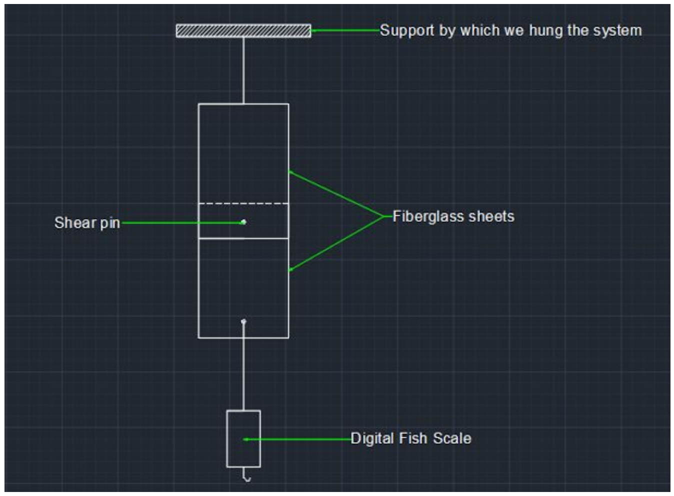


Figure 4.6.1.1: Experimental setup for shear pin stress test.

4.6.2. Future tests

Through the remainder of the 2016-2017 competition year, the Recovery Subteam will verify mechanism designs and fabrication procedures using numerous tests, detailed in Table 4.6.2.1 below. These tests will also help us converge on more realistic values for the Risk/Impact table:

Table 4.6.2.1: Test Plan

Test Name	Description of Procedure	Success Conditions	Expected Date
Parachute Construction Test	Open up the parachute in a natural or artificial wind tunnel to observe performance.	Parachutes inflate with no observable anomalies including but not limited to: tears, holes, weak seams, and tangled cords.	Spring 2017 for new chutes
Nichrome Pin Test	Apply upwards force on the nichrome pin (approximating the force from the inflated drogue) and activate the mechanism.	The nichrome wire disintegrates. The spring force is able to retract the pin, overcoming the forces of binding and friction.	November 2016
Ground Test	Horizontally separate the payload and recovery tubes using the CD3 system.	The amount of black powder used in the CD3 system is sufficient to puncture the CO_2 canister. The tubes separate and the payload-recovery bulkhead drags the drogue chute out of the rocket.	November 2016

Parachute Opening Test	Connect a dogbone strain gauge to the drogue or main parachute of a high power rocket to measure the force on the rocket caused by opening the chute.	No success conditions--this data will be used to verify mechanisms and that the structure of the rocket can withstand the opening force of an arbitrary parachute.	Late 2016
Complete Flight Test	Complete integration of all recovery-related systems for a to-scale flight test on the first iteration of Raziel.	The rocket is recovered in re-flyable conditions.	Late 2016

4.7. Going Forward Plan

Following the 2016 flight test, the Team anticipates that there will be significant modifications and improvements that can/must be made to the system. Any mission-critical modifications will serve as the focus for the Recovery Subteam during MIT's January term (IAP).

The Recovery Subteam's next focus will be on developing a more customized CO_2 system based off of the existing CD3 system. Among the anticipated improvements, the Team hopes to investigate the possibility of refillable gas canisters (for cost, consistency, and convenience) and the complete elimination of pyrotechnics from the system. The latter will likely involve a significant redesign from Rouse-Tech's commercial product and may require special attention to Internal Requirement 4.5.1 (DTEG 4.2.2 & 4.2.4.1).

Although the Recovery Subteam plans to use pre-existing chutes owned/created by the Team for the 2016 flight test, the winter and spring semesters will also see the fabrication of at least two new full sized parachutes for Project Raziel (a main and a drogue). The Recovery Subteam plans to continue using semi-ellipsoidal parachutes for the 2017 SAC due to familiarity with their fabrication and properties. In preparation for future years, however, new parachute shapes will be researched with a focus on stability and reducing shock loading. This research includes ballutes, which will be necessary at the high altitudes that the Team plans to reach in future projects.

Featured below in Table 4.7.1 is a Gantt Chart illustrating the timeline for Recovery-related projects leading up until the SAC in June.

Table 4.7.1: Recovery Subteam Gantt Chart

	Nov.	Dec.	Jan.	Feb.	Mar.	April	May
Nichrome Pin							

Fabrication and Testing							
Ground Test 1							
Flight Test 1							
Post-Flight modifications							
Custom CD3 improvements							
Parachute fabrication and research							
Ground Test 2							
Flight Test 2							
Flight Test 2 Modifications							
Ground Test 3							
Flight Test 3							
Flight Test 3 Modifications and SAC Preparation							

5. Avionics

5.1. Overview of Requirements

5.0	DTEG 3.3.1	Rocket shall have redundant electronics, at least one of which shall be a COTS flight computer.	Avionics
5.0.1	RRD 2.6	Rocket shall contain a COTS flight computer with on-board data storage for an official record of apogee for scoring.	Avionics
5.1	DTEG 3.4	All safety-critical wiring shall conform to ESRA Wiring Rules. (See DTEG Appendix B)	Avionics
5.2	DTEG 4.1.1-2	All arming features shall be externally accessible, such that the personnel arming them is safe.	Avionics
5.3	PDR 7	Rocket shall maintain a link via telemetry.	Avionics

In addition to the above, the primary considerations during the final design of the avionics system are as follows:

(I) The system needs to concurrently gather, record, and transmit data. It also needs to accurately determine the state of the rocket based on the data and control the recovery and payload systems based on the state.

(II) In order to use the team's time as efficiently as possible, code and hardware from last year's avionics system are being reused when possible.

(III) The system should be adaptable for use in many roles in this year's project and also in other rockets. Changing flight modes and parameters should be possible without changing code.

5.2. Design Process

5.2.1. Hardware

This year we designed a custom flight computer utilizing a 32-bit 96MHz microcontroller, numerous motion sensors, pyrotechnic launch ports, onboard storage, and wireless communication. The board is an iteration of last year's design but includes a variety of updates to decrease size, add features, and make a more flexible and efficient system. The biggest update on this year’s flight computer was the removal of the BeagleBone Black interface and subsequent change in form factor. All data collection, processing, and transmission is achieved through one powerful microcontroller. In addition, a new accelerometer was incorporated to accurately record high-g portions of flight, and onboard flash memory was added to record data. An indication buzzer was also added to aid with rocket locating and launch pad diagnostics. The GPS module was also updated to a higher quality unit capable of better resolution and greater reliability under high stresses. Other minor updates were also made to ease the testing, debugging, and integration process of the new flight computer. The 9-axis IMU, barometer, pyrotechnic ports with continuity checking, XBee wireless communication module, SD card slot, and onboard power management were maintained from the previous year’s design.

The design of the flight computer PCB was done entirely in EagleCAD, a free schematic and board layout tool. The board, measuring 2.75” x 1.75”, was first conceptually laid out in EagleCAD’s schematic editor. The board was then routed on 2 layers in EagleCAD’s board layout editor. After several design reviews were completed to ensure that the board met functionality and manufacturing specifications, the board layout files were sent to OSHPark, a prototype quantity PCB fabricator. Upon receiving the unpopulated boards from OSHPark, team members used a stencil to apply solder paste to the boards, then hand-placed the 100+ components on the board. The boards were then reflowed in a modified toaster oven, and all reflow errors were corrected by hand. Larger through hole components were soldered by hand after the reflow process was complete. Components on the board range in size from 0402 SMD capacitors and 0.4mm pitch QFN chips to larger, through-hole components.

5.2.2. Firmware

Because the Teensy microcontroller is Arduino-compatible, we use the Arduino IDE and the C++ language. Use of this environment allows the team to utilize existing libraries for some of our sensors, which saves significant time.

Much of the firmware that was written for last year’s version of the flight computer is still relevant this year. This includes the libraries for interfacing with sensors, the state machine, and some of the telemetry code. Other parts of the firmware need to be rewritten to work on the new board, or modified to meet our new system objectives. The telemetry protocol is being reworked to support packet acknowledgement, which will make it more robust to noise. The simple averaging filters from last year are going to be replaced with a Kalman filter. Once properly tuned, this filter will fuse data from all of the system’s sensors to produce a single estimate of the rocket’s position and attitude. Another improvement is in the configurability of the system. A new configuration system will allow conditions for state transitions to be set without having to upload new firmware to the microcontroller, and arbitrary conditions can be set to trigger different flight events. Finally, data logging is being enhanced and will be significantly more efficient in both time and space.

5.2.3. Software

Last year's ground station application code was written by two members and was functionality-oriented. Although the ground station achieved its objective, the code was very obfuscated and inaccessible to members who hadn't participated in its programming. Additionally, its UI was very cluttered and not user-friendly. We have thus decided to rewrite the ground station from scratch in order to achieve these two objectives.

Based upon feedback from last year's station, we will reorganize the user interface of our new ground station to be much more accessible and useful. For instance, previous ground station applications displayed the logged data transmitted from the rocket in a small text box. This required users searching for a specific piece of data to read through a lot of unrelated material. This year, we identified the four most important pieces of information we need - status, velocity, altitude, and time since last packet - and placed them in large, easily readable text boxes on the top of our status page.

Additionally, the ground station will be programmed to higher standards, utilizing more abstraction of concepts and modularity to facilitate code readability and improve both accessibility and modification of targeted features. We chose to utilize Python because of the versatile UI libraries and other useful resources found online. Whereas the previous ground station code was contained in a single file, we now have upwards of six files each separated by specific functionality. This distribution and compartmentalization of data allows team members to quickly identify where they need to make modifications to elements of our ground station.

5.3. Technical Design/Analysis

5.3.1. Microcontroller

We have decided to continue using a Teensy 3.2 microprocessor for the Pyxida flight computer. This decision was based off of the device's compatibility with the Arduino IDE, which allows us to leverage existing libraries for our project. The Teensy 3.2 was chosen over conventional Arduino microcontrollers due to its superior processing power. The Teensy runs at 96MHz versus the relatively meager 16MHz for the Arduino Mega. The Teensy has less EEPROM (2kB versus 4kB), however. This will be mitigated via the use of onboard flash memory. All Arduino peripheries are compatible with the Teensy. This, once again, allows us to utilize the ecosystem of Arduino sensors, transmitters, and communication protocols while using the superior speed and memory of the Teensy.

We have incorporated the Teensy into our custom designed PCB. The Teensy 3.2 will read sensor data, log it to onboard storage, and send it to the ground station via radio. It will also coordinate all events critical throughout the launch, including launch detection, apogee detection, and recovery deployment. Furthermore, it will be able to listen for custom commands from the ground station, allowing for a degree of active control over the rocket during flight.

5.3.2. Barometer

Competition and payload requirements dictate that we include a barometric altimeter. This will provide additional altitude data and apogee confirmation for the system. We have decided to use the Adafruit BMP180 Barometric Pressure, Temperature and Altitude Sensor for this task. One advantage of this barometer is that it has a very extensive library for usage with Arduino projects, making integration into our flight computer very simple. Another advantage is that this same barometer was implemented in last year's project. We are confident in our ability to use this chip not only because of the team's experience with it, but also due to the reliability of its measurements.

5.3.3. GPS

The Pyxida flight computer includes a U-Blox Max-6 GPS module for positioning. The Max-6 offers two serial interfaces, one UART and one I^2C interface – we will be using the UART interface as it is compatible with our existing codebase. It has a 5Hz update rate for its navigational data, a cold start time of 26 seconds, and a hot start time of 1 second. It is accurate to 2.5 meters. U-Blox GPS units are standard on many off-the-shelf avionics units for model rockets. One of the reasons for this is that U-Blox GPS are very good at re-acquiring GPS lock after they have lost it, which happens regularly during boost. They also do not come with an integrated antenna, which allows us to choose our own antenna.

5.3.4. Accelerometer

In addition to its IMU, the Pyxida will include an ADXL375 accelerometer. We have included this additional accelerometer in order to accurately measure the thrust curve of the rocket. The accelerometer included in the IMU cannot measure accelerations of more than ±12g. Our rocket will likely exceed this acceleration. The ADXL375 can read up to ±200g acceleration. This will help with the characterization of our custom motors.

5.3.5. IMU

The IMU chip utilized in the flight computer will be the InvenSense MPU-9250. The MPU-9250 is a 9 DOF (degree of freedom) chip that combines a 3-axis gyroscope, an accelerometer, and a magnetometer. This chip offers multiple advantages over other units; the chip has an onboard Digital Motion Processor™ that processes the raw data separately from the microcontroller, outputting the final values. This saves processing cycles for the Teensy, while eliminating a source of error. The chip communicates over I^2C so it is easily integrated with the Teensy. In conjunction with a Kalman filter, we will use the IMU to detect the orientation of the rocket and assist in calculating the rocket's altitude and velocity. This will allow us to coordinate deployment events with the rocket's orientation.

5.3.6. Telemetry

This year, we have decided to continue using the XBee-Pro 900 XSC S3B radio from DigiKey for communication between the rocket and the ground station. This radio will allow us to get live telemetry data from the rocket during flight, along with control over certain aspects of the rocket (i.e., pyro channels) in emergencies. We chose to continue using the XBee radio for several reasons. First, we have built a significant code base for the XBee over the years, and many of our team members have experience working with XBee brand radios. Second, the XBee has a very compact form factor and offers a wide variety of firmware customization options, allowing us to choose, among other things, our broadcast frequency and data baud rate.

The XBee sends and receives data from paired units using a serial interface. In previous years, we have developed our own packet-protocol for sending data over this serial network. This utilizes the XBees in transparent mode, which simply treats the XBees as a pass-through serial network. This year, we have decided to adopt XBee's Application Programming Interface (API) protocol for sending data packets. This firmware level feature offered by the XBee imposes a predefined packet structure on data being sent through the XBee network. Observing this protocol allows the XBee to automatically perform functions like sending packet receipts and providing a Received Signal Strength Indicator (RSSI). We believe that using this protocol will allow us to distinguish more easily between critical and non-critical data and to make sure that critical data reaches the rocket and vice versa.

There will be two XBees used in the project. One will be attached to the Pyxida flight computer in the rocket. The other will be connected to the ground station via USB. The ground station XBee will use a patch antenna and the XBee on the rocket will use a ducky antenna. Data transmitted from the rocket is received and displayed at the ground station and logged to a file location specified by the ground station user.

5.3.7. Ground Station

The ground station is a Python application used for three main tasks. The ground station is able to configure the altimeter, which modifies and dictates when certain actions will be performed by the rocket. Users will be able to read altimeter configurations from either the rocket or some external source and also write configurations they wish to save to a new file for future use. If the currently displayed configuration passes basic requirements, then the user will be allowed to upload the configuration to the rocket.

Additionally, the status tab displays the current status of the rocket and other relevant information. All data received and displayed from the rocket will also be logged to a file the user can specify. Important values like velocity and altitude are displayed at the top of the page, while more detailed information is listed on the side. There are also three buttons used to send critical commands to the rocket. Two buttons allow users to arm and disarm the rocket, while the third button governs pyro channel firing.

5.3.8. Deployment

The Pyxida flight computer uses a system of N channel MOSFETs to control the firing of pyro channels. These allow the Pyxida to control the various separation events that need to occur during descent. The MOSFETs are connected to a shift register which allows all the MOSFETs to be controlled over an I^2C interface. This means that we don't need to use an individual GPIO pins for each MOSFET, leaving our GPIOs open for auxiliary use.

5.3.9. Logging

Flight configurations and log data will be stored on a Micron N25Q256A NOR flash memory chip. The device can store 256Mb of data and interfaces over SPI.

5.3.10. Printed Circuit Board

All of the components, with the exception of the radio, are mounted on a single printed circuit board. This allows the system to be easily moved between rockets as a single unit and is a compact, robust design. Below is an image of the most recent iteration:

5.3.11. Power

The flight computer was designed to be powered by any source capable of outputting at least 5 volts.

Power to the flight computer system will be provided via a Lithium Polymer Battery (LiPo). A LiPo battery was selected since the amount of stored energy provides a sufficient amount of runtime for the system. More specifically, the power supply to be used is a 7.4V, dual cell LiPo that holds 9Wh (Watt-hours) of energy (equivalent to 1200 mAh). Because our microcomputer (the Teensy) and all our sensors operate at a voltage of 3.3V, the PCB will have a LD1117-3.3 Semiconductor to step down the input voltage (7.4V) and output a safe 3.3V.

5.3.12. Redundant Systems

Although we will test extensively, we do not feel that we can assure the reliability of our flight computer to the point where it can be the sole flight computer on the rocket. Therefore, we will include a TeleMetrum, a commercial flight computer, on the rocket. This computer comes with an integrated ground station and can handle many of the most essential functions that Pyxida performs. This system will take over to make sure that the rocket can be recovered if Pyxida fails.

5.4. Key Technical Issues/Risk

The flight computer is on its second full iteration after Project Therion. Even though the risks are high-impact, the working previous iteration, test plan, and implementation of a COTS flight computer in parallel reduce the likelihood of avionics failures.

5.4.1. Risk & Mitigation Tables

Unmitigated Risk Table						
Risk	20%					
	10%					
	5%			2		1
	1%					
	0.1%					
		1	2	3	4	5
Avionics		Impact				

	Risk	Risk Reduction Plan
1. Failure to deploy parachutes	Errors in the software programming, physical packing of parachutes, and preparation of deployment hardware - eMatches, nichrome pin releases - can all lead to a partial or complete failure of parachute or rover deployment. Any failure in the deployment of recovery systems is a very high risk that may lead to total recovery failure.	The Pyxida flight computer has an extensive test plan. It is also implemented in parallel with a COTS flight computer. The most risk-prone part of the avionics system is the implementation of the flight computers in the avionics bay, which is tested prior to flight integration.

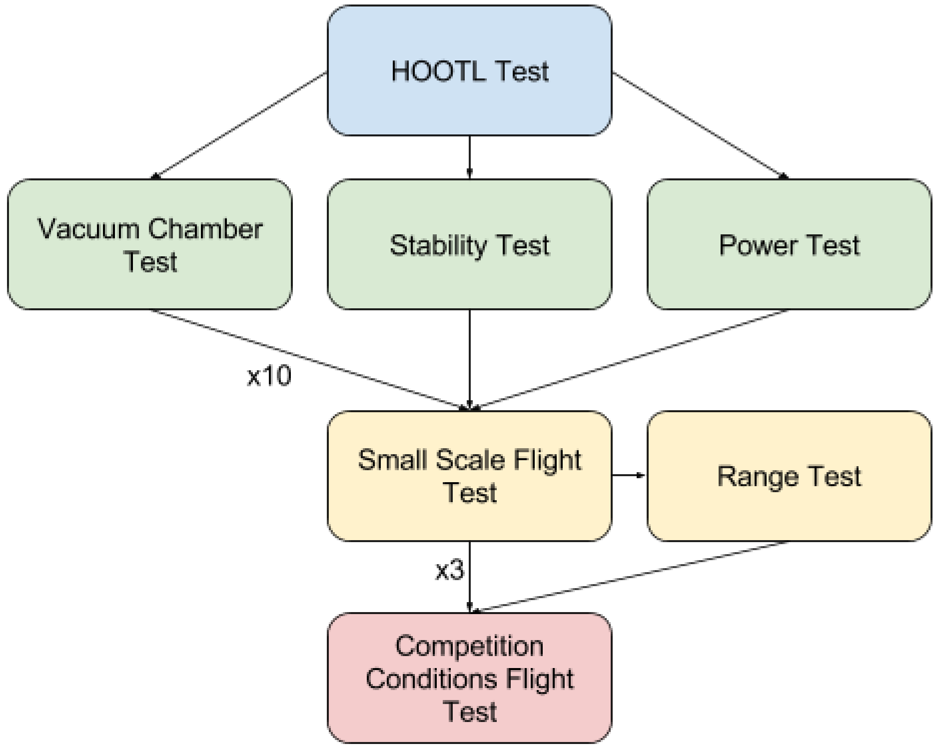
Guest User: can the COTS board supply enough current for long enough for the nichrome cutters to work?

2. Sensor failure	Sensor failure can be caused by errors in the software programming or assembly of the PCB. The wire connections can break, or the sensors could yield false data. This could result in a failure to deploy, but it could also result in a more minor failure to record data.	The Pyxida flight computer has an extensive test plan. It is also implemented in parallel with a COTS flight computer.

	20%					
	10%					
	5%					
Risk	1%					
	0.1%			2	1	
		1	2	3	4	5
Avionics		Impact				

5.4.2. Test Plan

Below is a diagram of the test plan for the avionics system.



The tests are ordered by risk level:

No risk, tests can be completed at any time without special equipment or hardware. Code must pass this test to be committed to a feature branch	Low risk, tests require Pyxida hardware and must be completed in lab A feature branch must pass these tests to be merged to the testing branch	Medium risk, tests need access to a launch site and a compatible rocket The testing branch must pass these tests to be merged into the master branch	High risk, tests require a complete competition rocket and access to specific launch sites. The master branch must pass this test to be used at a competition.

5.4.2.1. HOOTL Test

Compile a version of the software for x86 and run it through a set of scenarios to prove functionality and demonstrate stability.

Requirements
Testing branch meets style guide
Testing branch compiles on x86

Testing branch passes all unit tests

Success Conditions
No crashes
All tests test scenarios produce acceptable results

5.4.2.2. Stability Test

Power the board with a bench supply and leave it in a state, such as disarmed or armed, for several hours. The test should last longer than the maximum battery life of the device and the device should be logging data for the entire duration of the test. At the conclusion of the test, verify that the device is still responsive by issuing a command and checking the response.

Requirements
Testing branch has passed HOOTL
Testing branch compiles and flashes successfully

Success Conditions
Device responsive at the end of test
Device remained in the state that it was left in
Device changes to proper state following actions after the test

5.4.2.3. Power Consumption Test

Run through a variety of situations with the board powered by a bench supply. Keep track of current usage throughout the tests. This test should be completed after every hardware revision or major software changes to give us an idea of battery life.

Requirements
Testing branch has passed HOOTL
Testing branch compiles and flashes successfully

Success Conditions
Calculated battery life meet specifications

5.4.2.4. Vacuum Chamber Test

This test should be completed many times with many different flight configurations. Attach Christmas lights to every output for each flight configuration, including the outputs that you don't intend to fire for this particular test. Run the altimeter in the vacuum chamber following the procedure that you would for normal operation and verify that all of the outputs that were expected to fire did so at the proper time and that the others did not. After each "flight", check both the log on the device and the telemetry data to confirm that they agree with the pressure that was reached in the vacuum chamber.

Requirements
Testing branch has passed HOOTL
Testing branch compiles and flashes successfully

Success Conditions
The Pyxida responds to all commands as expected in every test
Every deployment event is timed properly
Telemetry data is correct and apogee is close to the estimated value
Data in flight log looks reasonable and matches the inputs that were applied

5.4.2.5. Small Scale Flight Test

Use a Pyxida as a backup or primary altimeter system on a rocket significantly smaller and lower risk than the competition flight. A minimum of 3 test flights should be made to prove major revisions to hardware or software, and these tests should cover as wide a range of flight profiles as possible. At least one flight should must enter a transonic regime.

If the recent changes to the system directly affect the function of deployment channels, it is acceptable for the first test of the sequence to be a "dry test" with the Pyxida connected to e-matches without deployment charges and a COTS altimeter system handling deployment. This kind of test represents less of a risk for the rocket as the chance of an early deployment is eliminated. The results, though less meaningful than a full flight test, can still be determined by checking if the e-matches were fired and verifying the timings recorded in the flight log.

Requirements
Ten successful vacuum tests

Completed stability tests
Completed power test

Success Conditions
All deployment events were timed properly
Accurate telemetry data is received at a reasonable rate
Flight log data agrees with simulation of rocket flight

5.4.2.6. Range Testing

Range testing is similar to power testing because it has less strict criteria for passing and is mostly for determining the capabilities of the system. To do so, set up a ground station at one end of a field and position a Pyxida at the other at some landmark a known distance away. Record the rate at which data is coming in and what percentage of sent commands are received, and then move the Pyxida to a further landmark and repeat the data collection.

Continue this process until both sending and receiving drop off. To improve the accuracy of the test, mount the Pyxida in a container that is similar to the competition avionics bay in materials and construction.

Requirements
Ten successful vacuum tests
Completed stability tests
Completed power test

Success Conditions
Sending commands works flawlessly at the safe distance for our motor, preferably further.
Telemetry data comes in several times a second up to half the target altitude, and once every several seconds above that.

5.4.2.7. Competition Conditions Flight Test

Fly the system in a test flight of the competition rocket both to prove the integration of the system with the rest of the vehicle and to demonstrate that it performs as expected when subjected to a competition flight.

Requirements
At least three successful test flights, one transonic
Range testing completed with results that suggest that telemetry will work on the planned flight profile.

Success Conditions
All deployment events were timed properly
Accurate telemetry data is received at a reasonable rate
Flight log data agrees with simulation of rocket flight

5.5. Interfaces

5.5.1. Payload

The flight computer will control the deployment of the payload. The Avionics team will also provide support for the Payload subteam as they develop the electronic systems for the payload.

5.5.2. Recovery

The flight computers will be actuating the recovery system, including the separation of the rocket and the resulting deployment of the drogue parachute as well as the release of the main parachute.

5.5.3. Structures

The Structures subteam will be constructing the avionics bay that the flight computers will be mounted in. They will fabricate and assemble the structural components of the bay, while the Avionics subteam will install and wire the flight computers and other electronics into the bay.

5.5.4. Propulsion

The Avionics subteam will provide the Propulsion subteam with motor performance data after each test flight, in the form of a thrust curve calculated from the acceleration measured during flight. This data, when combined with measured properties of the rocket, will allow the Propulsion subteam to generate a thrust curve for the experimental motor and evaluate its performance.

5.5.5. GSE

The Ground Support Equipment subteam will develop a system for mounting telemetry antennas and pointing them at the rocket when it is in flight. The system will feature automatic tracking of the rocket, so it will need a stream of data from the ground station on the last known position of the vehicle.

5.6. Going Forward Plan

The majority of the first semester this year has been spent reworking the PCB design from last year. We changed the form factor, replaced the GPS, and added an accelerometer, battery voltage monitor, buzzer, and new connectors for expansion. While development on this revision of the PCB was taking place, the new members of the team became acquainted with the development tools while others revised the telemetry protocol and enhanced the ground station application. Since the PCB was finished, the team has started to add firmware support for the new features that this revision of the board has. This process will continue for the remainder of the semester. The team has also revised our telemetry protocol which is going to be tested and integrated before the end of the semester. The team is in the process of developing a Kalman filter which should also be ready to test before the end of the semester. Finally, development of the ground station application will continue with the goal of configuration and graphing being functional by the semester's end.

The Independent Activities Period that takes place between first and second semesters often leaves students with significant time to further develop the rocket. We hope to use this time for testing and refinement of the Pyxida system. We also hope to finalize the hardware design before IAP so that the final revision of the PCB can be ordered and assembled before second semester starts.

Second semester will be spent testing the system and fixing any issues that emerge. Our goal is to complete as many test flights with the system as possible. We would like to test it in many different flight profiles to prove that the system works in general, but our primary objective is to prove that it works in competition conditions. To ensure that development isn't rushed, we hope that no new hardware will be needed during the second semester. Additionally, we are going to freeze development of the software and firmware by the end of April because any work past that point can not be flight tested. We will spend the remaining time before the competition by continuing to test the system to further build our confidence in it.

6. Propulsion

6.1. Overview of Requirements

6.0	RRD 2.0	Raziel shall achieve an apogee of 10,000 feet +/- 300 feet.	Propulsion
6.1	Internal	Propulsion shall be single-stage.	Propulsion
6.2	DTEG 2.1	Propulsion shall use non-toxic propellants.	Propulsion
6.3.1	DTEG 2.2.1	Propulsion shall have a two-step arming system which can only be armed when all personnel are at least 50 feet from the Rocket.	Propulsion
6.3.2	DTEG 9.2	Ignition system shall require no more than 15A at 12V to function.	Propulsion
6.4.1	DTEG 2.4	Propulsion testing shall comply with ESRA requirements.	Propulsion, GSE
6.4.2	DTEG 4.2.4.1	The combustion chamber shall be designed for at least twice the maximum chamber pressure. The chamber shall be tested to at least 1.5 times the maximum chamber pressure.	Propulsion
6.5	DTEG 2.4.3	Propulsion shall have a successful static fire test prior to a test launch. Propulsion should have two successful static fires prior to a launch.	Propulsion, GSE

6.2. Design Process

6.2.1. Desired properties for case material

A material trade study was performed to select an appropriate material for the motor case. The desired properties include specific strength, ductility, moderate temperature tolerance, fracture toughness, chloride resistance, manufacturability, cost, and availability. The case mass should be minimized by finding a material with high specific strength. It is also important that the hoop stress be less than the yield tensile strength. In the chance that the case fails, it should be ductile so it yields before breaking and thus fails more safely. The case will be exposed to elevated temperature for a short period of time upon firing, and thus must be able to withstand heat exposure up to 400 K, the likely wall temperature due to the insulative layer in the case. high fracture toughness is desirable to prevent the growth of cracks and other defects as much as possible. Other critical considerations were making sure the case material was easily accessible for acquiring and/or purchasing, was not too expensive, and could be machined and manufactured using the Team's available resources. As will be discussed, there are materials that were more exceptional than others, but the high cost associated with and potential difficulty in ordering these materials made it desirable to find a less advanced case material that still had all the necessary properties and was cheap and easy to manufacture - ultimately saving us time needed for rocket testing.

6.2.2. Materials considered

Type	**Alloy**	**YTS [MPa]**	**Specific strength [kN m kg^{-1}]**	**Elongation at break [%]**	**YTS at 400 K [% of room temperature value]**	**Fracture toughness [MPa m$^{1/2}$]**	**Chloride resistance**	**Cost for 4in x 1 ft rod [USD]**
Aluminum	2024-T3	290	105	14%	97%	25	--	203 McMaster
	6061-T6	276	102	12-17%	87%	29	No SCC	91 McMaster
	7075-T6 [5]	480	173	7%	85%	20-29	SCC Corrosion [6]	202 McMaster
Titanium	Ti-6Al-4 V [5]	800 (ann) 950 (ST)	180 (ann) 214 (ST)	8-10% (ann) 6% (ST)	82%	75	No SCC No corrosion [6]	1460 Online Metals
Steel	1018 [7]	370	47	15%	100%	--	--	128 Metals Depot
	4140 [8]	415 (ann) 1100 (HT160)	53 (ann) 140 (HT160)	26% (ann) 13% (HT160)	92%	88 (HT160)	--	177 McMaster

	4340	1100 (HT160)	140 (HT160)	13%	92%	132 (HT160)	SCC Corrosion [9]	245 TW Metals
	4340M (300M) [10]	1240	158	6%		60	--	--
	C300 maragin g	830 (ann) 1970 (HT)	104 (ann) 246 (HT)	16% (ann) 8% (HT)	96%	175	--	1930 Online Metals
	316	290	36	50%		112-278	SCC No corrosion	246 McMaster
	17-7PH TH1050 [5] [11]	1030	135	6%	94%	76	Good	230 OnlineMetals
Composite	Carbon fiber T300 / Toray 350F epoxy [12]	1760	1130	1.3%	--	--	--	~100 USD/kg Easy Composites
	E-glass / BPA epoxy [1]	1000	510	1.4%	70% [13]	--	--	--

Of the alloys examined, those with the highest specific strength were C300 maraging steel (246 kN m kg^{-1}), Ti-6Al-4V (214 kN m kg^{-1}), and 7075-T6 (173 kN m kg^{-1}). Considering material cost, maraging steel and titanium alloy were an order of magnitude more expensive than the other alloys examined, and were thus disqualified from consideration. Once these high-cost alloys were eliminated, Al 7075 offered the greatest specific strength.

1018 steel and Al 6061 were particularly cheap while the other metal alloys had fairly similar costs to one another (around 200 USD). Interestingly, the per-unit materials cost for a composite case would be about the same (or possibly somewhat less) than a metal case. However, a composite case would also require a capital investment in a filament winding machine, which would be unnecessary considering the number of cheaper, likewise effective options. A composite case would also be more difficult to design and manufacture. In trying to quickly develop a custom motor, an easy and simple design is important. Additionally, because the strength and failure criterion of metals are simpler and better understood than composites, it was estimated that a metal case would have a lower probability of structural failure.

A benefit of the aluminum alloys was that they are sold already heat-treated and can be easily machined in that condition. The high-strength steel materials are sold and machined in an annealed condition. We would have to perform heat treating ourselves to achieve the desired strength. Although the team has access to facilities for heat treating, this process adds a step to the manufacturing process, increasing complexity and production time. As such, and due to regulations surrounding the use of steel in experimental motor cases, aluminum will be used in the case.

6.3. Technical Design/Analysis

6.3.1. Propellant Formulation

The first step in developing the solid motor was determining a stable propellant formulation that has good burn properties. These properties can be discovered by combustion tests or through use of a Crawson strand burner. For Project Xaphan, which is the project name for development of the flight motor., a baseline formulation used on a separate project a couple of team members participated in was used as a starting point for propellant development. Research was also done on the experimental board on The Rocketry Forum to find published formulations and recommendations to improve the mechanical and chemical properties of propellants. Combined with combustion simulation using the Rocket Propellant Analysis (RPA) software, the following chemicals were used in the team's propellant:

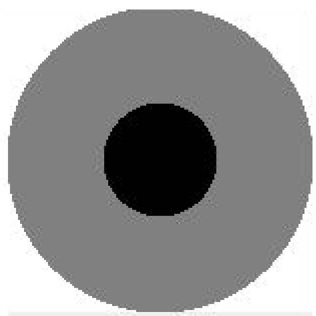
Component	**Role**
Ammonium Perchlorate	Oxidizer
Hydroxyl Terminated Polybutadiene	Binder
HX-752	Matrix Stabilizer
CAO	Antioxidant
Isodecyl Pelargonate	Plasticizer
Modified MDI	Curative
Aluminum Powder	Fuel
Copper Chromite	Catalyst
Copper Oxychloride	Colorant/catalyst
Castor Oil	Cross-linking promoter
Polydimethylsiloxane	Surfactant

Triton X-100	Surfactant

The maximum density this formulation can have is approximately 2000 kg/m^3. Achieving this density is unrealistic, however, as air pockets can be introduced into the mixture at nearly every stage of propellant fabrication, especially while the propellant is being mixed and poured. As such, a density of 1750 kg/m^3 is desired.

6.3.2. Grain Geometry

Due to the nature of solid propellants, the grain geometry is critical for setting the thrust profile of the engine. Because of the relatively short burn time of the engine and the weight profiles of the engine, a neutral or slightly regressive thrust profile is desirable. For this reason, a slightly modified Ballistic Test and Evaluation System (BATES) grain will be used. These grains use a cylindrical core with uninhibited ends to keep burning surface area constant.



For the flight motor, 4 identical grains will be used. Each of these will have the following dimensions:

Length	7 inches
Core Diameter	1.25 inches
Outer Diameter	3.27 inches
Core Taper	.5 degrees

The taper is required to allow the propellant mold to be release from the propellant after casting. This will result in protrusions in the core of the propellant that will erode during the start of the burn. This erosive burning will create additional thrust early in the burn, increasing the vehicle's velocity off the rail and improving stability early in the flight.

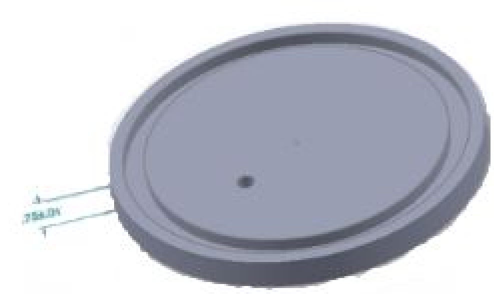
6.3.3. Propellant Fabrication

6.3.3.1. Chemical Measurement

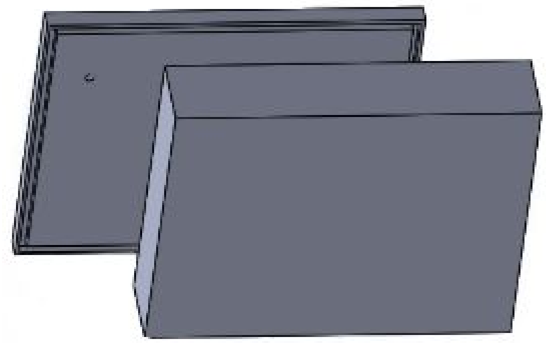
To enable very precise ratios of compounds in the final propellant, a system was developed to facilitate precursor mixing at the correct time. The precursors are poured into appropriately sized stainless steel bowls and then poured back into their original containers. This coats the bowls with the appropriate quantity of residue that will be left behind in the bowl after pouring the chemical out. The bowl is then tared on a scale with the residue. The actual quantity of precursor is then measured into the bowl. The bowl is then covered with plastic wrap to prevent contamination before use. The chemicals are measured approximately 40 minutes before mixing begins.

6.3.3.2. Vacuum Processing

To ensure uniform propellant density and consistent performance, all voids must be removed from the propellant mix during manufacturing. Voids are primarily created by gas pockets in the propellant. They are introduced by the mixing process and by ammonia off-gassing as the HX-752 reacts with the Ammonium Perchlorate. The ammonia generation falls off to imperceptible levels after 3 hours, but mechanical gas introduction is a constant problem. To mitigate gas pockets introduced by mixing, a low-speed, low shear mixer head is used. Vacuum processing is used to remove the remaining volume of trapped gas. Two pieces of equipment are used to facilitate the vacuuming of fuel. The first is a lid for the mixing bowl (figure below) that seals onto the bowl via a channel with silicone rubber. It has a fitting to attach a standard male quick-detach vacuum hose end.



The processing of propellant in the bowl loses effectiveness with propellant depth. When mixing batches of propellant in excess of 1 kg, there is insufficient vacuum to remove bubbles in the bottom of the propellant mixing bowl. To supplement this process, a shallow tray with a vacuum lid will process the propellant after it is thoroughly mixed. The shallow depth allows for effective and complete void removal. Because the volume of propellant is 5 quarts, we designed a roughly 10 quarts pan of dimensions 0.50 x 0.40 x 0.05m to allow the propellant to form a thin layer, encouraging the removal of air bubbles. The same male quick-detach and silicon seal is used for this as for the bowl lid, although the material is slightly thicker (1 in) to account for the larger surface area (see figure below).

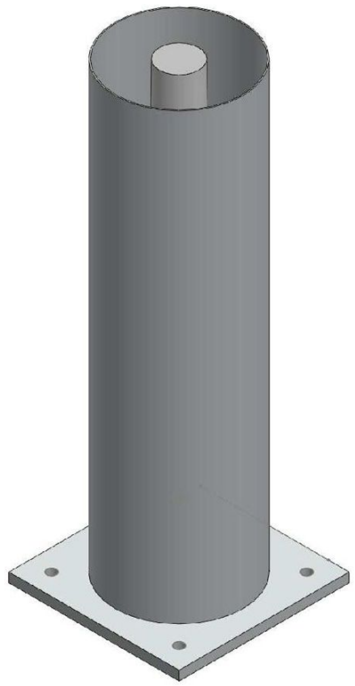


6.3.3.3. Casting

The first grains of the propellant were made by pouring the liquid mixture into commercial casting tubes for 54mm propellant grains and waiting for them to cure. Their core was then bored out with a drill press. However, this method produced large amounts of waste propellant and did not guarantee alignment of the core inside the grain. Considering that larger grains would intensify these problems and that drilling a larger core could create too much heat through friction, this method was impractical and another system, consisting of a mold made of aluminum and Teflon, was developed.

The mold consists of two aluminum bases that hold a 3.37” diameter casting tube coaxially aligned to a tapered Teflon rod to create the propellant grain core. The casting tube and the Teflon rod seat into the bottom aluminum base and the resulting mold is placed on a horizontal surface. In the next step, the propellant is cast into the tube and the top aluminum base is added to keep the frustum aligned during the curing process.

Initially, an aluminum rod instead of a Teflon frustum was considered to core the propellant, but the material was switched to Teflon and a 0.5 degree slope was created in its side in order to make the disassembling process easier and to prevent damaging the grain. Because the frustum is sloped, the central hole in the top aluminum base is smaller than the corresponding hole in the bottom plate. The diameters of the top and bottom holes in the plates are respectively 1.04” and 1.25”.



6.3.4. Motor Casing

The motor casing consists of all components necessary to assemble and fire the motor. These include the insulative liner that protects the case from the heat of combustion, the forward closure that forms the cap of the pressure vessel, the retention ring that prevents the forward closure from being ejected, the insulation disk protecting the forward closure from the flame, the case wall, the nozzle, the support structure for the nozzle, and the thrust ring, which retains the nozzle and transfers loads from the case into the airframe of the vehicle. Each of these sections will be discussed in more detail below.

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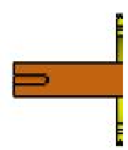

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6.3.4.1. Case Material

The original materials case study recommended Al 7075-T6 as the primary motor case material. It has high specific strength (compared to other metals), reasonable cost, and is easy to design with. It is sold in the T6 condition, so we will not need to perform heat treating. It is easy to machine. Al 7075-T6 does have a low fracture toughness and is susceptible to stress corrosion cracking, which could limit the lifetime of a 7075 case, but these concerns were expected to be mitigated with proper handling.

However, in re-evaluating the case study, Al 6061-T6 was selected for the case material. Looking at the design, the motor case is threaded, so it must be at least 0.2” thick. At this thickness, both Al-7075 and 6061 have sufficient tensile strength to withstand the pressure of the motor, estimated with margin to be 1500 psi. This was determined using Barlow’s formula for the internal pressure that a pipe can withstand:

$$P = \frac{2ST}{DF}$$

where *P* is internal pressure*, S* is material yield strength in psi, *T* is the thickness of the case, *D* is the outer diameter of the case, and *F* is the safety factor. Using an outer dimension of 4” and safety factor of 1.5, the working pressures of 6061 and 7075 were determined to be 2670 psi and 4640 psi, respectively--both well above the motor’s working pressure of 1500 psi. Therefore based on the case design thickness, even though Al 6061 had less resistance to pressure than Al 7075, it could function adequately under the expected conditions. Because the Team already had Al 6061 in its possession and it generally costs less than Al 7075, Al 6061 was selected for the case material.

6.3.4.2. Forward Closure

The forward closure is an airtight seal on the forward end of the motor. This plate provides the mounting architecture for positive retention, preventing the motor from being ejected from the rocket after burnout. For the purposes of this project, the forward closure being discussed will be the one used during motor testing, as this is more complex than the closure that will be flown in the vehicle. Specifically, due to considerations of the mass of the case, the need for additional data, the ease of assembly of the propulsion system, and available volume for the case, the flight closure will not have mounting points for any sensors.

The forward closure provides the mounting point for the thermocouple and pressure transducer that will be used during static fire tests of the motor. The closure, therefore, needs to have mounting holes for both of these sensors that are airtight seals. Moreover, the closure needs to be designed such that no thrust will be supported by the pressure transducer, as the sensor is fragile and difficult to replace.

Finally, the forward closure needs to be able to support the pressure of the motor without failing or causing significant deformation that could compromise the shaft seal between itself and the case wall. As such, the closure needs to be able to support at least 1500 psi, or be at least 0.5” to provide enough material to effectively mount the pressure transducer, thermocouple, and O-ring.

The minimum thickness of the forward closure was calculated using the following formula:

$$F = G * (A_i * T)$$

$$\frac{P_c*(\pi*R_e^2)}{G*(2\pi*R_i)} = T$$

$$T = \frac{(1500psi)*(1.81in)^2}{(30000psi)*2*(1.31in)} = 0.062\ in$$

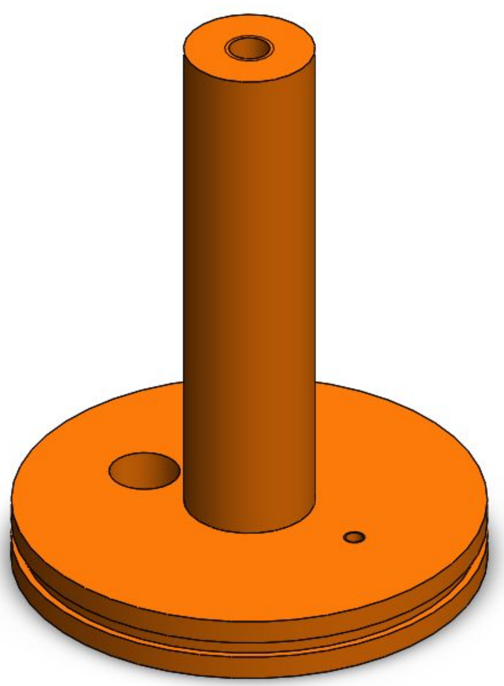
F = force exerted on the bottom of the forward closure
G = yield shear strength of Aluminum 6061-T6 = 30000 psi
T = minimum thickness of the forward closure
R_i = Internal Radius = 1.31 in
A_i = Internal Area
R_e = External Radius = 1.81 in
P_c = Chamber pressure = 1500 psi

The minimum thickness of 6061-T6 aluminum required to barely support 1500 psi is 0.062”. This is too thin to feasibly support the sensors and O-ring. As such, the closure will be 0.5” thick.

At this thickness, the forward closure will not fail until it is put under a pressure of:

$$P = T * G * \frac{R_i}{R_e} = 10860\,psi$$

This gives the forward closure a 7.24x safety margin over the design requirement of 1500 psi.

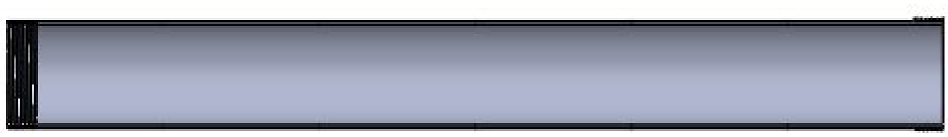


6.3.4.3. Case Wall

The case wall will be made of Al 6061-T6 with a thickness of 0.25”.

A threaded closure would requiring cutting threads into the case wall. Custom-cut threads can easily spall and bind together if both the type A and type B threads are of the same (soft) material. This is called thread galling or cold welding. To avoid this, there are several options. The simplest fix is to soften the threads using common materials such as pencil graphite, oil, wax, or WD-40. More complicated options are making the other threaded part (besides the case wall, e.g. the aft closure ring) from a harder material like steel or, if both parts are aluminum, anodizing the threads on one or both sides. However, because anodization would cause a slight dimension increase and requires special lab treatment, it will likely not be performed.

The wall’s outer diameter is 4”. The threads mating with the forward closure are internal 3-¾” x 10 UN and the threads mating with the thrust ring are external 4” x 8 UN.



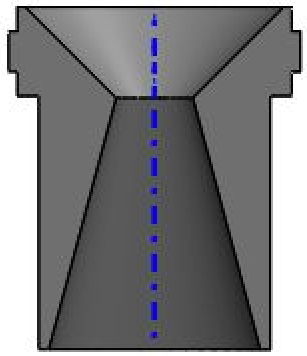
6.3.4.4. Nozzle

The nozzle directs the flow of exhaust out of the motor, expanding the flow from the chamber pressure to ambient pressure. For a given chamber pressure, the motor will produce the most thrust by expanding the flow to the the ambient pressure according to the following equation:

$$F = mdot * U_e + (P_e - P_a)/A_e$$

F: Thrust
Mdot: mass flow rate of exhaust exiting the motor
U_e: Actual exhaust velocity at the exit of the motor
P_e: Nozzle Exit Pressure
P_a: Ambient atmospheric pressure
A_e: Nozzle Exit Area

For the purposes of the competition, the nozzle will be designed to expand the flow to the ambient pressure at ground level in Las Cruces, NM, which is 12.7 psi. This sets the ratio of the nozzle exit area to the nozzle throat area.

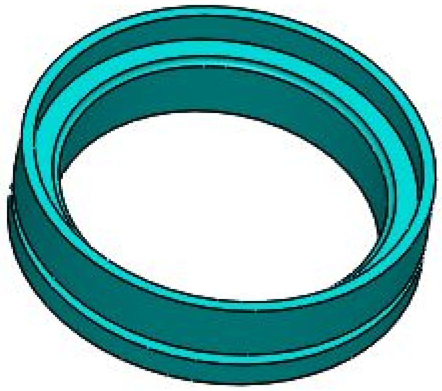


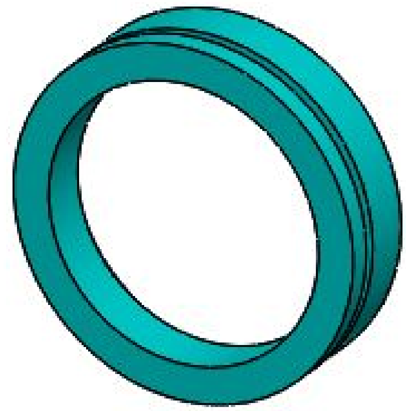
Nozzle Component	Dimension
Throat Diameter	0.95”
Exit Diameter	2.62”
Length	4.24”
Inlet Diameter	3.2”

The nozzle will be made of machined graphite. Graphite was chosen because it retains its shape well when exposed to extreme temperatures and the nozzle receives the most heat of any component in the case. As the nozzle heats up and the hottest parts break away, the graphite will be ablatively cooled.

6.3.4.5. Nozzle Carrier

The nozzle carrier supports the nozzle, allowing for less material to be used on the nozzle and thus reducing the weight of the case. The nozzle carrier also transfers the pressure loads to the thrust ring, protecting the nozzle from damage during the firing of the motor. The nozzle carrier also needs to provide sealing, keeping combustion gases from flowing around the nozzle and damaging the case.



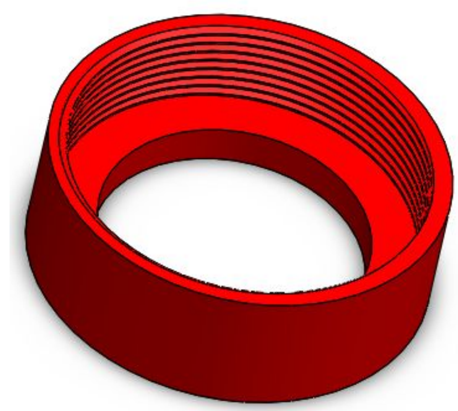


The dimensions of the nozzle carrier are:

Length	0.93”
Inner diameter	2.88”
Outer diameter	3.62”
Inner Step diameter	3.4”

6.3.4.6. Aft Thrust Ring

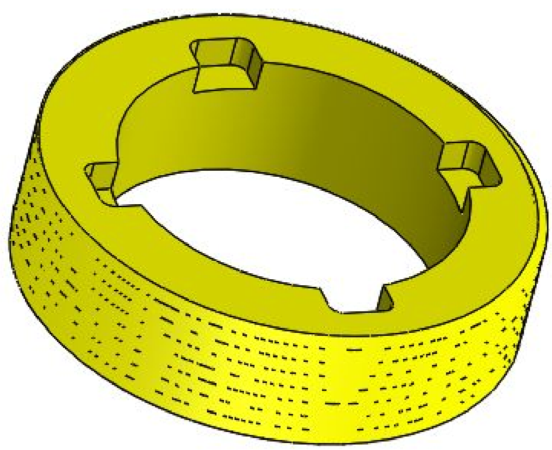
The thrust ring will be made of Al 6061-T6 with a thickness of 0.20”. This value is dependent on the estimated shear strength of the ring, providing significant margin between the yield load of the thrust ring and the expected chamber pressure.



The outer diameter is 4.06” and the diameter of the extruded cut is 1.18”. The threads are 4”-8 UN Type 1B threads, and mate with rear end of the case wall.

6.3.4.7. Forward Retention Ring

The forward retention ring threads into the front end of the case, and provides the load that keeps the forward closure in position. This part needs to have adequate thread strength to hold the forward closure in position while also being narrow enough to not inhibit integration of the pressure sensor. The part is shown below:



Inner Diameter	2.62”
Thickness	1”
Thread	3 ¾” -10

The notches on the upper surface of the part are needed to fully tighten the part into the case. These match a commercial screwdriver that is designed for a very similar part, allowing us to use this tool for tightening the forward retention ring. If needed, a custom tool can also be made using a 3D printer.

6.3.4.8. Liner

Solid propellant burns at a very high temperature, greatly exceeding the melting point of the aluminum casing. There are several steps that keep the case from melting or softening, however. For one, as the propellant burns from inside of the grain to the outside, the unburned propellant acts as an insulator for the surrounding casing and components. In addition, an ablative liner is installed between the propellant inhibitor and the outer casing. When the burn reaches the outer layers of propellant, the liner absorbs the heat and prevents the case from reaching extreme temperatures. The liner also supports the propellant grain stack and helps all the pieces fit together.

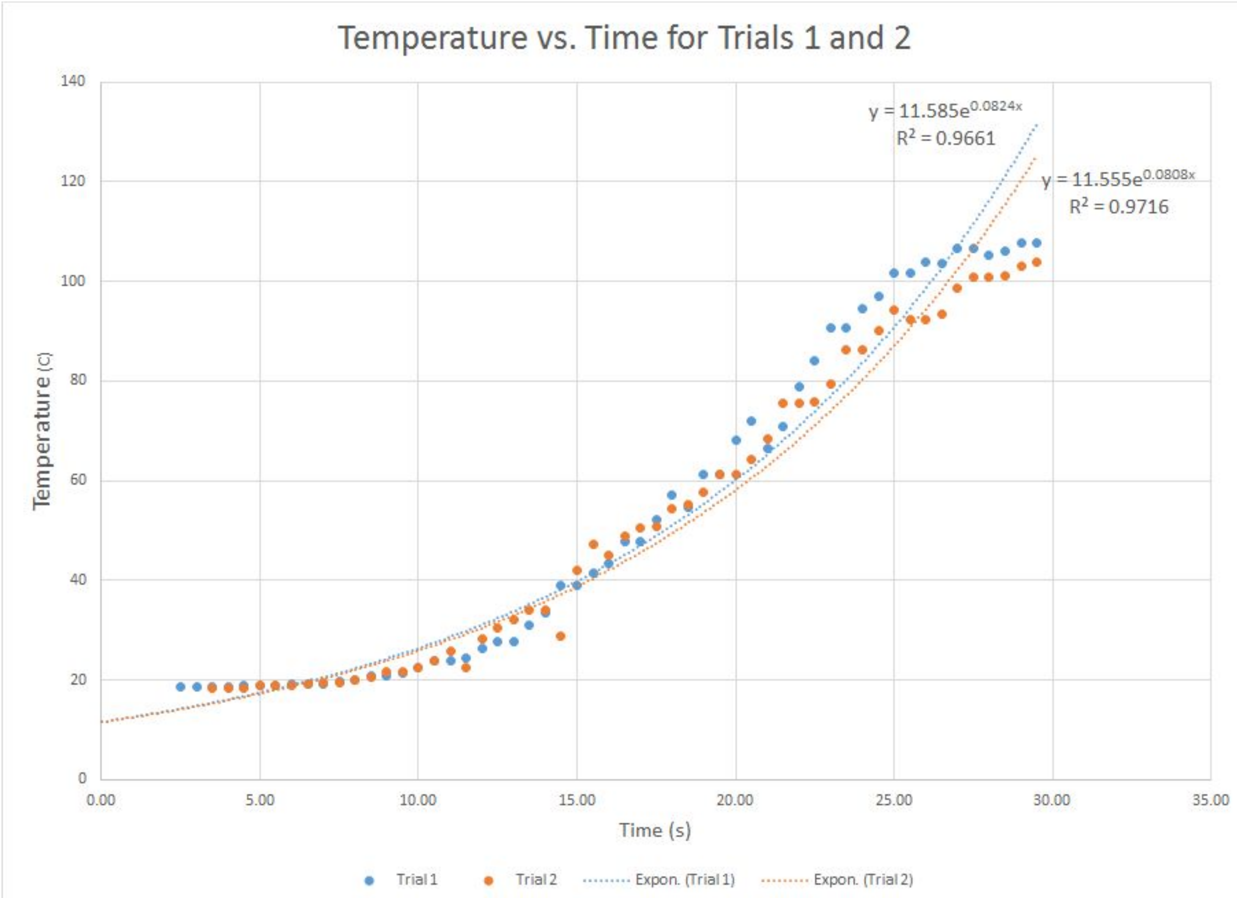
Liners are constructed from a convolute wound cotton phenolic material. The phenolic resin is very strong and can resist high temperatures, making it an ideal choice for motor liners. In order to manufacture liners in-house, a convolute winder and curing oven are required as well as the stock phenolic and epoxy materials.

Commercial solutions are also available for motor liners. These liners benefit from their reliability and simplicity of integration and eliminate the need for expensive equipment. For the solid propulsion system under development, a commercial liner was chosen due to material and budget constraints.

6.3.4.9. Insulation Disk

Cement board was selected for the motor insulation disks due to its low cost, accessibility for the team, low density, and excellent short-term resistance to heat flow. The insulating properties of this material were tested by exposing a 0.25”-thick sample to a propane torch flame to simulate the combustion of fuel. Propane was chosen to simulate the combustion of fuel because its flame temperature is 1967 ^{O}C, which is comparable to the burning temperature of the motor of 2000 ^{O}C. Over a 30-second interval of continuous exposure to the flame, the temperature of the other side of the board was recorded with a video recording device. The temperature of the opposite side was plotted with respect to time and an exponential trendline was imposed on the data, in accordance with the expectation that the rate of temperature increase with respect to time would be proportional to the temperature of the material, that is,

$$\frac{dT}{dt} = kT$$



Although the data deviated from this relationship, the temperature of the cement board increased less than this model predicts during the first 10 seconds. This result is especially encouraging because the expected burn time of the motor is approximately 4 seconds. According to these test results, cement board is capable of keeping AL-6061 safely below its softening temperature of 200^{O} C for at least 30 seconds of continuous exposure to flame, exceeding the requirements for a motor insulation disk. Shown below are insulation disks for use in development motors.

6.3.5. Ignition

6.3.5.1. Igniters

National rocketry safety codes require rocket motors to be started by electrical means. The igniter consists of an electric match dipped in a pyrogen mixture, which is inserted into the motor, contacting the propellant. When current is applied across the leads of the electric match, it flows through a bridge wire in the tip and generates heat. This heat ignites the pyrogen, which burns at an extremely high temperature for a short time, igniting the main propellant grain.

The development of a custom propulsion system necessitates a custom ignition system tailored to the design criteria of the motor. Specifically, igniters for 54 mm and 98 mm motors are required. The igniters must be capable of producing extreme heat over a period long enough to ignite the propellant. In testing, different casting methods were employed to a commercial pyrogen mixture to determine the ideal pyrogen and igniter construction for the rocket motor.

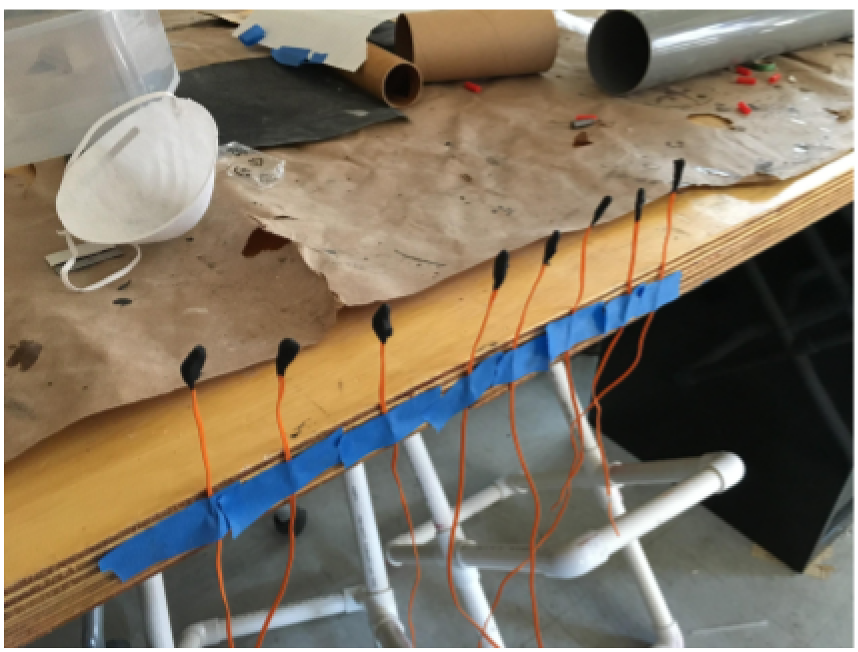
Commercial electric matches were used throughout all of the tests due to their reliability and simplicity. Testing began with commercial electric matches dipped in a commercial pyrogen mixture. The mixture consisted of a fuel and oxidizer that were combined with an acetone binder to obtain an even consistency. For the first round of tests, an electric match head and the pyrogen mixture were cast into one inch lengths of a plastic drinking straw. The intent was to obtain a uniform cylindrical shape that would fit inside the motor nozzle. Ten igniters were manufactured using this method.

It was discovered in the first tests that the curing pyrogen shrunk to an appreciable degree, enough to expose the electric match head and reduce the burn time and uniformity. For the second round of tests, the electric match heads were dipped in the pyrogen mixture without a casting straw. The heads were dipped in the pyrogen once and then dipped again to achieve substantial pyrogen buildup at the tip while still fitting inside the motor nozzle. Thirty igniters were produced in three batches using this method.

Igniters were tested in batches in a blast chamber using a 9V launch controller. Video footage was obtained for post-burn analysis. Ten igniters, using the double dipped method, were tested on propellant scraps. All ten igniters successfully ignited the propellant. Since the igniters ignited the propellant at open air pressure, it is very likely that ignition in a pressurized motor casing will be successful. Based on these results, the double-dipped igniters were chosen for use in the solid motor tests.

Manufacturing of the double-dipped igniters in the custom pyrogen mixture

An igniter manufactured with the double-dipped method under test in the blast chamber.



A batch of finished igniters curing after dipping.

6.3.5.2. Igniter location

In choosing igniter location, the team considered forward internal ignition, rear internal ignition, and external ignition.

External ignition involves wiring an igniter into the bottom of the rocket to a manual ignition switch just prior to the launch.

Internal ignition involves wiring the igniter to the flight computer, which performs pre-programmed checks prior to executing the launch command. Recent problems occurred by launching via external ignition without arming the flight computer, so internal ignition was favored to avoid these problems in the future.

Forward internal ignition would involve running a short signal wire from the flight computer through the forward closure and into the case during rocket assembly. In contrast, rear internal ignition would involve running a signal wire from the flight computer to the rear of the rocket, where the igniter would be attached just prior to launch.

A pro/con list for each igniter location was developed, and rear internal ignition was selected.

	Pros	**Cons**
Forward Internal Ignition	Minimal signal wire length Minimal interference with other internal components Allows self-ignition Allows automatic pre-ignition checks by flight computer	Difficult to access in case of igniter failure Safety concerns regarding pre-launch transportation with igniter on board Requires additional hole in forward closure Possibility of fastener melting Possibility of fastener leak during burn
Rear Internal Ignition	Allows self-ignition Allows automatic pre-ignition checks by flight computer No need for additional fastener Ease of access in case of igniter failure	Longer signal wire length Possible interference with other internal components
External Ignition	No interference with internal components No need for additional on-board fastener Ease of access in case of igniter failure	No pre-ignition checks by flight computer No self-ignition Longer signal wire length

Guest User: This sounds like the igniter was installed prior to arming the flight computers? If this was the case, the problem could have been solved by revising and using a checklist to arm the flight computer prior to installing the igniter.

For safety reasons it is common to only install the igniter after arming the flight computer. the few exceptions I have seen are for multi-stage rockets. And in these cases onboard ignition has only been used for the upper stage motors.

Guest User: Before or after turning on and arming the flight computers?

Guest User: What pre-ignition checks are you talking about?

Why is self-ignition a benefit?

	Substantially greater current can be applied	

Rear internal ignition was chosen since it offers the benefit of internal ignition while avoiding the mechanical complexity of and mitigating the major risks associated with forward internal ignition. However, since some launch sites require external ignition, external ignition compatibility will be maintained to meet this requirement.

6.4. Key Technical Issues/Risk

6.4.1. Major Identified Risks

6.4.1.1. Erosive Burning

Solid propellant motors of this size can exhibit the phenomenon of erosive burning. As the mass flux through the core of the motor increases, parts of the propellant grain can be broken apart by the exhaust. These broken pieces of the grain have a very high surface area to volume ratio and therefore significantly increase the chamber pressure and thrust of the motor. Should these increases be massive enough, the motor case can overpressurize.

Erosive burning itself is not necessarily a concern, as it can increase thrust early in the burn, resulting in increased velocity off the rail and therefore increased stability. The grain geometry is intended to take advantage of this, with the taper caused by manufacturing the grain constricting the flow as it passes through the core near the base of each grain. The risk is that the erosive burning is more substantial than anticipated, which, while unlikely, could result in a failure of the motor case.

6.4.1.2. Propellant Density

Due to the amount of variation possible during the fabrication of a batch of propellant, there is a substantial risk of significant variation between grains. This poses 2 considerable risks:

1) If the propellant has a much higher or lower density than expected, the motor may provide too much or too little impulse to carry the vehicle to precisely 10,000 feet AGL.
2) Low density propellants have substantial voids that can result in sudden increases in the burning surface area, increasing the chamber pressure and threatening to overpressurize the case.

6.4.1.3. HCl exposure

When the motor fires, the exhaust is simulated to be 21.4% HCl by mass. Due to the corrosive nature of HCl, this poses a considerable risk to equipment and personnel. In an environment without substantial ventilation, the concentration of HCl in the air can be high enough that the test stand and motor case can corrode if exposed to this atmosphere for a significant amount of time. Likewise, inhalation of high concentrations of HCl can result in damage to the lungs of those who are exposed.

6.4.1.4. Case Failure

One concern that exists with the nozzle is heat transfer into the case. Graphite conducts heat fairly well and will thus transfer a lot of heat to the case. Due to the positioning of the nozzle, much of this heat will transfer first to the threads on the thrust ring. Should these soften, the thrust ring could fail under pressure, resulting in the motor ejecting the nozzle. Alternatively, the threads on the case and the thrust ring may meld together, effectively welding the thrust ring to the case.

6.4.1.5. Igniter Failure

Possible failure modes were considered during igniter development. For one, the electric match may fail to fire, or the pyrogen may fail to ignite the propellant. The second scenario was mitigated by switching casting methods and adding two layers of pyrogen. In either case, these failures are easily remedied by installing another igniter in the rocket motor.

Guest User: There are probably more risks associated with your chosen ignition system. I'd like to see them explored further.

Below is a diagram detailing the probability and impact of the various risks.

	20%					
	10%		4		2	
Risk	5%	7		6		5
	1%			3	1	
	0.1%					
		1	2	3	4	5

Risk ID	**Risk Description**
1	Erosive burning causes motor failure
2	Propellant density variation over/underpowers the motor
3	Propellant density variation causes motor failure
4	Equipment exposure to HCl
5	Personnel exposure to HCl
6	Heat transfer from the nozzle causes the threads to fail
7	Igniter does not fire

6.4.2. Mitigation Plan

Due to the substantial impacts possible from the risks outlined in the previous section, plans are in place to mitigate each of these risks and protect all team personnel. These mitigations are detailed below.

6.4.2.1. Erosive Burning

In order to predict the behavior of the motor, simulation softwares are used. The software the team has used up until this point is Burnsim. This software allows custom propellants and grain geometries to be simulated to estimate the properties of the motor. However, this software has no mechanism for simulating erosive burning. As such, work is in progress on writing simulation software that accounts for erosive burning, allowing simulations to be run prior to motor combustion. In addition, substantial safety procedures will be observed during all testing to protect team members in case of motor failure.

6.4.2.2. Propellant Density

In order to mitigate the risk inherent in the variation between propellant densities, a few measures have been put in place.

1) The case length was sized to provide 10000 Ns of impulse with a propellant density of 1500 kg/m^3. Motors can be tuned to provide the necessary impulse based on the measured density of each grain, and spacers can be used to hold the forward closure down if a shorter motor is needed.
2) The density of each propellant grain is checked once the propellant is cured, and a propellant grain with a density below 1500 kg/m^3 will not be accepted for a test motor.
3) Excess propellant is cast into each grain so that sections near the end of each grain that are particularly uneven or full of voids can be trimmed away.

The margins of safety that the case has on operating pressure should be sufficient that the case will not fail at densities above 1500 kg/m^3.

6.4.2.3. HCl exposure

Mitigation for this risk has 3 major steps:

1) Increase ventilation: testing outside at a launch field or a test range will keep the concentration of HCl in the surrounding atmosphere well below acceptable limits.
2) Thoroughly clean equipment: after each test, the case, test stand, sensors, and other equipment will be thoroughly cleaned with cleaning wipes and solution.
3) Use respirators: in situations where the ventilation is not ideal, team members approaching the motor must wear a respirator with filters that are compatible with HCl.

6.4.2.4. Case Failure

Without more sophisticated FEA modelling, it is unlikely this situation will be able to be modelled prior to testing. As such, thorough safety procedures will be followed during every test, with tests occurring in either a blast chamber or outside while observing 1.5 times the NAR recommended minimum safety distance.

Should the case fail, a second case will be maintained ready for use to prevent schedule slip. Design work will also need to be done to insulate the case from the nozzle.

Should the thrust ring fuse with the case, the case will continue to be used. Assembly and disassembly of the motor will be done from the front end of the case. While this situation is not ideal, after the two parts weld together, heat transfer from the nozzle is no longer as significant a concern.

6.4.2.5. Igniter Failure

This is a minor enough risk that no mitigation plan beyond having replacement igniters is required.

In summary, after applying the above mitigation strategies, the risk matrix shows that all risks are within acceptable bounds. Certain risks cannot be mitigated in impact, but testing and the design mitigations discussed above can lower the probability of these risks occurring to acceptable levels.

Risk	20%					
	10%	4				
	5%	7				
	1%			6		
	0.1%			1, 3	2	5
		1	2	3	4	5

6.5. Interfaces

6.5.1. Ground Support Equipment

The propulsion team interfaces heavily with GSE during the development of the flight motor. In order to test and characterize motors of all sizes, test stands and a Data Acquisition Computer (DAC) is needed. GSE will be designing and manufacturing equipment and sourcing sensors for static fire tests of the motor. These include thermocouples, pressure transducers, and load cells that are appropriate for the size motor that is being tested.

6.5.2. Structures

One of the most important interfaces the propulsion system has with the rest of the vehicle is the fin can. This structural component holds the fins in alignment for aerodynamic stability, provides scaffolding that aligns the motor with the axis of the rocket, and transfers the thrust from the thrust ring of the case to the rest of the vehicle.

In order to be compatible with commercial propulsion systems, the mount for the motor needs to be able to support a motor case with a diameter of 98mm, with positive retention using a ⅜”-16 threaded rod that is attached to the forward closure of the case. As such, this interface sets the outer diameter of the case up until the thrust ring and determines the mounting architecture on the forward closure.

6.5.3. Avionics

If the igniter is fired from inside the rocket (i.e. Rear Internal Ignition is used over External Ignition), avionics will provide the command for the rocket to fire. The propulsion system will therefore take 1 pyro channel from Pyxida for ignition.

6.6. Going Forward Plan

6.6.1. Development Test Goals

The goals for motor testing are as follows:

- Characterize the chosen propellant formulation, determining the burn rate coefficient and exponent
- Verify that propellant combustion is stable as the amount of propellant in a motor increases
- Verify that propellant grain manufacturing is consistent and reliable
- Characterize the fidelity of simulation models
- Determine the expected thrust profile of the motor

In order to meet these goals, the motor will go through 4 stages of testing. These are:

- Strand burning
- 54 mm 1-grain motor
- 98mm 1-grain motor
- Flight-like 98mm 4-grain motor.

Each of these stages will be discussed in more detail below.

6.6.1.1. Strand Burning

In strand burning tests, short, thin samples of propellant are cast and burned at various pressures using a Crawson strand burner. Each strand contains roughly 10 grams of propellant.

By burning the propellant at different pressures and measuring the burn rate of the strand, the relationship between burn rate and chamber pressure can be determined. This relationship takes the form:

$$r = a * P_c^{\,n}$$

where r is the burn rate in mm/s, P_c is the chamber pressure, a is the burn rate coefficient, and n is the burn rate exponent. Determining the burn rate coefficient and exponent are important in simulating how the motors using the propellant behave as well as in analytically determining the properties of various motors.

To date, similar formulations have been tested in the strand burner, providing a burn rate coefficient of

$$a = 4.19\ mm\ s^{-1}(MPa)^{-n}$$

and a burn rate exponent of

$$n = 0.428$$

These values have been used for initial simulations. However, the final formulation has yet to be characterized in the strand burner due to malfunctions with the equipment.

6.6.1.2. 54mm 1-grain motor

This test involves mixing a 54mm diameter motor with 250g of propellant and successfully static firing it. In order for a test to be successful, thrust data for the motor should be collected in some manner and be similar to the simulated thrust curve. Should the empirical thrust curve differ substantially from the simulated thrust curve, further analysis will be needed to determine the cause of the difference.

The purpose of this test is to verify the propellant in a static fire situation as well as to establish good manufacturing and casting techniques. The accuracy of the team's simulation software can also be verified in this test. In order to not confound propellant function with case function, a commercially available case will be used.

To date, 6 motors of this size have been produced, with the following densities:

Grain number	Batch number	Density [kg/m^3]
54-1	1	1533
54-2	1	1539.1
54-3	2	1650.3
54-4	2	1664.4
54-5	2	1654.4
54-6	2	1654.4

While these motors have yet to be tested, the progressive increase in density suggests that the fabrication process has been improved over the last couple batches and further improvements can enable grains with a density of 1750 kg/m^3.

6.6.1.3. 98mm 1-grain motor

Upon successful firings of a 54mm motor, a 98mm motor with 1.2 kg of propellant shall be produced. This motor shall be static fired at an adequate facility. For this test to be considered successful, the motor must produce a thrust curve similar to the simulated thrust curve for this motor. Should the empirical thrust data differ significantly from what is expected, further analysis will determine the appropriate course of action.

The purpose of this test is to establish consistent mixing, vacuum processing, casting, and post-processing procedures to produce reliable propellant grains of this diameter. The case design can also be verified at the flight diameter in this round of testing. Establishing reliable manufacturing techniques in this stage is essential, as these motors will be the grains used in the flight-like motor.

6.6.1.4. Flight-like motor

Once several successful 1 grain tests are complete, manufacturing of flight motors shall begin. These motors will be a stack of four 98mm grains, as manufactured in the previous tiers. The total propellant mass for a flight motor shall be 5.2 kg. This is the largest amount of propellant that the mixing equipment is capable of processing. Casting and post-processing shall be performed in the same way as in the previous tier of testing. A couple days before the motor is to be fired, the grains shall be bonded into the liner to ensure that they are not ejected from the motor during the burn.

The purpose of this test is to verify that the motor and casing are ready to be flown on the vehicle. This test shall be conducted with a flight-ready case with the addition of some sensors for data collection. The expected thrust curve and accuracy of simulations should be determined by the end of this round of testing.

6.6.2. Test Schedule

Week	11/13	11/20	11/27	12/4	12/11	FINALS	1/8	1/15	1/22	1/29
Strand Testing										
54mm 1 grain										
98mm 1 grain										
Flight-like motor										

Propellant characterization and initial motor tests will be finished in the next 2 weeks, allowing for time to fix flaws in the equipment. 98mm grains will begin to be manufactured on the week of November 13th, allowing for tests to commence as early as the week of the 20th. Over the Thanksgiving break, should a sufficient number of grains be produced and pass density requirements, tests can be conducted at a launch field in New York during the team's first attempt at a test launch. It is possible that the full set of 98mm 1 grain motors can be tested at this time. Should this be the case, additional grains will be cast in the remaining weeks of the semester, allowing for testing of a flight-like motor immediately in January. No testing will occur during the week of final exams or over the Christmas holiday.

7. Ground Support Equipment

7.1. Overview of Requirements

From the Combined Requirements Table in Appendix A:

6.4.1	DTEG 2.4	Propulsion testing shall comply with ESRA requirements.	Propulsion, GSE
6.5	DTEG 2.4.3	Propulsion shall have a successful static fire test prior to a test launch. Propulsion should have two successful static fires prior to a launch.	Propulsion, GSE
7.0	DTEG 10.1	All Ground Support Equipment shall be man-portable over a short distance (~500 feet).	GSE
7.1	Internal	The launch-ready pad lifetime of Raziel shall be at least 2 hours.	GSE

These requirements are manifested in GSE's design and construction of a solid rocket motor test stand (6.4.1 and 6.5) and rocket cooling system (7.1). Requirement 7.0 is inherent to the design of various GSE components. The GSE subteam is also designing a self-controlled, 2 degree-of-freedom antenna ground station to reliably receive real-time telemetry from the rocket. This will enable safe and timely recovery of the rocket after landing. GSE also plans to acquire a trailer for team use during launches. The trailer will provide a base to enable smooth and timely integration of the rocket onsite.

7.2. Design Process

7.2.1. Solid Test Stand

The GSE subteam has focused most efforts on designing and constructing the solid test stand based on the needs of the propulsion subteam for a testing platform.

From the start, vertical inverted motor orientation was chosen for ease of construction and simple thrust measurement. The inverted orientation (nozzle firing into the air) was chosen to minimize risk to persons and property in the event of a forward enclosure ejection.

One of the driving factors of the design was the capability to use the same stand for various diameters and lengths of motors: 54mm 1-grain motor, 98mm 1-grain motor, and 98mm full-scale motor (4 grains). The solid motor test stand was inspired by a vertical-fire test stand design[17] found during online research. Dimensions for the test stand were driven by the case size of the largest motor (98 mm 4-grain motor) and the safety requirement that the majority of the casing be safely contained by the test stand in the event of a burn-through or uncontrolled propellant burn. Several sets of centering rings are used in the test stand in order to accommodate different load cells and motor sizes within the same test stand. The plate to which the load cell is mounted can be adjusted to different heights such that the nozzle of each motor under test will protrude just enough to be visible to surrounding cameras.

Several designs for the load cell plate and centering rings were considered, including flat plates, stool-like structures, and solid aluminum pieces. The team decided to pursue flat plates based on low material costs, reliable availability of materials, good machinability, and ease of mounting the centering rings and load cell mounting plate into the test stand. The centering rings and load cell mounting plate are constrained vertically and radially in the test stand with 3 and 6 ¼-28 bolts, respectively. The centering rings will experience negligible load levels from the motor firing, permitting a symmetric 3-bolt radial pattern to provide symmetric and fault-tolerant positioning. The load cell plate will be experience the full thrust of the motor. Anticipated loads vary from approximately 30 to 800 pounds, depending on the motor being tested. ¼-28 bolts have a shear strength of about 4000 lbs, so 6 bolts can withstand approximately 24000 lbs of shear, giving a safety factor of approximately 1.5.

7.2.1.1. Data Acquisition System

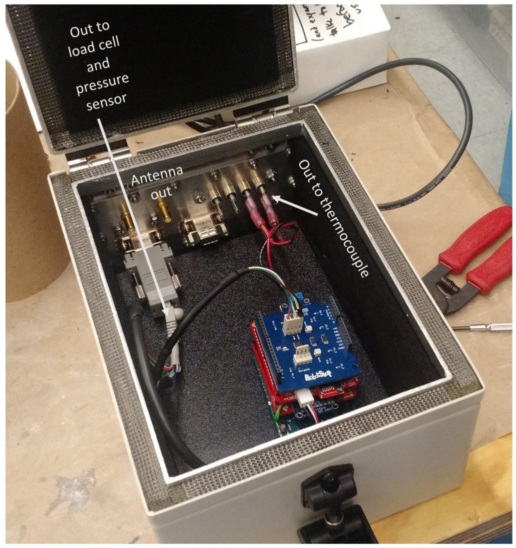
Reliability and simplicity of the Data Acquisition (DAQ) System are essential to ensure accurate and consistent motor test data. The basic architecture of the DAQ system is an Arduino with various amplifiers and breakout boards to collect data from a load cell, pressure sensor, and thermocouple. At a basic level, the DAQ system must capture and save the data read from these sensors. Because of the risks of the test stand, the DAQ system must also be rugged and contain a failsafe in case the SD card onboard the system is damaged. To do this, the system is designed to simultaneously transmit the sensor data to a computer at the base station in real time, and the entire DAQ system itself will be stored in a protected DAQ box.

The load cell signal needs to be amplified, likely with adjustable gain to fine tune the readings. The pressure sensor also needed a high precision amplifier. At first, the thermocouple amplifier was supposed to take K type wires, but rough estimates temperatures given by propulsion were as high as 2500 degrees Kelvin. Even the highest temperature S or R type thermocouples only go up to roughly 2000K, so temperatures readings may either need to cut off at this 2000K range, or be discarded completely. High-temperature probes or non-contact methods would be too bulky to fit inside the motor, but may work for an exterior mount.

For first design of the DAQ, an Arduino Leonardo was chosen as the microcontroller because of its extra peripherals speed compared to the Uno. The subteam also chose a 50 lb, 200 lb, and 1000 lb Wheatstone Amplifier Shield load cells, an ADS1115 16-Bit ADC for the pressure sensor, and a Thermocouple Amplifier MAX31855. These are all reliable, and fairly low cost

[17] Inspiring Design by Dewayne Doud, http://aeroconsystems.com/cart/test-stand-pictorial

components that integrate well with the Leonardo. For live wireless data transmission, an XBEE Pro module and an XBEE shield were chosen, and simply stack on top of the Leonardo. Lastly, a Sparkfun OpenLog was selected for SD logging. This simplified the data-logging code and moved the SD writing off the Leonardo's processor and to the OpenLog itself.



DAQ System in the DAQ Box

7.2.1.2. Ignition System

The purpose of the ignition system is to supply power to the igniter in order to fire the rocket motor propellent. The wires attached to the igniters from the ignition system must have sufficient current flowing through them to cause the pyrogen to combust, which will then ignite the solid rocket motor propellant.

The ignition system is designed to avoid accidental ignition, ignite when the ignition system is properly connected, and minimize launch failure by effectively delivering battery power to the igniter.

The system is powered by a 12V battery connected to an LED toggle switch, a spring-loaded launch button, and the igniter itself. The LED toggle switch acts as both a safety switch before the launch button as well as a continuity test. The purpose of the safety switch is to prevent accidental ignition of the rocket with a single press of a button. The continuity test allows users to ensure that there are no problems with the circuitry before launching the system. The igniter is then attached separately to the ignition system through alligator clips that are extended from a 300 ft wire in order to keep users as far away from the launch site as possible.

7.2.2. Antenna Ground Station

The antenna ground station is a gimballing structure that autonomously tracks the rocket in flight to gather live telemetry data. The system will need to automatically control a 2 degree of freedom (DOF) gimbal to point antennas and other accessories at the rocket in flight. At present, The Team has 2 yagi antennas which were successfully utilized with Project Therion. The antenna ground station is designed primarily to facilitate the automatic pointing of these specific antennas. As a secondary goal, a universal accessory tray is being incorporated into the gimbal assembly to allow future instruments to easily be added. The antenna brackets will be a permanent structure with adjustable counterweights.

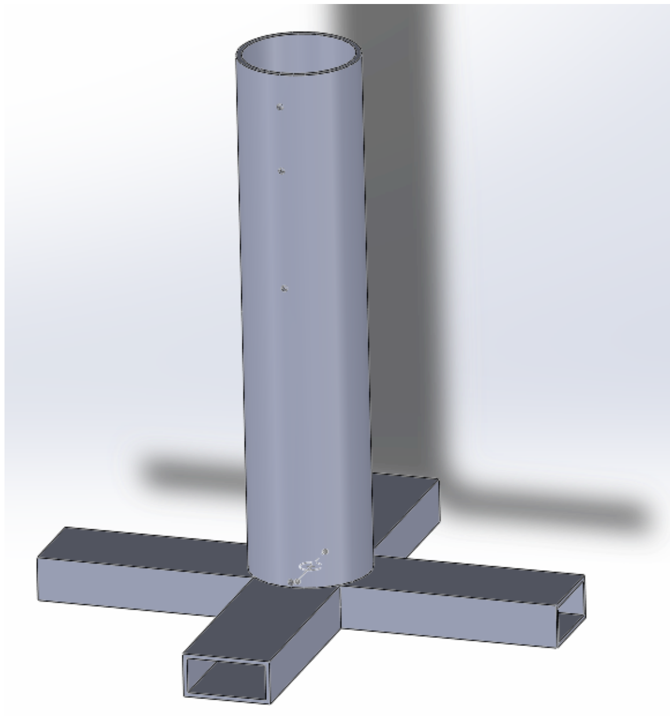
The accessory tray and antenna brackets will pivot up and down on an axle, which will be driven by a timing belt and servo motor. A single, solid axle was considered, but such a design would eliminate the ability to position accessories at the center of rotation in vertical direction, which may be desirable for some instruments. The vertical supports will be mounted to a platform which is rotated about the vertical axis by a second timing belt and servo motor. The use of timing belts allows considerable flexibility in choosing motor position, gearing ratios, and by extension, sizing motor torques and speeds. The using of timing belts rather than v-belts guarantees no slippage between the servo motors and driven elements. The advantage of timing belts over sprockets and roller chains are reduced cost and weight. Using a gear-driven design has proven to be far too costly.

The use of v-belts and allowing slippage was considered, but this would introduce the need for custom servo motor control hardware and software. While this is very feasible, it will not be worth the additional manpower.

The controller will utilize a combination of reported telemetry from the rocket's sensors, along with simulation data in an unscented Kalman filter to estimate the rocket's relative position. As a stretch goal, active sensing elements such as passive infrared (IR), radar, laser detection and/or ranging (LADAR) might be used to determine the rocket's relative positioning more reliably.

7.3. Technical Design/Analysis

7.3.1. Solid Test Stand

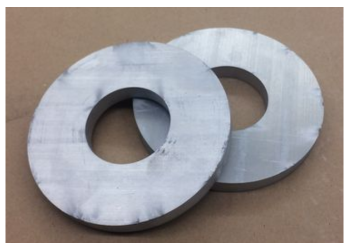


CAD Model of Solid Motor Test Stand Structure

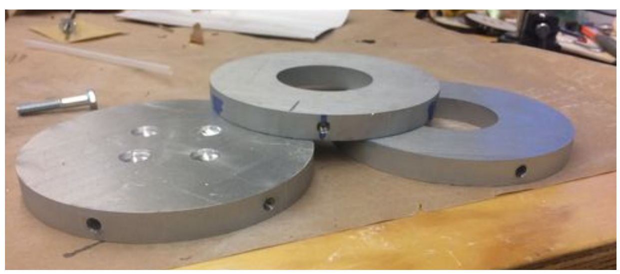
The solid motor test stand is comprised of a TIG-welded steel structure, two sets of two centering rings each (one set for a 54 mm motor and one set for a 98 mm motor), and a load cell mounting plate. Extra stability will be provided by sandbags on each leg of the base.

Welded Solid Test Stand Structure

The solid motor test stand is constructed from .25” thick steel. The base is made from two rectangular tubes with a 2” by 5” profile and 0.1875” wall thickness. Each leg is 12” long. The tall cylindrical tube is 24” tall with a 5.05” inner diameter. The 54 mm centering rings are .5” thick, 2.126” inner diameter, and 5.04” outer diameter. The 98 mm centering rings are .5” thick, 3.879” inner diameter, and 5.041” outer diameter. The load cell mounting plate is .5” thick and 4.997” diameter. The load cell mounting plate features a series of holes for mounting various load cells, as well as two “finger holes” to allow for the plate to be easily manipulated and set up inside the test stand. A series of holes are drilled in the steel body tube to enable the use of centering rings and the load cell mounting plate. The centering rings are secured with 3 radial holes (spaced evenly around the circumference of the tube), and the load cell mounting plate is secured with 6 evenly spaced radial holes.



Load Cell Mounting Plate, 54 mm and 98 mm Centering Rings During Machining



Load Cell Mounting Plate and 54 mm Centering Rings with Radial Holes

For the 54 mm motor, the two centering rings are mounted 1.5” and 4.5” from the top of the circular steel tube. The load cell mounting plate will be mounted 9.15” from the top of the tube (currently, holes are drilled 10” below the top based on originally estimated dimensions; a new series of holes will be drilled at 9.15” below the top).

For the 98 mm motor, the two centering rings are mounted 1.5” and 10” from the top of the tube. The load cell mounting plate will be mounted 12.25” below the top of the tube. (The holes for the load cell mounting plate will also be added.) For the full-scale 98 mm motor, the two centering rings are mounted 1.5” and 10” from the top of the tube, and the load cell mounting plate is constrained radially at the bottom of the steel tube.

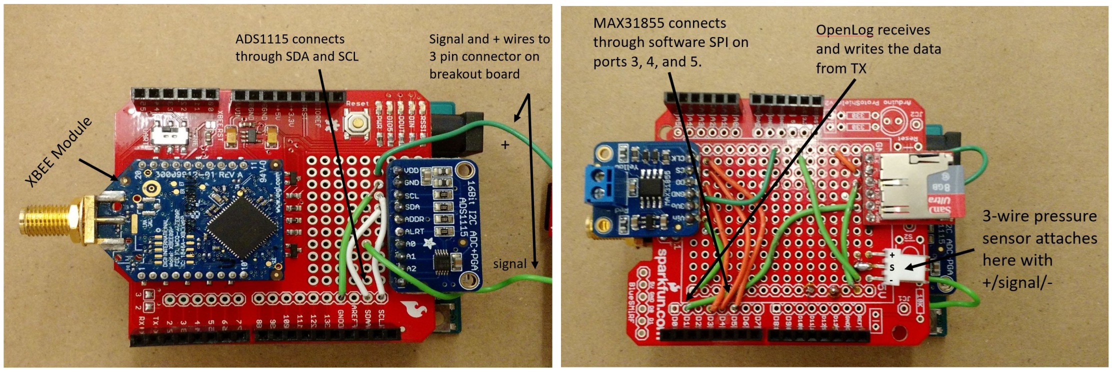
The 54 mm 1-grain motor will be tested on a 50 pound load cell. The 98 mm 1-grain motor will be tested on the 200 pound load cell, and the 98 mm 4-grain motor will be tested on a 1000 pound load cell. The 98 mm motors will also have a pressure tap and thermocouple integrated in the forward closure.

Solid Test Stand Mock Integration with Load Cell Mounting Plate and Load Cell

7.3.1.1. Data Acquisition System

Assembling the DAQ system, the XBEE shield stacks first, followed by a breakout board shield and the Wheatstone shield. This XBEE shield includes pins for SCL and SDA ports, which are used for connecting the ADS1115 module. The pressure sensor signal wire from the ADC connects to the breakout board shield. The MAX31855 connects through SPI, which was convenient because of the pin layout on the Leonardo.

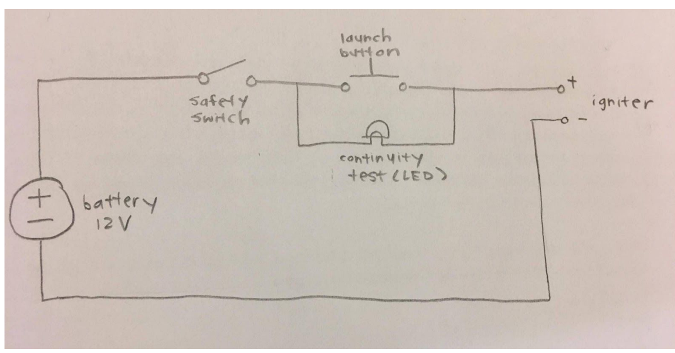
When first initializing, the serial port from the Leonardo prompts the user to calibrate the load sensor. If no calibration is needed, the original calibration is used. If calibration is needed, a known load is placed in the sensor rig, and the program rewrites force and analog read constants used for the linear interpolation. Additionally, the program writes a header to the serial port with the time and reading type (load, pressure, temp) of each number that will be recorded. Essentially, the Leonardo takes readings from each of the sensors every time interval and outputs each reading on the serial port. Once there, the XBEE outputs this same data on a computer, and the OpenLog writes this data to the SD card as a text file, which can be converted to CSV. The OpenLog takes care of file creation and file saving with an onboard system.



DAQ Arduino Board with Breakout Board

7.3.1.2. Ignition System

The ignition system consists of four parts: the 12V battery, LED toggle switch, the launch button, and the igniter. Current flows from the 12V battery, through the LED toggle switch, then to the launch button, and finally to the igniter, which then connects to the battery's ground voltage, thereby completing the circuit. In order for energy to reach the igniter, there must be a complete electrical connection between one end of the battery, through the igniter and pyrogenic material, back to the other end of the battery.



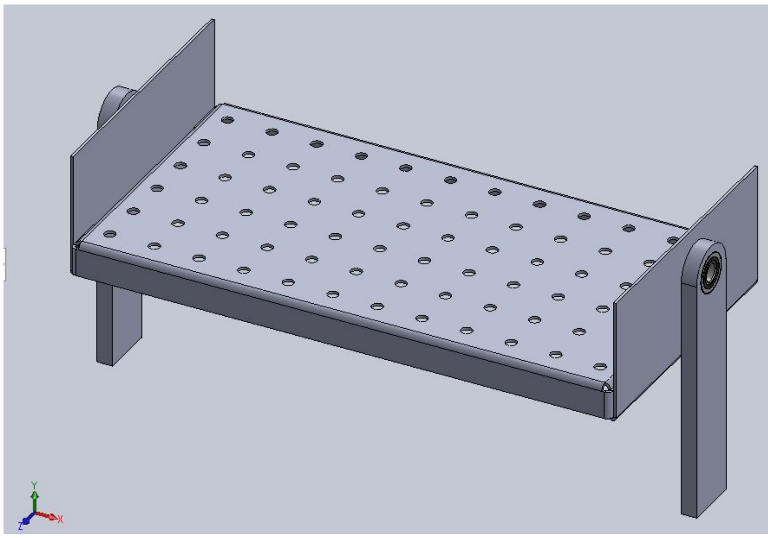
Ignition System Schematic

Current is provided by the 12-volt battery, and is first verified by the LED toggle switch. If the switch is not first turned on, the circuit remains open and no current is able to flow through. This acts as a safety and inhibits unintended ignition in the event that the launch button is pressed prematurely. When the LED switch is toggled, the LED lights up, confirming that the circuit is closed, that there are no problems with the circuitry, and that the ignition system is ready for launch. The current passing through the LED is not enough to heat up the igniter because of the LED's high resistance.

When the launch button is pressed, a low resistance circuit is closed, which allows high current to pass through the igniter. The high current through the ignitor heats up the pyrogen, thereby igniting the rocket motor.

7.3.2. Antenna Ground Station

The gimbal mechanism will consistent of three assemblies: a stationary platform, an assembly which pitches up and down (primary rotation), and a turntable assembly which rotates about the vertical axis (secondary rotation). The structure of the stationary and secondary rotation assembly will be made mostly of aluminum. The primary rotation assembly will consist of a 12” x 6” single-piece stainless steel accessory tray mounted to 2024 aluminum keyed shafts on either end. The shafts will pass through the bearings in the secondary assembly and terminate at the yagi antenna brackets on either end.



Accessory tray and supporting arms

A timing pulley will be mounted to the shaft which is on the same side as the 450 MHz antenna. The mating pulley and servo motor will be affixed to the secondary gimbal assembly. The driven pulley for the secondary assembly will be mounted to the bottom of the assembly, and mates to a servo motor via timing belt. This servo motor will be affixed to the stationary assembly.

The technical specifications for the servo motors, belts, and controller housing are to-be-determined at this time.

7.4. Key Technical Issues/Risk

7.4.1. Operational Failure Modes

1. Cooling system
    1.1. Structural failure
    1.2. Retraction failure
    1.3. Simultaneous retraction and interlock failure
2. Antenna ground station
    2.1. Vertical slew
        2.1.1. Degraded
        2.1.2. Fail

2.2. Horizontal slew
  2.2.1. Degraded
  2.2.2. Fail
2.3. Controller failure
2.4. Downlink failure

3. Trailer
   3.1. Structural failure
   3.2. Mechanical failure

Unmitigated Operational Risk Table						
Risk	20%					
	10%	1.2	2.1.1, 2.4			
	5%		2.1.2, 2.2.1		3.2	
	1%		2.2.2			
	0.1 %			1.1	1.3	3.1
		1	2	3	4	5
		Impact				

7.4.2. Testing Failure Modes

1. Solid test stand (During test firing)
   1.1. Superstructure failure
   1.2. Internal structure failure
   1.3. Sensor failure
   1.4. DAQ failure
   1.5. Ignitor malfunction
      1.5.1. Unintended ignition
      1.5.2. Failure to ignite

Unmitigated Test Risk Table						
Risk	20%					
	10%	1.5.2		1.4		
	5%					1.5.1
	1%			1.3		
	0.1%				1.2	1.1
		1	2	3	4	5
		Impact				

7.4.3. Operational Risk Mitigation

1. Cooling system
   a. The system will be designed with a safety interlock which will interrupt ignitor continuity until it is stowed.
2. Antenna ground station
   a. Quick-disconnect pins can be included to disengage the servos so the gimbal can be pointed manually.
   b. The system will include an independent flight model which will be used in tandem with a pointing spiral to reacquire signal in the event of downlink failure.
3. Trailer
   a. The trailer will undergo a thorough inspection before departure and during routine stops while en route to launch sites.

7.4.4. Test Risk Mitigation

1. Solid test stand
   a. Wireless transmission provides redundant data in the event of catastrophic motor failure.

b. Each new data capture series is saved to a new file, protecting old data in the event of power loss.
c. The safety switch shall be confirmed to be in the ‘safe’ position before personnel approach the rocket motor.

7.5. Interfaces

7.5.1. Summary Chart (Organized by Subteam)

Interface with:	Scope/Project
Propulsion	Solid Test Stand, Trailer
Structures	Rocket Cooling System, Trailer
Avionics	Rocket Cooling System, Antenna Ground Station, Trailer
Payload	Trailer
Recovery	Antenna Ground Station, Trailer

7.5.2. Organized by Project

7.5.2.1. Solid Test Stand

The main purpose of the test stand is to provide a testing platform for experimental motors developed by the propulsion subteam. More specifically, it provides:

- Lateral and vertical support for experimental motors during testing.
- A mounting point for a temperature sensor (i.e. thermocouple).
- A mounting point for a pressure sensor.

It is GSE’s responsibility to provide a reliable test apparatus to house and test-fire said rocket motors. Furthermore, the test stand should provide consistent test conditions so the propulsion team can obtain reliable and repeatable data, subject to their own constraints. The two sensors (thermocouple and pressure sensor) will provide combustion chamber data to be passed back to the propulsion subteam. GSE will be responsible for providing the necessary hardware for measurements and data collection and handoff. Data analysis will be conducted by the propulsion subteam. It is the propulsion subteam's responsibility to modify the forward enclosure or other motor casing hardware as necessary to facilitate the inclusion of these sensors. All reasonable measures possible will be made by the GSE subteam to limit the need for such modifications.

7.5.2.2. Rocket Cooling System

The cooling system design will require input from all other subteams, particularly structures and avionics. Although cooling of the entire rocket would be ideal, it is likely that such a system would be excessively complicated. As such, the GSE subteam should coordinate with other subteams to determine and prioritize cooling needs. Avionics cooling is by far the highest priority, and the GSE subteam should ascertain both minimum and ideal cooling needs. Structural integrity of the rocket is also a high priority. If asymmetric cooling methods are to be used, thermal gradients should be checked by structures to ensure material responses to uneven heating won't be an issue.

If the cooling device uses any kind of moving mechanisms which must clear the rocket before launch, then the GSE subteam must coordinate with launch control personnel to provide a launch interlock. Such an interlock shall prevent the rocket motor from igniting until the cooling device is clear of the rocket.

7.5.2.3. Antenna Ground Station

The primary goal of the antenna ground station is to provide reliable downlink capability for the rocket's radios and telemetry systems. Therefore, the ground station shall provide automatic pointing capabilities for antennas specified by the avionics team.

The gimbal pointing controller will rely in part upon the rocket's self-reported telemetry. In order to facilitate this, GSE will need to coordinate with Avionics to receive and parse radio signals transmitted by the flight computer.

The pointing controller will also use simulated data which will be provided by Structures. It is GSE's responsibility to ensure flight data is satisfactory for the purposes of reliable pointing algorithms.

7.5.2.4. Trailer

The trailer acquired by the GSE subteam will interface with every other subteam by providing valuable long-term storage space, and transportation of equipment and parts to and from launches. Subteams will collaborate to determine the appropriate layout and allocated storage space per subteam within the trailer.

7.6. Going Forward Plan

7.6.1. Solid Test Stand

The solid test stand is ready for the first round of 54 mm motor testing. Configuring the test stand for different sized motors will be as simple as repositioning centering rings, switching out load cells, and repositioning the load cell mounting plate.

The Team is able to test one motor at a time in a blast chamber on campus. However, for ventilation and safety reasons, testing rates cannot exceed more than one motor every hour. Thus, GSE is pursuing alternate off-campus testing locations that will enable the team to safely test motors at a faster rate.

7.6.2. Rocket Cooling System

Following the completion of the solid test stand, the next highest priority project will be rocket cooling. Currently, the team is discussing high-level design considerations. It is clear that whatever design is used, it must move out of the launch trajectory via remote operation at launch, or be statically positioned as not to interfere with the rocket during launch. The following overall designs have been considered include:

- Static, adjustable shade clamped to the launch rail.
- Partial and full clamshell launch rail enclosures.
- Full launch rail enclosure with a penetrable upper cover.
- Forced air cooling for the avionics section.
- Water-cooled fuselage.
- Refrigerated modules.

In coming meetings, the GSE subteam will analyze potential designs with respect to overall subteam goals and the needs of other subteams. Once initial design decisions are made, the cooling system will immediately move into drafting.

7.6.3. Antenna Ground Station

The antenna ground station is being worked in tandem with other GSE projects. It is currently in the physical design stage. The current plan calls for a modular gimbal which will readily accept new bolt-on accessories as deemed appropriate by GSE or other subteams in the future. Once the gimbaling mechanism has been built and characterized, the controlling software will be written.

8. Payload

8.1. Overview of Requirements

The mission payload consists of two distinct systems: a *rover*, and the *payload segment*.

The *rover* must comply with the following requirements:

1. Official requirements
    a. Payload shall weigh at least 8.8 lbs.
    b. Payload shall be removable from rocket.
    c. Payload shall stow within a CubeSat Standard geometry (1U, 2U, 3U, etc.), and mechanically approximate a cubesat while stowed (i.e. roughly cuboidal)
    d. Any radioactive substances within Payload shall be encapsulated and limited to 1μC or less of radioactivity
2. Mission requirements
    a. Rover shall exit payload segment and land safely on the ground
    b. Rover shall withstand pad and launch environments
    c. Rover shall store data, and should transmit data
    d. Rover should make environmental and ground measurements
    e. Rover should drive at least 5ft

The *payload segment* must comply with the following requirements:

1. Official requirements
    a. Deployable payloads shall comply with recovery requirements 5.1-5.7
2. Interface requirements
    a. Payload segment shall allow rover to exit and land safely on the ground
    b. Payload segment shall incorporate and recover the nosecone
    c. Payload segment shall be attached to the recovery tube by #4-40 nylon bolts, and shall provide at least 0.25 in^2 of *ledge* within the body tube for the payload-recovery bulkhead to rest on

8.2. Technical Design/Analysis (Rover)

The rover is being divided into the following subsystems:

- Chassis
- Drive system
- Computer
- Camera
- Nuclear Experiment
- Atmospheric Sensors
- Mechanical Release

Table 1: Mass estimates and volume percentages for rover subsystems. The percent volume is given for percentage of space within the chassis; not all components are mounted in the chassis and are not included in these estimates. Total payload mass meets mission requirements of 9 pounds of payload within a CubeSat outer mold line.

System	**Wheels**	**Shell**	**Nuclear Sensor**	**Environ. Sensor**	**Batteries**
Mass (lbs)	1	1	0.75	0.25	3
% Volume	-	-	25	7.5	30
System	**Camera**	**Phone**	**Motors**	**Chassis**	**Total**
Mass (lbs)	0.5	0.5	1	1	9
% Volume	7.5	15	15	-	100

8.2.1. Chassis

8.2.1.1. Requirements

The chassis must:

- Contain all components necessary for the operation of the rover and execution of the mission
    - Drive motors
    - Wheels
    - Computer
    - Battery
    - Camera
    - Nuclear experiment
    - Atmospheric sensors
    - Mechanical release mechanism
- Be strong enough to withstand the forces subjected to the rover during:
    - Launch
    - Recovery
    - Landing
- Comply with cubesat size requirements
- Be light enough to meet payload weight requirements

8.2.1.2. Design

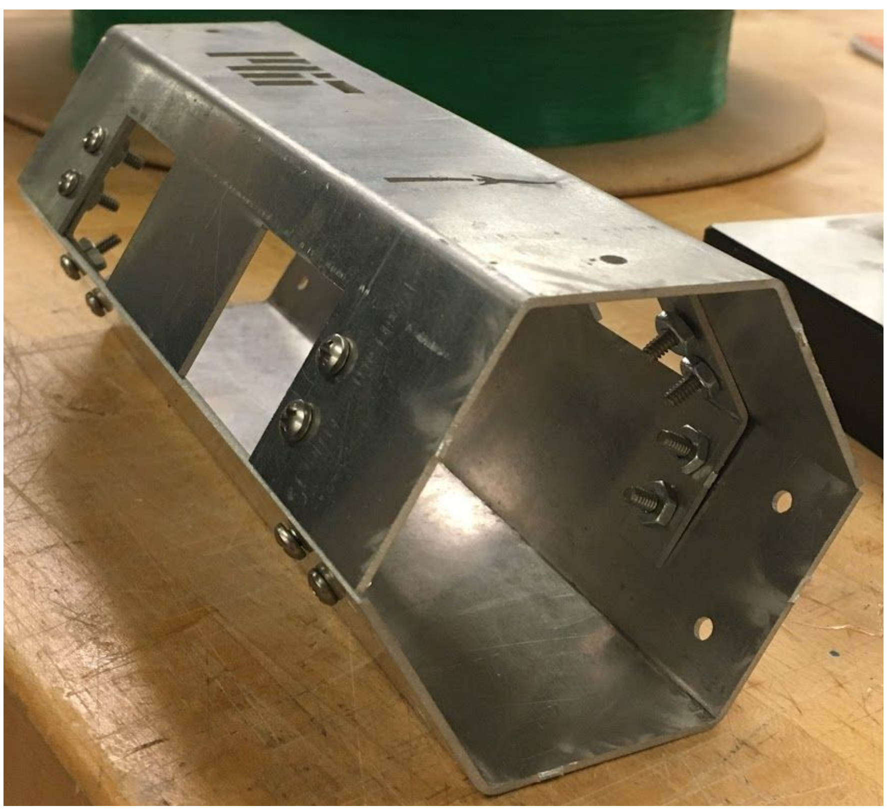
A hexagonal prism design was chosen to take full advantage of the available space and protect the components during launch, recovery, landing, and rover operation. Two 3.93” diameter custom wheels, one on each end of the hexagonal prism, will fit in the cubesat skeleton and maximize use of available space. These designs also curb the danger of landing “upside down” because the rover will still be able to drive as long as both wheels are in contact with the ground.

The diameter of the hexagon is limited by the diameter of the wheels, which in turn is limited by the cubesat requirement and diameter of the rocket. The hexagon’s diameter must also provide enough clearance to traverse obstacles such as pebbles, loose sand, and low vegetation without becoming stuck.

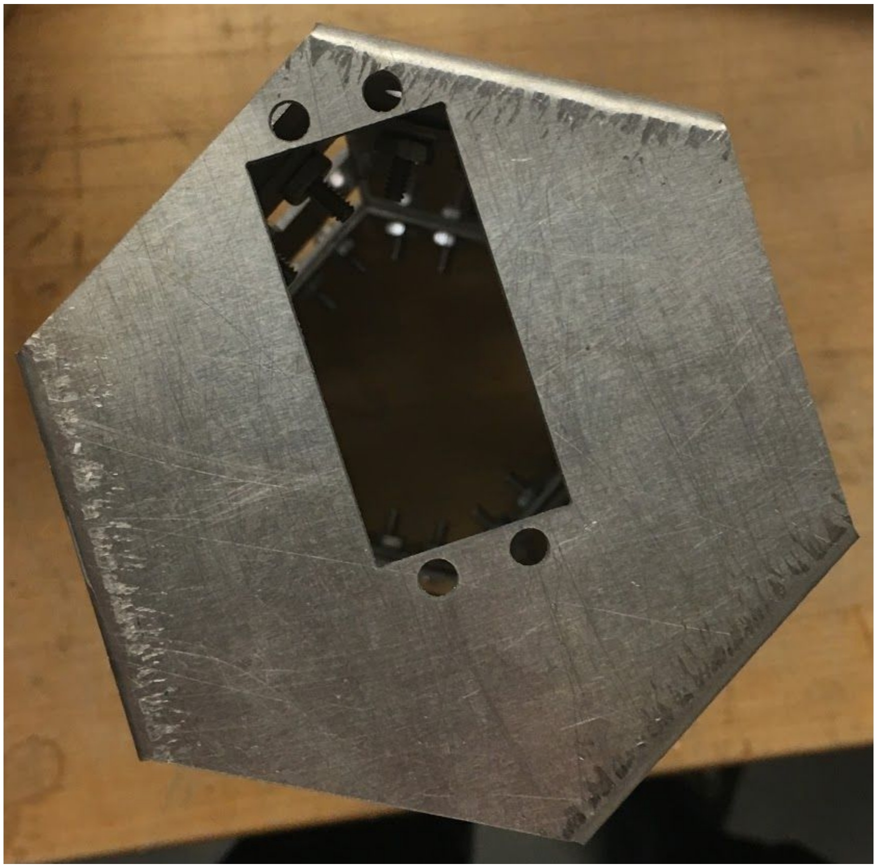
The chassis will be constructed with 1/16” aluminum sheet metal. The hexagon had to be split in two halves, top and bottom, to make sure the sheet metal could be bent into the desired shape on the brake press. Endcaps were designed for each end of the hexagonal prism to provide a mounting location for the drive motors, and brackets provide additional reinforcement between the halves.

The process of bending the sheet metal was not as accurate as predicted, so future designs will need to account for looser tolerances. Additionally, future revisions will have to include access holes to allow easy access to fasteners.

The center of mass of the rover must be below the axle for there to be a torque to drive the rover effectively; this may require the use of ballast.



The two halves of the hexagonal prism, joined by brackets. Access holes allow the tightening and loosening of nuts inside the body when the endcaps are in place (see below).



The endcap on the hexagonal prism, which provides a location to mount the drive motors. The dimensions of the design and bends were not precise enough to allow the endcap to fit into the hexagon, so future designs will need looser tolerances (perhaps by using brackets).

8.2.2. Drive System

8.2.2.1. Requirements

There are four main requirements for the drive system:

- Fit inside cubesat specifications
- Withstand the forces of launch and landing
- Get to a driving position from any landing position
- Drive the rover 5ft
    - Apply sufficient torque to ascend inclines
    - Drive despite rough terrain

8.2.2.2. Design

8.2.2.2.1. Mechanics

The chassis design uses two, 4 inch diameter wheels that are positioned on either end of the rover so the wheels will always be touching the ground. The chassis center of mass must be below the axle to keep the chassis from spinning. The further down the center of mass is, the more torque the motors can supply without spinning the rover.

8.2.2.2.2. Wheels

The wheels will be 3D printed with a combination of flexible and rigid filaments, to help to absorb some of the force of landing. We want the wheels to compress only when the rover is dropped and not during normal driving. The flexible filament will also provide some grip to the wheels, acting like rubber tires and giving the rover more traction. We will test different combinations of flexible and rigid filaments to get the best performance for our purpose.

The two wheel design allows the rover to get to a driving position from any position it lands in. If the rover lands with two wheels on the ground it can simply start driving and the chassis will right itself. If the rover lands on the face of one wheel, it can be programmed to spin that wheel back and forth until the rover tips over onto two wheels.

8.2.2.2.3. Motors

The motors should be able to provide enough torque to equal the torque done by gravity keeping the center of mass down. There is no point in having motors that are stronger than the torque due to gravity, because if the wheels get stuck, the motors will spin the rover, not the wheels.

We discussed both stepper and DC motors. Stepper motors provide very good torque at low speeds and can be precisely controlled, but high speeds they lose almost all torque. They also require additional circuitry to control the steps. DC motors were chosen because they could still provide good torque with a gearbox and don't need as much circuitry or software control.

8.2.3. Computer

8.2.3.1. Requirements

Requirements for the controller include

- Providing adequate power to all systems (with the assistance of onboard battery)
- Processing and relaying RF signals to the drive system
- Processing data and images from atmospheric sensors and the camera
- Processing and relaying data for the nuclear experiment

8.2.3.2. Considerations

Two microcontrollers were considered: the Arduino Due and the Teensy 3.1. Both controllers are functionally capable of the rover's requirements and interface well with third party sensors, but the Teensy beat out the Arduino in the end. While the Teensy and Arduino are very similar in operational capacity, the Teensy is half the cost and allows for coding in 'C', which will allow for more efficient control of the rover's hardware.

The team is considering using an additional controller (a smartphone), because the nuclear experiment requires additional processing. The smartphone could also be used as the wireless receiver and transmitter for the rover.

8.2.3.3. Specifications

The drive system, camera, and atmospheric sensors will be controlled by the Teensy's onboard I/O pins and data pins.

8.2.4. Camera

8.2.4.1. Camera requirements

The camera must:

- Be securely attached within the rover
- Weigh less than 0.5lbs
- Withstand launch and landing conditions
- Obtain pictures without draining the battery

8.2.4.2. Considerations

There are several drone and robotics cameras that are suitable given the space and volume constraints, but it is difficult to determine how a commercial camera will function after a launch.

The major risks for the camera include:

- Breaking on launch or landing
- The rover being oriented such that the camera is face down
- The camera or the motors drawing too much power to complete the mission

8.2.5. Nuclear Experiment

8.2.5.1. Theory and background

Surface analysis has been identified as a goal of the deployable rover. Among the many techniques available for soil analysis, nuclear methods such as x-ray fluorescence (XRF) offer rapid and inexpensive material characterization using small radioisotope sources.

These devices function by subjecting material samples to beams of ionizing radiation and detecting the x-rays emitted back. These re-emitted x-rays are primarily K-shell x-rays, which are the result of electron transitions in the innermost atomic orbitals (1s-2p). These emission lines (and their corresponding cross-sections) are unique and well characterized, and therefore can be used to identify the composition of a sample. This method has been in broad use since the 1950s, and has been employed with outstanding success on all three of the past Mars rover missions. However, broad application of this technique has been hindered by the cost of high-resolution x-ray detectors. Recent improvements in low-cost CZT (cadmium-zinc-telluride) detectors, however, have allowed the development of handheld x-ray fluorescence detectors for use in metallurgical quality control, jewelry valuation and lead paint detection.

Further, intrepid amateurs have demonstrated that XRF/APXS devices can be built using detectors that do not employ direct conversion (like CZT), albeit with limited sensitivity. Several successful efforts have been documented in which commercial thin-crystal NaI(Tl) detectors have been used to ID elements as light as Fe (7.1keV k-alpha). While these devices lack the high resolution of their commercial counterparts, they are nonetheless serviceable for a variety of material ID tasks.

In addition to characteristic x-ray analysis, the backscatter of ionizing radiation can be used to estimate the density and composition of a material. The relationship between the backscattering cross section and atomic number (for beta particles) scales as log(Z)[18], while the density relationship scales linearly. Using XRF data, one can identify the average Z of the soil and use the log(Z) relation to come up with an approximate total backscatter coefficient, then use the observed count rate to determine the approximate density.

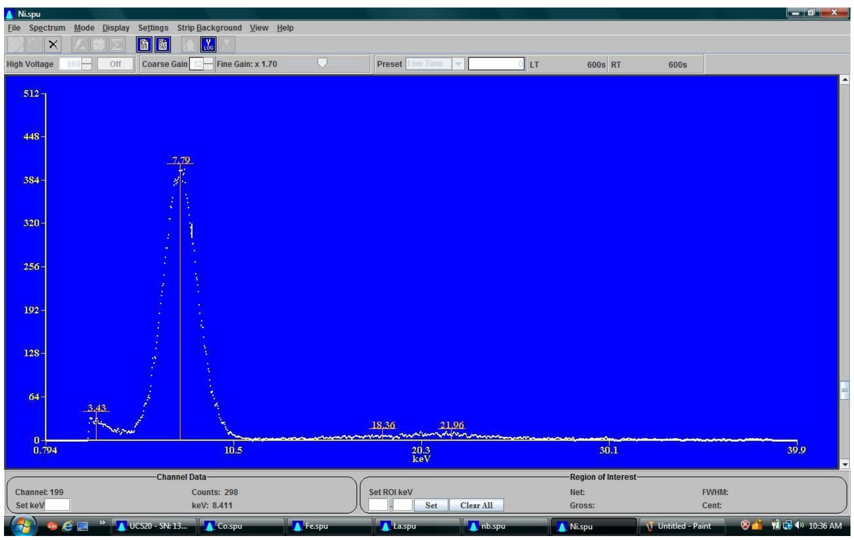


Fig. 0: Pulse-height histogram from a thin-window NaI(Tl) detector being used in a homebuilt XRF device assaying a sample of Ni (k-alpha: 7.47keV). Image from George Dowell, posted on fusor.net on 10/20/2012.

8.2.5.2. Instrument Description

To determine soil density and composition using the backscatter/XRF technique described above, the rover will use a beta source and a scintillation detector arranged to detect both backscattered electrons and x-ray photons. The scintillator (6x6x6mm LYSO) and photosensor (SensL 6mm silicon photomultiplier) have been chosen such that the system will be able to detect k-alpha lines from elements Z>26. Likewise, the location of the scintillator with respect to the source has been chosen such that it can perform rudimentary backscattering studies from a standoff distance of several tens of millimeters. The photosensor and scintillator will be cooled by a thermoelectric cooler (TEC) to reduce thermal noise and prevent gain drift.

[18] Kuzminikh, V. A., and S. A. Vorobiev. "Backscattering of beta-particles from thick targets." *Nuclear Instruments and Methods* 167.3 (1979): 483-488.

The other major component of the system is a ^{55}Fe beta source (3.7MBq) from Spectrum techniques behind a graded-Z shutter which can be remotely controlled. The source is aligned within the collimator such that a beam of beta particles ~10mm diameter impacts the soil approximately normal to the surface below the rover. The scintillation detector is offset from the source axis and aligned such that the axis of the scintillator points towards the area where the beta particle beam impacts the soil (to detect brem and XRF photons).

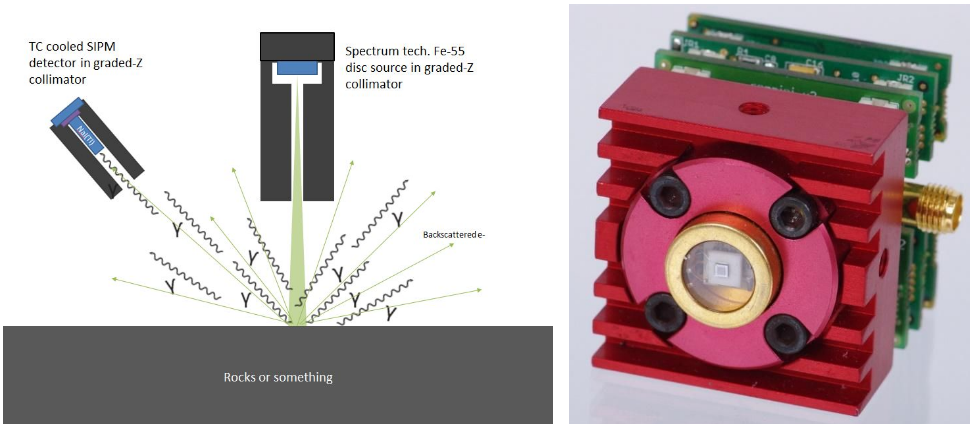


Fig. 1: (Left) Cartoon diagram of the physical layout the source and detector array. The angle between the source and the detector has been exaggerated for clarity. Note that the detector will see both pulses from fluorescence x-rays (labelled “gamma” in the diagram) and scattered electrons. **(Right)** SensL 3x3 C-series SIPM with built in TEC and control circuitry. Image courtesy of SensL.

When the source shutter is open, beta particles impact the soil, scattering and producing photons via bremsstrahlung and x-ray fluorescence. A minority of these particles will scatter such that they impact the detector volume, causing a light pulse proportional to the deposited energy. The SiPM detector then translates each light pulse into a ~100mV, 100ns signal. Pulses from the scintillation detector are then coupled out through a capacitor and processed using a Cremat CR-113 charge sensitive amplifier and a Cremat CR-200 10us shaping amplifier. This translates the ~100mV, 100ns pulses into 1V, ~10us Gaussians. To ease digitization, these pulses will then be stretched using a passive pulse-stretching circuit to ~100mV/~100us. At this point in the circuit, a soundcard connected to a RasPi or a smartphone with an audio jack can be used to digitize and histogram the pulses. While software exists to do this, we may have to develop our own version.

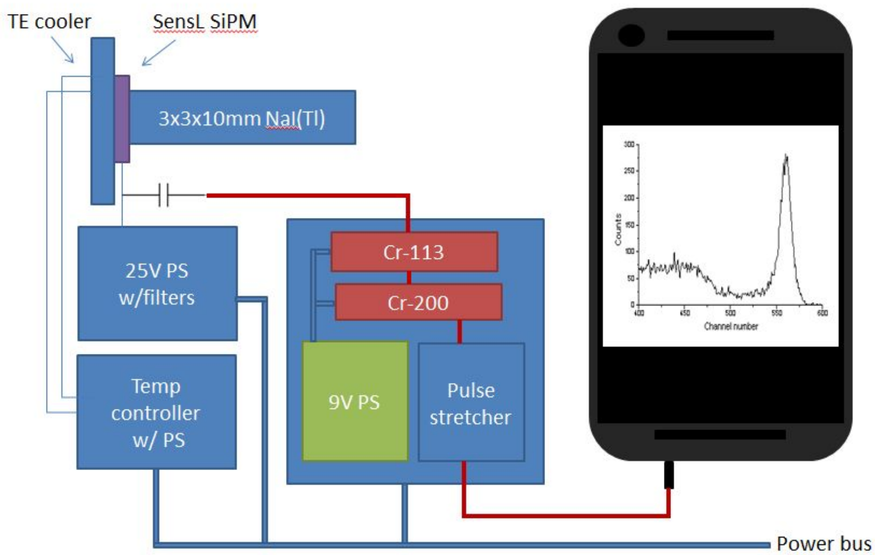


Fig. 2: Rough block diagram showing the electronic hardware needed on the detector side (assuming a phone is used for pulse processing rather than a RasPi/soundcard). Note that blue lines indicate power connections, while red shows the signal path.

8.2.5.3. Instrument Operation

The device operation consists of three distinct phases: blank acquisition, data acquisition and processing. In the first stage, the rover has been deployed and is facing wheels-down on an interesting patch of dirt. The operator (or an automated routine) ensures that the shutter is closed, turns on the scintillation detector and gathers a background spectrum for a specified time. This data is then saved. Next, the rover repeats the prior measurement routine, but with the shutter open. If everything is working correctly, this dataset will have both backscatter information (total counts under the 1/E curve on the pulse-height histogram) and XRF data (the peaks in the pulse-height histogram). Following data acquisition, the rover will verify its connection to the base station, and send both data sets. Operators at the base station will then use MatLab tools to estimate the density and composition of the soil.

8.2.5.4. Performance Simulations

Monte Carlo simulations have been performed to verify that the XRF and backscatter rates are sufficient to build this instrument within the existing constraints on source size and detector volume. These simulations have not yet been extended to test sensitivity to soil composition and density, but exist purely as a proof that we can detect soil and see XRF signals. Further work is in progress to improve the results, but these simulations are extremely time-consuming. For example, the images shown below were the result of 12h of computation time on a laptop.

According to Monte Carlo simulations performed with MCNP6[19], backscattered electrons will make up the majority of the signal, outnumbering photons ~5:1. Analysis based on the following parameters shows the general feasibility of the detector instrument:

- 3.7MBq 55Fe source

[19] X-5 Monte Carlo Team, "MCNP - Version 5, Vol. I: Overview and Theory", LA-UR-03-1987 (2003).

- 0.4pi-sr collimator aperture
- Aperture 3mm above surface
- 6x6x6mm LYSO scintillator
  - 1cm above soil surface, 10cm laterally displaced from beta beam impact site
  - 100% efficient scintillator-detector chain
- Average western soil (from PNNL material database[20])

Mode-e simulations in MCNP6 show that this arrangement will result in a total count rate of 3.8 (+/-.40) counts/sec in the detector volume. This represents a signal-background ratio of ~15 overall, and ~3 for photons alone[21]. Note that this is for the overall signal, and that some regions of the pulse-height spectrum may have significantly better or worse SNRs based on the source of background counts.

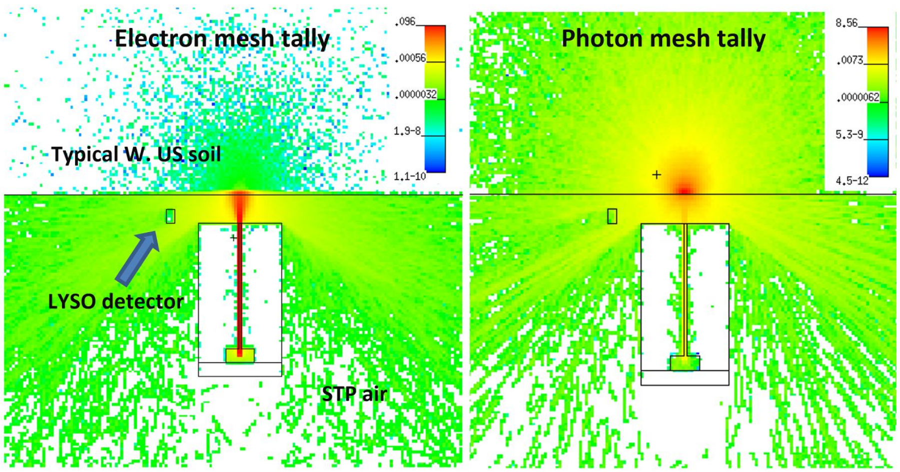


Fig.3: (Left) MCNP6 mode-e mesh for electrons. **(Right)** Photon mesh tally generated by the same mode e input file. Note that in both images, the block to the left of the source collimator is an idealized LYSO detector that represents the active area of the SiPM based device. The units of both are particles/cm^2 per source particle, and are averaged across 2x2x0.1mm voxels. Although detailed simulations will require scheduling NSE cluster time, some initial investigations of pulse height spectra in the detector have been performed using 12h of processor time on a Lenovo X1 laptop. The results indicate that the electron backscatter spectrum (1/E) dominates the photon contribution. Further design work will be done to determine if a plastic beta-shield could be used to separate the two signal sources more clearly.

[20] PNNL material database PDF: www.pnnl.gov/main/publications/external/technical_reports/PNNL-15870Rev1.pdf

[21] This is based on an average background count rate of ~.25/sec, which has been extrapolated from an ongoing experiment with a 3x3x10mm LYSO scintillator attached to a 3x3mm SensL C-series SiPM. This rate can vary based on altitude, soil composition and detector temperature. From limited research on Spaceport America, I do not expect the background count to be more than a factor of two higher.

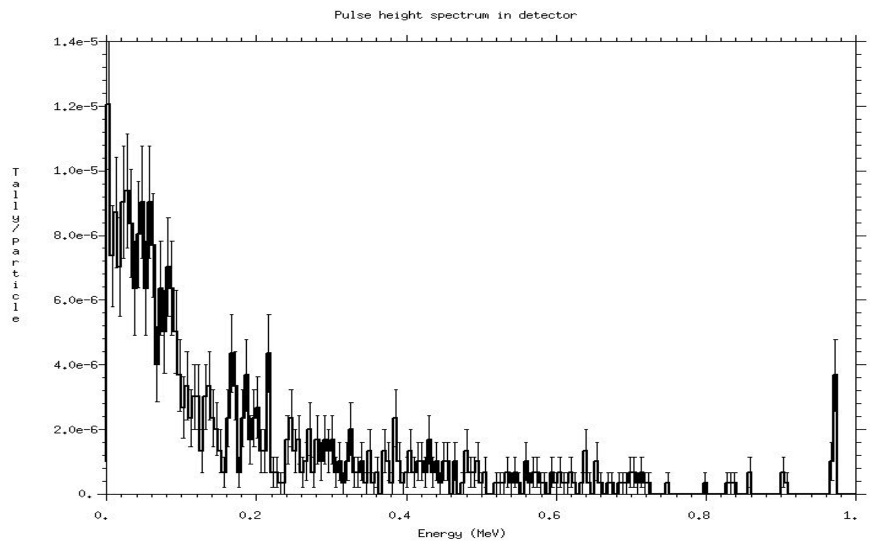


Fig. 4: Pulse-height histogram in the idealized scintillation detector showing the anticipated 1/E distribution due to electron scattering along with peaks due to fluorescence in collimator and scintillator materials. Due to limited computation time, the spectrum has large error bars. Future simulations will use the NSE cluster, which will give a factor of ~50 improvement in computing speed, and can be scheduled for up to 72 hours continuously.

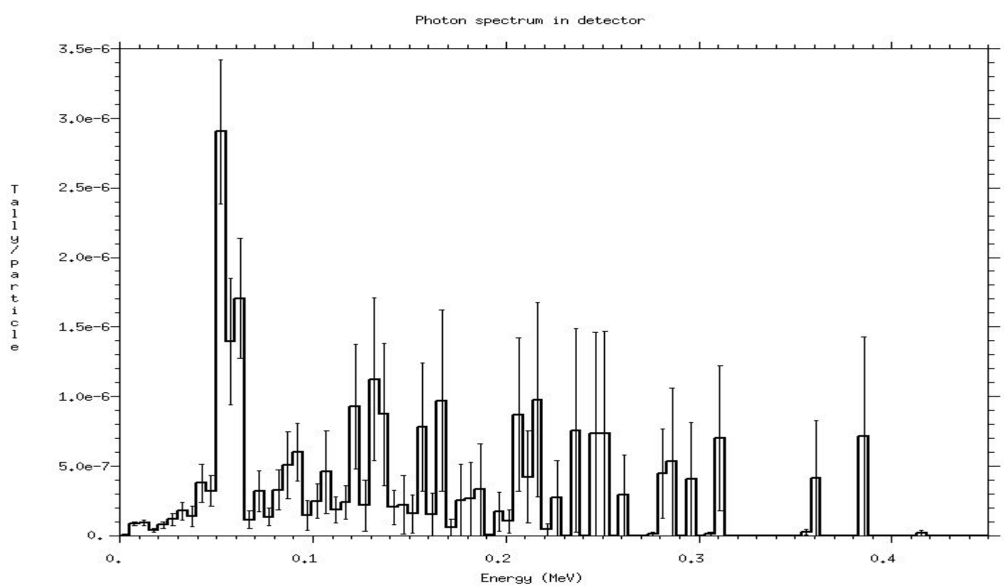


Fig. 5: Pulse-height histogram counting only photon-induced pulses. Note the clear presence of fluorescence from Lu (63.3keV) in the scintillator.

8.2.5.5. Mission Requirements

8.2.5.5.1. Power

The main power draw in the system will be the shutter solenoid (not yet designed) followed by the amplifier circuits. According to the datasheet, the CR-113/200 will draw a total of 10mA at 9V. While it has not been decided yet, the processing (RasPi or cell phone) may create a significant power draw.

8.2.5.5.2. Vibration

SiPMs have been tested in a variety of extreme shock environments (oil well logging, for example), all of which greatly exceed the stresses to be encountered in the mission. Vibration may cause issues with solder joints, but further research must be done before the exact risk can be characterized.

8.2.5.5.3. Space

The Cremat CR-113/200 modules and the associated power supply should take up no more than 25cm^2. The detector unit itself should measure no more than 1cm on each side, and will likely be integrated into a collimator measuring no more than 2cm OD. The source disk measures 2.5cm OD, and will be integrated into a shield measuring no more than 3.5cm OD. Depending on rover design, the source-detector offset can be changed from 5-10cm.

8.2.6. Environmental Sensors

8.2.6.1. Sensor requirements

The environmental sensors must:

- Be securely attached within the rover
- Weigh less than 0.25lbs
- Withstand launch and landing conditions
- Obtain data without draining the battery

8.2.6.2. Considerations

The sensor suite should include:

- Atmospheric sensors
    - Temperature
    - Pressure
    - Humidity sensors

Many temperature, pressure, and humidity sensors are available online separately and in combination. They tend to draw up to 4 microamps of current.

8.2.7. Mechanical Release

8.2.7.1. Overview

The lander section (containing the rover) will separate from the main rocket body at apogee. The lander will then descend under parachutes, with the rover fully contained. At an altitude of 1,000ft, the rover will drop out of the tube (event a), supported by a tether. Upon touchdown, a second separation event will cut the rover free from the lander assembly (event b). The following section outlines the mechanical release system responsible for events (a) and (b).

8.2.7.2. Initial Design Concepts

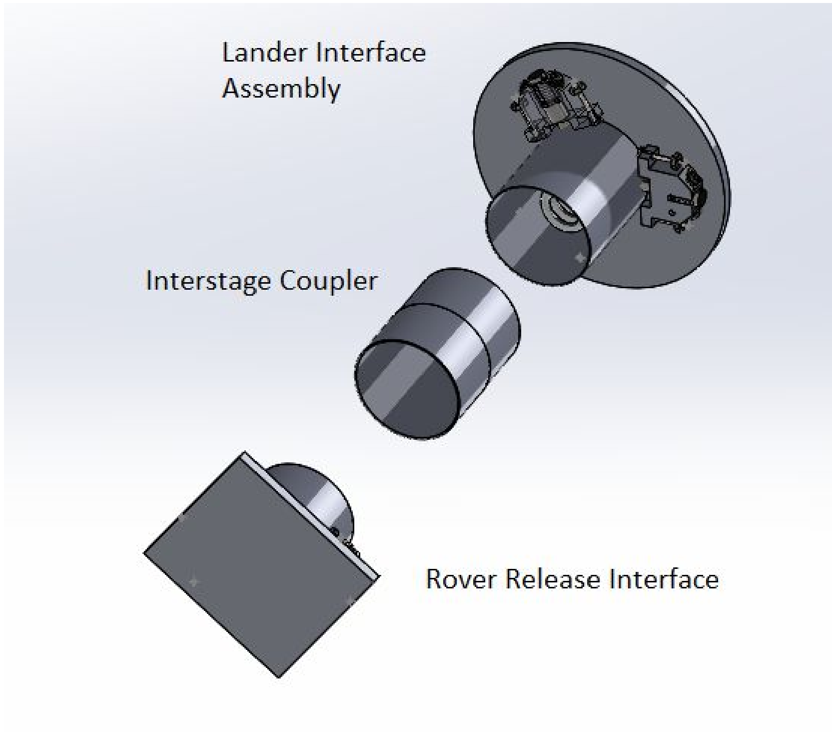
The initial design for the release mechanism was called “direct nichrome wire release”. This system involved directly supporting the rover using a metal wire (specifically nichrome). Upon applying a large current to the wire, its tensile strength decreases and the wire breaks, releasing the payload.

After evaluating, several weaknesses were discovered. First, only one wire would be supporting the rover, and the premature failure of this wire would cause the rover to be released early. Second, the tension in the wire would be depend on acceleration, so any thrust anomalies or deviations from the predicted ascent profile could exceed the breaking strength and lead to premature failure. Third, only one separation event was possible.

8.2.7.3. Final Design Concept

A second concept to address these issues was developed. It features a tethered dual-event separation, with the load being held by nichrome release pins mounted perpendicular to the axis of release. The release components are composed of interconnecting tubes that fit together in the stowed configuration but successively separate, expanding the length of the stack. The successive release reduces the force of deployment at any given stage on the rover and provides for a softer touchdown.

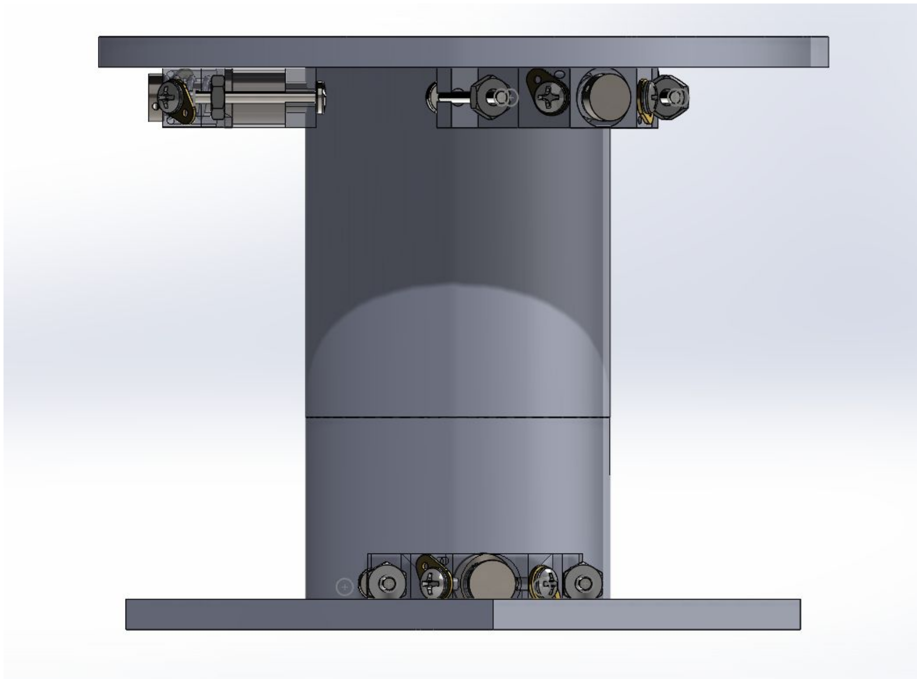
8.2.7.3.1. Tether assembly



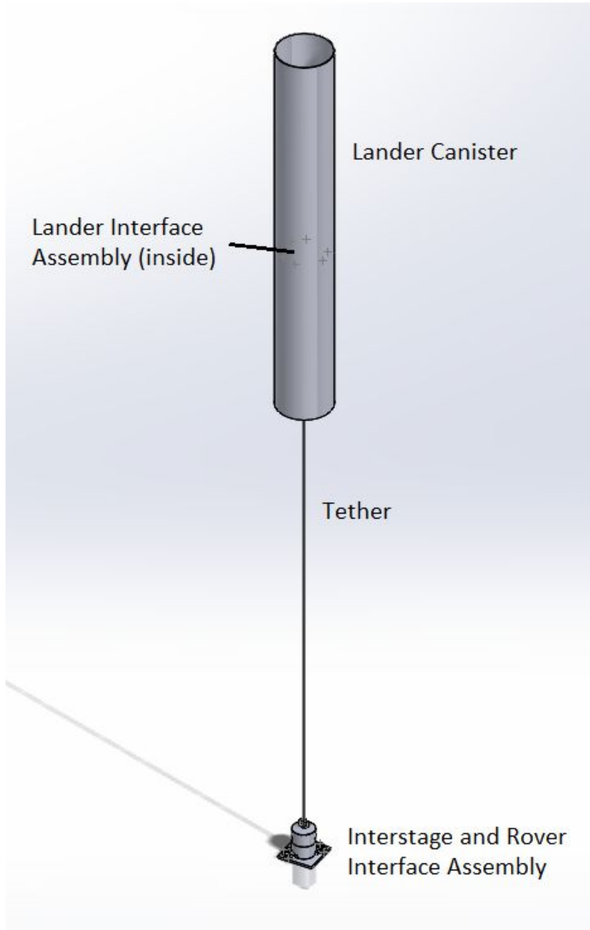
Exploded View of Release Assembly (tether omitted for clarity) - forward is toward top right

The *rover release interface* will be secured to an *interstage coupler* by a pair of *Nichrome Pin Release Modules* (detailed in the next section). This *interstage coupler* will be attached to the *lander interface assembly* bulkhead by three more *Nichrome Pin Release Modules*, and additionally by a six foot long cord. The deployment sequence is as follows:

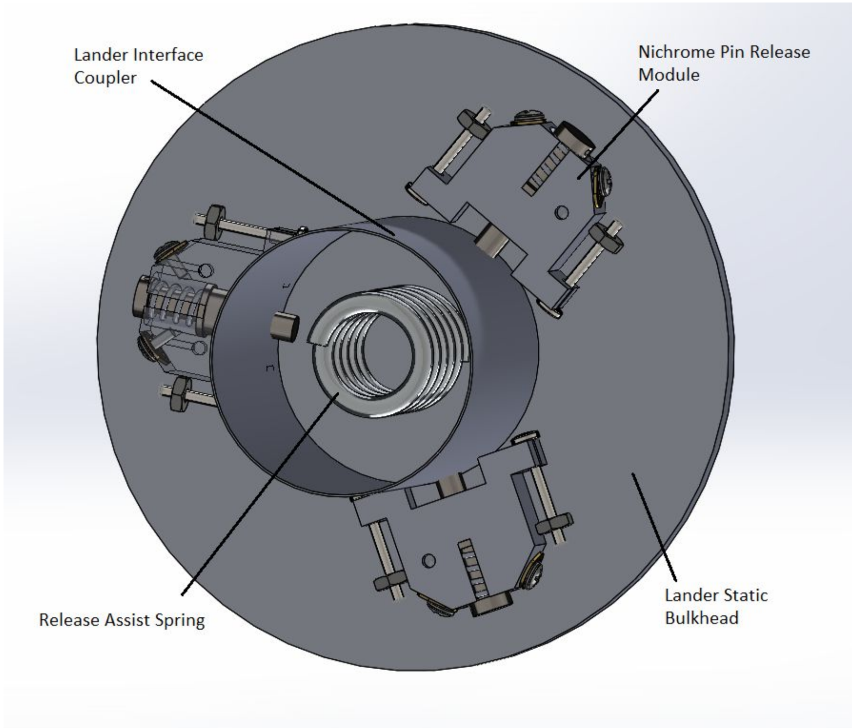
1. The three *Nichrome Pin Release Modules* on the *lander interface assembly* free the interstage coupler from the lander interface. A spring forces the *interstage coupler* and *lander interface assembly* apart.
2. The *interstage coupler* and *rover* are ejected from the lander, but the fall is broken by a six foot length of tubular Kevlar and nylon cord. The nylon, attached between the Kevlar and the lander interface, aids in shock absorption and minimizes disturbances to the rover or lander caused by the forceful ejection. The rover and interstage coupler now hang out of the tube.
3. The landing canister and tethered rover descend under parachute until the rover senses surface contact.
4. The two *Nichrome Pin Release Modules* on the *rover release interface* release the interstage coupler from the rover, assisted by a spring acting in the axial direction. The rover is now on the ground and completely detached from the lander compartment.



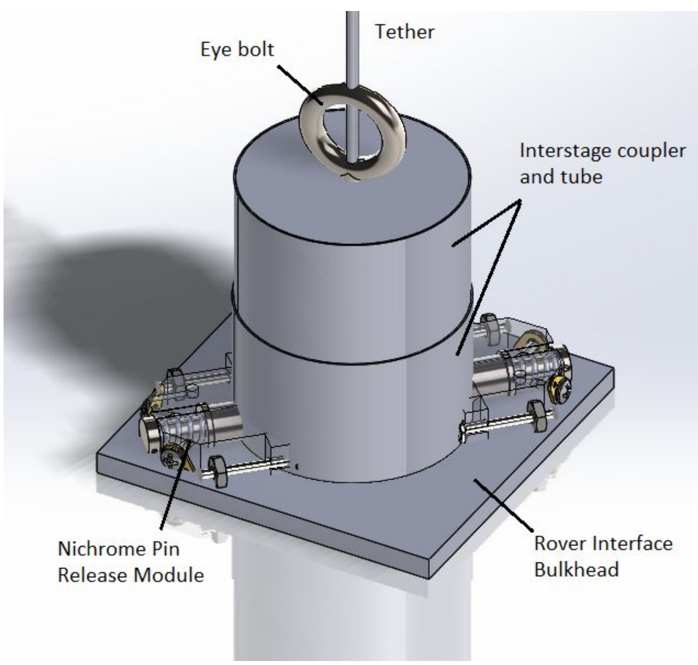
Stowed View of Release Assembly - forward is toward top



First separation event - rover hanging on tether



Detail of Lander Interface Assembly - forward is out of page



Detail of mated Rover Interface Assembly
And Interstage Assembly

Released rover upon touchdown -
spring assisted nichrome pin

8.2.7.3.2. Nichrome Pin Release Module

The *Nichrome Pin Release Module* (see figure below) starts with a small Delrin block with a hole bored in the center. A spring loaded pin is inserted into the hole such that the spring is compressed. It is held in this stored state by a 26AWG stainless steel wire running across the bolt head (stainless steel wire performed more reliably than nichrome wire, but the name stuck). When the wire is heated by electric current, its tensile strength is exceeded, and the spring forces the pin out of the hole.

In the above application, the pin will restrain the interstage coupler in single shear (the *Nichrome Pin Release Module* extends through the *lander interface assembly* then the *interstage coupler*). The act of the pin pulling away will free the *interstage coupler*.

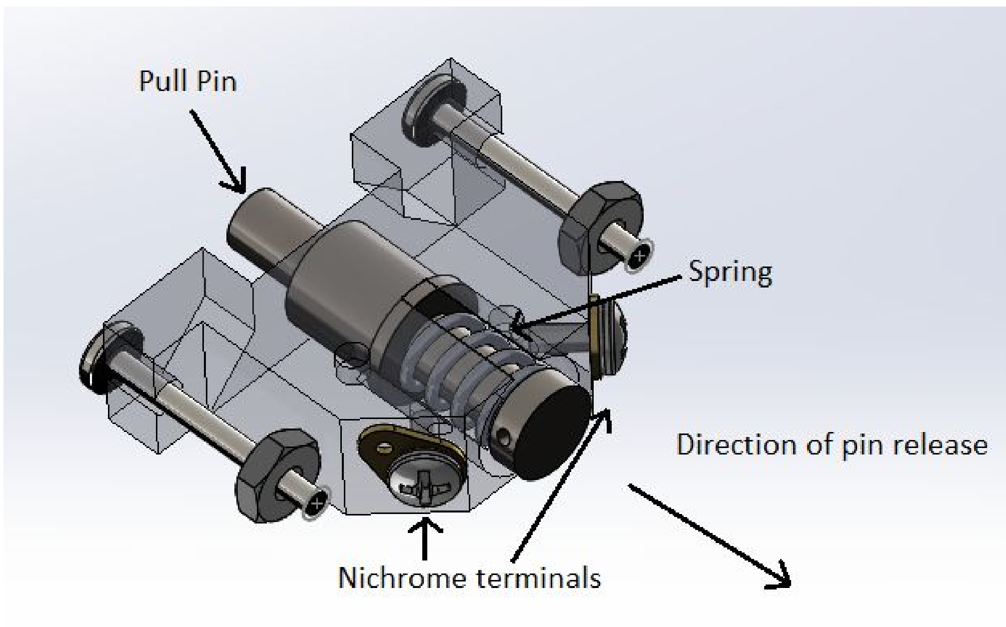
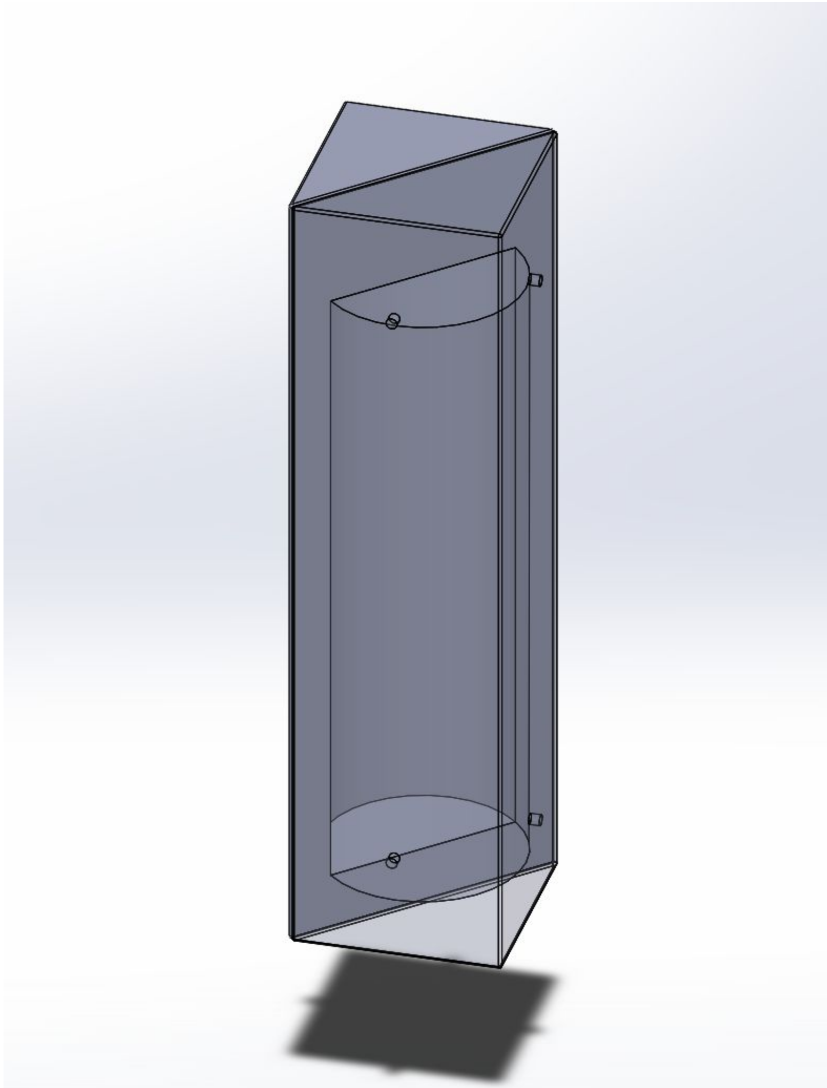


Diagram of Nichrome Pin Release Module

8.2.7.4. Rover Shell

The rover must be surrounded by a shell, because the rover will not conform to the cubesat standard. The shell will be constructed from sheet plastic via vacuum thermoforming: the sheet of plastic will be heated up, then pulled over the rover with a vacuum, and finally allowed to cool.



A mockup of the shell assembly.

We will vacuum-form each half of the rover independently, which causes the shell to split when released from the rails. The shells will be tethered to the lander, so they are not lost.

The shell will be made from ABS, because it has high strength, high impact resistance, high formability and low cost (source).

8.3. Technical Design/Analysis (Lander)

8.3.1. Computer

The lander segment requires a flight computer to deploy the reefed parachute at apogee, and unreef it at 1,500 feet AGL. In addition, the flight computer will initiate the deployment of the rover.

The Pyxida flight computer, developed by our Avionics subteam, will be used for all deployments; a COTS flight computer will be used for redundancy on parachute events. The Pyxida is solely responsible for deploying the rover and shell assembly at 200 feet AGL.

The flight computers will be mounted on the bulkhead on the same side as the rover, which is separated from the black powder charges above, and from the CO2 pressurization in the recovery section. The payload segment is vented to the atmosphere, to prevent premature pressurization and to allow the barometric sensors to record the altitude.

8.3.2. Lander structure

8.3.2.1. Overview and Proposals

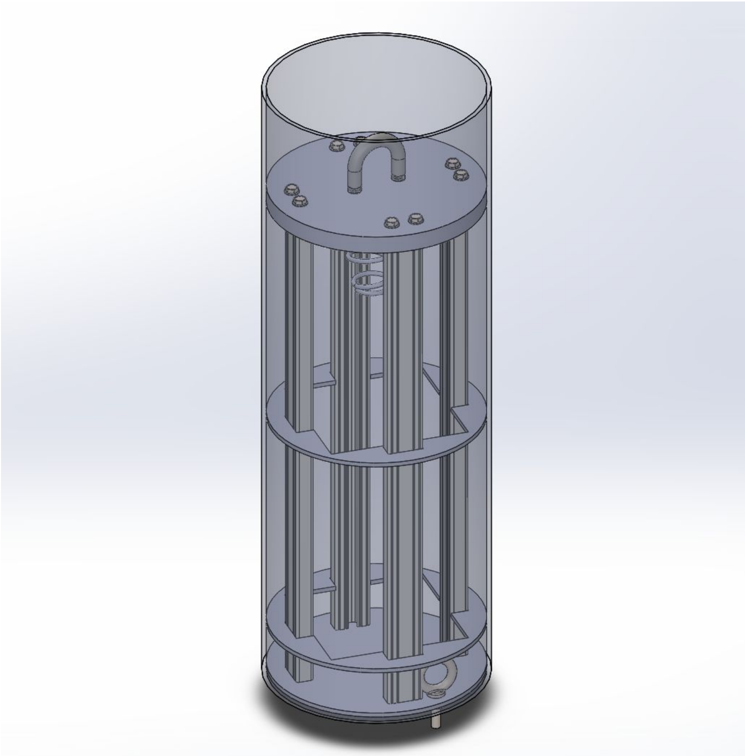
The payload subteam has developed a lander to protect the payload as it descends. The requirements for this device are:

- It must fully contain the payload and prevent its motion during ascent
- Regulate and protect its descent following separation from the main vehicle
- Reliably eject the payload at the desired altitude without jamming

In order to minimize the mass and complexity of the lander, the outer tube of the rocket will be an integral component.

Two initial ideas were considered: one in which the entire lander structure would split down the middle after deployment to release the rover, and one which would eject the payload through the aft end of the tube as the deployment event. The first was rejected because the risk of the body splitting during ascent was unacceptable.

The resulting design consists of an 18” long tube with a forward bulkhead, which will have mounting points for the lander parachute, guide rails, and a Pyxida flight computer.

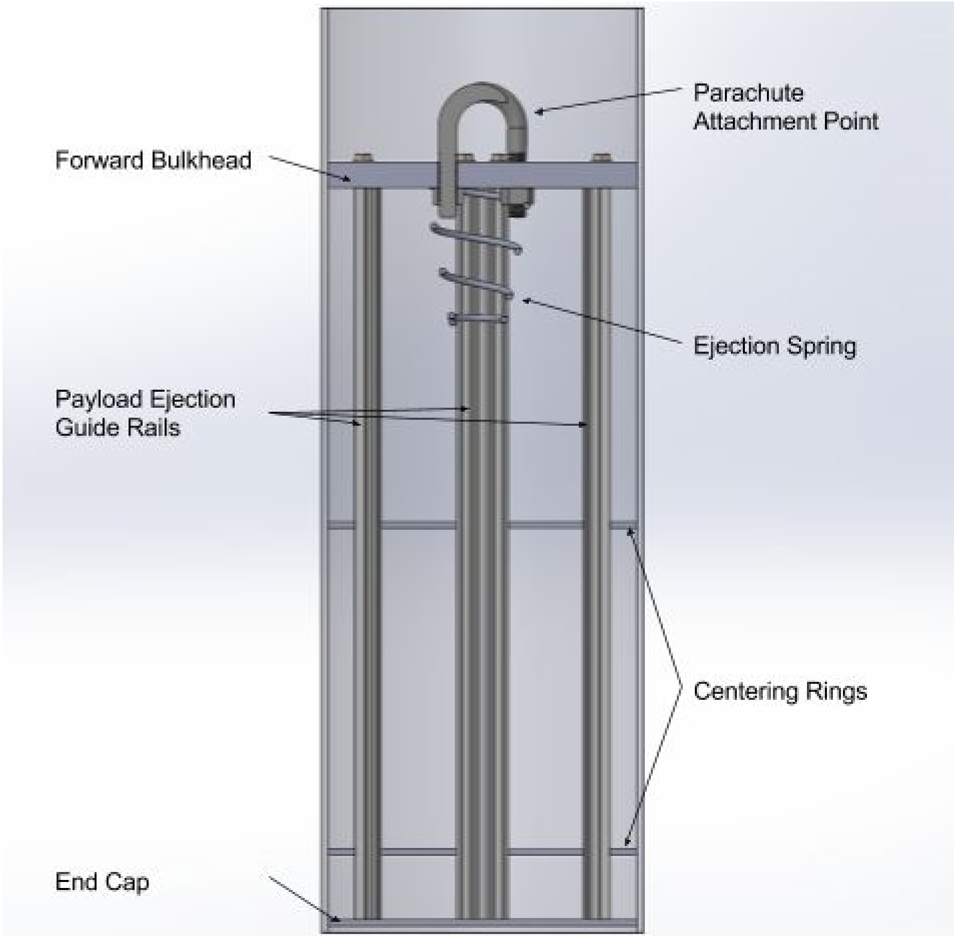


Rev. 1 Lander design.

8.3.2.2. Features and Reasoning

This system consists of four main components:

1. The ejection spring, which is a 20.6 lb-load conical spring clamped to the forward bulkhead by the parachute u-bolt mounting plate.
2. Four 1050 aluminum rails at the midpoints of the cubesat square, along which the payload will be able to slide with standard rail buttons. This type of system provides a robust and low-friction method of constraining the payload while allowing it to easily exit the chamber upon deployment.
3. Two centering rings, which are mounted at equidistant intervals down the length of the rails. These provide a rigid support between the outer tube and the guide rails.
4. The mechanical release detailed in section 8.2.7.



Lander side cutaway view with labelled components.

8.3.3. Parachutes

The lander requires its own parachute, approximately 1m in diameter. The parachute will be sewn by the Recovery subteam.

The parachute is initially stowed between the payload and the nosecone. At apogee, a black powder charge will be detonated using an electric-pyrogen igniter, separating the nosecone and ejecting the parachute. However, the parachute will be reefed at this point with a *Jolly Logic Chute Release*, which uses an elastic band wrapped around the parachute.

At 1,000ft, the Jolly Logic Chute Release will release the elastic band, unreefing the parachute. When the parachute inflates, the lander will slow significantly.

8.4. Key Technical Issues/Risk

The diagrams below are stoplight diagrams of risks. Green risks are deemed “acceptable,” whereas red risks must be mitigated prior to flight. The table outlines the risks, probability, and impact on project. To minimize the risks, the Payload team created a risk reduction plan for each risk.

8.4.1. Rover

8.4.1.1. Chassis and drive system

Risk	20%					
	10%		1			
	5%					
	1%					
	0.1%			2		
		1	2	3	4	5
Chassis		Impact				

	Risk	Risk Reduction Plan
1	The get stucks on pebbles, loose sand, low vegetation, etc.	Drive around significant obstacles. If stuck, back up.
2	Rover does not survive launch or landing	Test by dropping the rover.

8.4.1.2. Computer

Risk	20%					
	10%					
	5%		1			
	1%					
	0.1%			2		
		1	2	3	4	5
		Impact				

	Risk	Risk Reduction Plan

1	Communication with rover is lost or never initiated	Choose a reliable communication technology.
2	One or more sensors are disconnected	Securely attach connectors in rover.

8.4.1.3. Camera

Risk	20%					
	10%					
	5%					
	1%		1			
	0.1%		2			
		1	2	3	4	5
Rover: Camera		Impact				

	Risk	Risk Reduction Plan
1	Camera is obstructed	Locate camera high on rover.
2	Camera damaged by debris	Protect camera within chassis

8.4.1.4. Nuclear Experiment

The scintillation detector system from the SiPM sensor to the RasPi interface has been prototyped and tested over the last two years as part of an unrelated project undertaken by a Rocket Team member. So far, the system has shown exceptional robustness and reliability, even under challenging environmental conditions (including outdoor use for radioactive material identification). Likewise, the technology required to build a source with a moveable shield is well understood, and will not present a manufacturing or implementation issue.

The largest uncertainty in the system is in the pulse-processing hardware and the radio equipment needed to return the spectrum information to base. While it is difficult to estimate exactly how much development work will be necessary to establish a robust, high-speed data link, it will no doubt be a challenge. While using a cell phone for pulse processing and a data link could reduce this hurdle, the size of the device may be prohibitive for such a constrained payload space.

8.4.1.5. Mechanical Release

Risk	20%					
	10%					
	5%			1		
	1%			2		
	0.1%					
		1	2	3	4	5
Rover: Mechanical Release		Impact				

	Risk	Risk Reduction Plan
1	Wire does not break	The mechanism and flight computer should be tested extensively.
2	Rocket spinning jam rails	Tabs should be rigid.

8.4.2. Lander Segment

Risk	20%					
	10%					
	5%		1			
	1%				2	
	0.1%					
		1	2	3	4	5
Payload Segment		Impact				

	Risk	Risk Reduction Plan

1	Failure to deploy payload at desired altitude	Ground test system.
2	Failure to deploy parachute	Use redundant flight computers, separated from high pressures and vented to atmosphere. Ground test system. Pack parachute carefully. Pack black powder carefully

8.5. Interfaces

8.5.1. Payload-Avionics

Avionics is responsible for the Pyxida flight computer, which deploys the parachute and the rover. Avionics is responsible for installing a redundant COTS altimeter.

8.5.2. Payload-Recovery

The main vehicle recovery system (adjacent to payload) uses compressed CO2 to push on the payload segment rails through a bulkhead and separate the payload segment. The payload segment will separate from the vehicle, and the bulkhead, which is a slip fit in the recovery tube, is pulled out by the vehicle drogue, leaving the bottom of the Payload segment open.

8.5.3. Payload-Structures

Structures is responsible for providing a fiberglass tube for the lander, and the nosecone.

8.6. Going Forward Plan

The lander will be flight tested in late November or early December, using a “rover mass simulator” (a block of similar weight).

The rover will be prototyped by January, then ground tested and drop tested. Then the rover will be integrated with the rocket for subsequent flight tests.

9. Summary of System Level Risks & Concerns

The Team's goals are ambitious, and go beyond the scope of the competition in several ways. Given that the Team is composed of students, schedule slip is a critical risk that we always face. However, since we are flying multiple flight-tested systems, some of the risks are significantly lower.

Overview of Systems Level Risks						
Risk	20%				1	
	10%				2	4
	5%				3	
	1%					
	0.1%					5
		1	2	3	4	5
		Impact				

9.1. Schedule Slip

As with any complex system, the development of Project Raziel may take longer than predicted. To mitigate schedule slip, we can reduce the scope of the project to ensure we fly at competition. For instance, a custom casing for propulsion could be cut (our fin can and propellants are designed such that the case can be exchanged for a CTI 98mm casing). Similarly, the CO_2 recovery system could be exchanged for a black powder recovery system.

9.2. Testing opportunities

Test opportunities after December 3rd are strongly subject to weather, and schedules for the launch sites we go to have not released their winter/spring schedules. To mitigate this, we are planning a test flight of Raziel before the end of fall semester. Other launch sites are also available, though require significant travel time. Finally, we can usually request a special launch from the clubs that run the launch sites.

9.3. Loss of rocket

Flight tests risk losing the flight vehicle, which is a significant setback to budget and schedule. To mitigate this risk, the team has planned for a flight test before the end of semester to minimize sensitivity to vehicle loss. The team plans to manufacture or purchase redundant copies of parts that have long lead times. In addition, whenever possible, the manufacturing of parts will be streamlined for multiple rockets (e.g., water-jetting many sets of bulkheads at once).

9.4. Insufficient funds

The Team relies on industry and MIT sponsorships in order to complete its projects; the team may be unable to complete the project with insufficient funds. This risk can be mitigated by asking members to help pay for travel costs, asking for sponsorships in kind, and reducing the scope of the project.

9.5. Injury to members

Injuries to team members would have a significant impact on the team. As mentioned in Section 2.3, the Team works among many hazards. To mitigate this risk, we use the aforementioned safety plan, and abide by the rules defined by ESRA and this document, including, but not limited to:

- Wearing PPE around energetic devices
- Operating machinery safely
- Handling and storing flammables

10. Conclusion

This Preliminary Design Review accomplished the following:

- Defined requirements for the competition and for the mission
- Presented a design expected to meet all requirements
- Identified key risks and strategies to mitigate
- Defined requirements for system interfaces
- Set out a plan for the near future

After passing this PDR, the team will begin testing of several key components, and continue to build the first flight vehicle. The next review will be an internal review of testing results, scheduled for February-March of 2017.

Appendix A: Combined Requirements Table

The requirements for the project are based on the requirements for the Spaceport America Cup (REF) and the team's goals (Table REF). The master requirement numbering is based on the order of this document and how the responsibilities were divided between the subteams.

This requirements table is designed to contain all currently relevant requirements to Project Raziel. If the project design changes, and requirements beyond those in the table are needed, we will look to the ESRA DTEG and RRD for guidance, as both are more comprehensive than the below table.

RT #	Source	Requirement	Flowdown
1.0	DTEG 1.2	Shall be safe for all personnel involved	Safety
1.1	PDR 1	Personnel shall stand, at minimum, 400ft from the launch pad during launch.	Safety
2.0	RRD 2.7	Shall provide ESRA with deliverables in accordance with the IREC master schedule.	Exec
2.1	PDR 2	All major subsystems shall be student-designed and student-built.	Exec
2.2		Technologies tested should be scalable to higher-altitude flights.	Exec
3.0	DTEG 6.2	Structure shall withstand all loads from flight, landing, and transportation.	Structures
3.1.1	PDR 6	Rocket shall be able to withstand bending moment from aerodynamic pressure at expected maximum angle of attack.	Structures
3.1.1.1	DTEG 6.2.3	All coupling tubes shall extend at least one "tube diameter" through each adjacent tube	Structures
3.1.2	PDR 6	Structure shall be able to withstand 7,000N of thrust.	Structures
3.1.3	PDR 6	Structure shall be able to bear deployment loads.	Structures

3.1.3.1	DTEG 6.2.2	All eye bolts/nuts shall be closed-eye, forged steel	Structures
3.1.4	PDR 6	Structure shall bear the maximum landing loads due to rocket landing under main parachute.	Structures
3.2	PDR 5	Rocket shall have at least one place to adjust ballast weight for purposes of altitude tuning.	Structures
3.3	DTEG 8.2	Rocket shall use the ESRA-provided launch rails	Structures
3.3.1	DTEG 8.1	Rocket shall launch at a nominal elevation angle of 84°±1°, and as low as 70°.	Structures
3.3.2	DTEG 6.2.4	Rocket shall incorporate a minimum of two rail guides. These rail guides shall support the vehicle's fully loaded launch weight when suspended horizontally, and the aft most rail guide must support the launch vehicle's fully loaded launch weight while vertical.	Structures
3.3.3	DTEG 8.2-8.3	Rocket shall have a stable angle of attack upon launch rail exit, and shall remain stable for the entire ascent.	Structures
3.3.4		A person shall stand no higher than 4 feet on a ladder to access the rocket on the launch pad.	Structures
3.4		To allow for radio to be placed freely inside the rocket, the body material shall be radio transparent.	Structures
3.5	PDR 4	Structure shall be designed for accessibility. Any subsystem shall be accessible within 5 minutes of disassembly.	Structures
3.5.1		Fins shall be designed to be removable.	Structures
3.6.1	DTEG 6.1	Airframe shall be adequately vented to prevent unintentional separations due to pressure.	Structures
3.6.2	PDR 6	Structure shall tolerate temperature and heat flux from motor and environmental conditions.	Structures

3.7.1	DTEG 6.3	The team ID shall be visible from all sides of the rocket.	Structures
3.7.2	DTEG 6.3	The rocket shall be painted with the project name and academic affiliations.	Structures
3.7.3		The rocket shall be painted with all Inconel and Gold level sponsors.	Structures
4.0	RRD 2.8.1.4	Rocket shall not be “excessively” damaged during recovery (i.e. could be launched again safely with consumables replaced).	Recovery
4.1	DTEG 3.1	Rocket shall follow a dual-event CONOPS.	Recovery
4.1.1	DTEG 3.1.1.1	The initial deployment event shall occur at apogee and significantly lower the descent velocity.	Recovery
4.1.2	DTEG 3.1.1.2	The main deployment event shall occur at an altitude no higher than 1500 feet AGL, and decrease the descent velocity to less than 30 feet/second.	Recovery
4.1.3	DTEG 3.1.3	Drogue and main parachutes shall have dramatic color differences.	Recovery
4.2	PDR 8	Recovery should use non-pyrotechnic methods to initiate and complete all deployment events.	Recovery
4.2.1	DTEG 3.1.2	Recovery shall protect cords, parachutes and other vital components from any hot gases.	Recovery
4.3	RRD 2.5	All independent sections of the rocket shall carry a radio beacon or similar transmitter aboard.	Recovery
4.4	DTEG 3.6.1	The recovery system shall undergo a functional test prior to IREC, replicating flight conditions for sensors.	Recovery
4.4.1	PDR 3	Recovery shall have a successful flight test prior to competition.	Recovery
4.5	DTEG 4.1	Any energetics used by recovery shall be “safed” until the rocket is in the launch position.	Recovery

4.5.1	DTEG 4.2.2 & 4.2.4.1	Any non-COTS pressure vessels shall be designed to withstand twice their rated pressure. They shall be tested to 1.5 the maximum operating pressure, and contain a relief device set to open at no greater than the proof pressure.	Recovery
5.0	DTEG 3.3.1	Rocket shall have redundant electronics, at least one of which shall be a COTS flight computer.	Avionics
5.0.1	RRD 2.6	Rocket shall contain a COTS flight computer with on-board data storage for an official record of apogee for scoring.	Avionics
5.1	DTEG 3.4	All safety-critical wiring shall conform to ESRA Wiring Rules. (See DTEG Appendix B)	Avionics
5.2	DTEG 4.1.1-2	All arming features shall be externally accessible, such that the personnel arming them is safe.	Avionics
5.3	PDR 7	Rocket shall maintain a link via telemetry.	Avionics
6.0	RRD 2.0	Raziel shall achieve an apogee of 10,000 feet +/- 300 feet.	Propulsion
6.1		Propulsion shall be single-stage.	Propulsion
6.2	DTEG 2.1	Propulsion shall use non-toxic propellants.	Propulsion
6.3.1	DTEG 2.2.1	Propulsion shall have a two-step arming system which can only be armed when all personnel are at least 50 feet from the Rocket.	Propulsion
6.3.2	DTEG 9.2	Ignition system shall require no more than 15A at 12V to function.	Propulsion
6.4.1	DTEG 2.4	Propulsion testing shall comply with ESRA requirements.	Propulsion, GSE
6.4.2	DTEG 4.2.4.1	The combustion chamber shall be designed for at least twice the maximum chamber pressure. The chamber shall be tested to at least 1.5 times the maximum chamber pressure.	Propulsion

6.5	DTEG 2.4.3	Propulsion shall have a successful static fire test prior to a test launch. Propulsion should have two successful static fires prior to a launch.	Propulsion, GSE
7.0	DTEG 10.1	All Ground Support Equipment shall be man-portable over a short distance (~500 feet).	GSE
7.1		The launch-ready pad lifetime of Raziel shall be at least 2 hours.	GSE
8.0	RRD 2.3.1	Payload shall weigh at least 8.8 lbs.	Payload
8.0.1	RRD 2.3.1	Payload shall be removable from rocket.	Payload
8.0.2	RRD 2.3.4	Payload shall stow within a CubeSat Standard geometry (1U, 2U, 3U, etc.)	Payload
8.1	DTEG 7.1-7.1.1	Deployable payloads shall comply with recovery requirements 5.1-5.7	Payload
8.2	RRD 2.3.5	Any radioactive substances within Payload shall be encapsulated and limited to 1μC or less of radioactivity	Payload

Appendix B: Additional Resources

- ESRA wiring rules: can be found here, on page 20
- Spaceport America Cup Rules and Requirements: can be found here
- Spaceport America Cup Design, Test, and Evaluation Guide: can be found here