ROCKETRY TEAM

PER ARDUA AD ASTRA

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Room 636

Critical Design Review
NASA Student Launch 2018-2019
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1 Summary

1.1 Team Summary

The University of Pittsburgh's Rocketry Team, or the Pitt Rocketry Team (PRT) consists of approximately 50 members that contribute to one of four sub-teams: systems, mechanical, avionics, and payload. Team members either build and design the rocket or rover, control the on-board electronics, or manage the team's finances, logistics, and community presence. To aid students the team is working with Tripoli Pittsburgh, TRA Prefecture #001, and Pittsburgh Space Command, NAR Chapter #473. The Pitt Rocketry Team can be contacted at 3700 O'Hara St, Pittsburgh, PA 15213.

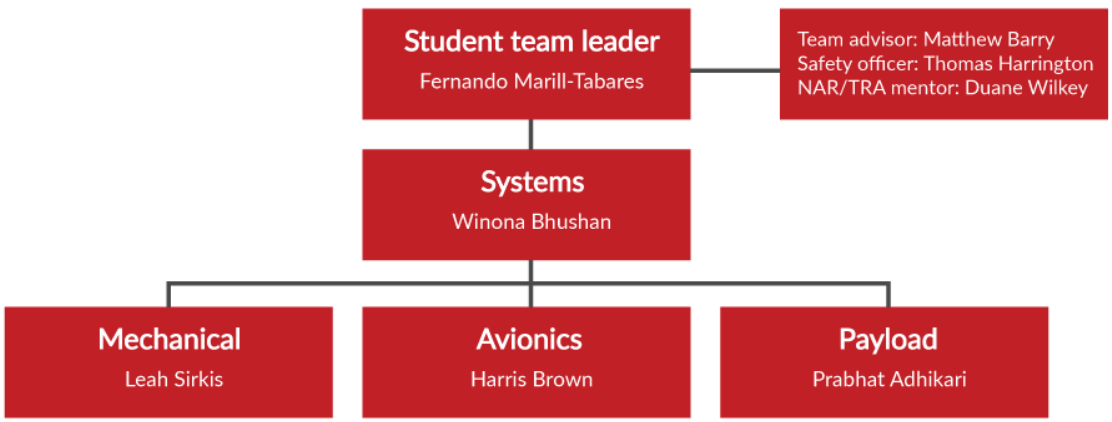


Figure 1.1a: Team organization.

Name	Team Role	Contact Information	Additional Information
Professor Matthew Barry	Team Advisor	mmb49@pitt.edu	(412) 624-9031
Fernando Marill-Tabares	Student Team Leader	fjm17@pitt.edu	(412) 313-1133
Thomas Sullivan Harrington	Safety Officer	tsh25@pitt.edu	(267) 421-1399
Duane Wilkey	NAR Mentor	duane@velocity.net	NAR Level 3 #6342

Table 1.1b: Important contact information.

1.2 Launch Vehicle Summary

Title: **PRT-1**

Mass	Length	Diameter	Motor choice	Target altitude	Recovery system
21 lbs	95in	4in	CTI PR75-2W-G K555 Reusable Motor	4750 ft	Dual deploy

Table 1.2: Launch vehicle summary.

1.3 Payload Summary:

Title: **WALL-E**

Experiment: Autonomously travel at least 10 feet away from the launch vehicle and collect 10 mL of soil.

2 Changes Made Since Preliminary Design Review

2.1 Changes Made to Vehicle Criteria

Mechanical	
Change:	**Reason:**
4” -75mm Thrust Plate will be used	The motor chosen is considered a high power, K-Class motor which means a thrust plate is highly recommended to reduce the forces that will be applied to the bulkhead, retainer, and motor mount tube. As a result, it removes the chances for parts of the rocket to fail due to the motor. Such failure includes any epoxied centering rings and bulkheads inside the rocket tubes to shear under the thrust of the motor.
Motor changed to K555 motor	Our expected mass increased, so we chose a more powerful motor to bring our rocket to the correct apogee.
Added spin tabs to fin design	This will allow us to better control the apogee of our rocket by changing the angle of the tabs.
Changed fin geometry to clipped delta	This allows the fin tabs to be mounted onto the fins.
Main Parachute shape changed to Elliptical	This parachute shape has a lower coefficient of drag so that the rocket lands within the 90 second time limit, while also being large enough to keep the kinetic energy at landing of the rocket sections under the maximum 75 ft-lb

Table 2.1a.

Avionics	
Change:	**Reason:**

Altimeters to have two electric matches per ejection charge.	The design with both altimeters lighting the same two matches introduces needless complexity into the system. Additionally, there will be less likelihood of damaging an altimeter if they are electrically separate.
Transceiver choice switched to RFM9X LoRa Packet Radio Breakout	Two sets of HC-12 transceivers from different vendors were not functioning within the parameters suggested by the datasheet. The RFM9X is a more reliable device according to our tests.
Second transceiver pair removed from avionics bay and ground system	The LoRa units have the capability to perform both GPS and rover release system functions
Remote ignition system removed from avionics bay	Including this system in the avionics bay would add excessive complexity to the design of the rocket. Our mentor asserted that the firing system provided at launch will be more than adequate for a safe launch.
Quick disconnect used to route black powder charge leads through bulkhead	This method of wiring charge leads will be quicker and more organized than continuous wires being sent through the bulkheads, as they were for the subscale launch
Waterproofing included	Our team captain encouraged the addition of waterproofing to protect the altimeters and PCB, making the rocket safer and more reusable

Table 2.1b.

2.2 Changes Made to Payload Criteria

Payload	
Change:	**Reason:**
Soil collecting wheels installed backwards	This ensures compliance with the requirement that the rover must travel 10 ft before soil collection can begin. The rover will then drive in reverse with an anchor deployed to begin collecting soil.
Transceiver choice switched to RFM9X LoRa Packet Radio Breakout	Ground system transceivers were switched to RFM9X from HC-12 due to better range and reliability.

New anchor arm mechanism added that also functions as the flipping mechanism in case of off-nominal orientation of the rover after deployment	Preliminary tests with the wheels showed that they would not collect much soil on their own, so a deployable arm was added that will provide drag to help the wheels dig into the soil
The ultrasonic sensor will be mounted on a servo	This helps the sensor scan for obstacles better.

Table 2.2.

2.3 Changes Made to Project Plan

Change:	**Reason:**
Timelines for design and fabrication teams were updated to reflect progress	The current Gantt charts produced by our sponsor, Workzone, most accurately reflect our timeline
Dates set for STEM engagement outreach	The participating schools sent their preferred dates for us to complete the STEM engagement

Table 2.3.

3 Vehicle Criteria

3.1 Mission Statement and Success Criteria

3.1.1 Mission Statement

PRT's launch vehicle will fly to an apogee of 4,750 feet, safely land, and deploy a rover that collects 10 ml of soil. The mission will comply with all requirements of the NASA Student Launch. Throughout the duration of the project, all members will adhere to the highest standards of precaution and safety.

3.1.2 Success Criteria

PRT will meet all the success criteria stated in the NASA SL 2018-2019 handbook. To achieve this, PRT's launch vehicle, PRT-01, will be ready for launch and able to stay on the launch pad fully functioning for 2 hours on launch day. A 12V direct current will be able to ignite the motor for launch without any outside circuitry. The motor selected will be commercially available and use ammonium perchlorate composite propellant (APCP) as propellant. Impulse will not exceed 5120 Ns. Our launch vehicle will have a stability margin greater than 2 upon rail exit and accelerate to no more than 52 fps. PRT's rocket will fly to an apogee of 4750ft using a single stage. PRT-01 will be recoverable and reusable. The launch vehicle design will include less than 4 separate sections with couplers and shoulders at least one body diameter in length.

Our rocket will also utilize spin tabs to allow our rocket to spin so we can better control the apogee of our rocket.

3.2 Selection, Design, and Rationale of Launch Vehicle

3.2.1 Vehicle Body Design Review

As a first year team, we do not yet have the resources to manufacture our own air frame. Therefore, we have chosen to use and modify the Wildman Darkstar Extreme air frame for our launch vehicle. This kit uses a 94 inch airframe with spiral wound G12 Fiberglass as the material. This particular kit was chosen because G12 Fiberglass is a lightweight, strong material. Carbon fiber bodies were also considered. As illustrated in table 3.2.1a below, carbon fiber is stronger than fiberglass, but significantly more expensive, so ultimately the fiberglass kit was chosen.

Fiberglass	**Carbon Fiber**
Price: 24.3-34.4 USD/kg	*Price*: 37.4-41.6 USD/kg
Strength: 138-241 MPa (T) 128-207 MPa (C)	*Strength*: 550-1050 MPa (T) 440-840 MPa (C)

Table 3.2.1a.

The dimensions of each piece of the rocket were measured in order to: 1. Satisfy NASA's requirements, and 2. Purchase the correct-sized parts. These dimensions (in mm) can be seen below in the following figures, which were taken from a SolidWorks Drawing. The mass of each component was also calculated and can be seen in Tables 3.2.1g.

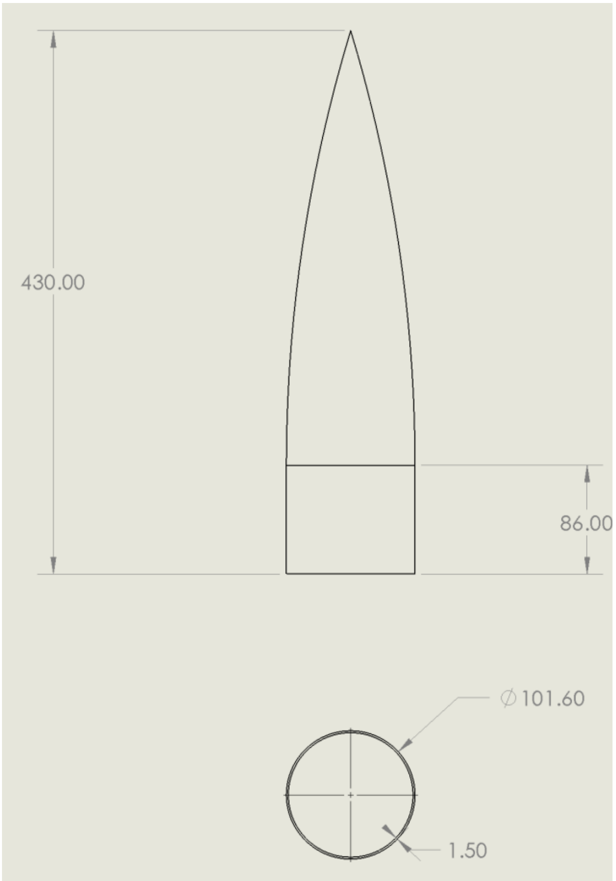


Figure 3.2.1b: Nose cone.

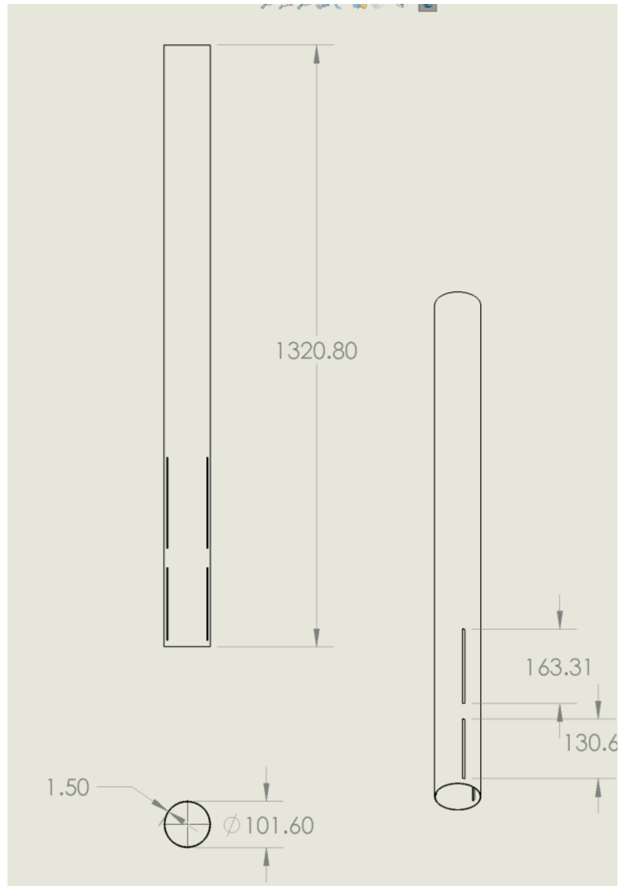


Figure 3.2.1c: Booster.

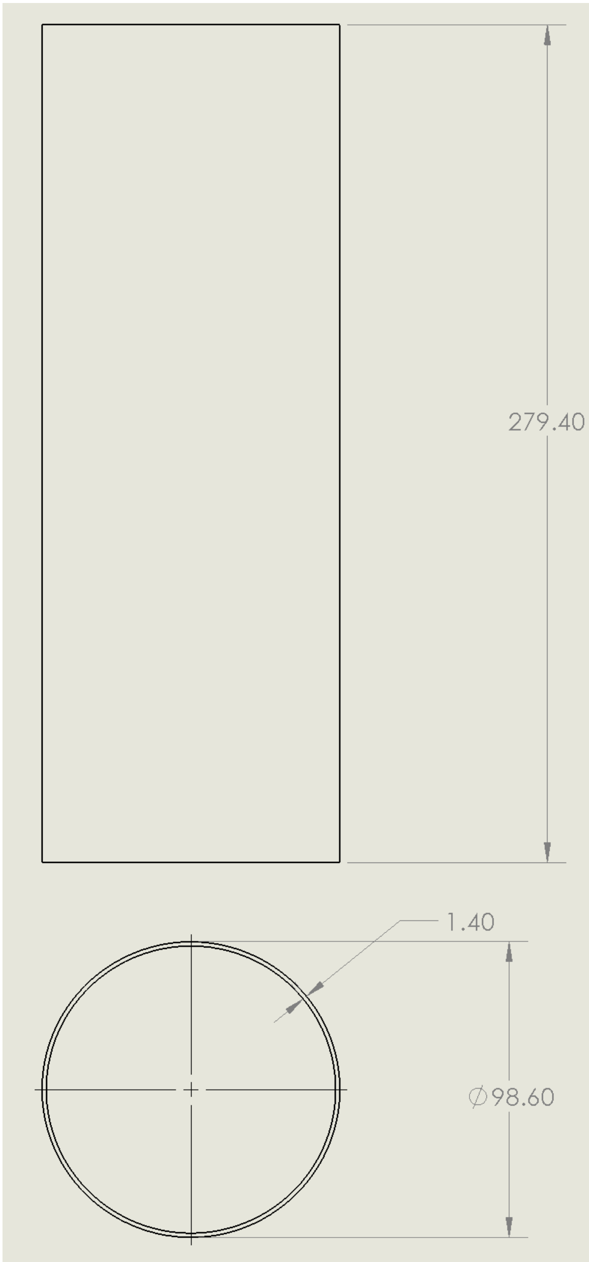


Figure 3.2.1d: Coupler.

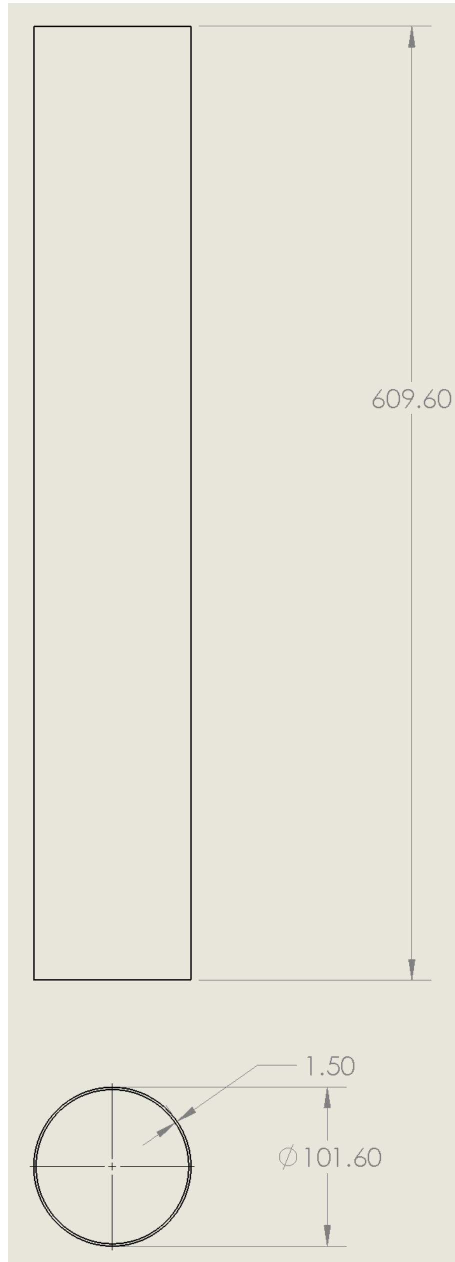


Figure 3.2.1e: Payload section.

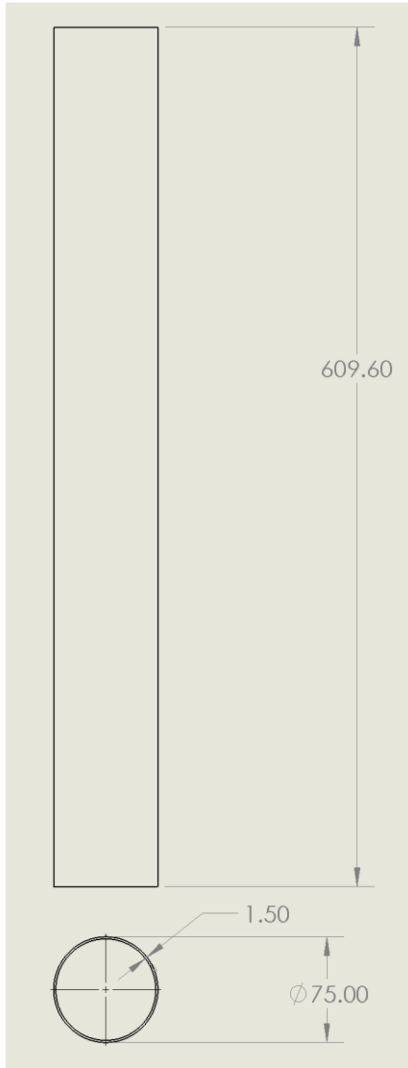


Figure 3.2.1f: Motor mount.

	Booster	Payload Section	Motor Mount	Coupler	Nose cone
Mass [g]	1284	499	367	337	353

Table 3.2.1g: Mass of each part of the rocket.

Our rocket body will experiences forces throughout the launch, flight, and landing. We determined that the rocket will undergo maximum stress in one of the following stages: takeoff, flight, parachute deployment, or landing. Our rocket body, a Wildman Darkstar Extreme 4, is made out of spiral-wound G12 fiberglass. Although the material properties for G12 fiberglass are not available, we were able to obtain the flexural, compressive, and tensile stress of G9, G10, and G11 fiberglass, as shown below in Table 3.2.1h.

Material	Flexural Strength (psi)	Compressive Strength (psi)	Tensile Strength (psi)
G9 Fiberglass	50,200-60,400	45,000-70,000	39,000
G10 Fiberglass	45,000-55,000	35,000-68,000	45,000-55,000
G11 Fiberglass	59,600-76,700	32,900-63,000	59,600-76,700

Table 3.2.1h: Strengths for different types of fiberglass.

With these properties, we can make a safe estimate for the material properties of G12 fiberglass. We can determine that the flexural strength is most likely between 45,000 and 76,700 psi; the compressive strength is likely between 32,900 and 70,000; and the tensile strength is likely between 39,000 and 76,700 psi.

Additionally, the hoop, or cylindrical, strength of G12 fiberglass is ~50,000 psi, according to the manufacturer of the rocket body. Using the lower limit of these material properties for our simulations and calculations, we can accurately determine if the G12 fiberglass body is strong enough to withstand any force that it could experience during the run.

During takeoff, the airframe, or body, will experience a compressive force from gravity and the motor thrust. It will also experience a negligible drag force. Our motor, a Cesaroni K555, has a maximum thrust of 646.7 Newtons, or 145.38 lbf, according to the manufacturer. The gravitational force on the rocket is just the mass times the acceleration, which comes out to the weight of the rocket -- 22.046 lbf. This weight is exactly 10 kg. For the purposes of simulations and calculations, we are using an overestimation for our weight since the exact weight may be modified in the future. Since these two forces are working in opposite direction, the maximum stress the rocket will undergo during takeoff will be the sum of the max thrust and weight, which totals 167.43 lbf.

To determine the stress on the rocket, we divide the force by the cross sectional area using the following equation:

$$\sigma = \frac{F}{A}$$

Our rocket has an outer diameter of 10.16 cm (r = 5.08 cm) and inner diameter of 10.01 cm (r = 5.00507). Using the following equation, we can determine the cross-sectional area of the body.

$$A = \pi(r_{outer})^2 - \pi(r_{inner})^2$$

The cross-sectional area is 0.935 cm, which means that the maximum stress endured during takeoff is 3136838.78 Pa or 454.96 psi. This is significantly less than the estimated minimum compressive stress of G12, 32,900 psi, so the rocket will easily be able to withstand takeoff.

During flight, the rocket will experience a drag force due to the air around it. This creates a compressive force on the airframe. According to our simulations, the maximum drag force, which was calculated with the maximum wind speed that we are supposed to test with, is 393.99 lbf. While this is a sizable number, the stress (F/A) only comes out to 1070 psi, still well below the lowest estimate for compressive strength of G12, 32,900 psi.

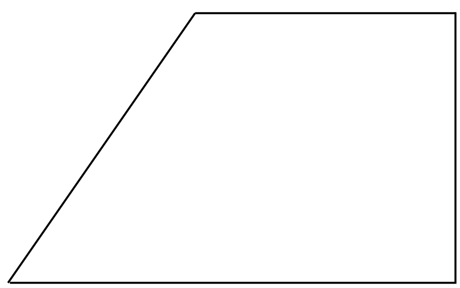
To show that the rocket will be able to withstand drag forces during takeoff as well, we will assume that the maximum drag force will occur when the motor is providing maximum thrust. Although this is not true, this situation will give us a stress that is higher than any possible stress during takeoff. The drag force and motor thrust together create a compressive force on the body. This force will total 561.42 lbf (167.43+393.99). Divided by the cross sectional area, the stress is 1525.60 psi, again well below the estimate of 32,900 psi strength. Since the rocket will be able to withstand max drag force and max motor thrust at the same time, it will be able to stand those forces occurring at different times.

When the parachutes deploy, a tensile stress is placed on the body. The force that is created is from the difference in drag force from the rocket and main parachute. According to our simulations, the maximum deceleration occurs when the main parachute deploys. This deceleration is 643.04 ft/s^2. Assuming this deceleration acts on the entire mass of the rocket, it comes out to a force of 14,176.46 lbf. Divided by the cross-sectional area, this yields a stress of 38,522.99 psi. Although this is close to the lowest estimate for tensile strength of G12 at 39,000 psi, the deceleration lasts only for an instant and the true mass of the rocket will be lower than what is used in these calculations, which show that the rocket will be able to withstand all forces exerted by both parachutes.

During landing, the rocket will experience a compressive force from the motor as well as from impact with the ground. We have calculated, with our rocket, that the landing force will be 69.6 lbf. However, for the purpose of maximizing stress we will assume that the rocket landing will be at the maximum permitted with respect to the regulations, which is 75 lbf. To analyze if the airframe can handle the stress during landing, we will perform calculations for two different scenarios; the first of which is if the rocket lands vertically, landing on the entire cross-sectional area and creating an axial compressive force. From the manufacturer of the body, we know that the axial compressive strength of the tube is 20,000 psi. To calculate the stress on impact, we divide the force by the cross sectional area: 75 lbf/0.368 in^2. This yields a stress of 203.804 psi, well below the limit of 20,000 psi. For the second scenario, we will assume that the rocket does not land upright and instead lands at an angle, therefore putting more stress on a smaller section of the cross-section. Although we can't find the peak stress, we can find the smallest area that the rocket can land on within the lowest estimated compressive strength of the body, 32,900 psi. To find the area, we divide the force by the strength: 75 lbf/32,900 psi, which comes out to 0.00228 in^2. This is an area that is approximately 1.47 mm^2. The rocket coming down on an area that small is virtually impossible, so it is extremely unlikely that the rocket will fail on landing. Overall, the rocket will be able to withstand any forces exerted on it during takeoff, flight, parachute deployments, and landing.

3.2.2 Fin Design Review

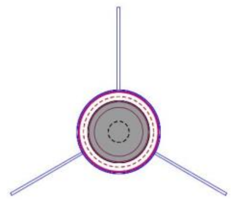
The fin design for the launch vehicle has changed in many ways since the PDR report. Firstly, we have opted for three clipped delta shaped fins instead of the previous trapezoidal shaped fins. Part of the rationale is that we intend to use a system of spin tabs during the launches of our rocket to experiment with roll of the vehicle during flight, and this system relies on the geometry of the clipped delta fins. This is the general shape of each fin with the following dimensions shown in Table 3.2.2.



Dimension	Measurement
Root Chord	15.494 cm
Tip Chord	7.010 cm
Height	10.008 cm
Sweep Length	8.484 cm
Sweep Angle	40.4°
Thickness	0.318 cm

Table 3.2.2: Dimensions of the fins.

The fins will be spread 120° apart along the body tube as shown below:



With this fin shape, our rocket will have a center of pressure located at 173.99 cm from the forward-most point. The center of mass is at 152.4 cm from the same location. This will produce a static margin of 2.21, which is above the 2 that is required by NASA.

In addition, our stability during flight is increased by the spin tab system which is one of the key reasons we chose this design over the stock fins that we already owned. The tabs create a gyroscopic effect which will cause the rocket to resist changes in its orientation during our flight, further increasing our stability.

The spin tabs also allow us to control our apogee, which is extremely important for this mission as we are required to be as close to our target as possible. The angled fins do this by creating more drag, thus reducing the amount of lift the engine provides. The larger the angle, the more severe this effect.

We plan on testing several tab angles (4, 7, 10, 13 degrees) to collect data about how each angle value reduces our apogee. From that, we will fit the data to an equation that will used to extrapolate or interpolate the angle needed to reach our target apogee.

Fin Angling Design:
The fin adapters will be designed as follows:

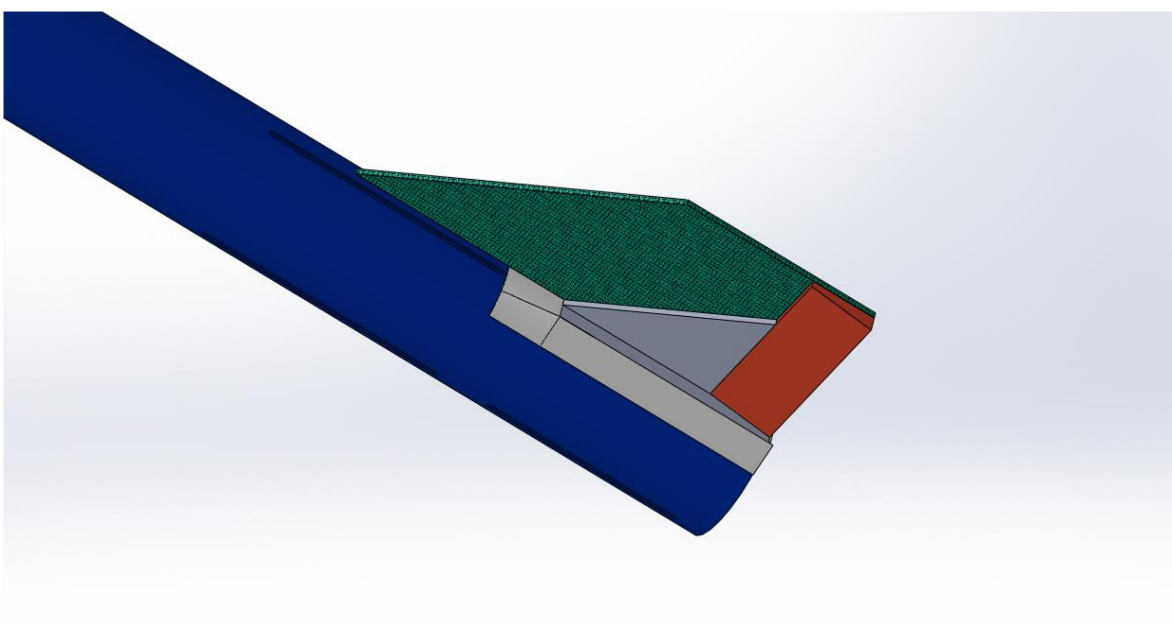


Figure 3.2.2a: Assembled fin system.

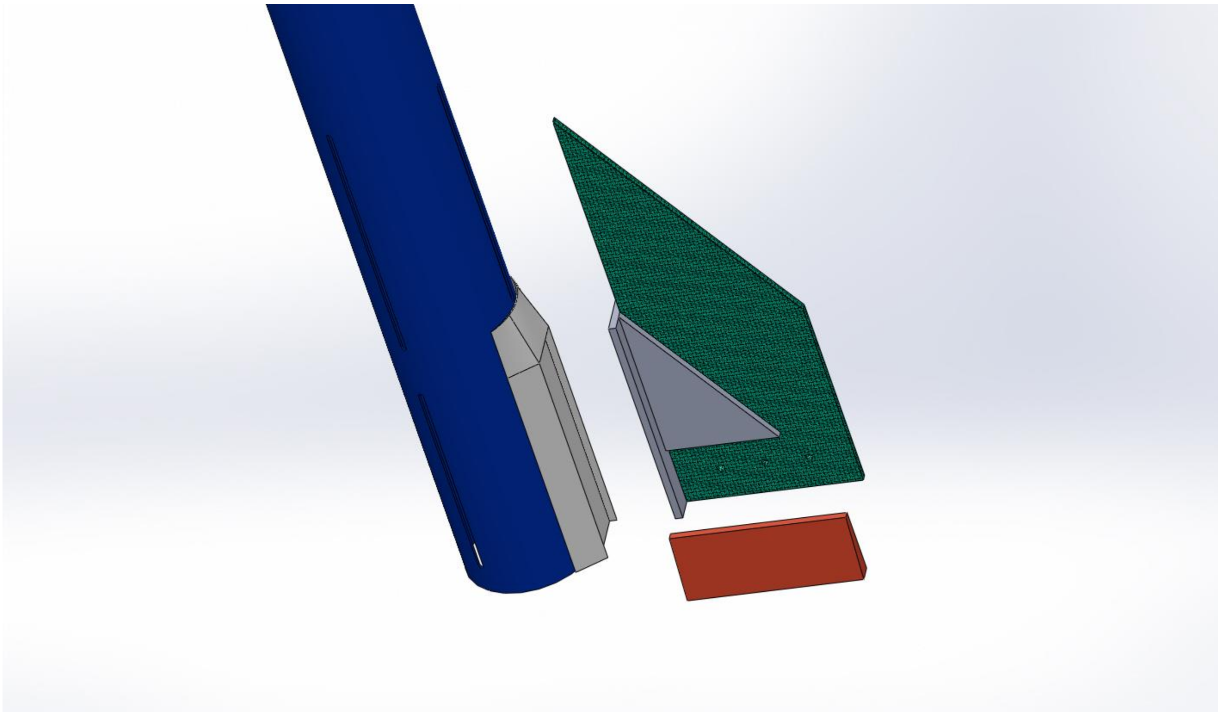


Figure 3.2.2b: Exploded view of the fin system.

The spin tabs (shown in red) will be 3D printed out of ABS and be attached to the fins via three small fasteners along the bottom of the fin. This will allow us to easily interchange tabs after each flight.

Fin Performance:

To ensure a strong attachment of the fins to the airframe, and to ensure that the placement with respect to all axes is correct, we have designed our system such that the fin can be easily slotted in and epoxied to the airframe. This is done by printing parts that attach to the booster section and the fin respectively, and can then be epoxied together to create a good fit (see Figure 3.2.2b).

To mitigate the risk that fins may break during flight due to aerodynamic forces, the fin flutter equations can be used. In order to use the fin flutter equation, the the aspect ratio of the fins must be calculated first. This is done with the following equation:

$$AR = b^{2}/A$$

where b is the height of the fin and A is the planar area of the fin. To calculate the planar area, we will use the following equation:

$$A = (Ct + Cr)/2 \times h$$

Where Ct and Cr are the root and tip chords respectively and h is the fin height. When we evaluate this equation, we find that the planar area for the fin is 112.580 cm^2. Substituting that into the previous equation, we find that the aspect ratio of the fin is 0.89. This is close to 1, which means that the fins produce only slightly more drag than lift. To minimize this effect, we will also bevel the top edge of the fins to make the surface more aerodynamic. In addition, proper sanding will also mitigate the drag caused by the fins.

Now that the aspect ratio has been calculated, the maximum flutter felt by the fins can be determined.

In the following equations, a is the speed of sound, P is pressure, G is the shear modulus of fiberglass, E is the elastic modulus of fiberglass, v is Poisson's Ratio, AR is aspect ratio, $\lambda$is the ratio of the tip chord/root chord, t is the fin thickness, and c is the root chord.

at T = 70°F

$$a = \sqrt{1.4 \times 1716.59 \times (T + 460)} = 1128.59 \text{ ft/s}$$
$$P = 2116/144 \times (T + 459.7/518.6)^{5.256} = 16.43 \text{ lb/in}^2$$

$$G = E/(2 \times (1 + v))$$
$$G = 2.7 \times 10^6/(2 \times (1 + 0.12)) = 1{,}205{,}000 \text{ psi}$$

$$V = a \times \sqrt{(G/(1.337 \times AR^3 P(\lambda + 1))/(2(AR + 2)(t/c)^3}$$
$$V = 1128.59 \times \sqrt{1205000 \div (1.337 \times 0.87^3 16.43(0.452 + 1))/(2(0.87 + 2)(0.125/6.1)^3}$$
$$= 1899.56 \text{ ft/s} = 1295.15 \text{ mph}$$

As given by our simulations with OpenRocket, the maximum velocity of our rocket is 447.387 mph as shown below in Figure 3.2.2c.

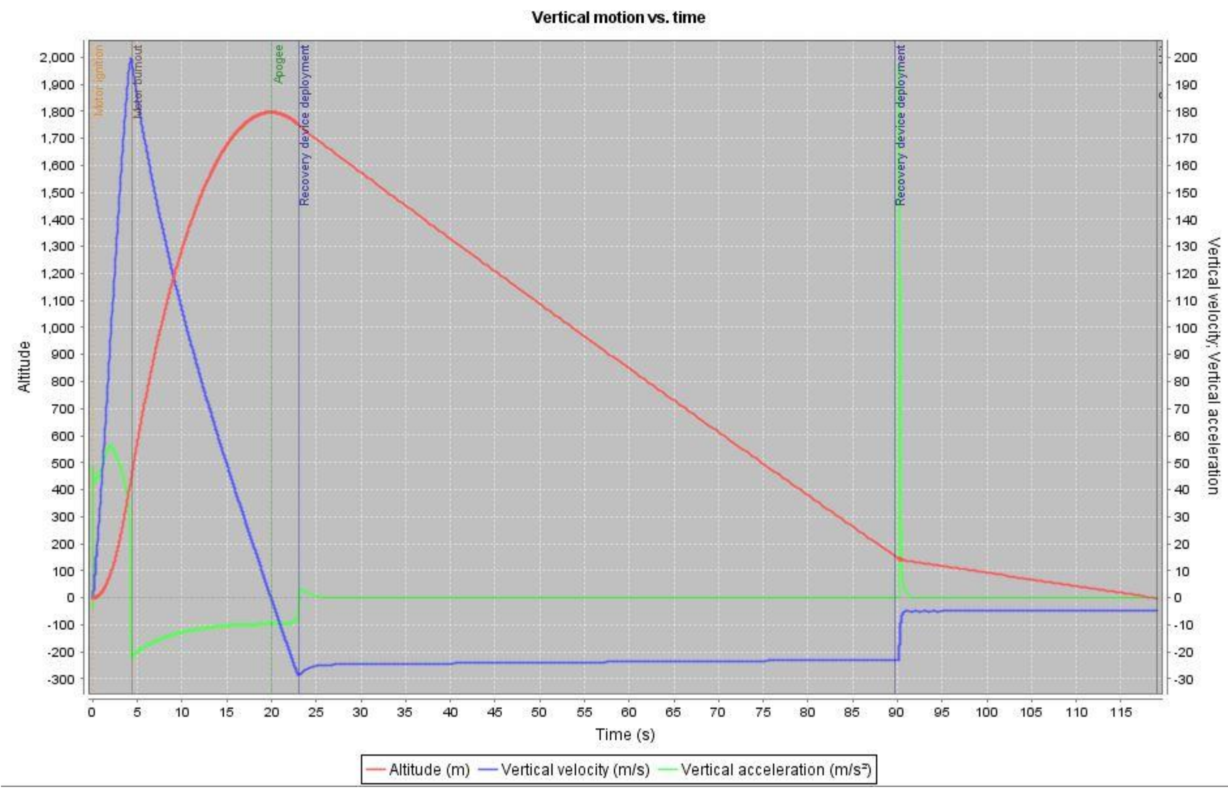


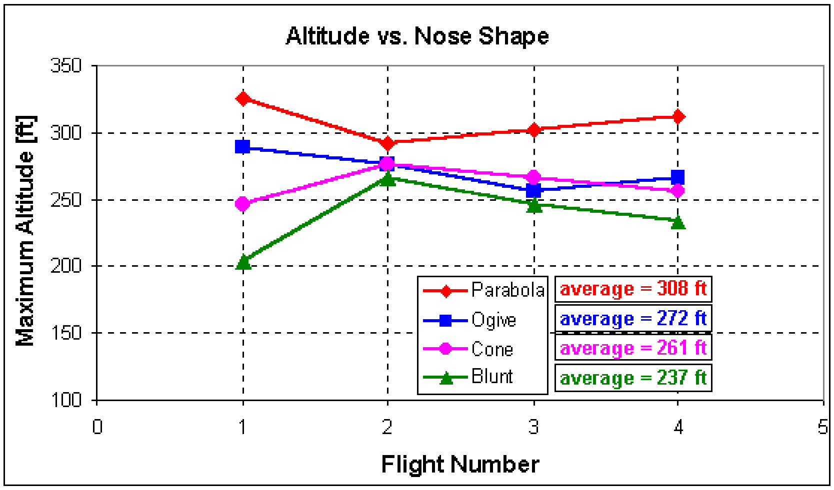
Figure 3.2.2c: OpenRocket simulation results.

3.2.3 Nose Cone Design Review

The team looked into many different possible nose cone shapes and ultimately decided to use a fiberglass Von Karman Ogive (LD-Haack) nose cone with an aluminum tip.

Nose Cone Shape:

For subsonic high speeds, the top nose cone options that provide low coefficients of drag include Parabolic, Cone, and Von Karman (Ogive). We have been provided with a Von Karman nose cone with our airframe, so the Von Karman shape is the most cost effective option. The performance of different nose cone shapes on the same rocket type were tested by AeroSpaceWeb and the resulting data is shown below in Figure 3.2.3.



Altitudes achieved using different nose shapes on the same rocket

Figure 3.2.3.

From Figure 3.2.3, we can see that the parabolic nose cone shape allows the rocket to reach the highest altitude of all the shapes tested so it is the most aerodynamic option. The ogive shape has the second highest average altitude but is also the most cost effective and easiest to manufacture since we already have this nose cone and therefore do not need to purchase or manufacture it. We have chosen the Von Karman nose cone shape because it has a low coefficient of drag as evidenced by Figure 3.2.3 while also being cost effective since it is provided with our air frame.

<u>Material Selection:</u>

We have chosen a fiberglass nose cone to maintain the same material as the airframe for ease of use. We have selected a nose cone with an aluminum tip to give the tip extra strength so that if it hits the side of the airframe or ground the tip is not damaged. We considered using a nose cone without an aluminum tip, as it would be lighter, but the extra strength provided by the aluminum tip is important to ensuring our rocket is durable enough to be reusable.

3.2.4 Motor Selection and Retention System Design Review

<u>Motor Selection:</u>

We have selected the Cesaroni Technologies PR75-2W-G K555 Reusable Motor. This has changed from our past decision of a Cesaroni Technologies Pro54 K570 motor chosen in our PDR due to our expected increase in mass and our subscale launch that showed the OpenRocket simulations to be overly optimistic in terms of altitude. We chose Cesaroni Technologies as the company for our motor because their motors are readily available, come in many sizes, and are fairly easy to assemble. We chose a reusable motor because while single use motors would make motor mounting easier, single use motors are more expensive given the number of times we plan to launch our rocket.

<u>Motor Retention:</u>

The motor retention system of our rocket design is made of a motor retainer, a thrust plate, and centering rings. Our motor retainer was carefully chosen based on three main factors: cost, weight, and ease of application. Each of these factors was given a weighted percentage to quantitatively define its importance. The most important factor was the weight of the motor retainer, with a value of 50%. This was given highest priority because it's essential to not overload the rocket with excess weight. Next highest was the cost, with a value of 30%. This was ranked second because it is not ideal to spend more than necessary on our final design. Lastly was the ease of application, at 20%. This was given its value because it was assumed prior to the purchase that the method to apply would be done with ease.

The weighted importance of the three criteria were applied to three different manufacturers' motor retainers: Apogee Components, Rocketarium.com, and Giant Leap Rocketry. The results and how the importance of factor affected each motor retainer can be seen below in Table 3.2.4.

Criteria	Apogee Rockets	Rocketarium	Giant Leap Rocketry
Weight (50%)	9	6	4
Cost (30%)	5	7	9
Ease of Application (20%)	10	10	6
Total	8.0	7.1	5.9

Table 3.2.4: Results from decision matrix of motor retainer along with weighted percentages and scoring breakdown.

Based on the decision matrix, it was was concluded that the 75 mm motor retainer from Apogee Rockets is the best option for our rocket since it has the highest total weighted score. While it is the most expensive of the three and it has the same method of applying it to the rocket, the weight played the largest role in the final calculation because it held the lightest weight.

The centering rings we selected, which act as a complement to the motor retention system, are fiberglass centering rings that were included with our rocket kit. We looked at options such as plywood and fiberglass as centering ring materials but we decided that the centering rings provided with our kit are the best option based on the criteria of strength, cost, ease of manufacturing, and mass. They were premade and provide a perfect-fit inside the 75 mm tube. These will be used to maintain the motor retainer's position in the appropriate tube and transfer force from the motor to the airframe which will ultimately allow the motor itself to be remain stationary as needed in our design.

We have chosen a thrust plate made of aluminum, based on simulations of the forces we project the motor retention system will feel. These simulations are described in detail below.

Motor Retention System Test Simulations:

Using the SolidWorks simulation program, we were able to simulate the motor retainer under a similar scenario that it will undergo during a launch. The material of the retainer is Aluminum 6061 (T6) and its model is the AeroPack 75 mm Flanged retainer. Its CAD design can be see below in Figure 3.2.4a. Note that the green arrows represent the support whereas the purple represent the applied load. The flanged retainer was chosen for the final design due to our needing of a thrust plate. This will enable a better connection to the thrust plate, and to the thrust plate to the rocket through the usage of #6 and #10 screws.

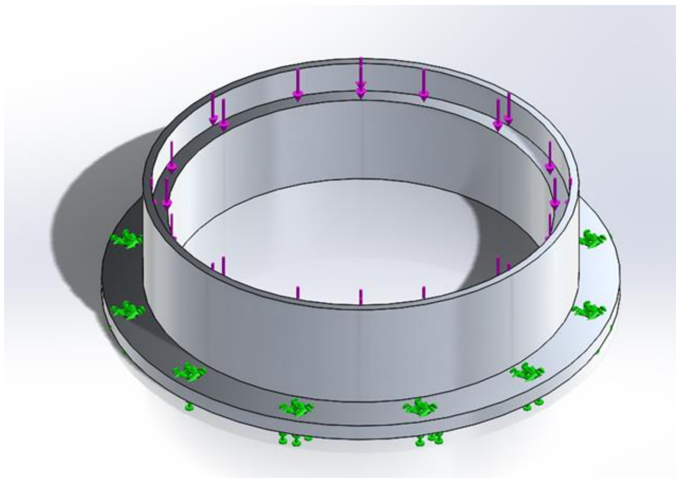


Figure 3.2.4a: Flanged motor retainer CAD design.

The motor chosen for our full-scale has maximum thrust force of 646.7N, or 145.38lb (obtained from the manufacturer Cesaroni). Inducing this load on the lip of the motor retainer symbolized the force applied from the motor itself during a launch process. The fixture was placed on both the bottom side of the retainer and the threaded holes. This was chosen as such since those sections will be rigidly attached to the rocket itself.

Running the simulation gave a minimum factor of safety of 285.2 and a maximum displacement of less than 0.2 μm, which meets our standards of performance for the design. Similarly, based on the von Mises stress theory, the maximum stress that the retainer undergoes is about 964 kN/m^2. Since the yield strength of the retainer is about 285 times greater than this (275000 kN/m^2) the retainer will not fail under the maximum load of the motor. These results can be seen below in Figures 3.2.4b and 3.2.4c. Note that the deformation shown is not to scale.

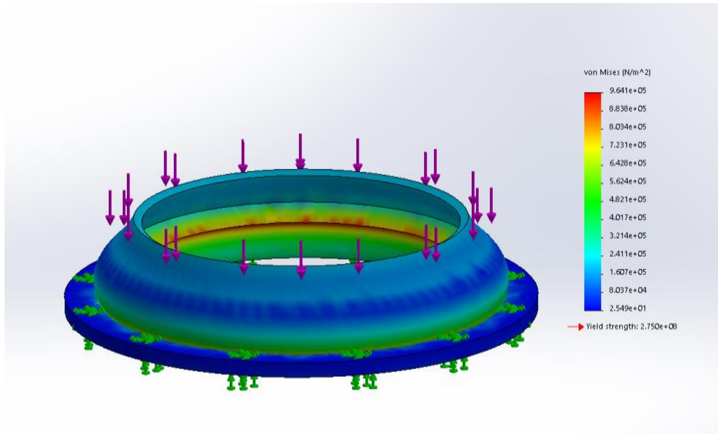


Figure 3.2.4b: Stress via von Mises theory for motor retainer.

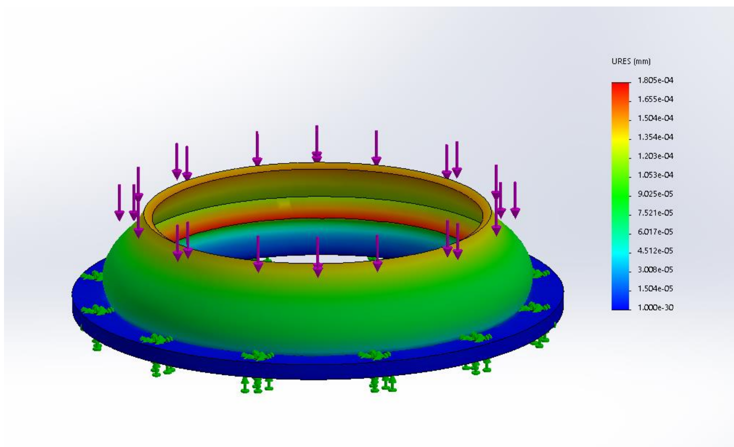


Figure 3.2.4c: Deformation of the motor retainer.

Next, the thrust plate must be simulated to show what will happen under the thrust of the motor. The thrust plate is a 4 inch - 75 millimeter manufactured by SC Precision, and it is composed of Aluminum 6061 (T6). This can be seen below in Figure 3.2.4d. The thrust plate was rigidly attached at the outer lip and the three #10 screw holes. This will represent the plate both against the motor mount tube (outer lip) and screwed into the bulkhead (screw holes). Its load will be induced on the opposite surface of the lip which will be the outer side that the motor retainer is fastened.

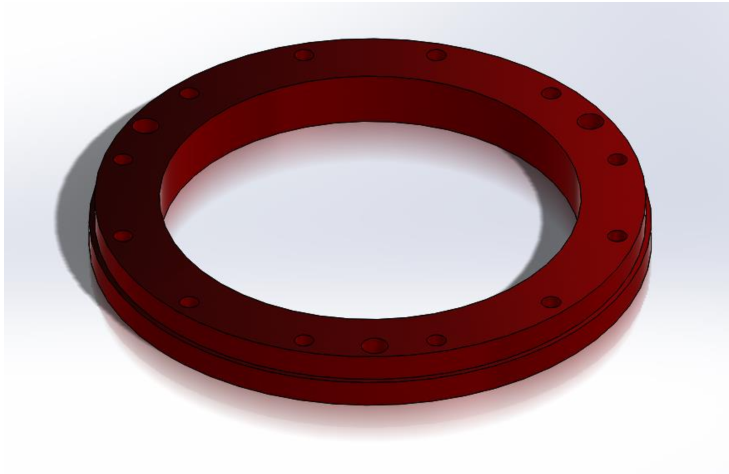


Figure 3.2.4d: Thrust plate CAD design.

Under the maximum thrust force of the motor, 646.7 N, the maximum displacement was about 1.2 μm with a minimum factor of safety of 55.6. Both of these are adequate for our purposes. Lastly, under the von Mises stress theory, the highest stress on the part is about 4950 kN/m^2 which means the part will not fail considering the yield strength of the part is 275000 kN/m^2. These results can be seen below in Figures 3.2.4e and 3.2.4f. Again, note that the deformation is not to scale.

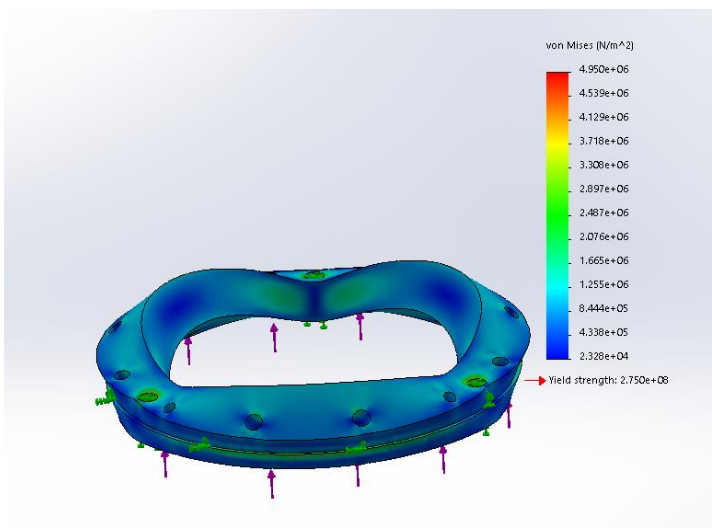


Figure 3.2.4e: Stress via von Mises theory for thrust plate.

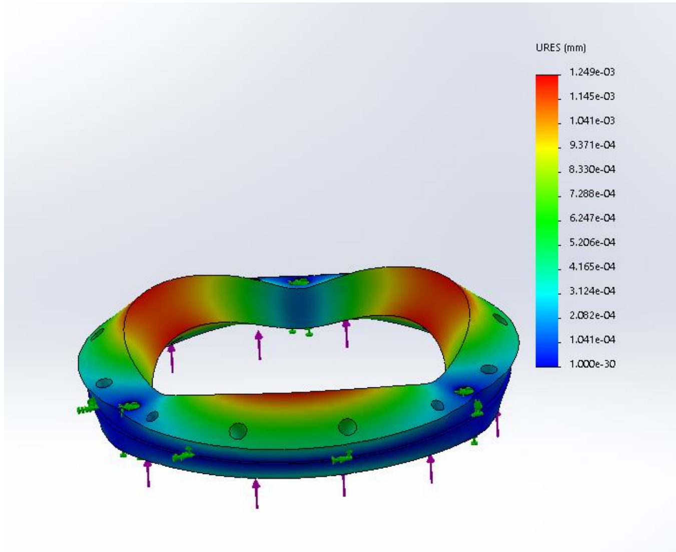


Figure 3.2.4f: Deformation of the thrust plate.

3.2.5 Avionics Bay System Design Review

After carefully reviewing all avionics bay construction methods and commercially available avionics bays, we have determined that manufacturing our own with 3D printing is the optimal solution. The ability to construct an avionics bay specific to our components is crucial in the decision. This will eliminate any wasted space within the avionics bay, reducing its mass.

Furthermore, 3D printing permits us to design an avionics bay using geometries that would otherwise be impossible for us to achieve using more traditional construction methods and materials that we have available to us. Another added benefit to 3D printing our bay is that we can make changes to the design and simply print a new one with minimal effort. This greatly speeds up the prototyping process as compared with typical manufacturing methods. While the print itself may take longer to complete than various other methods, it can be printed while the designer is working on other tasks, thus freeing up a substantial amount of manpower that typically would be lost to manufacturing.

The avionics bay consists of a long avionics sled mounted between two bulkheads. The total length of the avionics bay system is 79.032 centimeters. The avionics bay acts as a coupler between the forward and booster sections. The aft of the avionics bay is tethered to the main parachute and the booster section.

The avionics sled is printed using NylonX. This material was chosen because it's easy to 3-D print with and is considerably tougher and more durable than traditional 3-D printer filaments (ABS, PLA, and PETG). Brass inserts imbedded in the avionics sled, tape, and zipties are used to securely fasten all the avionics. The dimensions of the avionics sled are 24.49 cm x 8.26 cm x 0.66 cm. The total mass of the AV bay including the avionics is 18.32 ounces.

There were multiple limiting factors and constraints taken into account when determining the layout of the avionics bay electronics. One primary concern was to eliminate the need to have holes through the avionics sled for power and other connections. The reasoning behind this was twofold. Having no holes through the avionics sled helps minimize the difficulty of manufacturing, and secondly, makes it easier to RF shield the recovery system from the rest of the avionics components, specifically the LoRa transceiver. Another consideration was the position of the LoRa transceiver. The LoRa transceiver was placed in a location that will allow the antenna to be directly attached by SMA connector and then extent lengthwise down the avionics bay. The batteries and voltage regulator were placed with wire control in mind, to enable the wiring of the avionics bay to be organized. Finally, the PCB was placed in order to ensure that the Adafruit GPS patch antenna would be facing away from the avionics sled. This will decrease the probability of the GPS failing to get a fix due to antenna orientation. The design explanation for the placements of the electronics on the PCB is discussed in Section 3.4.3.

The bulkheads are made of sheet nylon and will be made using a CNC router. This material was chosen because of it's high strength, as sheet nylon is stronger than 3D printed nylon. The diameter of the bulkhead is 3.9" and the thickness is 0.37". The bulkhead closest to the booster section contains a 2.5" long steel eye bolt to hold the tether between the forward and the booster section. This eyebolt includes a shoulder so that we can use angular loading, and the eyebolt is closed to ensure it will not open up when the forces from the shock cord and parachute pull on it.

The avionics sled is supported by 2 partially threaded 6061-T6 aluminum rods. The rods go through the bulkheads and connect to aluminum nuts. Dampened washers are also utilized to absorb additional force that will be caused by the recovery system. All of the edges of the avionics sled are filleted to reduce stress concentrations.

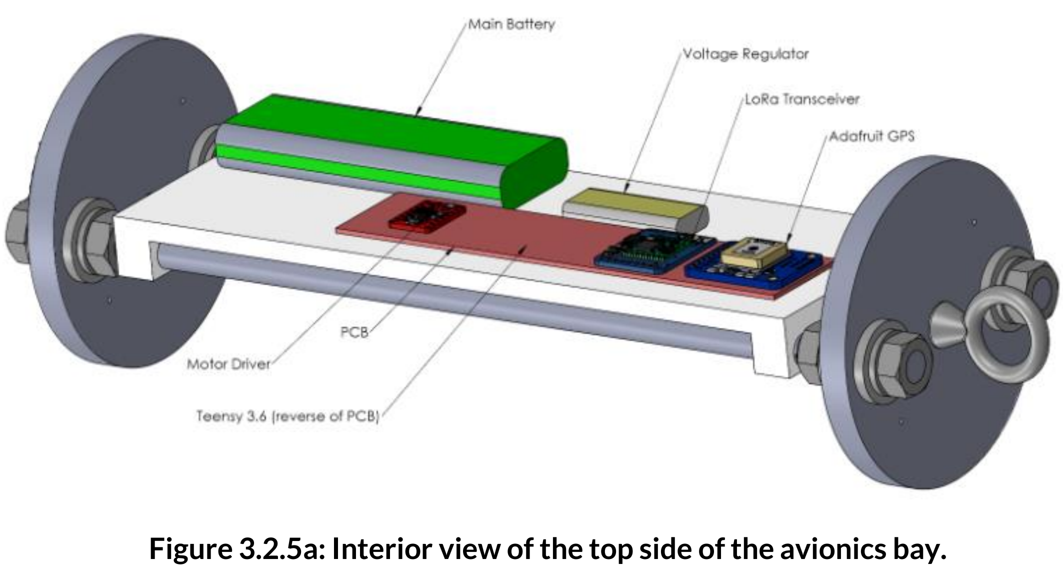


Figure 3.2.5a: Interior view of the top side of the avionics bay.

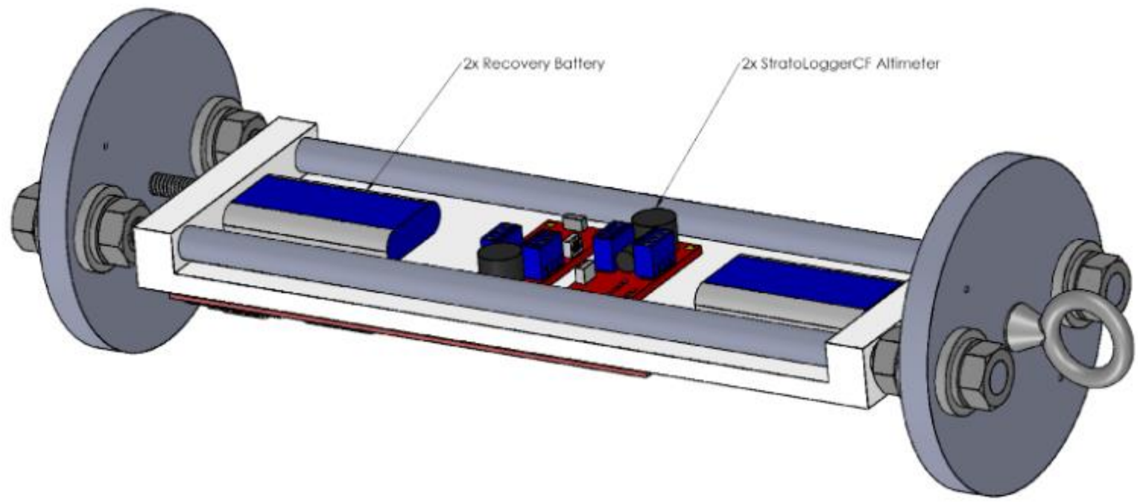


Figure 3.2.5b: Interior view of the bottom side of the avionics bay.

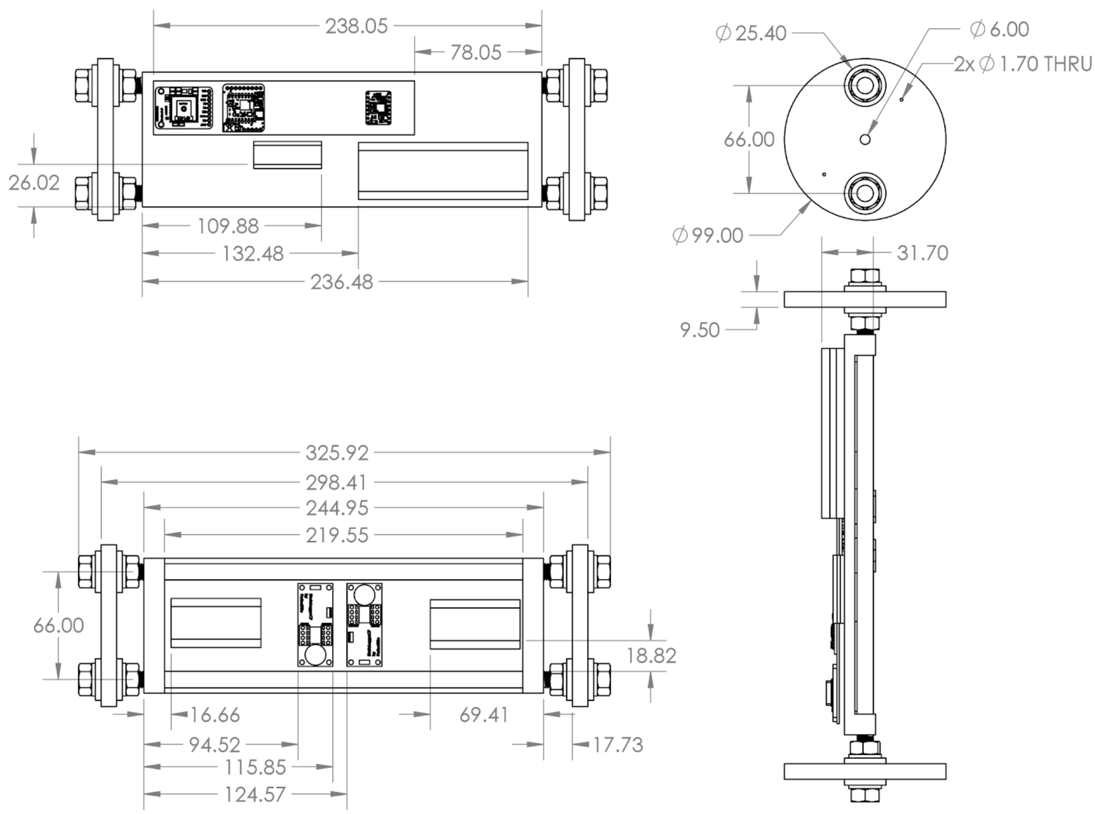


Figure 3.2.5c: Dimensional drawing of avionics bay interior.

Component	Mass (g)
Taoglas Limited TI.15.3113 - Heaviest Antenna	21
RFM9X LoRa Packet Radio Breakout	3.1
Adafruit Ultimate GPS Breakout	8.5
Teensey 3.6	4.9
Zippy Compact 2200mAh LiPo - Main Battery	144.5826
StratoLoggerCF (2)	10.773
Rotary Switches (2)	3.69
Turnigy nano-tech 460mAh - Recovery Battery (2)	31
Total (g):	**227.5456**

Table 3.2.5a: Mass list for the electrical components in the avionics bay.

For ease, the mass components list in Section 3.5 lists a total value for the avionics devices mass. A mass list for the non-electrical components of the avionics bay system is shown below in Table 3.2.5b.

Component	Mass (g)
Bulkhead (x2)	71.92
Washer (x4)	0.5
Hex Nut (x4)	1.47
Aluminum Rod (x2)	24.5
Sled Base	181.4
Abrasion Resistant Washer (x4)	1.5
Eyebolt	27.1
Total	**415.2**

Table 3.2.5b: Mass list for the non-electrical components in the avionics bay.

3.3 Subscale Flight Results

3.3.1 Basic Requirements

The design and fabrication of our subscale vehicle revolved around the following requirements of the NASA Student Launch:

- 2.19.1. The subscale model should resemble and perform as similarly as possible to the full-scale model, however, the full-scale will not be used as the subscale model.

- 2.19.2. The subscale model will carry an altimeter capable of recording the model's apogee altitude.
- 2.19.3. The subscale rocket must be a newly constructed rocket, designed and built specifically for this year's project.
- 2.19.4. Proof of a successful flight shall be supplied in the CDR report. Altimeter data output may be used to meet this requirement

3.3.2 Vehicle Design Review

For our full scale vehicle we elected to buy to the Darkstar Extreme kit from Wildman Rocketry. The same supplier sells a kit called the Darkstar Jr., which is a close 1:2 subscale of the extreme. Using the Jr. kit allowed us to create a subscale rocket with dimensions that matched our desired ratio more closely than if we had tried to piece together fiberglass parts from different manufacturers.

As a close subscale of the Extreme, the Darkstar Jr. has a matching Von Karman nose cone shape and a 1:2 cross sectional area, which ensures that to a decent approximation the aerodynamic effects we expect on our full scale rocket will be seen during the subscale launch. Additionally, the positions of our CP and CG points with respect to the forward-most point were half of the same locations in the full scale to emulate the dynamics of the final vehicle as much as possible. Aside from these parameters, the difference in mass distribution resulting from the interior placement of components such as electronics should have no bearing on the flight performance.

There are several notable implications resulting from the difference in the dimensions of the subscale vehicle and the dimensions of the full scale vehicle. For example, since the subscale avionics bay was 0.167 the volume of the full scale avionics bay (compared to a scaling factor between 0.5 and 0.704 for every other measurement), the contents of the subscale avionics bay had to be dramatically reduced from the contents present in the full scale avionics bay. In the subscale avionics bay, only one stratologger and one lithium-polymer battery were present instead of the two stratologgers and lithium-polymer batteries that will be present in the full scale avionics bay. Additionally, the avionics bay was not large enough to support the GPS subsystem.

Dimension	Subscale Measurement	Full scale measurement	Scaling factor
Total length	147.3 cm	241.05 cm	0.611
Length of avionics bay	17.77 cm	28.0 cm	0.635
Width of sled	17.25 cm	24.49 cm	0.704
Length of sled	4.82 cm	8.26 cm	0.584
Inner diameter of avionics bay	5.08 cm	10.16 cm	0.500

Diameter of bulkhead	5.42 cm	9.906 cm	0.547
Volume of avionics bay	360.17 cm^3	2157.97 cm^3	0.167

Table 3.3.2a: Scaling factors between the subscale vehicle and the full scale vehicle.

Not all subscale variables are different than full scale. The same airframe, fin, and parachute materials were used f with them for subscale which can now be applied for full scale fabrication. Had these variables not been constant, future full scale fabrication would include a steeper learning curve and not allow us to teach other members in full scale fabrication techniques, leading to longer production times. In learning the manufacturing process earlier on, less mistakes will be made on the materials intended for the full scale airframe and avionics bay.

3.3.3 Flight Analysis

Launch of our subscale rocket occured on Saturday January 5th, 2019 in Dayton, OH. Predicted apogee height from an OpenRocket simulation was 2,904 feet (Figure 3.3.3a). Apogee height during out flight was 2,620 feet. The complete log recovered from the onboard stratologger can be found in Appendix 7.3.

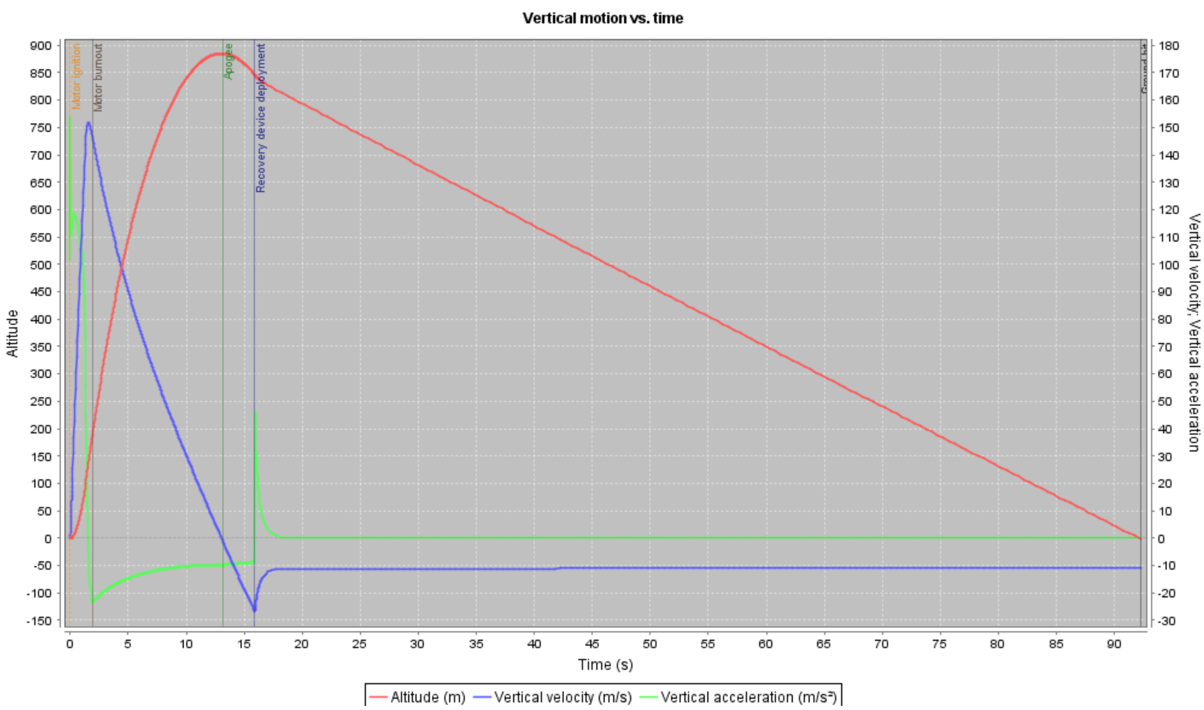


Figure 3.3.3a: Subscale flight simulation.

By performing the altitude backtracking method described in Tim Van Milligan's paper on determination of drag coefficients from altimeter data, we arrived at a coefficient of drag of 0.865 compared to the 0.745 calculated by Open Rocket.

This result, however, cannot be extrapolated for the full scale rocket. Due to inexperience with sanding and epoxying practices, we failed to produce a smooth finish in our subscale rocket. The final product was a representation of the aerodynamics of our full scale rocket only insofar as it had the same proportions. In many places around the airframe there were protrusions that could have been avoided with a more careful approach to the manufacturing methods we used.

After extensive conversations with our mentor, we have gained a better understanding of how to proceed with full scale fabrication. Our current manufacturing plan details all the necessary steps to create a smooth finish, and the airframe of the final rocket should reflect the best practices known in model and high-power rocketry.

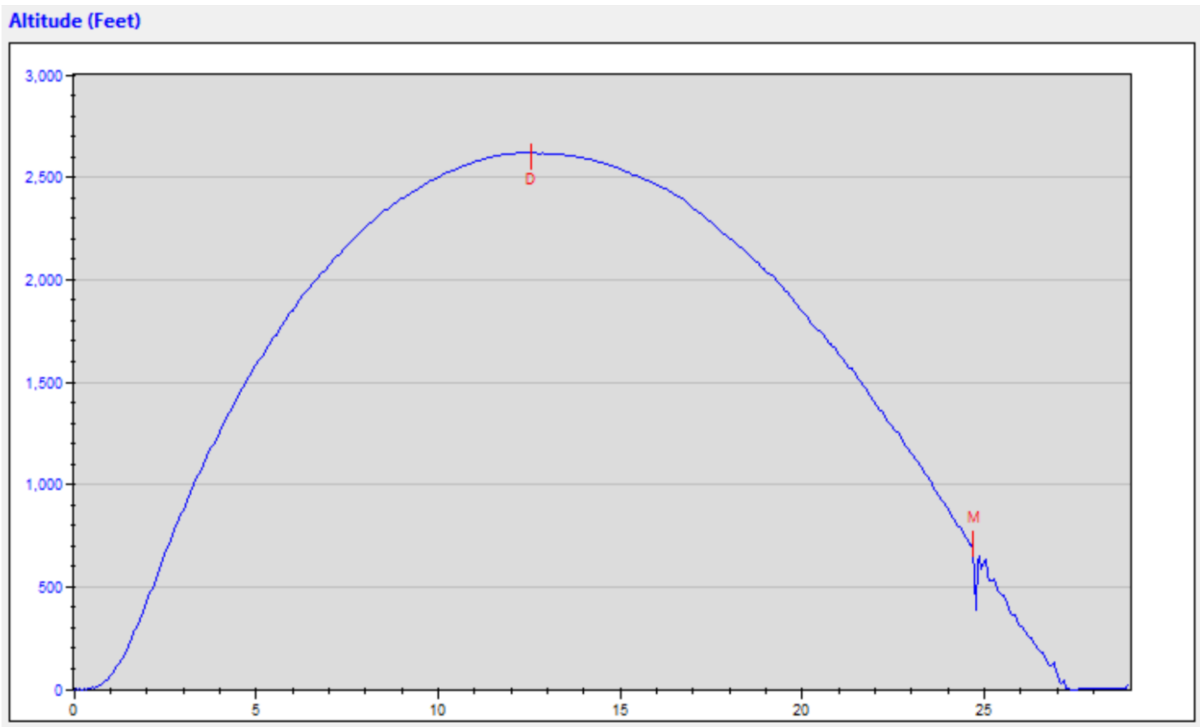


Figure 3.3.3b: Subscale flight, altitude (ft) versus time (s).

Figure 3.3.3b above shows a plot of data recovered from our stratologger.

3.3.4 Flight Performance Review

Observations: The altimeter was turned on using a rotary switch accessible from the outside of the airframe. After the pad was cleared, the ignition signal was sent. The upward trajectory adhered to our expectations. No ejection charged detonation was heard when the vehicle reached apogee, which indicated that the black powder had failed to detonate. The main parachute did deploy when the rocket fell to an approximate altitude of 500 feet, but the vehicle did not seem to slow down.

We successfully located all parts of the rocket. The shock cord tying the nose cone to the forward section broke at some point during the flight, so the main parachute remained with the nose cone while the rest of the rocket experience unbuffered freefall. Figures 3.3.4a and 3.3.4b show the state of the rocket when it was found.

Figure 3.3.4a: Landing site of the rocket (minus nose cone and main parachute).

Figure 3.3.4b: Damages to the subscale rocket.

List of damages:

- The shock cord connecting the nose cone to the forward section snapped.
- Main parachute ripped.
- Aluminum tip of the nose cone was lost.
- Forward tube was partly destroyed.
- Fins sheared off.
- Battery became disconnected at some point during the flight.

Causes:

- The main parachute was not designed to sustain the load created by attempting to slow down a vehicle that had not previously been slowed down by a drogue parachute, so it ripped when deployed.
- The shock cord connecting the nose cone to the avionics bay snapped for the same reason.
- Given that the nose cone's freefall was not slowed down, it must have lost its aluminum tip upon impact with the ground.
- The fins were found directly next to the rocket in the landing site (Figure 3.3.4a) which indicates that they only broke off when the vehicle made impact with the ground.

Root cause:

- The failure of the drogue ejection charge to ignite is responsible for all the damages that occurred following that point. After inspecting our hardware, we found that one of the solid copper wires connecting the altimeter to the ejection canisters had an internal kink that would not have been visible externally. This likely resulted in lack of continuity for the cable, explaining the failure of the drogue ejection charge to ignite.

3.3.5 Design Impact

Mitigation of Risks:

In order to prevent the same problem from happening again, we will check continuity on all cables before flight. We should also use more flexible stranded wires that are less prone to kinks.

Additionally, we must add redundancy to several of our subsystems to improve the probability that ejection will occur. For that reason, as discussed in Section 3.4.3 we have decided to wire both altimeters in the avionics bay to each black powder charge. If one of the stratologgers is malfunctioning, or a cable is faulty, the ejection charge will still ignite.

We also intend to add redundancy to the harnessing system. It is possible, for each of the sections where shock cord is used, we will add a second shock cord of greater length. The second cord will only come into play in the case where the shorter one snapped and given that the shorter one received the full load of ejection, the longer cord is unlikely to break.

Although we can minimize the probability that parachute deployment will fail, we can also make key structures mo rubber spaces can be added near the bulkheads to absorb impact forces on landing, mitigating the risk of damaging critical avionics systems.

For convenience, we have decided to make the fin system of the full scale rocket removable, which would allow us to swap them for new ones in the case where a fin breaks during landing.

From a safety perspective, our subscale launch has been very instructive. We practiced all launch safety procedures and best practices for assembly of the rocket.

3.4 Avionics System Design

3.4.1 Objectives

The rocket's avionics is split into two systems: the recovery avionics system and the main avionics system. The recovery system is entirely independent from the main avionics system and is discussed in the following section. Its purpose is to ensure the safe recovery of the rocket following apogee. The main avionics system is comprised of an onboard system and a ground system. The purpose of the main avionics system is to locate the rocket after landing and to deploy the rover following a successful landing. The main avionics system transmits all data collected from the GPS unit to the ground system and the recovery avionics system logs all data collected from the altimeters for use in both competition judging and future analysis.

3.4.2 Success Criteria

The avionics system will be considered successful if the rocket is fully recovered and the rover is successfully deployed. The recovery avionics system must ensure that the rocket makes a safe descent following apogee and the main avionics system must ensure that the rocket is located following landing. The recovery avionics system must fire ejection charges at the correct time to deploy the drogue and parachute. The main avionics system must transmit GPS coordinates to the ground system so the rocket can be found following landing and release the rover when it receives a command to do so. In addition, the system must receive the signal to release the rover and properly liberate it from the rocket.

3.4.3 Main Avionics System Overview

Figure 3.4.3a depicts the block diagram for the entire avionics system (revised to reflect changes to the avionics system since the Preliminary Design Review) including the main avionics system, the avionics recovery system, and the avionics ground system. The revisions to this diagram, as mentioned under Section 2.1 are (1) the replacement of the HC-12 transceivers with the more reliable RFM9X LoRa Packet Radio Breakout transceivers, (2) the elimination of one of the original two transceiver pairs, and (3) the elimination of the unnecessarily redundant remote ignition safety system.

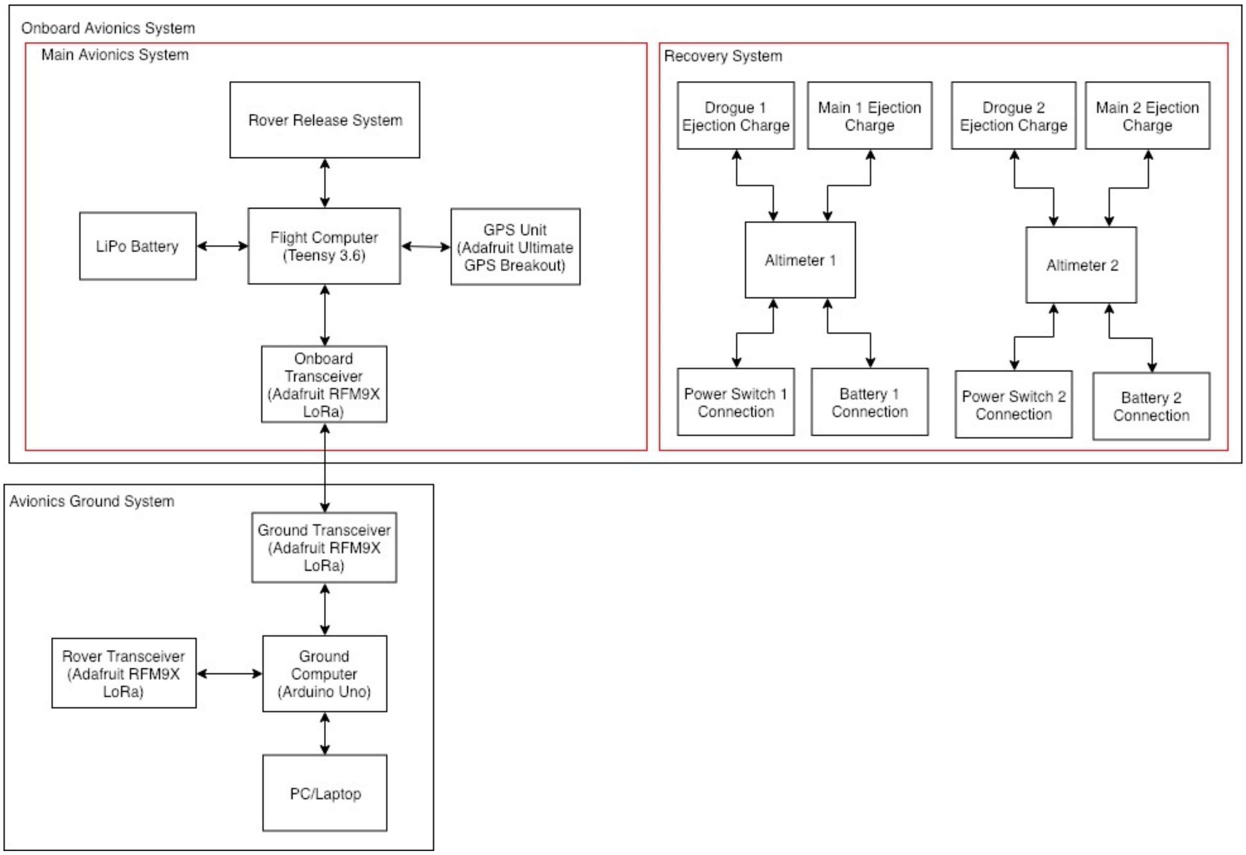


Figure 3.4.3a: The revised block diagram for the avionics system.

A revision that cannot be seen in the new block diagram is the decision to put our avionics system on a single printed circuit board. During testing, it became clear that the wiring of components would be nontrivial and that effort should be made to minimize the number of exposed wires on the final flight assembly. For this reason, we decided to manufacture the main avionics system as a printed circuit board. In addition to the minimization of loose wires, this configuration saves valuable space in the avionics bay and on the avionics sled for a larger, more robust antenna to increase strength and range of the transceiver signals. The flight computer, transceiver, GPS, and stepper motor driver will all be integrated into a single unified PCB as shown in Figure 3.4.3b.

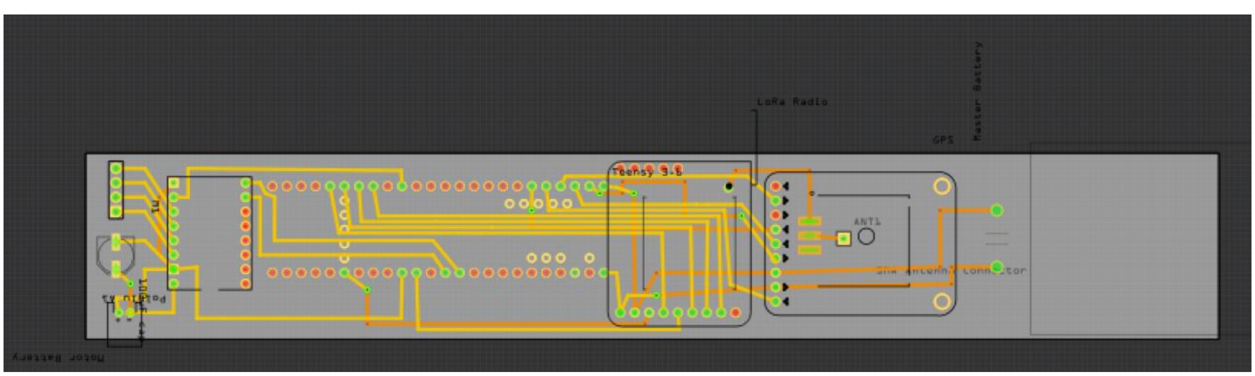


Figure 3.4.3b PCB routing diagram.

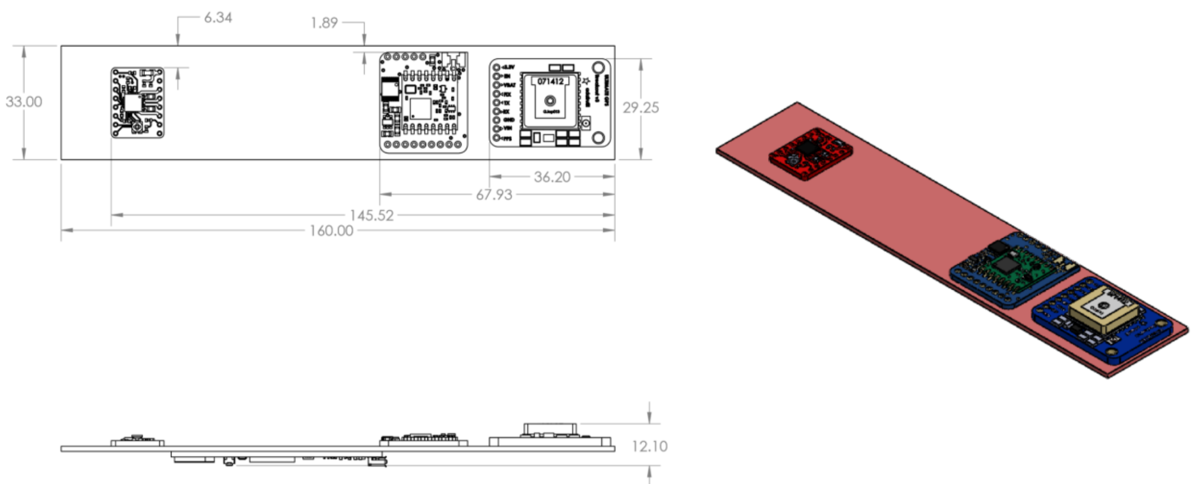


Figure 3.4.3c: CAD model of the PCB integrating the GPS module, the transceiver module, and the stepper motor driver with the flight computer. Units in millimeters.

Voltage regulator and battery will accompany the PCB on one side of the avionics sled and the other side will support the recovery system electronics. The sled will be coated with 219.98 cm^2 of nickel spray RF shielding to protect the recovery system from RF interference produced by the GPS transceiver and other components. CAD models depicting the avionics bay are below:

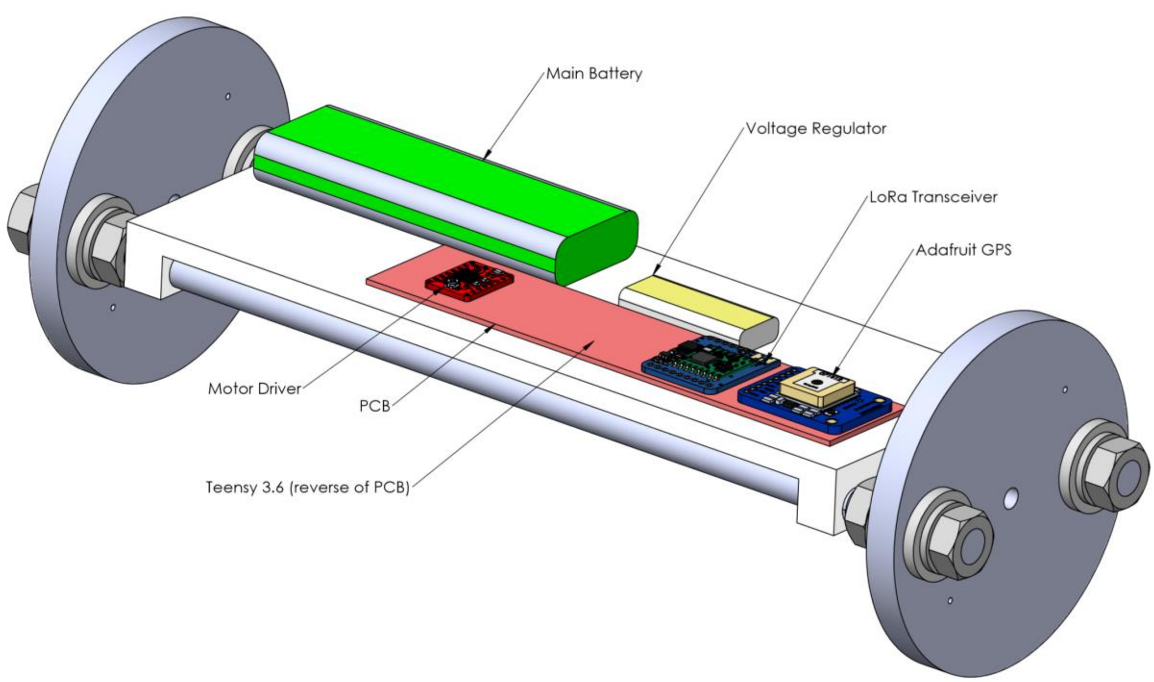


Figure 3.4.3d: CAD model of the top side of the avionics sled inside the avionics bay. Note that since final antenna selection is ongoing there is no antenna connected to the transceiver, however, the final CAD model will also feature an omnidirectional antenna.

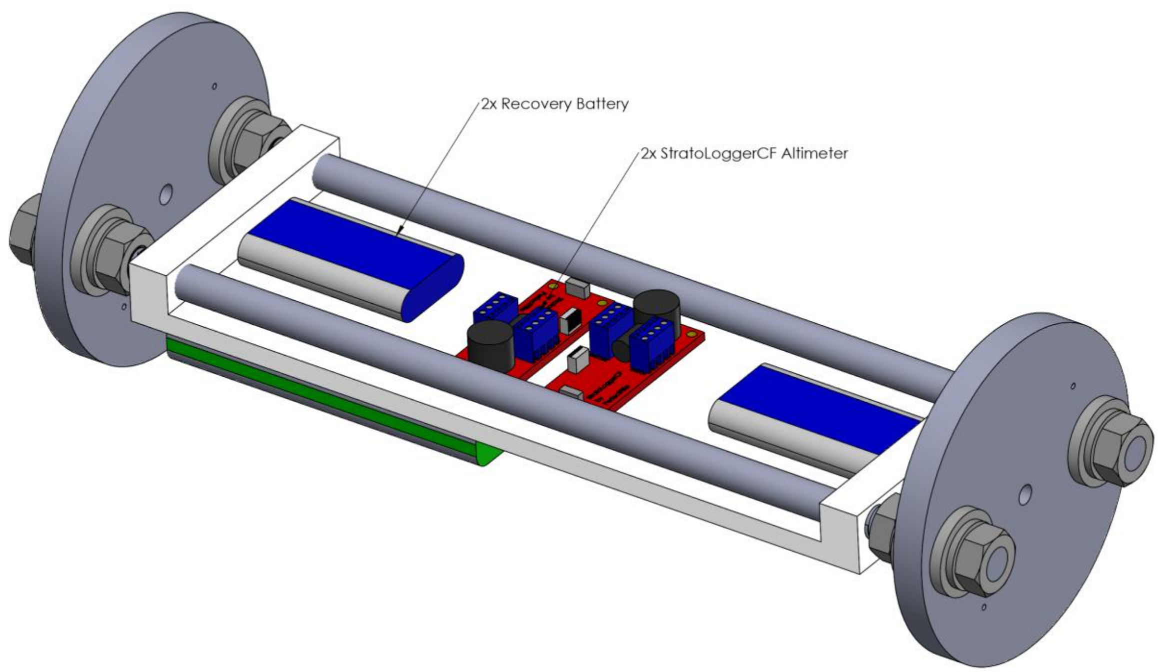


Figure 3.4.3e: CAD model of the bottom side of the avionics sled inside the avionics bay.

3.4.4 Power

The PCB containing the main avionics system will be powered by a Zippy 2200 mAh 2S 25C lithium-polymer battery with a HexTronik 5/6V 3A UBEC switch-mode voltage regulator. The chosen battery has a nominal voltage of 7.4 volts, a rated energy capacity of 16.28 watt-hours, and a rated current output of up to 55 amps sustained. The chosen voltage regulator has an input voltage range of 5.5-23 volts which fully encompasses all safe operating voltages supplied by our battery. The regulator will operate in its 5V-output mode since all on-board electronics are capable of operating at this voltage. The actual measured output-voltage for this particular regulator is 5.39V, however this still falls within spec of all the electronics it will be powering.

Figure 3.4.4a: Zippy 2200 mAh 2S lithium-polymer battery.

The sum of the rated peak current-draws for all avionics-bay electronics (with the exception of the stepper motor) should be 0.483 amps. This falls well within the rated 3 amps that can be supplied by our voltage regulator. At 5.39 volts, this 0.483 amp current draw translates to a power draw of 2.603 watts. Our regulator does not have any official efficiency ratings, however testing showed its power-conversion efficiency to be roughly 85% under similar loads. Assuming 85% efficiency, the power pulled from our battery would be 3.062W.

The stepper motor used for the rover release mechanism will be powered directly by the battery, bypassing the 5V regulator. Our chosen stepper motor has 2 phases with 20Ω resistance per phase. At 7.4V, this equates to a power draw of 5.476W. Assuming the stepper motor must run for 5 minutes to release the rover (this is likely an overestimate), the act of releasing the rover should consume roughly 1.642kJ or 0.4561 Wh.

If 0.4561Wh is to be consumed during rover release, 15.824Wh remains to power the remainder of the on-board electronics for the duration of the flight, recovery, and pre-flight wait time. Assuming the on-board electronics constantly draw 3.062W from the battery, the system could run for 5.17 hours before the battery is fully depleted.

This 5.17 hour figure was calculated assuming all electronics in the avionics bay were operating at peak current simultaneously for the entirety of the duration. The actual *average* power draw will likely be significantly less than 3.062W. Thus, the avionics bay should be capable of running for much longer than our estimated figure (although such a prolonged run-time should not be necessary).

3.4.5 Main Control Unit

Figure 3.4.5a: A Teensy 3.6 microcontroller.

The Teensy 3.6 is the best microcontroller choice as it meets all the required specifications for running the GPS, transceiver, and stepper driver simultaneously. Compatibility with Arduino libraries make programming the board easy and familiar. The Teensy board has more capabilities than Arduino boards, and is perfect for final products as they can be directly soldered to printed circuit boards. Arduinos such as the Mega will be used for testing all sensors on the launch vehicle, and the Teensy will be on the final product. Teensies run off code from the Arduino IDE, making it much easier to use and setup than other advanced microcontrollers. The Teensy 3.6 also had the smallest area and mass of all the microcontrollers that were up for consideration. In the end, the teensy was chosen because of its satisfactory capabilities, familiarity and ease of use, and compact dimensions.

3.4.6 GPS Subsystem

The first design decision considered in PDR was whether to have two separate transceiver pairs (one of the rover release command and one for GPS telemetry) or one transceiver pair with a custom data framing scheme to determine whether telemetry received is for the rover release system or for the GPS system. These two network design options are shown in Figure 3.4.6a.

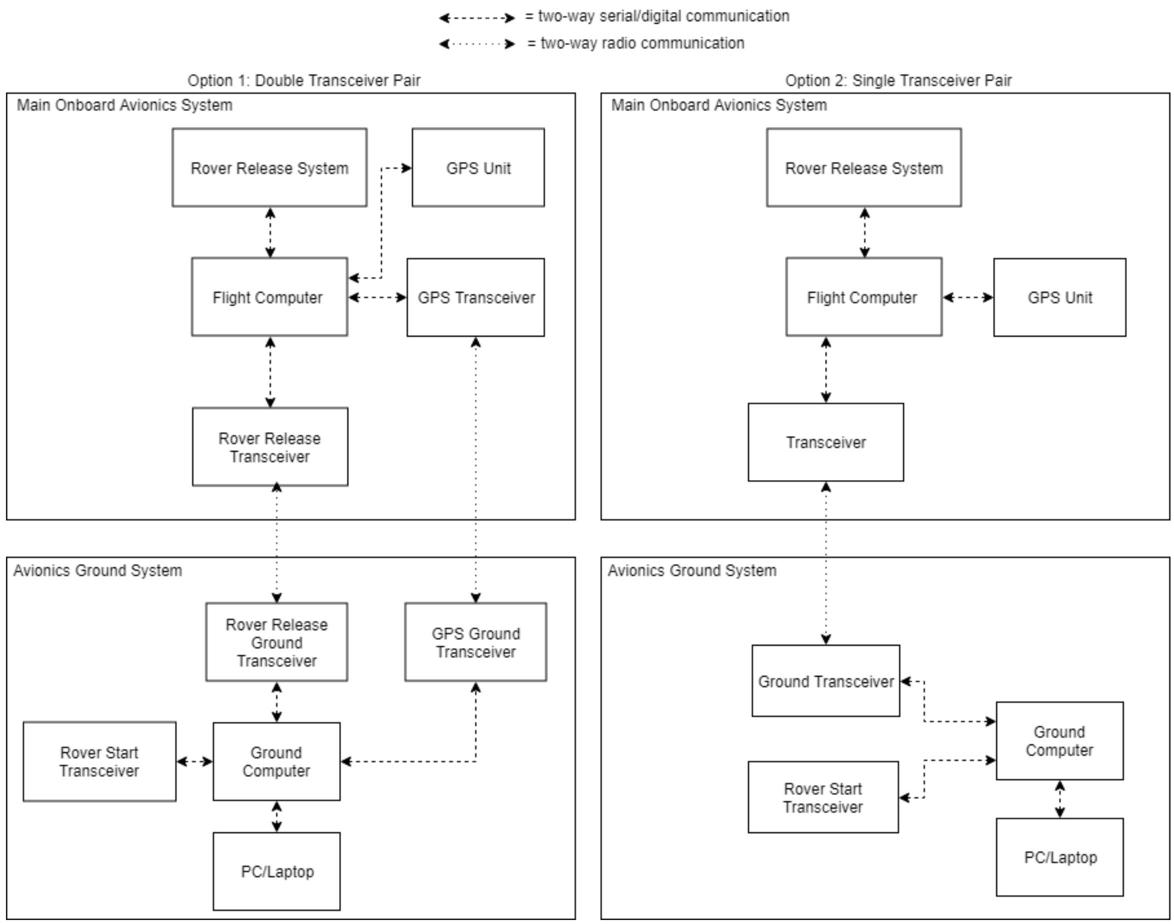


Figure 3.4.6a: Network diagram with one transceiver pair (left) and two transceiver pairs (right).

The double transceiver pair design was originally chosen for two reasons. The first was to minimize data framing complexity. There were concerns that if the data framing scheme somehow failed that this could result in either the rover not being released or the GPS data being corrupted, both of which are unacceptable. The second reason was because the transceivers that had been chosen in the design (HC-12 transceivers) are very inexpensive. The cost of doubling the number of transceivers in the system was deemed minimal compared to the risks to both the rover deployment and GPS functionality incurred with designing a single transceiver pair system. The risk incurred with this design, however, was that the two transceiver pairs could interfere with each other.

We initially chose the HC-12 transceiver, however, testing revealed that in practice the HC-12 transceivers do not meet our range requirements, only transceiving up to around 22.2 meters. After replacing the HC-12 transceivers with the RFM9X LoRa transceivers and performing several tests of the single transceiver pair design and the double transceiver pair design, the team determined that the single transceiver pair design is a much safer choice. The first reason the team originally chose the double transceiver pair design, to reduce complexity in the GPS data framing scheme, turned out to be a trivial concern once the the HC-12 transceivers were replaced with the RFM9X LoRa transceivers, which offer superior documentation and greater ease-of-use. The second reason, the relatively low cost of transceivers, became invalid when the team chose to use the RFM9X LoRa transceivers which are almost three times the cost of the HC-12 transceivers. While the team was not able to test whether or not the new transceiver pairs would interfere with each other, the team opted for the simpler design, the single transceiver pair, since the primary reasons for choosing the double transceiver pair were no longer valid.

Another decision the team faced was whether or not to rely on the GPS unit in the rover as the primary GPS for locating and recovering the rocket. While this would remove potential impedance of the fiberglass airframe on the antenna, it was decided that it is bad practice to rely on the performance of a non-critical component (namely, the payload) for such important functionality as the ability to locate the rocket after landing. If something were to happen to cause the rover to fail it would adversely affect not only the rover's operation but the ability to track the rocket after landing. For this reason, the main avionics system now has its own GPS unit.

Once these decisions were finalized, a final wiring diagrams for testing purposes, shown in Figures 3.4.6b and 3.4.6c, were created.

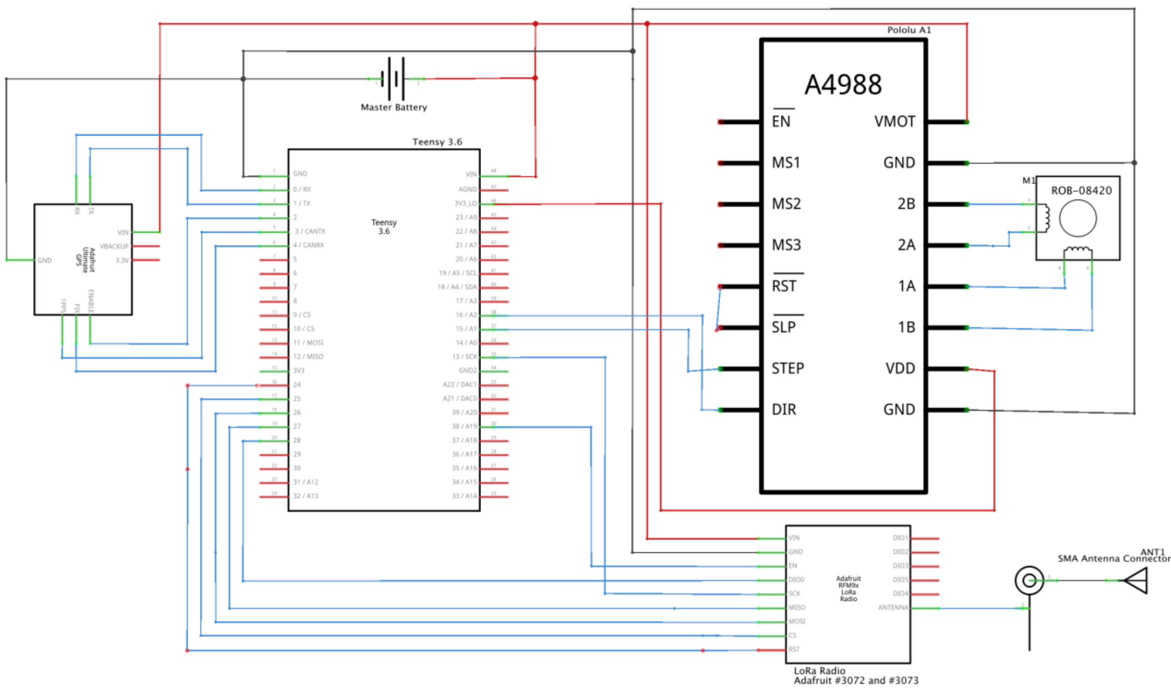


Figure 3.4.6b: The wiring diagram for the onboard main avionics system.

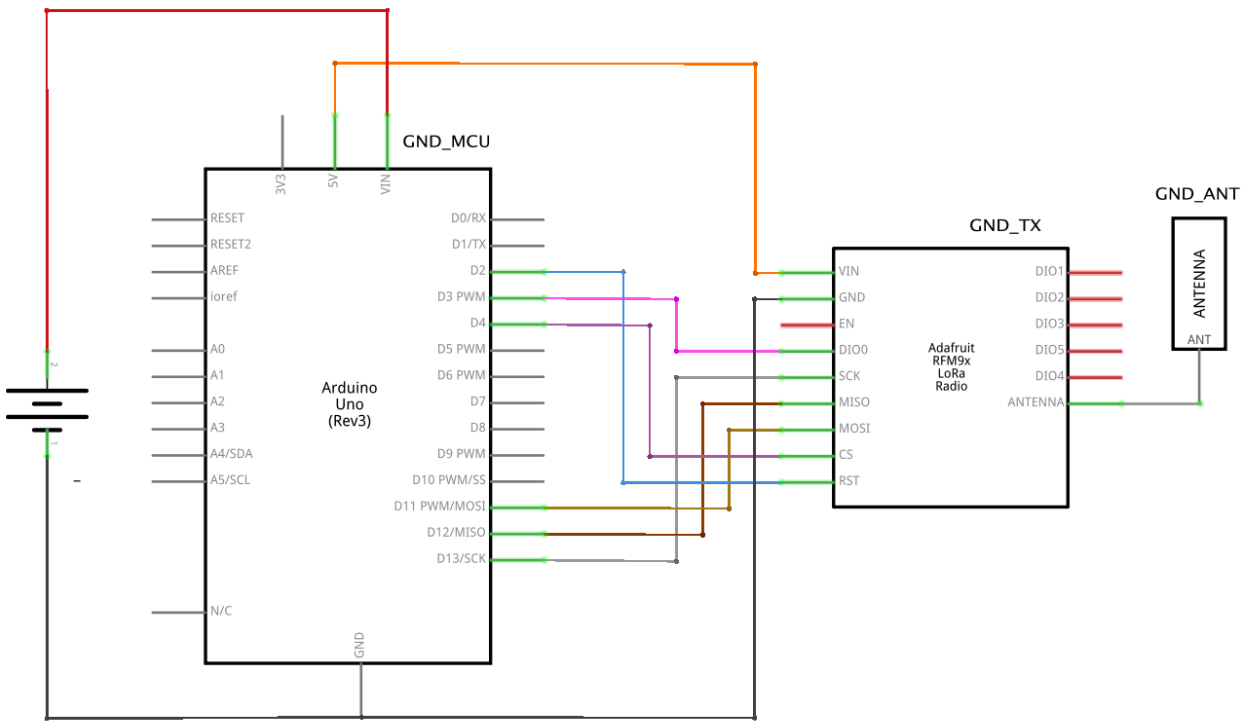


Figure 3.4.6c: The wiring diagram for the ground system.

Our GPS subsystem consists of an Adafruit GPS breakout board, a Teensy 3.6, and an Adafruit LoRa radio board. NMEA sentences from the GPS module are read by the Teensy and transmitted to the ground station by means of the LoRa module which is received by a paired LoRa module. The Teensy 3.6 will read NMEA sentences from the GPS breakout board, and feed it to the LORA module to transmit to the ground station. The communication procedure section further defines this process. The LoRa module in the ground station will receive the data and the Arduino Uno in the ground station will parse and then relay the incoming data to the ground system operator so the vehicle can be located following landing.

Through research and testing of the GPS modules and transceivers, we have concluded that the Adafruit GPS breakout board remains as the best choice for the GPS subsystem, as no stability, data integrity, or any other observations that could pose a threat to the mission critical nature of the subsystem were identified.

No change was made to the high level structure of our GPS subsystem. A GPS receiver module connects to our flight computer on the vehicle, and transmits through a transceiver module with an antenna to a matching transceiver on the ground station. The transceiver is responsible for sending GPS data and other telemetry from the rocket to the ground station, in addition to allowing the ground station to send a rover release command to the rocket.

The second major part of the GPS subsystem is the LoRa transceiver module. After performing tests to investigate the data integrity of the HC-12 transceivers at different distance increments, it was determined that the HC-12 boards were no longer a suitable option for the avionics subsystem. Table 3.4.6a compares the HC-12 to the Adafruit RFM95W. Immediately, the potential range and superior documentation make the Adafruit transceiver a better choice than the HC-12.

Model & Manufacturer	**HC-12 Wireless Serial Port Communication Module**	**Adafruit RFM95W LoRa Radio Transceiver Breakout 433 MHz**	**XBee Pro 60mW U.FL Connection - - Series 1**	**Adafruit RFM95W LoRa Radio Transceiver Breakout ~900 MHz**
Cost	$6	$19.95	$37.95	$19.95
Mass	2 grams	3.1 grams	3.4 grams	3.1 grams
Transmission Power	100 mW	100 mW	60 mW	100 mW
Max range (best case)	1 km (FU3 mode); 1.8 km (FU4 mode)	Up to 2 km	750 m	Up to 2 km
Operating frequency	433.4 MHz to 473.0 MHz	433 MHz	2.4 GHz	868 MHz - 915 MHz
Expected operating baud rate	1.2 kbps (FU4 mode)	Up to 300 kbps	Up to 250 kbps	Up to 300 kbps
Supply voltage	3.2V or 5.5V	3.3V or 5V	2.8V or 3.3 V	3.3V or 5V
Quality of documentation	Fair	Excellent	Excellent	Fair

Table 3.4.6a: Comparison of transceivers for use in GPS system.

The Adafruit RFM95W is a suitable replacement for the HC-12 transceiver because of its ability to operate within existing parameters for the transceiver section of the avionics subsystem. Both the HC-12 and the RFM95W operate at 433 MHz. This is essential for ensuring that the transceiver can be legally operated with a HAM technician's license. The operating voltage of the RFM95W is the same as the HC-12's operating voltages; as a result, no other parts of the avionics subsystem needed to be changed to implement the new transceiver. Initial testing showed that the the Adafruit RFM95W is capable of greater ranges than the HC-12 transceiver. This leads us to conclude that the RFM95W is a better choice of transceiver, yet testing the RFM95W to a more thorough extent will solidify this decision.

Communication Procedure

The Adafruit GPS module outputs GPS data in the standardized NMEA format. Because this format consists of character strings, the NMEA sentences are transmitted directly to the transceiver without requiring processing on the flight computer. All processing is done by the ground system which is further explained Section 3.4.8.

In addition to transmitting GPS data to the ground station, the flight computer is responsible for receiving the rover release command from the ground computer. Our current transmission protocol involves the flight computer sending data to the ground the computer. Upon receiving data, the ground computer sends a response to confirm receiving data. If a rover release is commanded, the response from the ground computer contains a command keyword. The flight computer is designed to look for this keyword and execute the proper actions.

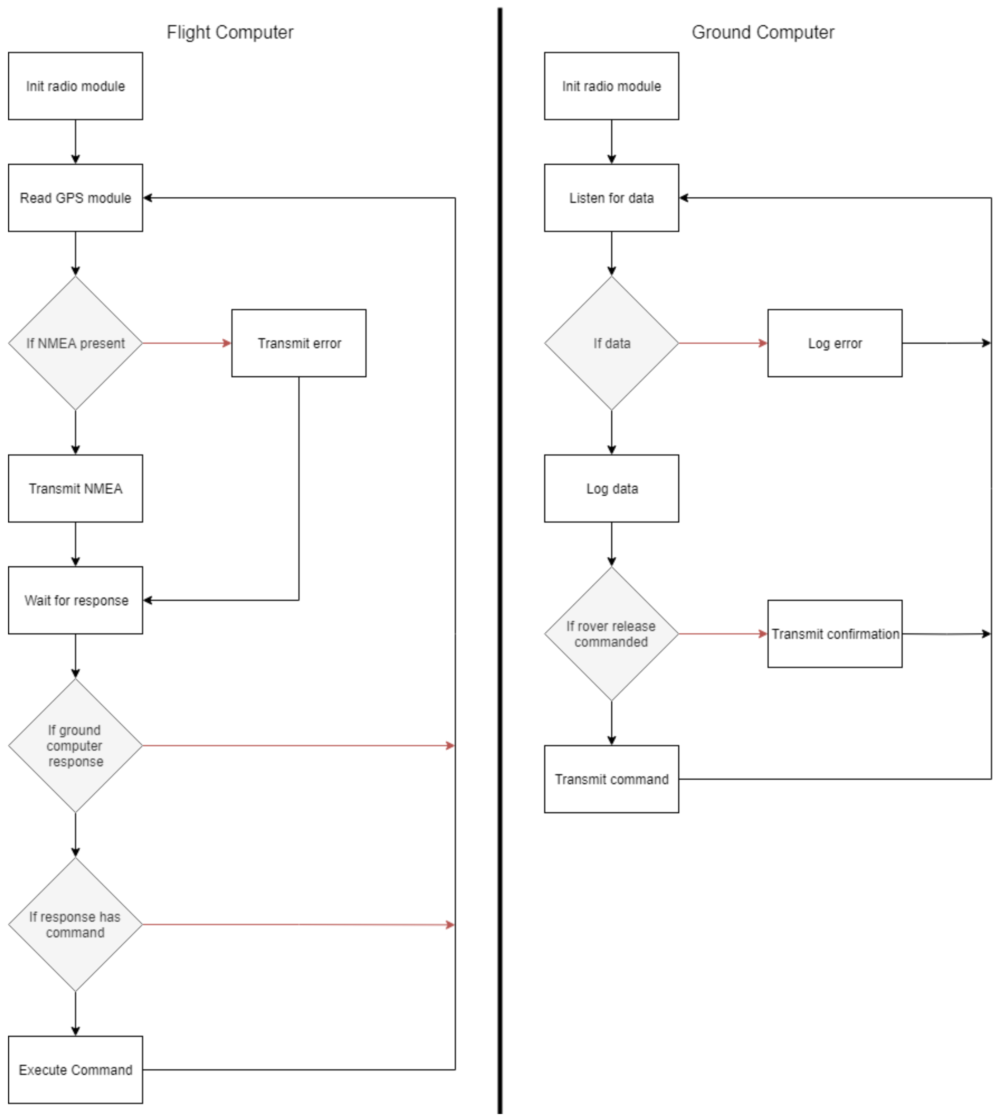


Figure 3.4.6d: Flow chart of flight computer operation (left) and ground computer operation (right).

The second consideration is related to the range of the HC-12 transceiver. Without any modifications, the maximum ideal range of the transceiver is 1 kilometer (about 3280 feet), but our own tests indicated much poorer performance, indicating max ranges of about 76 feet with default antennas. These findings posed concerns considering our target apogee is 4750 feet and the the recovery area is limited to a 2500 foot range from the launch pads. Without extending the range of the transceivers, they are guaranteed to disconnect prior to achieving apogee. This means two different problems need to be tackled: first, the range of the transceivers needs to be extended; second, the transceivers will need to be able to reconnect when they disconnect. Since the recovery area is within the transceiver base range, as long as the transceivers can reconnect upon landing and the transceivers can operate at the range claimed by the spec sheet, there will be no problems locating the vehicle. Upon researching the HC-12 transceiver further, it was discovered that the transceiver actually has an FU4 mode which supports a 1.2 kbps maximum baud rate at a 1.8 kilometer (about 5900 foot) maximum range. Since this mode's maximum range exceeds our target apogee and the low baud rate is sufficient for transmitting GPS data (capable of transmitting 60-byte packets with a minimum transmission time interval of at least 2 seconds), we initially believed the HC-12 transceivers to be sufficient for our needs. Our testing, however, revealed that the HC-12 modules were finicky, difficult to initialize, and generally unreliable, leading the team to look into alternatives.

Yet another consideration in designing the custom GPS system was whether to have the transceiver pairs perform a handshake transmission to initiate data transmission or frequency hopping. Both handshaking and frequency hopping can be used to resist interference and make data transmissions more difficult to intercept. Fortunately, the Adafruit RFM95W LoRa transceivers also use handshaking to initiate data transmission so this decision did not need to be reevaluated.

Antenna Selection

The final consideration in designing the custom GPS system was which antenna to choose. We initially planned on using a SMAKN 433MHz 11 cm Omnidirectional Antenna for both the on-board antenna and the ground system antenna. Two primary range tests using the SMAKN 11 centimeter omnidirectional transceivers paired with the RFM9X LoRa transceivers were performed: one test in which the antennas were unobstructed and one in which one antenna was obstructed by the same amount of fiberglass that will obstruct the antenna on the actual vehicle. To determine the strength of the signal between the two transceivers, we transmitted the received signal strength indicator (RSSI) as well as GPS coordinates. One transceiver remained stationary while another was walked across a flat field. We created a simple python script which uses the Haversine formula to calculate the distance between the stationary transceiver coordinates and the moving transceiver coordinates. This yielded RSSI values and GPS coordinates which we then analyzed. Plots of the data from these two tests as well as regression lines are shown below.

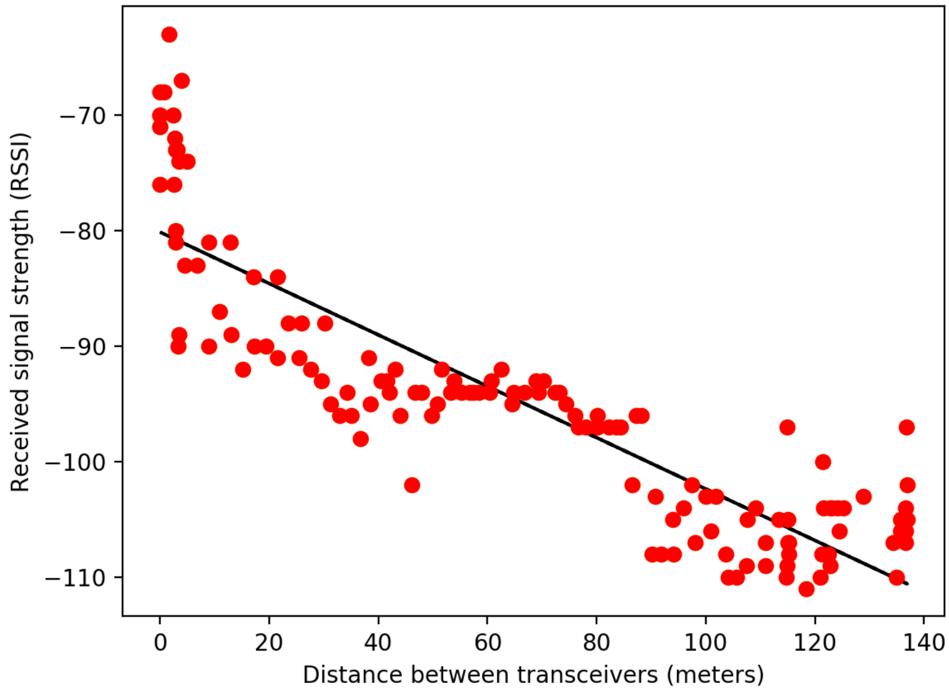


Figure 3.4.6e: Plot of data collected from unobstructed range tests of the RFM9X LoRa transceivers with the SMAKN 11 centimeter omnidirectional antennas.

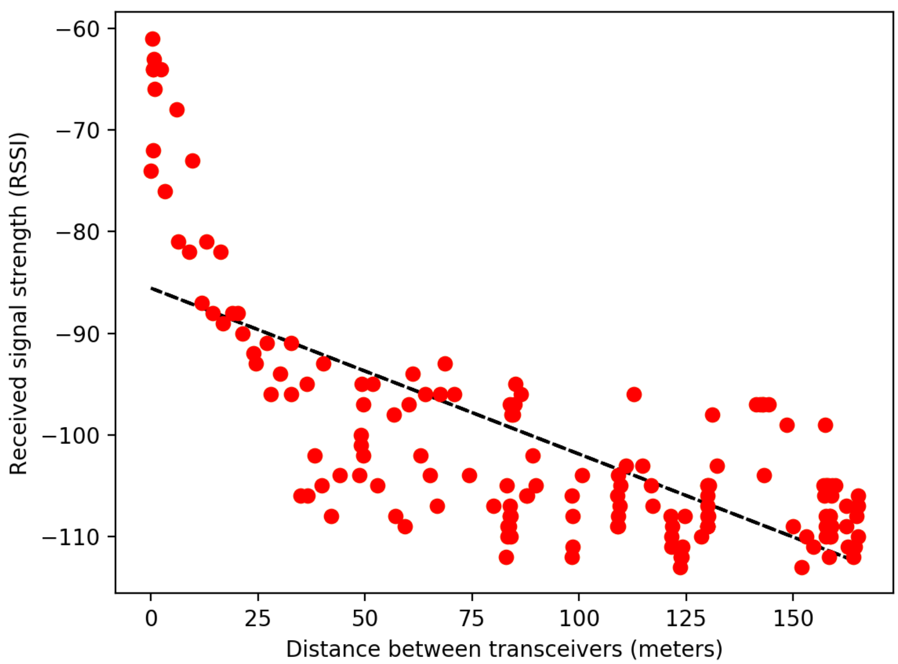


Figure 3.4.6f: Plot of data collected from fiberglass-obstructed range tests (using the thickness of the fiberglass of the avionics bay) of the RFM9X LoRa transceivers with the SMAKN 11 centimeter omnidirectional antennas.

As shown by Figures 3.4.6e and 3.4.6f, the RFM9X LoRa that we chose as a replacement for the HC-12s far surpasses the HC-12 transceivers' range both according to specifications and testing. The RFM9X LoRa has passed initial testing, transceiving while unobstructed at 38.1 meters with an RSSI of -95 and 38.1 meters obstructed with fiberglass (our rocket airframe material) with a RSSI of -86. With RSSIs of these magnitudes, we expect the LoRa to transmit longer distances than 38 meters, ultimately aiming to transmit over 726 meters (2500 feet). In both the unobstructed and obstructed range tests, the maximum range we were able to achieve with our current antennas was approximately 150 meters which is still significantly below the LoRa's capabilities, leading us to conclude that our antenna choice had to be reconsidered. To remedy the problem we decided that using two different antennas, one for the on-board system and one for the ground system, would allow us to achieve the longest range. The proposed solution is to have an omnidirectional antenna with a very low gain on the rocket itself, while a high gain directional antenna will be utilized by the ground system. Because we have no control over the final orientation of the on-board antenna, an omnidirectional antenna will eliminate the risk of having a weak radiation pattern in the direction of the ground system. The ground system antenna will be a high gain directional antenna as it can easily be pointed in the direction of the rocket, and weight is not an issue.

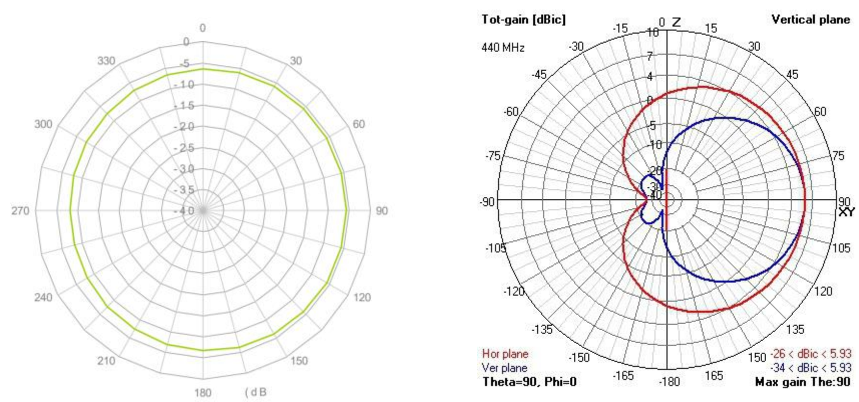


Figure 3.4.6g Radiation patterns of omnidirectional Taoglas Limited 433MHz Whip Tilt antenna (left) and directional 433MHz UHF Moxon Rectangle (right).

Once It was recognized that we would have to implement this new system of antennas, we looked into potential antennas to satisfy our requirements. A summary of the best options we found for the on-board and ground systems are below in tables 3.4.6b and 3.4.6c respectively. After extensive research it was determined that the best way to determine the final antennas would be to test each of the selected antennas. Antenna testing will be performed in the same manner as the range tests describe above. Instead of testing for the range of the system, we will be exchanging different antennas and performing the same range and RSSI tests to determine which antenna offers the best range and the strongest RSSI. Once testing is completed, we will utilize the combination of on-board and ground antennas that provides the longest range and best received signal strength as determined by those tests.

Manufacturer & Model	Taoglas Limited 433MHz Whip Tilt	Taoglas Limited 433MHz Whip STR	Linx Technologies 433MHz Whip STR
Cost	$13.35	$16.49	$9.08
Gain	-4.7 dBi	0 dBi	0.7 dBi
Frequency	432-434 MHz	433-434 MHz	420-445 MHz
Length (mm)	198.00	48.20	88.0

Table 3.4.6b: Specifications for omnidirectional antennas considered for on-board system.

Manufacturer & Model	VAS 433MHz UHF Moxon Rectangle	Zdacomm 433MHz Yagi antenna
Cost	$24.95	$89.00
Gain	5.75 dBi	8.0 dBi
Frequency	433 MHz	428-438 MHz
Length (mm)	304.80	600.00

Table 3.4.6c: Specifications for directional antennas for ground system.

3.4.7 Rover Release System

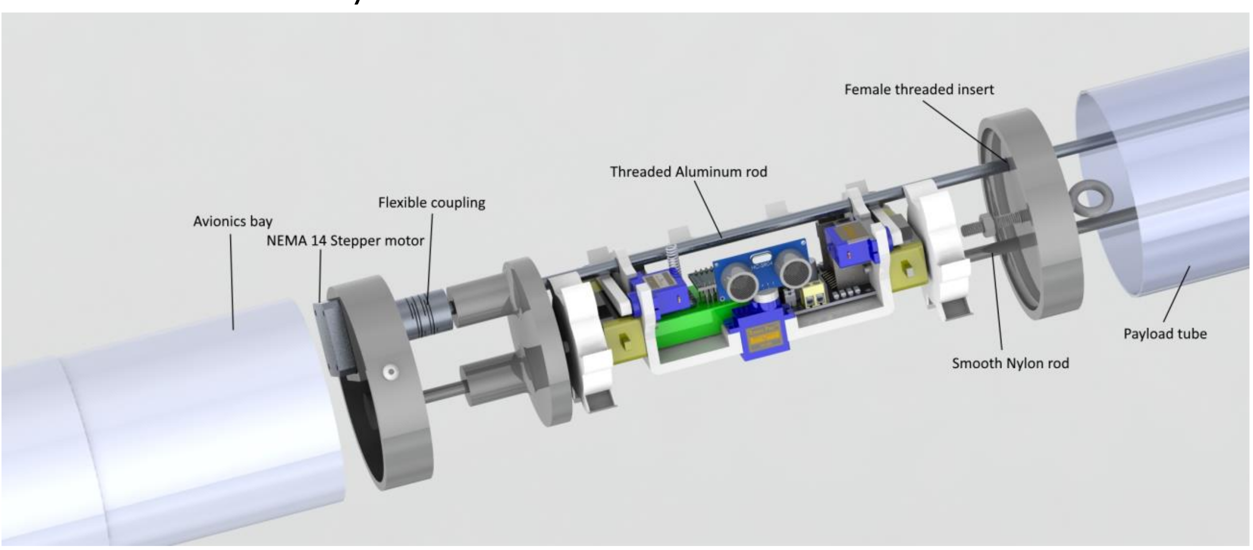


Figure 3.4.7a: CAD model of the rover release system.

The rover release system will consist primarily of a stepper motor spinning a threaded rod to expel the payload. Upon receiving the command from the ground station, the Teensy will control the A4988 Stepper Motor Driver Carrier to run a 7.4V stepper motor for a set of 320 rotations (64000 steps) to expel the rover from the rocket.

3.4.8 Ground System

To track the position of the rocket throughout to launch mission, Pitt Rocketry will be using a custom GPS and ground system. The ground system will obviously stay stationary on the ground. Throughout flight, landing, and after landing, the rocket will be transmitting position coordinates from the GPS to the ground system, which will then be interpreted and placed onto the ground station computer screen.

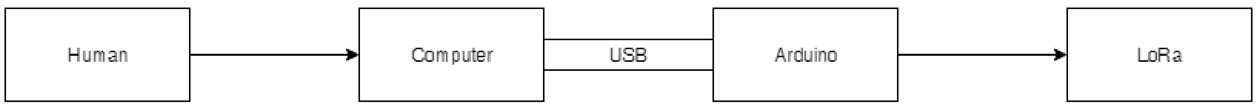


Figure 3.4.8a: Ground system data pipeline.

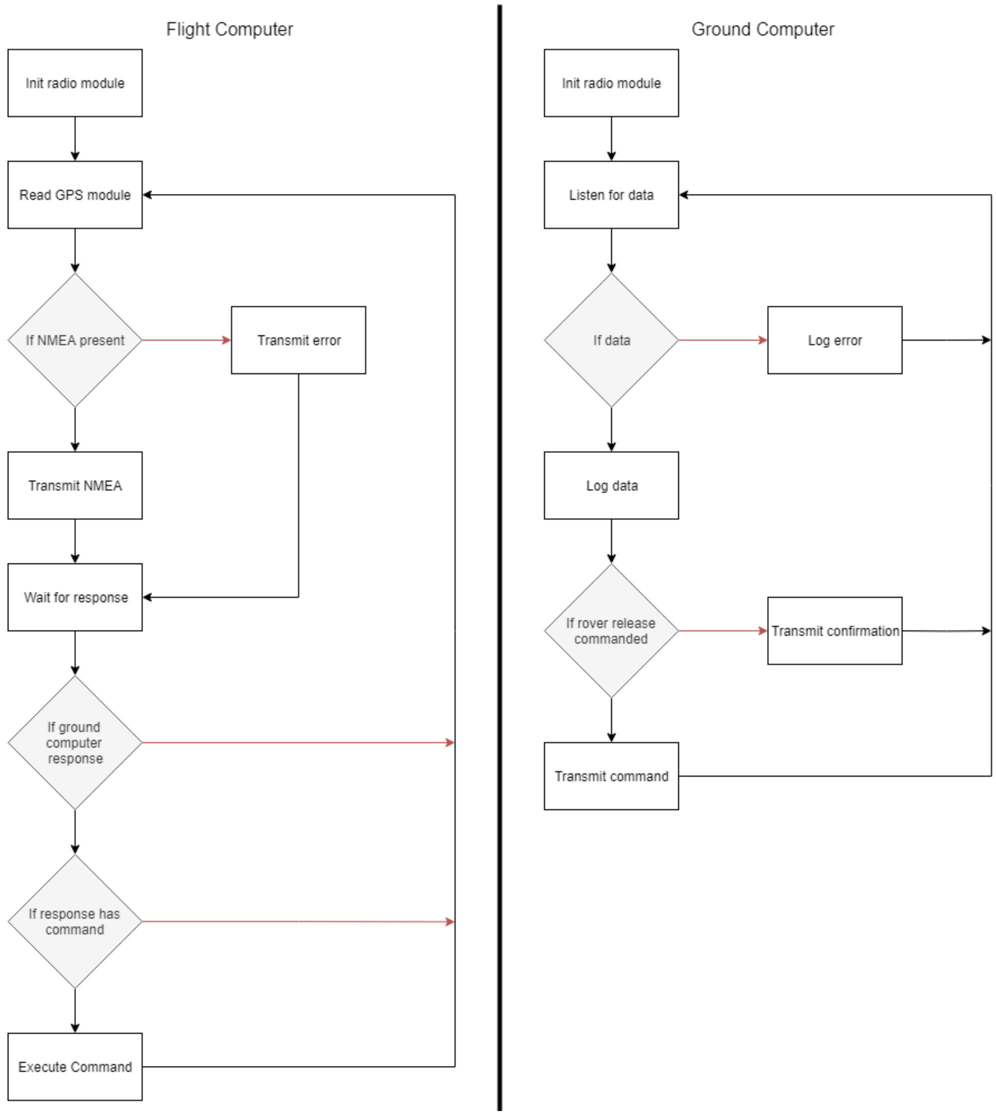


Figure 3.4.8b: Ground system and flight computer flow charts for operation.

The ground system portion of the avionics system communicates with both the vehicle and the payload. It displays the location of vehicle and provides an interface to deliver commands remotely to both the vehicle and the payload for rover deployment. The ground system consists of one LoRa transceiver, a Moxon antenna, an Arduino Uno, and a standard laptop.

Figure 3.4.8c: The Moxon antenna

The one transceiver will communicate with the transceiver in the avionics bay and the transceiver on the rover. We anticipate we will use a Moxon antenna for its significant range, however, this will first be confirmed by testing. The severe directionality of the antenna on the ground station does not matter, as there will be operators there to seek the best reception from the rocket. The ground transceiver receives the GPS coordinates of the vehicle, transmits the command to deploy the rover, and transmits the command to initiate rover movement. Both LoRa transceivers will be connected to the Arduino which will in turn be connected to a laptop via a serial connection. The computer will read serial data from the Arduino and display it using a LabView interface. LabView was chosen since it allows for easy construction of interfaces and because unlike some other platforms, the LabView file can be run as a Windows executable. The LabView executable will take the serial data from the Arduino, interpret said data from raw byte form, and display the data in a GUI. The GUI will consist of two separate windows. Both the rocket and payload windows will display a readout of the GPS coordinates, as well as time domain plots of latitude and longitude and a parametric plot of longitude vs latitude. These plots will help us get an idea for the state of motion of our rocket once it has landed. The payload window will also contain a button to deploy and activate the payload.

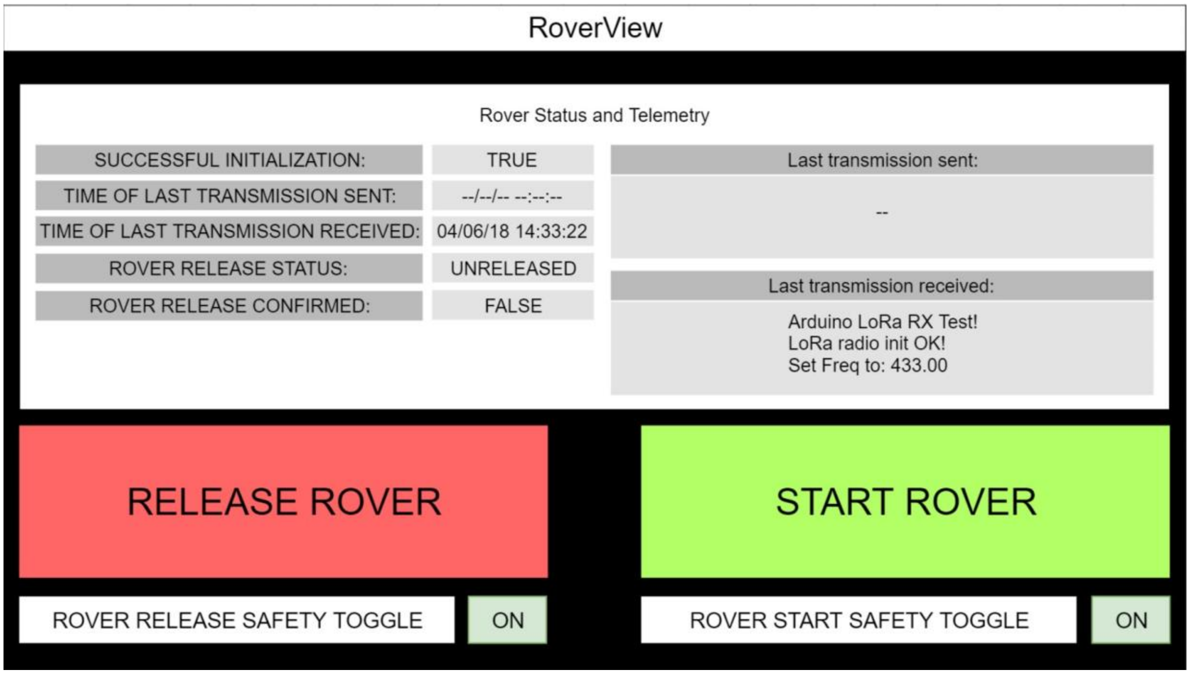


Figure 3.4.8d: This is a mockup of the RoverView GUI window which will be created using LabView. It displays important rover status and telemetry information and provides the ground control team with buttons to trigger rover release (after disengaging a safety toggle to prevent accidental rover release) and to confirm successful rover release upon landing. The "Trigger Rover Release" button sends a signal when pressed to the vehicle to begin releasing the rover. The "Confirm Successful Rover Release" button sends a signal to the rover when pressed to indicate to the rover that it has been successfully released and cleared to begin operation.

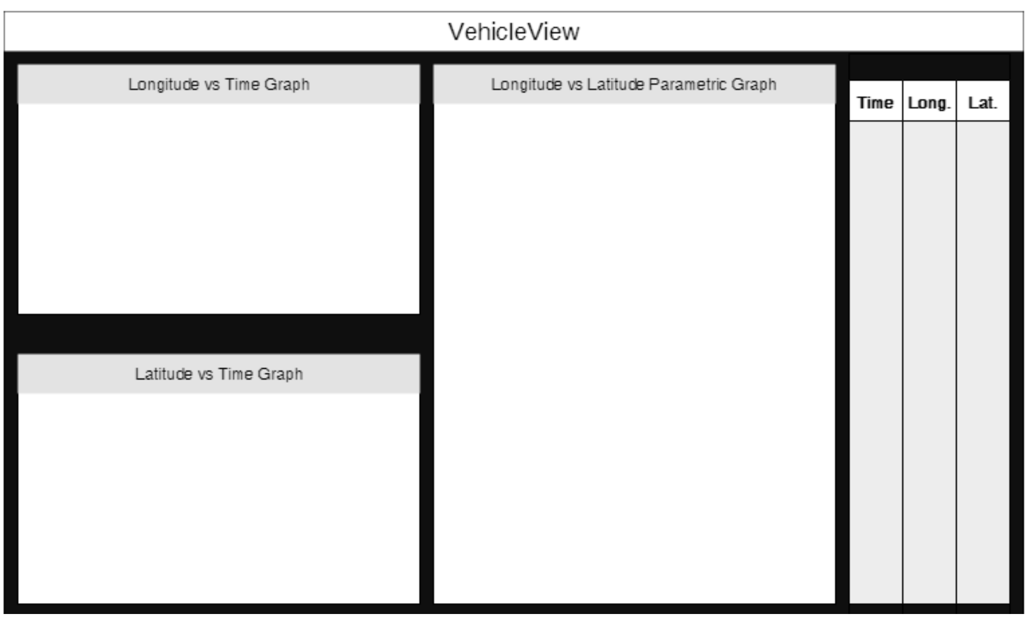


Figure 3.4.8e: This is a mockup of the VehicleView GUI window which will be created using LabView.

3.4.9 Water protection

Figure 3.4.9a: Silicone coating.

One of the requirements that our team captain requested for the avionics system was to include some waterproofing, especially for the expensive electronics such as the transceivers and altimeters. It was decided that silicone conformal coating would act as an appropriate membrane to protect the PCB and some exposed parts of the altimeters that can be safely insulated. This coating will also be used to mitigate the safety concern of the threaded rods spanning the avionics bay being conductive, as a thin layer of silicone coating will insulate the rods without changing their size in a significant manner.

3.5 Recovery Subsystem

3.5.1 Dual-Deploy Recovery System

In order to increase the success rate of our launch vehicle, our team decided to use a dual-deploy recovery system. The dual-deploy recovery system will require the launch vehicle to release a drogue parachute at apogee to prevent excessive acceleration upon main deployment. The main parachute will be deployed at a lower altitude to ensure the minimization of kinetic energy upon landing. Two independent sections allowing for separation via two ejection charges will hold the drogue parachute and the main parachutes. It is critical that the nose cone air frame and the Avionics bay booster section separate when the two ejection charges go off. The first ejection charge will separate the nose cone and the payload section releasing the drogue parachute at apogee which will be at an approximate height of 4,750 ft. The second ejection charge will separate the motor mount from the avionics bay releasing the main parachute at our estimated height of 550 ft. Since the PDR, we have finalized the mass of our rocket as well as finalized the parachute selections, the amount of black powder charge needed, and the descent path calculations.

The nose cone and air frame are tethered together by 1500# Kevlar shock cords to ensure strength in attachment. In order to maximize securement, we will use a shock cord that is 3 times the length of the entire rocket, approximately 7.22 meters. Both parachutes have 12x12 Nomex parachute protectors to allow for protection against the hot gasses created by the ejection charge. The 72" main parachute will be held in a deployment bag to ensure proper parachute release and minimize the chance of entanglement. The main parachute has a barrel swivel attached to the end of the suspension lines at the bridle. Not only will the swivel minimize entanglement, but it will be secured through a connection between a quick link and the eyebolt on the bulkhead. The 18" drogue chute will have a similar connection system involving the three components listed above as well. As a first year team, we will be purchasing Kevlar shock cords, parachutes, and recovery system components (swivels, quick links, etc.) from outside vendors due to the difficulty of fabricating the essential components. Since the PDR, we have determined the quantity of recovery system components needed to ensure proper deployment.

Black Powder Ejection Charge

In order to precisely deploy the main and drogue parachutes at their given altitudes, we will use ejection equipment consisting of black powder, shear pins, and wiring (including electronic matches). The black powder will be enclosed in ejection canisters attached at the ends of the Avionics bay and payload section. In order to determine the mass of the black powder needed to break the shear pins, we must look at four main components: pressure, volume, gas constant, and combustion temperature. With an airframe diameter of 3.9 inches, we determined the approximate shear force required to break the shear pins. With 4 shear pins, each pin shears at 44 lbf needing a total minimum force of 176 lbf. The additional 3.2 lbf acts as a buffer to ensure the shear pins will in fact break off. Listed below are the Black Powder Ejection calculations using the Ideal Gas Law. The calculations are necessary to find the correct amount of black powder needed to deploy the parachutes.

$Diameter\ of\ Rocket\ (D) = 3.9\ in$	$Force = 179.2\ lbf$
$Length_{Main\ Parachute\ Compartment}\ (l) = 18.75\ in$	$Black\ Powder\ Gas\ Constant\ (R) = 266\ \frac{in * lbf}{lbm}$
$Length_{Drogue\ Parachute\ Compartment}(l) = 5.51\ in$	$Combustion\ Temperature\ (T) = 3307^{\circ}\ R$

$$\underline{Area = \frac{\pi D^2}{4} = \frac{\pi * 3.9^2}{4} = 11.9\ in^2}$$

$$Pressure\ (p) = \frac{F}{A} = \frac{179.2\ lbf}{11.9\ in^2} = 15\ psi$$

$$Volume\ (V) = Area\ (A) * Length\ (L) = \frac{\pi D^2}{4} * l$$

$$pV = MRT$$

$$M = \frac{pV}{RT}$$

	Length (in)	Volume (in^3)	Mass (lbs)	Mass (g) per altimeter
Main Charge	$18.75\ in$	$223.125\ in^3$	$0.003805\ lbs$	$1.73\ g$
Drogue Charge	$5.51\ in$	$65.569\ in^3$	$0.001118\ lbs$	$0.507\ g$

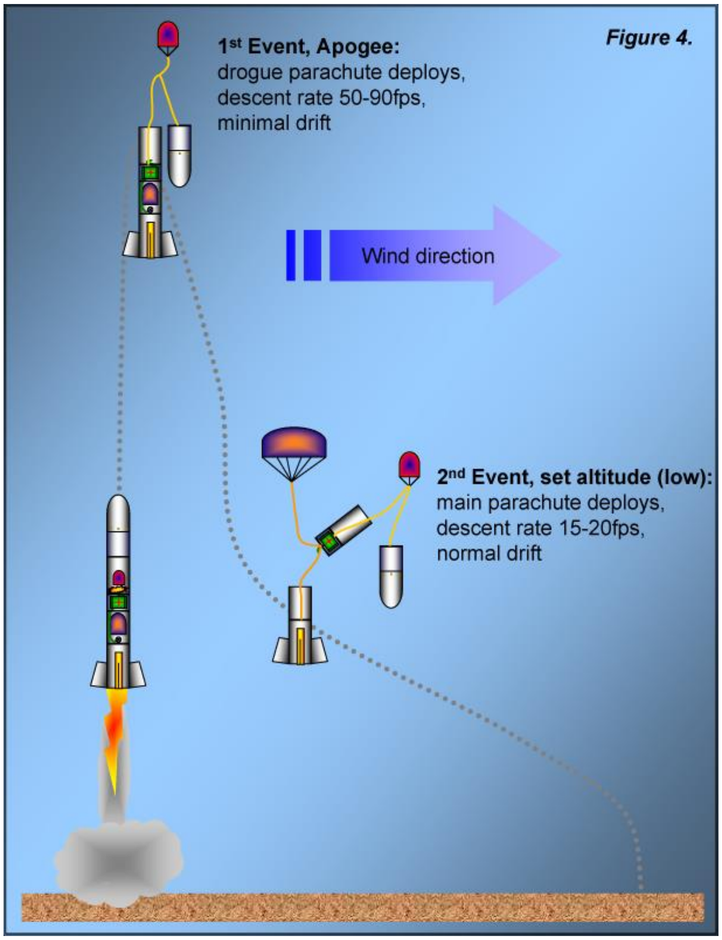


Figure 3.5.1: Plan for descent (courtesy West Rocketry).

3.5.2 Recovery Avionics System

To successfully deploy the parachutes and separate rocket sections, floating black powder ejection charges will be ignited by a pair of redundant altimeters setting off electric matches.

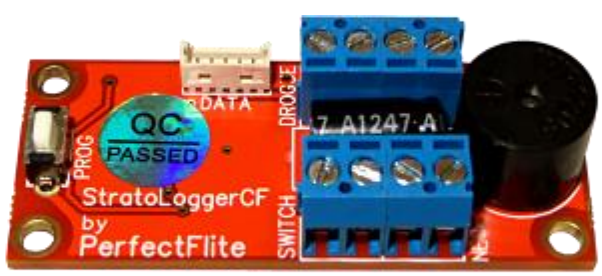


Figure 3.5.2a: A StratologgerCF.

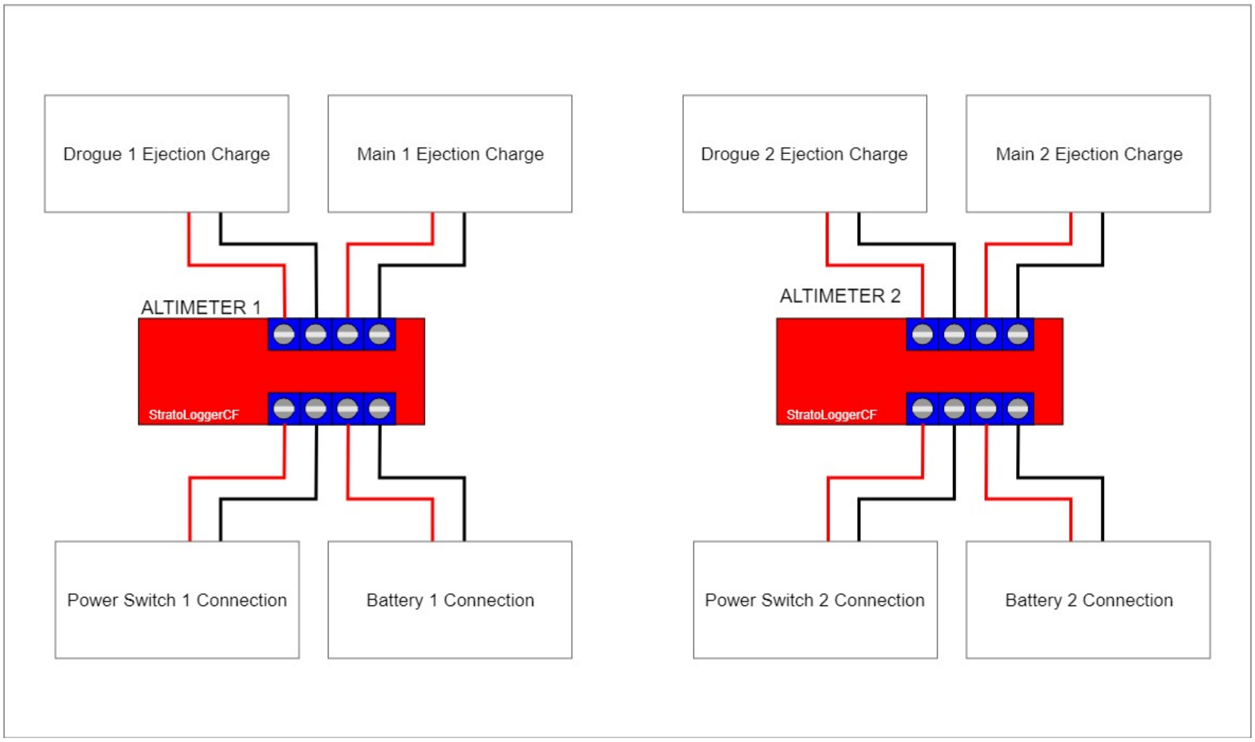


Figure 3.5.2b Altimeter wiring diagram.

The StratologgerCF combined barometric altimeter and flight computer was chosen as the altimeter for the avionics recovery system due to its relatively low price for the desired functionality and our NAR certified mentor having it on his list of recommended altimeters. It is designed for dual deploy recovery and can log data at a rate of 20 samples per second throughout the flight for at least 9 minutes. This data is stored and can be downloaded later on a computer.

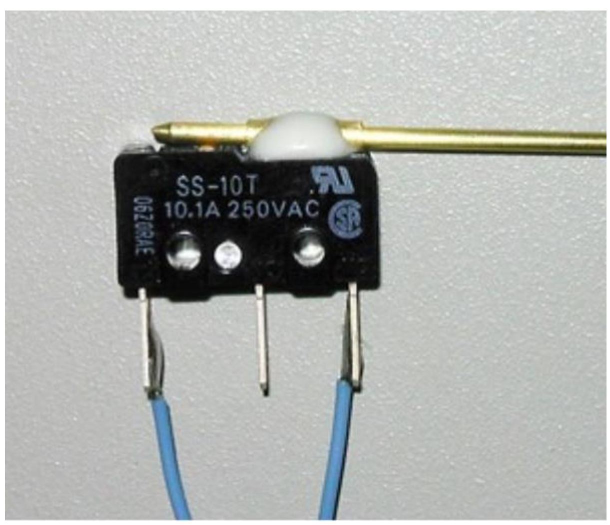


Figure 3.5.3c: A snap action switch modified to have a removable pin depress the button.

The primary purpose of the altimeters is to deploy the vehicle's drogue and parachute. The altimeters will deploy the drogue parachute no more than two seconds after apogee and the main parachute at 550ft. They will be powered by 9V alkaline batteries as suggested by the manual and connected to snap action safety switches that will be toggled by removing or inserting a pin. When the altimeters detect that the altitude events take place, the batteries discharge through the appropriate ejection charge output for one second. When tested with a 2ohm altimeter and the LiPo batteries being used, a constant output of 2.8A is produced for one second. This will be more than enough to ignite the electric matches that fire the black powder ejection charges.

Two StratologgerCF altimeters will be used to ensure the redundancy of the electronic components in the recovery system. They will both perform the same function of firing the ejection charges upon receiving triggers from the barometer sensing the apogee and 550ft altitudes. For additional safety, the altimeters will be shielded from radio frequencies by a modest coating of nickel conductive spray on the avionics sled. Tests will be performed to find the minimal amount that can be used to successfully protect the altimeters while minimizing interference with the transceivers' function.

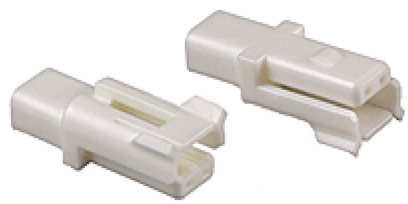


Figure 3.5.2c: Friction locking disconnects.

While the altimeters are contained in the avionics bay, the ejection charges are beyond the bulkheads. To securely route the wires through them, we will be using friction locking hermaphroditic disconnects. These will help speed up the assembly of the avionics bay before launches and do a better job of blocking the avionics bay from the black powder discharge than continuous wires would.

3.5.3 Parachute Choice

Parachute Location:

The drogue parachute will be placed between the nose cone and the payload section. Between the avionics bay and motor mount will be the location of the main parachute. We have chosen this set up as we expect the release of the drogue parachute to cause less stress on the payload bulkhead than the release of the main parachute. This will ensure the safe deployment of both parachutes.

Main Parachute Selection:

Since the PDR, we have finalized the main parachute selection. We have chosen the Rocketman elliptical 72" diameter parachute. After thorough research into main parachute options, the Rocketman elliptical parachute was deemed most suitable for our purposes. The Rocketman parachute is a multi-gore chute with an elliptical shape when inflated, which gives it a coefficient of drag of 1.6. The parachute has attached suspension lines, a heavy bridle, and barrel swivel to ensure maximum strength and security. It is brightly colored so that it will be easily visible as it falls, making it easier to track.

Drogue Parachute Selection:

Following the PDR, we have decided to stick with our prior decision and use the Apogee Fruity Drogue Parachute (18” diameter model). Without a drogue parachute, there would be significant drift away from the point of launch. In order to slow down our launch vehicle, we must have a drogue parachute deployed at apogee so the main parachute can be deployed at the lowest elevation point possible. In the process of selecting the drogue parachute, the three factors we are concerned with are the material, weight, and dimensions. The Apogee Fruity Drogue Parachute (18”) is not only bright in color, but it has an elliptical shape which is considered optimal for high drag and minimum weight and material. The typical drag coefficient is 1.5-1.6 which fits the estimated drag coefficients tested in our simulations. The strong nylon cloth material used for the parachute and the suspension lines that attach to a nylon bridle and a barrel swivel ensure for maximum strength and minimization of twisting. The 18” diameter is the proper size to minimize drift between its deployment at apogee and the deployment of the main parachute. Additionally, it will maintain a factor of safety in the force applied to the main parachute shroud lines and shock cords.

Numerical Analysis of Drogue Parachute:

_______By analyzing the surface area and coefficient of drag on the respective drogue parachute, we are able to find the terminal velocity for the drogue chute:

$$p\Delta = F_{net} \cdot \Delta t$$
$$0 = F_{gravity} - F_{drag}$$
$$F_{gravity} = F_{drag}$$

So,

$$g \cdot M_{rocket} = \frac{C_d \cdot \rho \cdot V^2 \cdot A}{2}$$
$$V_{terminal} = \sqrt{\frac{2Mg}{\rho C_d A}}$$
$$V_{terminal} = \sqrt{\frac{(2)(8.04\ kg)(9.81\ m/s^2)}{(1.225\ kg/m^3)(1.6)(0.164 m^2)}}$$
$$V_{terminal} = 22.2\ m/s$$

Where C_d is the drag coefficient, M is the mass of the system after motor burnout, g is the acceleration due to gravity, ρ is the air density, V is terminal velocity, and A is the cross sectional area of the parachute.

Numerical Analysis of Main Parachute:

To find the terminal velocity achievable by solely the contributions of the main chute, we use similar calculations to those above:

$$p\Delta = F_{net} \cdot \Delta t$$
$$0 = F_{gravity} - F_{drag}$$
$$F_{gravity} = F_{drag}$$
So,
$$g \cdot M_{rocket} = \frac{C_d \cdot \rho \cdot V^2 \cdot A}{2}$$
$$V_{terminal} = \sqrt{\frac{2Mg}{\rho C_d A}}$$
$$V_{terminal} = \sqrt{\frac{(2)(8.04\ kg)(9.81\ m/s^2)}{(1.225\ kg/m^3)(1.6)(2.63m^2)}}$$
$$V_{terminal} = 5.53\ m/s$$

Total Landing Kinetic Energy

To calculate the total kinetic energy at landing, the system is defined as the rocket and parachute. Since the PDR, we have calculated the total mass of our rocket resulting burnout mass of 8.08 kg. We can now calculate the total kinetic energy at landing with the following equations:

Using the terminal velocity achievable with solely the drogue parachute:

$$KE = \frac{1}{2}MV^2$$
$$KE = \frac{1}{2}(8.04\ kg)(22.2\ m/s)^2$$
$$KE = 1981.2\ J = 1461.25\ ft - lbs$$

Using the terminal velocity achievable with the main parachute:

$$KE = \frac{1}{2}MV^2$$
$$KE = \frac{1}{2}(8.04\ kg)(5.53\ m/s)^2$$
$$KE = 122.94\ J = 90.68\ ft - lbs$$

Landing Kinetic Energy Limit

_______To ensure we meet the kinetic energy requirement set by NASA, we performed calculations on the each subsection of our rocket to test that we meet the kinetic energy requirement of a maximum value of 75 ft-lbs (101.68 J). The burnout mass of our rocket is 8.04 kg and has been split to illustrate the mass of each section. Below, each subsection mass will be used to test that we fulfill the 75 ft-lb requirement per each subsection.

Booster Mass	4466.7 g
Payload Mass	3167.6g
Nose Cone Mass	403.0 g

Energy calculations using the drogue parachute terminal velocity:

Booster Mass

$$KE = \frac{1}{2}MV^2$$

$$KE = \frac{1}{2}(\ 4.47\ kg)(22.2\ m/s)^2$$

$$KE = 1101.5\ J = 812.4\ ft - lbs$$

Payload Mass

$$KE = \frac{1}{2}MV^2$$

$$KE = \frac{1}{2}(3.17\ kg)(22.2\ m/s)^2$$

$$KE = \ 781.2\ J = \ 576.2\ ft - lbs$$

Nose Cone Mass

$$KE = \frac{1}{2}MV^2$$

$$KE = \frac{1}{2}(0.403\ kg)(22.2\ m/s)^2$$

$$KE = \ 99.31\ J = 73.1\ ft - lbs$$

Energy calculations using the main parachute terminal velocity:

Booster Mass

$$KE = \frac{1}{2}MV^2$$

$$KE = \frac{1}{2}(\ 4.47\ kg)(5.53\ m/s)^2$$

$$KE = 68.3\ J = 50.4\ ft - lbs$$

Payload Mass

$$KE = \frac{1}{2}MV^2$$

$$KE = \frac{1}{2}(3.17\,kg)(5.53\,m/s)^2$$
$$KE = \ 48.5\,J = \ 35.8\,ft - lbs$$

Nose Cone Mass

$$KE = \frac{1}{2}MV^2$$
$$KE = \frac{1}{2}(0.403\,kg)(5.53\,m/s)^2$$
$$KE = \ 6.16\,J = 4.54\,ft - lbs$$

The above calculations confirm that with a successful main parachute deployment, each section will land with a kinetic energy less than the upper limit of 75 ft-lbs.

3.7 Mission Performance Predictions

Given the nature of the rail system, ignition and the first few meters of ascent will be a period crucial to mission success. The launch vehicle performance during the ascent phase can be accurately predicted perfected through testing and running simulations. Given this ability to mitigate the potential of failure through the design and testing process makes this stage of the mission relatively risk free. With proper engineering, design, and testing, we can ensure with a high degree of certainty that ascent will not be the greatest potential point of failure. The aspect of our mission profile and launch vehicle that is paramount to a successful launch and recovery is the recovery system. Any failure of the recovery system renders all other mission criteria unobtainable. The largest probability of recovery system failure will occur during parachute deployment, as numerous events must occur simultaneously, without hesitation, and reliably.

3.7.1 Target Altitude

The Student Launch guidelines require a launch vehicle to reach a maximum altitude between 4000 and 5,500 feet to receive points for altitude. To minimize the chances our launch vehicle is outside this range, we have chosen a target altitude of 4,750 feet, the center of the provided range.

3.7.2 Flight Simulations

Flight Profile Simulations:

The rocket's design was recreated in the flight simulation program OpenRocket by using the dimensions, component shapes and properties, and mass distribution of the actual launch vehicle.

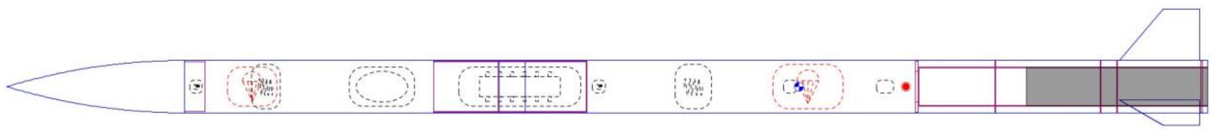


Figure 3.7.2a: Side view of the assembled vehicle model in OpenRocket.

The simulation was configured with the coordinates and elevation of the launch site in Toney, Alabama to improve accuracy. We found that the mean wind speed at the launch location in April is 6 mph, and this was incorporated into the simulation to determine the mass of the ballast needed to achieve our target altitude. The total mass of the rocket without ballast is 9.6 kilograms, and our simulations showed that adding a ballast of 960 grams would result in our target apogee of 5452.756 ft when the wind speed was set to 6 mph. We believe this is an underestimate, not being able to factor in the fin tab masses as they will require testing before we can confirm their mass.

Using a launch rail height of 12 ft, the expected rail exit velocity of the rocket is 17.4 m/s or 57 ft/s. The vehicle will experience a peak acceleration of 175.8 ft/s^2. The maximum velocity of the rocket is 625 ft/s and occurs 4.2 seconds into flight, at an altitude of 430 ft. The rocket is expected to reach apogee 19.8 seconds into flight. The total descent time from apogee to ground hit is expected to be 87 seconds.

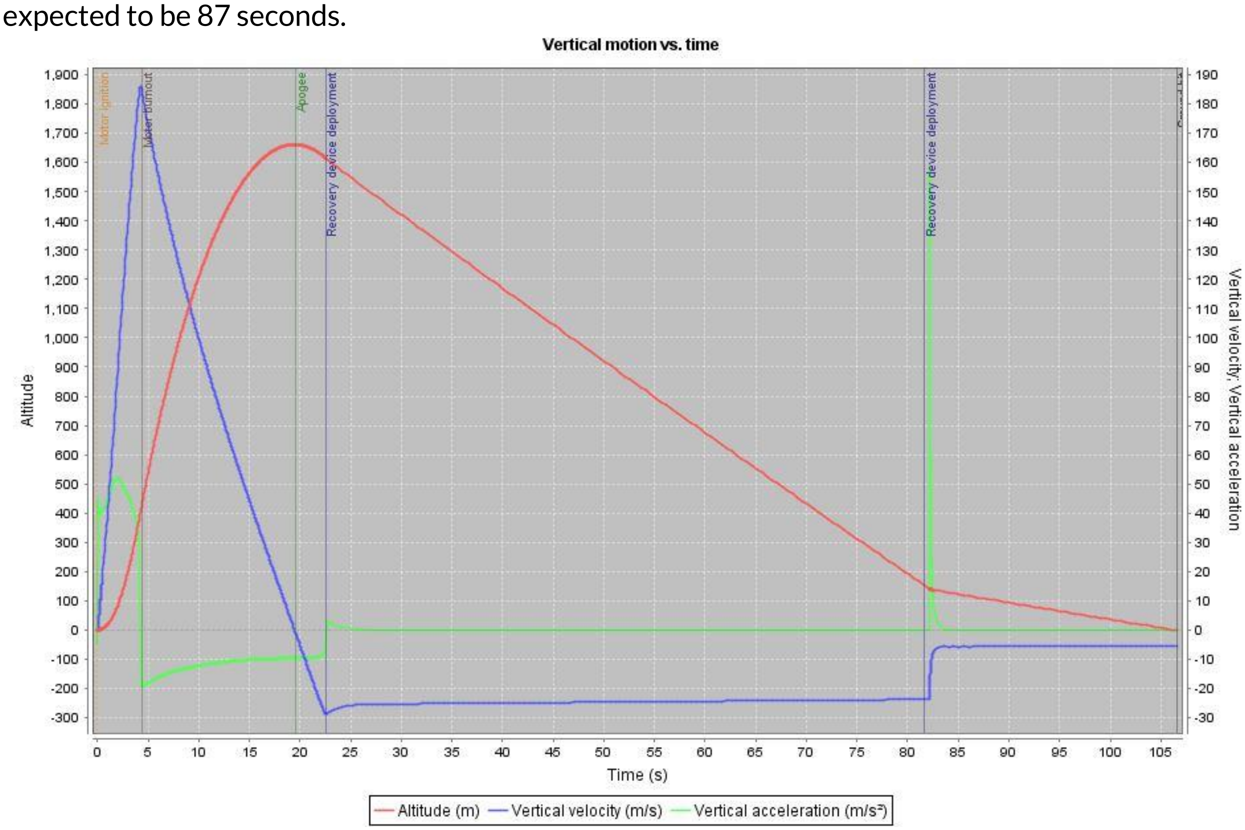


Figure 3.7.2b: Simulated plot of altitude, vertical velocity and acceleration at a wind speed of 6 mph

Using OpenRocket simulations with our maximum allowed ballast, our rocket has an apogee of 5547.9 ft, which is about 800 ft from our target apogee. However, OpenRocket is unable to model the effects of the angled fin system, which will lower our apogee to our target altitude once implemented on our rocket. We have also found through our subscale flight that the simulations in OpenRocket predict an apogee higher than the actual apogee.

Component	Mass (g)
Avionics Sled Assembly	415.2
Centering Rings	216.3
Airframe total	2940
Motor Fuel	1486
Motor	1273
Motor Retainer	139
Thrust Plate	210
Avionics Equipment	277
Fins	187
Rover	430
Rover Release Mechanism	566
Chute Protectors	60.1
Ejection Canisters	40
Quick links	39.7
Swivels	23.2
Drogue parachute	31.2
Main parachute	380
Eyebolts	47.4
Shock Cord Protectors	20

Shock Cord	47.4
Fin Mounts	250
Rail Button	6.7
Epoxy (estimated)	100
Total:	9623g 21.21lbs

Table 3.7.2a: Component masses.

Motor Thrust Curve:

The thrust curve of the K555 motor is shown in figure 3.5.2. The motor has a total impulse of 2406.2 Ns (540.94 lb-s). The launch mass of the motor is 2759 g and the burnout mass is 1273 g. The average thrust of this motor is 555.1 N, and the maximum thrust is 646.7 N, resulting in a Thrust-To-Weight ratio of 5.9.

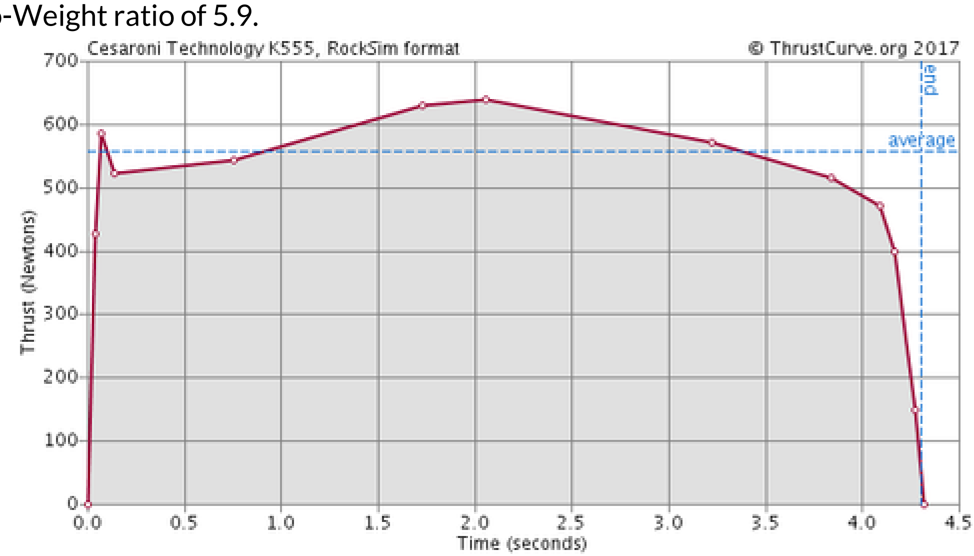


Figure 3.7.2c: Expected Thrust Profile of the Cesaroni Technologies K555 motor. Source: RockSim Motor Database.

3.7.3 Stability Data

Stability Margin:

As simulated, the Center of Gravity (CG) of the rocket lies at 152 cm from the top of the nose cone and the Center of Pressure (CP) lies at 174 cm. This results in a stability margin of 2.2 cal. This meets the NASA requirements for a stability margin.

Stability margin was also calculated using the equation shown below.

$$S = \frac{x_{cp} - x_{cg}}{d}$$

Where x_{cp} is the location of the Center of Pressure, x_{cg} is the location of the Center of Gravity, and d is the diameter of the rocket. Using our values listed above, we found the stability margin to be 2.2 cal, which corresponds with the simulation value found in OpenRocket.

3.7.4 Recovery Calculations and Simulations

<u>Vehicle Drift and Altitude Predictions:</u>

The following results were obtained for the landing site lateral distance at different wind speeds.

All simulations were run in OpenRocket multiple times.

Drogue parachute diameter: 18 in

Drogue parachute drag coefficient: 1.6

Main parachute diameter: 72 in

Main parachute drag coefficient: 1.6

Burnout mass of the rocket: 8.14 kg

Wind speed (mph)	Lateral Distance from launch site
0	0
5	435
10	880
15	1305
20	1740

Table 3.7.4: Landing Distance Results for Varying Wind Speeds.

From this table it is clear that the motor selected for the rocket will allow us to stay within the required drift radius of 2500 feet even in adverse weather conditions. We calculated the landing distance by multiplying the descent time by the wind speed for each wind speed. Ballast can be adjusted based on weather conditions to ensure we reach the correct height.

4 Safety

4.1 Forward, Safety Mission Statement

Safety is the primary responsibility for every member of the team; no single facet of the team's operation is of greater importance. It is each individual's responsibility speak up when uncertain, to work safely, and to ensure the safety of others.

4.2 Launch Concerns and Operation Procedures

The following section outlines the launch recovery preparation procedures, motor preparation procedures, launch-pad setup procedures, igniter installation, troubleshooting, and post-flight inspection.

4.2.1 Recovery Preparation Procedure

Two actions will be taken by the recovery team in the field on launch day. These actions include attaching the shock cord and properly folding the parachutes. The recovery lead will be responsible for overseeing all of recovery preparation checklists on the field. The checklists are as follows:

Preparation of the Shock Cord:

1. Place each shock cord with the proper length into the fuselage.
2. Securely tie and verify that the connections between the payload bay bulkhead and parachute are well secured.
3. Thread the Nomex parachute protector onto the shock cord and securely tie and verify that the cord between the main avionics bay forward bulkhead and forward eyebolt is secure to each other.
4. Securely tie and verify the attachment between the forward eyebolt and that the parachute is secure.
5. Thread the Nomex parachute protector onto the shock cord and securely tie and verify the cord between the avionics bay aft bulkhead to aft eyebolt is secure.
6. Securely tie and verify that the attachment between the aft eyebolt and the drogue parachute is secure.

Parachute Preparation:

Parachutes will be properly folded to ensure that they will be ready for deployment on launch day. Parachutes will be laid out flat and then folded from the corners into the center until the parachute is able to fit in the proper slots.

4.2.2 Motor Preparation Procedure

We will follow the Motor Assembly procedure as set by Cesaroni Manufacturing. The following will illustrate the steps needed to be taken to ensure the motor is ready for launch:

1. Inspecting the motor casing to ensure there is no damage
2. Gently twisting the delay/ejection module into the forward end of plastic liner
3. Remove nozzle from top of liner
4. If grains are not already assembled inside the motor, slide the grains inside the motor liner, making sure it is place beveled end first.
5. Insert the nozzle into the rear end of the plastic liner and remove nozzle cap

6. Insert liner assembly , including the delay/ejection module, grains, and the nozzle, into the rear end of the motor
7. Re install nozzle cap to protect nozzle and propellent from outside elements

Our mentor will be the person handling the motor since he is qualified to do so with his NAR and TRA certifications.

4.2.3 Set-Up on Launch Pad:

1. Carry the launch vehicle to the launch pad (safety lead).
2. Slide the launch vehicle onto the launch pad rail (safety lead).
3. Power on and arm the altimeters and verify correct initialization (avionics lead).

4.3.4 Igniter Installation Procedure

1. Igniter leads will come into contact with each other once the igniter is placed on the launch pad.
2. Igniter will be installed by the propulsion lead once team has been maintained to a safe distance.
3. The igniter will be placed into the motor
4. Tape will be applied to the igniter in order to prevent it from sliding out.
5. When the igniters are properly secure, the leads of the igniter will connect to the ignition terminals.
6. Team will check for electrical continuity and area will be cleared for launch.

4.3.5 Troubleshooting:

Altimeters are not responding to arming command when on launch pad:

1. Turn off altimeters (avionics lead).
2. Return Aviation Bay to workbench (avionics lead).
3. Check for continuity (avionics lead).
4. If continuity is not the problem, replace wiring and electronic matches (avionics lead).

Misfire:

1. Immediately turn off power to the launch area (safety lead).
2. Wait for five minutes after the last misfire before approaching the rocket or wait until the all clear is given (safety lead).
3. Remove the igniter (mechanical lead).
4. Check if igniter has discharged (mechanical lead).
5. Perform continuity test on ignitor (mechanical lead).
6. Replace ignitor with a new one if it is defective (mechanical lead).
7. If ignitor was not defective, examine cause of the misfire at the workbench (mechanical lead).

4.3.6 Post-Flight Inspection:

1. Use GPS information to locate the launch vehicle if needed (avionics lead).
2. Inspect launch vehicle for charges that have not gone off. If any explosives are still intact, immediately disarm them (safety lead).
3. Bring launch vehicle back to determine the official altimeter reading (safety lead).
4. Remove motor and ejection charges after cooling (mechanical lead).
5. Clean residue off of the motor (mechanical lead).

4.4 Safety Officer Identified-Responsibilities Defined

Table 4.4.1 Identifies the Safety Officer Identified-Responsibilities.

Requirement	Design Feature	Verification	Status	Req #
Each team will use a launch and safety checklist.	The team's safety officer is has created and will continuously improve a launch and safety checklist for launch day operations.	This requirement shall be verified through inspection.	Verified.	5.1
Each team must identify a student safety officer.	Thomas Harrington is the team's student safety officer.	This requirement shall be verified through inspection.	Verified.	5.2
Roles and responsibilities of safety officer shall be outlined.	Safety officer and/or appointed safety deputies will attend all team activities to monitor and emphasize safety	This requirement shall be verified through inspection.	Verified.	5.3.1
Safety officer shall monitor design of vehicle and payload.	Safety officer and/or appointed safety deputies will periodically review the overall design for safety.	This requirement shall be verified through inspection.	Unverified.	5.3.1.1

Safety officer shall monitor construction of vehicle and payload.	Safety officer and/or appointed safety deputies will be present during construction and fabrication processes.	This requirement shall be verified through inspection.	Unverified.	5.3.1.2
Safety officer shall monitor assembly of vehicle and payload.	Safety officer and/or appointed safety deputies will be present during assembly processes.	This requirement shall be verified through inspection.	Unverified.	5.3.1.3
Safety officer shall monitor ground testing of vehicle and payload.	Safety officer and/or appointed safety deputies will be present during ground testing processes.	This requirement shall be verified through inspection.	Unverified.	5.3.1.4
Safety officer shall monitor subscale launch test(s).	Safety officer and/or appointed safety deputies will be present during subscale launch.	This requirement shall be verified through inspection.	Unverified.	5.3.1.5
Safety officer shall monitor full scale launch test(s).	Safety officer and/or appointed safety deputies during full scale launch.	This requirement shall be verified through inspection.	Unverified.	5.3.1.6
Safety officer shall monitor team activities on launch day.	Safety officer and/or appointed safety deputies for launch day.	This requirement shall be verified through inspection.	Unverified.	5.3.1.7
Safety officer shall monitor recovery activities.	Safety officer and/or appointed safety deputies will be present during recovery	This requirement shall be verified through inspection.	Unverified.	5.3.1.8

	activities.			
Safety officer shall monitor educational engagement activities.	Safety officer and/or appointed safety deputies will be present during educational activities.	This requirement shall be verified through inspection.	Unverified.	5.3.1.9
Safety officer shall implement standard work procedures developed by the team for construction, assembly, launch, and recovery activities.	Safety officer and/or appointed safety deputies will strictly implement standard work procedures for construction, assembly, launch, and recovery activities.	This requirement shall be verified through inspection.	Unverified.	5.3.2

Table 4.4.1

4.3 Approach to Analysis of Failure Modes

After deliberation and further research of Failure Modes and Effects Analyses, the decision to redact the previous risk assessment code (RAC) matrix was made. This decision was made primarily for the creation of a "Risk Priority Number"(RPN) for the team's benefit to help them understand and plan for the most debilitating risks. The new grading system for risks includes severity, occurrence probability, and detection. The severity scale is outlined in The RPN is calculated as (Severity*Occurrence*Detection). The highest RPN is our most debilitating risk. Table 4.3.1 outlines the severity of the risk, Table 4.3.2 outlines the occurrence probability of the risk, and Table 4.3.3 outlines the detection probability of the risk.

Severity Scale

Effect	Criteria: Severity of Effect				Ranking
	Casualty	Equipment Damage	Project Plan	Environmental	
Catastrophic	Permanent disability or death	Irreparable damage to or loss of system, machinery, or equipment	Heavy delays or budget overruns resulting in failure to complete project.	Long-term or irreversible damage (>5 years)	5
Critical	Severe injury or illness temporarily preventing normal activites.	Signifigant damage to system, machinery, or equipment resulting in prolonged non-operation	Signifigant delays or budget overruns that severely limit project performance.	Medium-term (1-5 years)	4
Marginal	Injury or illness without effect on daily activities.	Damage to system, machinery, or equipment resulting in temporary non-operation	Delays or budget overruns that impact non-critical project performance factors.	Short-term (<1 year)	3
Neglegible	Insignifigant injury treated through basic first aid.	Minor damage to system, machinery, or equipment fixed by immdiate repairs	Minor Delays in non-essential operations.	Minor damage, readily repaired	2
None	No Effect				1

Table 4.3.1: Severity scale used in the calculation of the risk priority number of each possible risk in our failure mode analysis.

Occurrence Scale

Probability of Failure	Qualitative Definition	Quantitative Definition	Ranking
Very High	Likely to occur repeatedly during life of project.	Probability $>10^{-1}$	5
High	Likely to occur several times during life of project.	10^{-1} > Probability $>10^{-2}$	4
Moderate	Likely to occur sometime within life of project.	10^{-2} > Probability $>10^{-3}$	3
Low	Not likely to occur within life of project.	10^{-3} > Probability $>10^{-6}$	2
Very Low	Occurrence is not expected during life of project.	10^{-6} > Probability	1

Table 4.3.2: Occurrence likelihood scale used in the calculation of the risk priority number of each possible risk in our failure mode analysis.

Detection Scale

Detection	Criteria: Likelihood the existence of a defect will be detected by process controls	Ranking
Almost Impossible	No known controls available to detect failure mode	5
Low	Low likelihood current controls will detect failure mode	4
Moderate	Moderate likelihood current controls will detect failure mode	3
High	High likelihood current controls will detect failure mode	2
Almost Certain	Current controls almost certain to detect the failure mode. Reliable detection controls are known with similar processes.	1

Table 4.3.3: Detection scale used in the calculation of the risk priority number of each possible risk in our failure mode analysis.

4.4 Analysis of Failure Modes

Table 4.4.1 shows the preliminary failure modes and effects analysis performed by the Pitt Rocketry Team.

Process Step/Input	Potential Failure Mode	Potential Failure Effects	SEVERITY (1 - 5)	Potential Causes	OCCURRENCE (1 - 5)	Current Controls	Verification Plan	DETECTION (1 - 5)	RPN (1-125)
Rocket	*Altimeter failure*	*Recovery system is not deployed resulting in an uncontrolled free fall possibly leading to injury or death.*	*5*	*Loss of power to altimeter; hardware malfunction.*	*2*	*Perform rigorous testing of recovery system altimeter; purchase reliable altimeter.*	*Test the recovery system and altimeter during the subscale launch.*	*5*	*75*
	GPS failure	*Difficult or impossible to locate the vehicle after landing.*	*2*	*Loss of power to GPS; Satellite Fix failure; Transceiver signal loss;*	*2*	*Perform rigorous testing of GPS system; purchase reliable GPS.*	*Test the GPS system prior to and during the subscale launch.*	*2*	*12*

	Transceiver failure	*Failure to receive GPS coordinates and therefore failure to locate the vehicle after landing.*	*3*	*Transceiver exceeds range and is unable to operate upon returning to range;*	*2*	*Test transceivers at multiple ranges within and exceeding maximum expected range.*	*Test transceivers independent of system and during subscale launch.*	*2*	*18*
	Fin Detachmen t	*Fin falls from vehicle mid-flight causing potential injury or death; loss of fin causes vehicle to lose stability.*	*5*	*Structural weakness at fin attachment point.*	*2*	*Take special precaution when attaching fins to airframe to ensure quality of attachment.*	*Perform structural tests on fins to ensure proper attachment*	*2*	*30*
	Premature parachute deployment	*Parachutes deploy prior to apogee causing the vehicle to lose stability, rendering its flight path off-nominal and unpredictable.*	*4*	*Altimeters not properly calibrated; faulty deployment logic; improper packing.*	*2*	*Verify and test altimeter calibration procedure; verify and test deployment logic.*	*Test the recovery system during subscale launch.*	*5*	*40*

	Rocket motor explodes	*Vehicle is destroyed and explosion could cause injury or death.*	*5*	*Manufacturing anomaly; damage upon and/or during delivery and/or transportation.*	*2*	*Consult NAR advisor Duane Wilkey on the reliability of the rocket motor.*	*Test the viability for use of the motor.*	*1*	*10*
	Kevlar shock cord snaps	*Vehicle separates in flight; certain components have no parachute or safety mechanism, turning them into potentially deadly projectiles*	*5*	*Excessive force on shock cord during parachute deployment.*	*2*	*Appropriate calculations simulation, and testing can demonstrate that our choice of shock cord significantly reduces the likelihood of this event.*	*Develop tests and simulations to ensure that the risk of this occurring is at least extremely unlikely, but preferably entirely mitigated.*	*3*	*30*
	Battery overheats	*Battery explodes, catches on fire, damages internals.*	*5*	*Any damage to battery including dents, punctures, and heat/cold.*	*2*	*Rigorous inspection of batteries prior to installation; removal of batteries until final assembly.*	*A pre-flight checklist will include the battery installation step. Battery to be tested before installation.*	*4*	*50*

Payload	*Electrical failure*	*Failure or loss of power to one or more onboard electrical component(s).*	*2*	*Insufficient battery voltage; voltage ripple from 5v switching voltage-regulator*	*4*	*Fully charge all batteries prior to launch; Choose batteries with capacities suited for devices' power draw and expected run time; Choose a switching voltage-regulator designed for use with sensitive electronics.*	*Measure power draw of all components during typical use. Endurance test switching regulator with corresponding electronic devices.*	*5*	*40*
Payload Integration	*Premature rover release.*	*Recovery does not work; rover is not able to perform as expected.*	*3*	*Failing avionics; launch failure; separation failure.*	*1*	*Ensure that rover release mechanism adheres to rules outlined in Student Launch Handbook.*	*Test the rover release mechanism and the altimeters before flight.*	*5*	*15*

	Coupler tube does not separate at the correct time.	*Parachutes to do not deploy, rocket hits ground at terminal velocity.*	*4*	*Improper fastening techniques (cross threading, under-torqued, etc); improper shear-pin-selection.*	*3*	*The rocket will be assembled and the shear pins will be tested and sheared by hand to validate their strength*	*Assembly checklists will include proper fastening techniques for coupler tube*	*5*	*60*
	Coupler tube separates during flight (not at the correct time).	*Flight does not reach target altitude; endangers spectators if the rocket is still in powered flight.*	*5*	*Improper fastening techniques (cross threading, under-torqued, etc); improper shear pin selection.*	*1*	*The rocket will be assembled and the shear pins will be tested and sheared by hand to validate their strength*	*Assembly checklists will include proper fastening techniques for coupler tube*	*5*	*25*
Launch Support Equipment	*Rail button breaks during launch.*	*Trajectory alteration causing danger to anything or anyone on the ground level.*	*5*	*Improper material selection for rail buttons; insufficient fastening method for rail buttons; ambitious fin angle.*	*2*	*Rail buttons will be tested for security and integrity before launch*	*Visual and mechanical inspection will verify the security and integrity of the rail buttons*	*5*	*50*

	Launch rail breaks.	*Trajectory alteration causing danger to anything on the ground level.*	*5*	*Previous damage to launch rail; launch rail improperly assembled.*	*1*	*Consult NAR advisor Duane Wilkey on the reliability of the launch rail.*	*Consult NAR advisor Duane Wilkey.*	*5*	*25*
Launch Operations	*Igniter fails to ignite.*	*Launch failure*	*1*	*Manufacturing anomaly; damage.*	*2*	*Careful inspection and handling of igniter.*	*Consult NAR advisor Duane Wilkey.*	*1*	*2*
	Propellant fails to ignite.	*Launch failure*	*1*	*Manufacturing anomaly; damage.*	*2*	*Careful inspection and handling of motor.*	*Consult NAR advisor Duane Wilkey.*	*1*	*2*
	Non-uniform propellant burn.	*Launch failure; target altitude missed; ground danger.*	*5*	*Manufacturing anomaly; damage.*	*1*	*Careful inspection and handling of motor.*	*Consult NAR advisor Duane Wilkey.*	*5*	*5*

Table 4.4.1: Failure modes and effects analysis.

4.5 Personnel Hazard Analysis

Table 4.5.1 shows the preliminary personnel hazard and risk analysis performed by the Pitt Rocketry Team. For this section of the report, a similar scale to the FMEA was employed, as well as a similar format. Again, the "Risk Priority Number"(RPN) allows the team to devote the most attention to the most dangerous and present risks.

Potential Failure Mode	Potential Failure Effects	SEVERIT	Potential Causes	OCCURRE	Current Controls	Verification Plan	DETECTI	RPN (1-125)

		Y (1 - 5)		N C E (1 - 5)			O N (1 - 5)	
Injury from machinery	Bodily harm to team member(s); Damage to machine and/or rocket	*5*	Mishandling of machines, fatigue, failure to comply with safety guidelines, or machine malfunction.	*3*	Safety guidelines and instructions will be carefully observed with all machines used. Certification tests for SCPI and the SSoE Makerspace must be passed before using the facilities.	Access to machinery will be limited to those with certification; Powered machinery will be operated only while another certified user is present.	*2*	*30*
Chemical Injury	Bodily harm to team member(s); Damage to rocket or rocket parts.	*5*	Mishandling of chemicals and/or failure to comply with MSDS guidelines and warnings	*3*	Proper safety equipment will be worn by all team members working with any chemical. MSDS guidelines will be carefully observed with all chemicals used.	Chemicals will only be available to members who have read and understand the MSDS guidelines; Chemicals will only be used when there is at least one other member present.	*2*	*30*

Injury from erratic rocket flight	Injury to team members and/or bystanders resulting from rocket impact.	*5*	Poor rocket stability; Rail system malfunction.	*3*	Accurately calculate the center of pressure and center of mass; Perform simulations before fight.	Calculations and simulations will be performed multiple times and reviewed by team advisor Matthew Barry; A spotter will be assigned to watch each launch and alert others if another rocket is dangerous	*2*	*30*
Premature rocket ignition	Injury, including severe burns, to team members and/or bystanders.	*5*	Ignition malfunction; Failure to follow safety procedures.	*1*	Conduct briefings at pre launch meetings; Ensure reliability of ignition safety switch.	Launch day operations will be supervised by safety officer Thomas Harrington and NAR mentor.	*5*	*25*

Premature black powder ignition.	Injury to team members form explosion and resulting shrapnel.	*5*	Recovery system malfunction; Faulty testing procedures; Failure to follow safety procedures.	*2*	Perform black powder tests within a testing enclosure; Follow launch day safety procedures.	A safety officer or mentor will be present for all black powder tests; Launch day operations will be supervised by safety officer Thomas Harrington and NAR mentor.	*2*	*20*
Free falling rocket sections	Damage to rocket and/or injury to team members on the ground from free-falling projectiles.	*5*	Recovery system fails to deploy.	*4*	There will be ground and subscale testing of the entire recovery system and its components. Test launches will be carried out on days with optimal weather conditions and minimal clouds below projected apogee. All persons present during launch will be notified to	The recovery system will be declared as functional before test launch. All members at the launch will be reminded to stay attentive during the entirety of the flight, from launch to landing.	*4*	*80*

					remain attentive.			
Lithium battery fire or explosion	Heat and/or chemical burns to team members, damage to rocket and other equipment.	*5*	Overcharge, over-discharge, overheating, puncture, or physical impact to lithium cells	*2*	Care will be taken when charging, discharging, handling, and storing lithium batteries. Batteries will not be overcharged or over-discharged.	Batteries will be stored away from flammable materials. Voltage monitoring will occur before and after charging, and every flight to ensure nominal function. Personnel working with lithium batteries will check for shorts prior to powering circuit. Batteries will be kept away from sources of heat such as soldering irons.	*4*	*40*

						Batteries will be routinely checked for physical damage and swelling. Batteries will only be charged under supervision and with a proper Lipo charger. All lithium cells in the rocket will be wired in parallel with a Lipo low-voltage alarm. Batteries in the rocket will be secured-down and located away from potential points of impact.		

$PM_{2.5}$ emitted during production of parts	$PM_{2.5}$ can penetrate deeply into the lungs and affect respiratory system	3	Fumes of material created during laser cutting might contain $PM_{2.5}$	2	Turn on the ventilator and make sure people does not work alone	Use materials that creates less $PM_{2.5}$ and make sure only trained members have access to manufacturing equipment.	2	12

Table 4.5.1: Personnel hazard analysis.

4.6 Environmental Concerns

Table 4.6.1 shows the environmental concerns analysis performed by the Pitt Rocketry Team. T For this section of the report, again a similar scale to the FMEA was employed as well as a similar format. Again, the "Risk Priority Number"(RPN) allows the team to devote the most attention to the most dangerous and present risks.

Potential Failure Mode	Potential Failure Effects	SEVERITY (1 - 5)	Potential Causes	OCCURRENCE (1 - 5)	Current Controls	Verification Plan	DETECTION (1 - 5)	RPN (1-125)
Water pollution with perchlorate	Perchlorate inhibits NIS-Sodium Iodide symporters in thyroid, which NIS is essential for Iodine transport, which is needed for synthesizing T3(thyroxine) and T4	*4*	Perchlorate from the ammonium perchlorate composite propellants release to air	*2*	Dispose of all spent motors properly;	Safety officer and team mentor will confirm the proper handling of all spent motors.	*5*	*40*

	(triiodothyronine) hormones							
Parts of rover break off from system	Pieces of rover are littered into the environment.	*2*	Structural weakness in rover due to poor design choices or fabrication errors.	*2*	Verify quality of rover assembly and minimize number of separate, small (potentially detachable) components on rover.	Inspect rover before and after each flight to ensure no components were lost.	*2*	*8*
Parts from rocket become detached and become projectiles	Debris might be left in the environment.	*5*	Parts experience impact force from moment of vehicle acceleration until vehicle is stable after landing.	*3*	Pay close attention to potential detachment points during fabrication.	Thoroughly test potential detachment points	*4*	*75*

Battery rupture that spreads hazardous chemicals	Hazardous chemicals are spread into the environment.	*5*	Battery is punctured by a hard crash or by general mishandling.	*3*	Protect the batteries and locate them away from potential points of impact on launch vehicle. Properly recycle any non-functional or damaged batteries	Test battery enclosure to ensure it is sufficient to protect batteries from a hard crash. Team mentor and Safety Officer will oversee proper disposal of damaged batteries.	*2*	*30*
High temperature exhaust damages grounds around launching area	Ground of the launch site could be damaged or even catch fire.	*3*	The motor expels high temperature exhaust at the ground during launching.	*4*	Ensure the ground around the launch pad has no flammable material surrounding it.	Examine the ground condition before and after every launch.	*3*	*36*

Bird strike or animal strike	Birds might get killed and the vehicle's trajectory is might be affected.	*5*	Birds might get hit by the rocket.	*2*	Visual inspection of vehicle during launch to ensure no bird are hit.	Visually inspect number of bird in area prior to launch to validate risk is low.	*3*	*30*
High dB sounds during launch	Disrupt people living around the launching area	2	Sound created during motor ignition could potentially require hearing protection	5	Measure and monitor the noises during test launches. Provide hearing protection to team if required	Safety officer will ensure all team members will be wearing the required hearing protection	5	50
Sparks from the motor ignite rocket body.	Perchlorate from the ammonium perchlorate composite propellants release to air	4	A fire on or in the rocket would cause critical failure of numerous components.	3	Use non-flammable material for rovers and use waddings for parachutes to make sure they do not catch on fire after landing	Use fireproof wadding everywhere where the motor or black powder charges are found.	3	36

Table 4.6.1: Environmental risks and hazards.

5 Payload Criteria

5.1 Leading Rover Design Overview and Rationale

The final design of the rover consists of a three-wheel configuration in which the two drive wheels on each side will also perform the function of soil sample collection by means of scoops that are built into the wheels. One of the major changes made to the design since the PDR is the orientation of the wheels. This was done to comply with the requirement that the rover can only begin to collect soil after travelling a distance of 10 ft. With the new design, the rover collects no soil while moving in the forward direction. To begin soil collection, the rover will deploy an actuating arm that digs into the soil and provides extra drag as the rover begins to drive in reverse, thus aiding in the ability of the wheels to collect the soil sample.

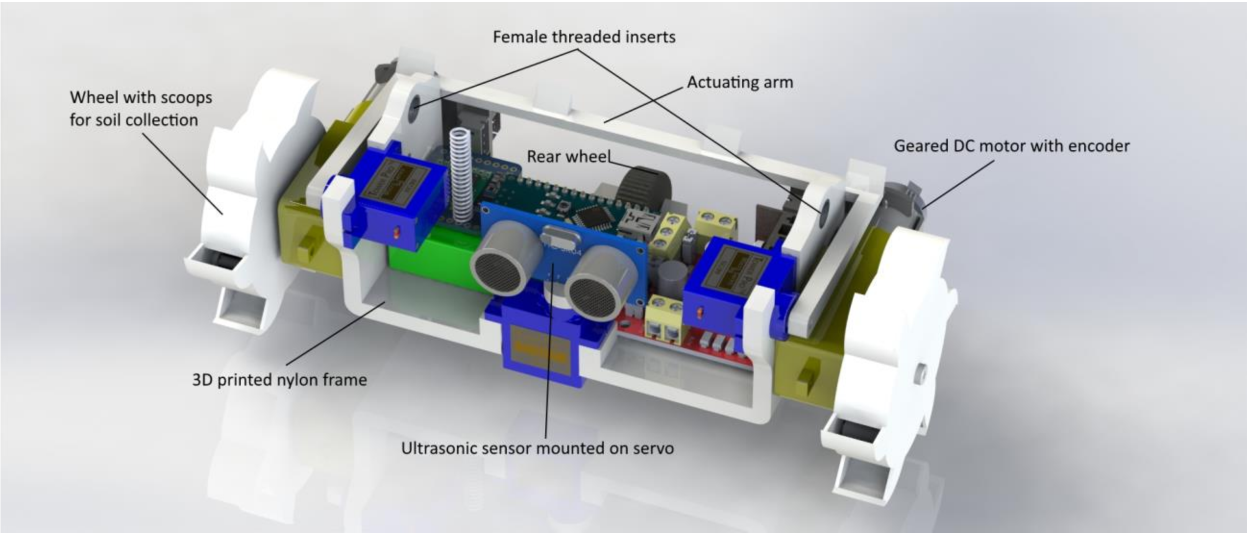


Figure 5.1a: CAD model of the rover with certain critical components labeled.

Figure 5.1a shows an overview of the final rover design highlighting some of the major subsystems. The frame is designed to be 3D printed with Nylon for its high strength and impact resistance, and will serve as the base for attaching all other components via metal fasteners or adhesives. Two geared DC motors drive the main wheels which double as the soil sample collection and containment devices. An actuating arm spans the length of the frame and is attached to a servo on each side, and serves two functions:

1. To correct the rover's orientation after deployment in case the rover is deployed upside down
2. To dig into the ground in front of the rover and provide resistance to movement once the rover begins to collect soil by driving in reverse

The rear wheel provides stability when the rover is in motion, and an ultrasonic sensor mounted on a servo scans the surrounding for obstacles and allows the rover to correct its course.

Control is provided by an Arduino Nano, which communicates with the ground system via an Adafruit RFM95W transceiver. The rover will be deployed using a threaded Aluminum rod that passes through the two female threaded inserts mounted on the upper part of the frame, guided and stabilized by a smooth Nylon rod that goes through the bottom of the frame.

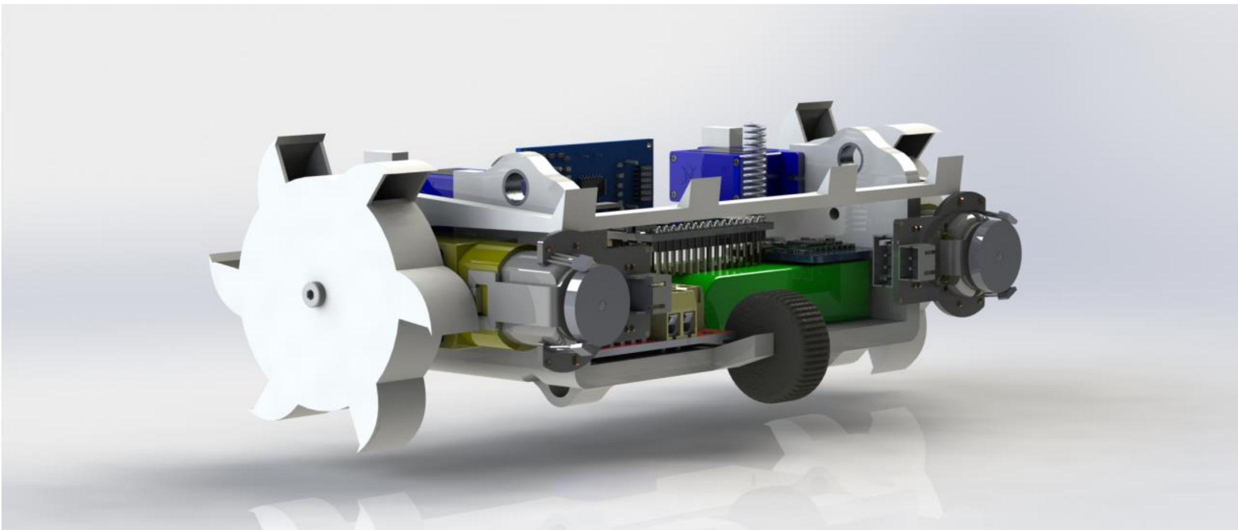


Figure 5.1b: Rear view of the rover

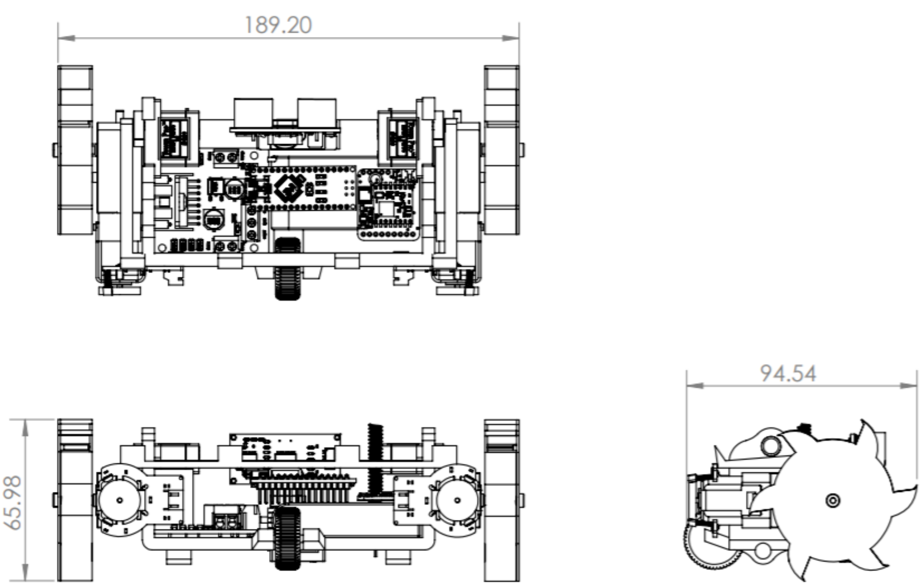


Figure 5.1c: Drawing of the rover showing the maximum dimensions.

As shown on Figure 5.1c, the rover has a total length of 189.20 mm, width of 94.54 mm and a height of 65.98 mm.

After the PDR, studies were done to evaluate the feasibility of alternative designs for the rover, and especially the soil collection mechanism. These alternatives included designs for collecting a core drilling sample, or a servo-powered robotic excavation mechanism as shown in Figure 5.1c below.

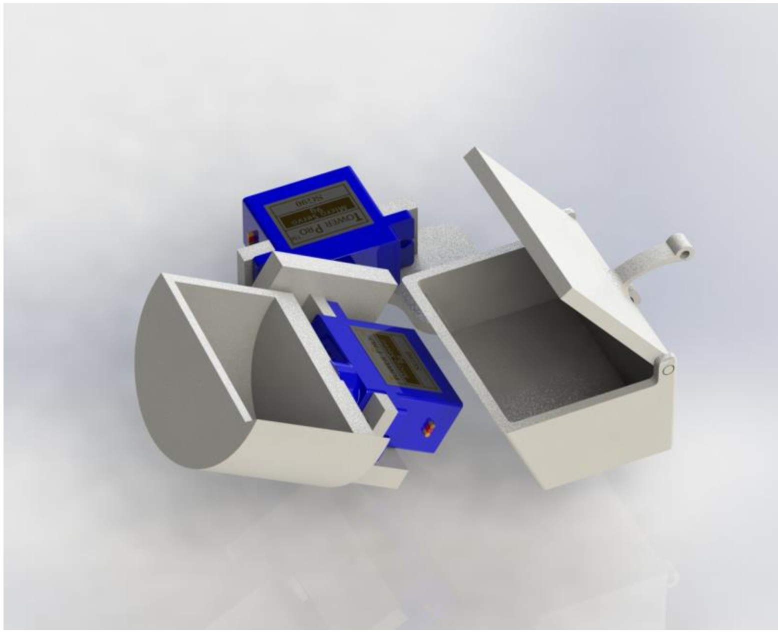


Figure 5.1c: A servo-actuated scoop-and-container mechanism concept. This design was ultimately abandoned due to the lack of adequate space and concerns about structural strength and reliability.

Ultimately, the biggest challenges theses designs faced was the limited space inside the payload bay and the addition of failure points due to the increased complexity. For these reasons, it was decided that the integrated wheel-based soil collection design would be retained in the final design, with some changes and optimizations made to improve its efficacy and meet all requirements.

5.2 Component-level review and specifications

5.2.1 Soil collection mechanism

Soil sample collection is achieved with hollow wheels which contain six scoops along the outside perimeter, and passive valves that allow the soil to be retained inside the body of the wheels, as shown in Figure 5.2.1a.

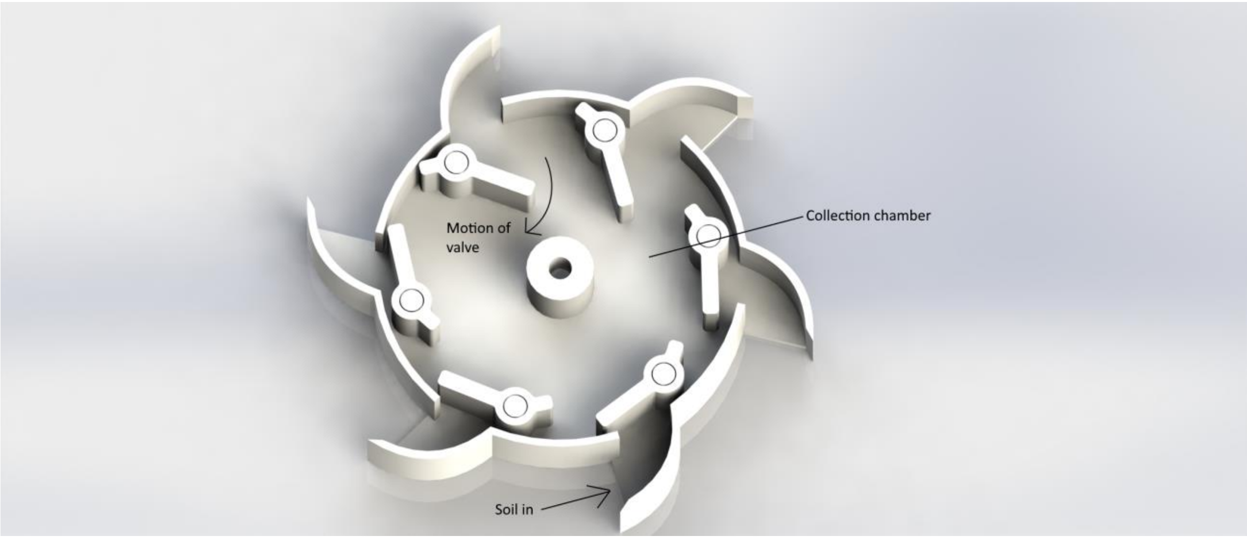


Figure 5.2.1a: Cutaway of the wheel showing the interior mechanism by which soil is collected.

All components of the wheel are 3D printed out of clear PETG filament which offers suitable mechanical properties for this application. The scoops are designed to have sharp edges which help them dig into the soil and the soil is channeled into the interior of the scoop. Once it approaches the top of the wheel, the gravity-controlled one-way valve rotates on its hinge and allows the soil to fall into the hollow interior. On its way down, the valve closes automatically as it turns due to its own weight, preventing the soil sample collected in the chamber from escaping.
The design of the valve was achieved by iterating between different shapes, positions and tolerances until a working prototype could be created.

Preliminary tests of the wheels using a commercially available rover kit showed that the mechanism could successfully collect dry and soft soil, but struggled with other ground environments.

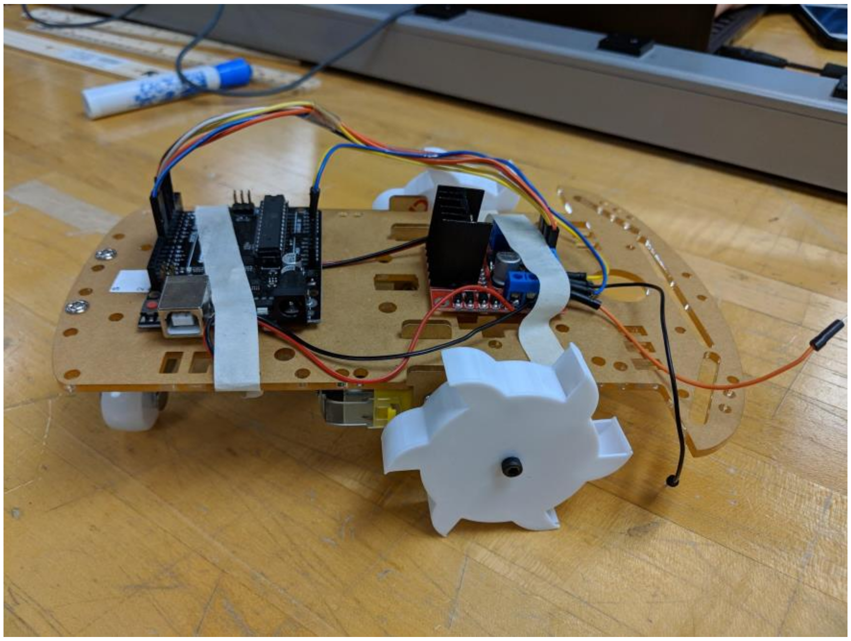


Figure 5.2.1b: Rover kit built for testing the soil collecting capability of the wheels.

To improve the chances of the rover working on various soil types, the design was changed to incorporate a servo-actuated arm that will dig into the soil in front of the rover, providing more grip to the wheels when they turn. Figure 5.2.1c illustrates this design.

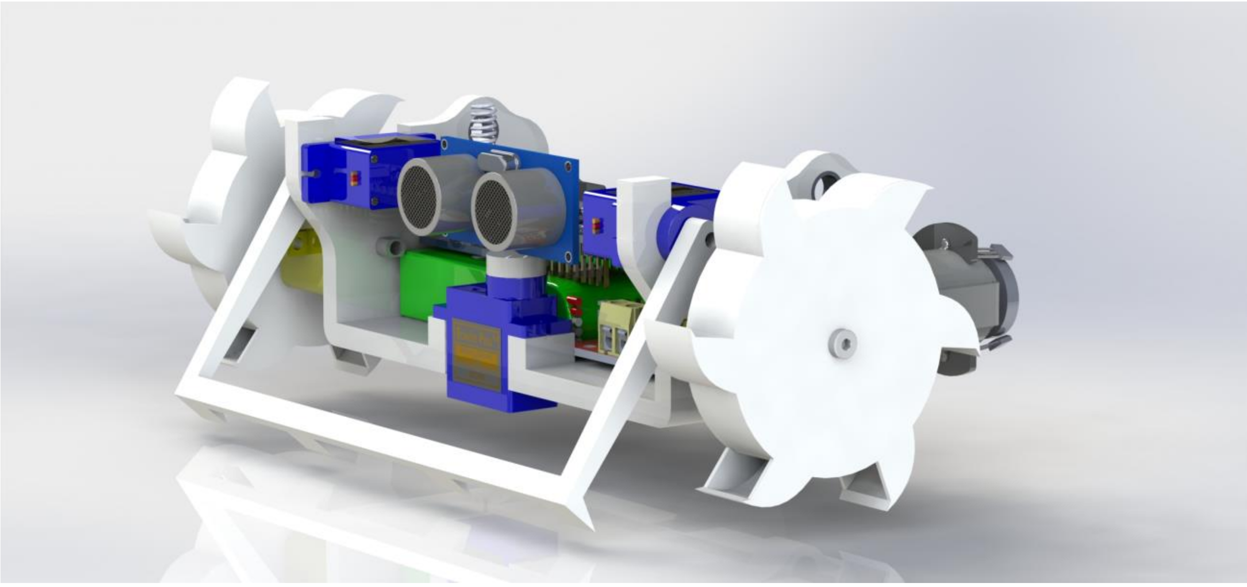


Figure 5.2.1c: Anchor arm in its deployed configuration. The arm is designed to provide extra drag as the wheels turn in reverse, helping the scoops dig into the soil.

The wheels are printed in two parts: body and lid. An M3 screw is used to secure the lid to the body, which also screws into the shaft of the motor through the wheel's axle, thus securing the wheel with the frame.

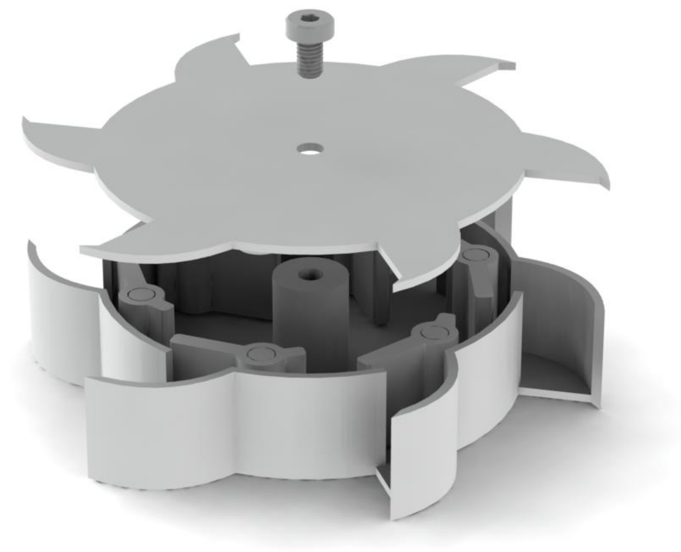


Figure 5.2.1d: Exploded view of the wheel showing the assembly of its components

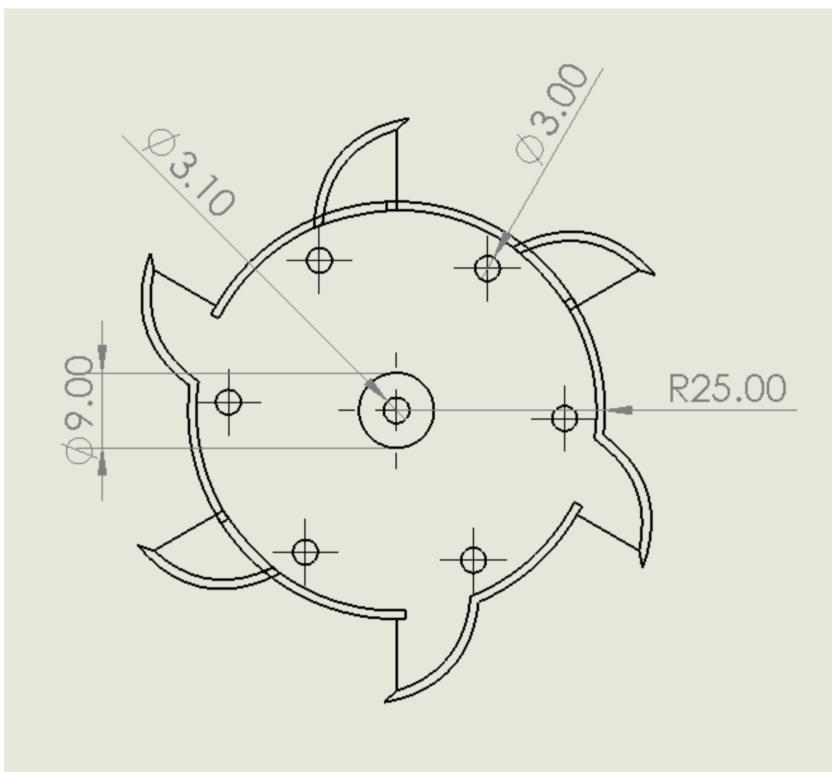


Figure 5.2.1e: Drawing of the wheel with the relevant dimensions

The wheel is 70 mm in diameter including the scoops, and the inner chamber has a diameter of 50 mm. The width is 13 mm on the outside and 12 mm for the interior. The volume of the space inside the chamber of each wheel is approximately 22 cm^3. The collected soil sample will be accessible by removing the screw and and the lid. Further testing is planned for the wheels in different soil environments, and is described in the testing section below. Figure 5.2.1f shows one of the development prototypes printed with PLA for tolerance and function testing.

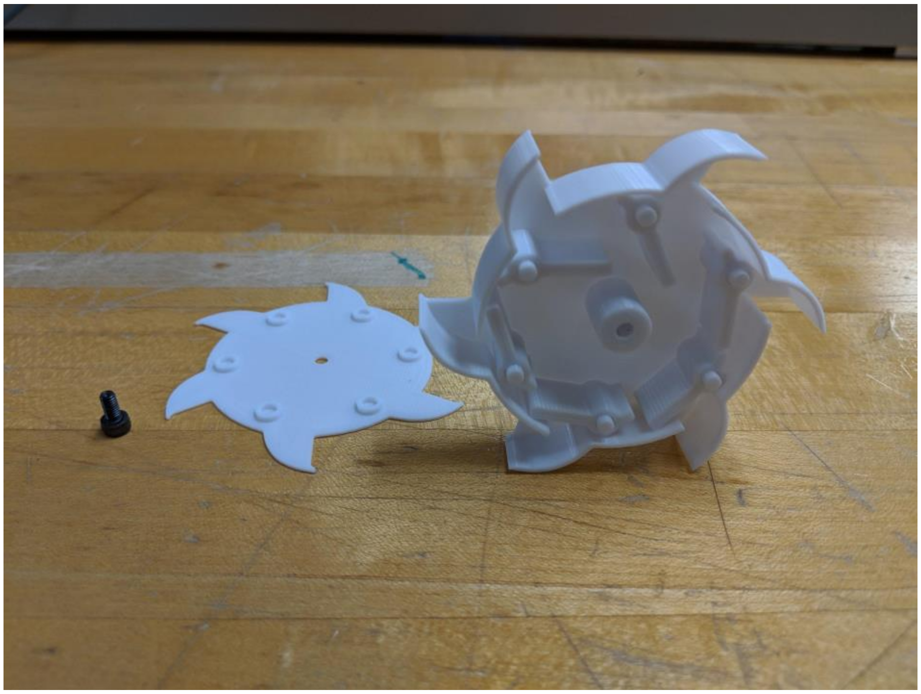


Figure 5.2.1f: Prototype print of the wheel

5.2.2 Frame

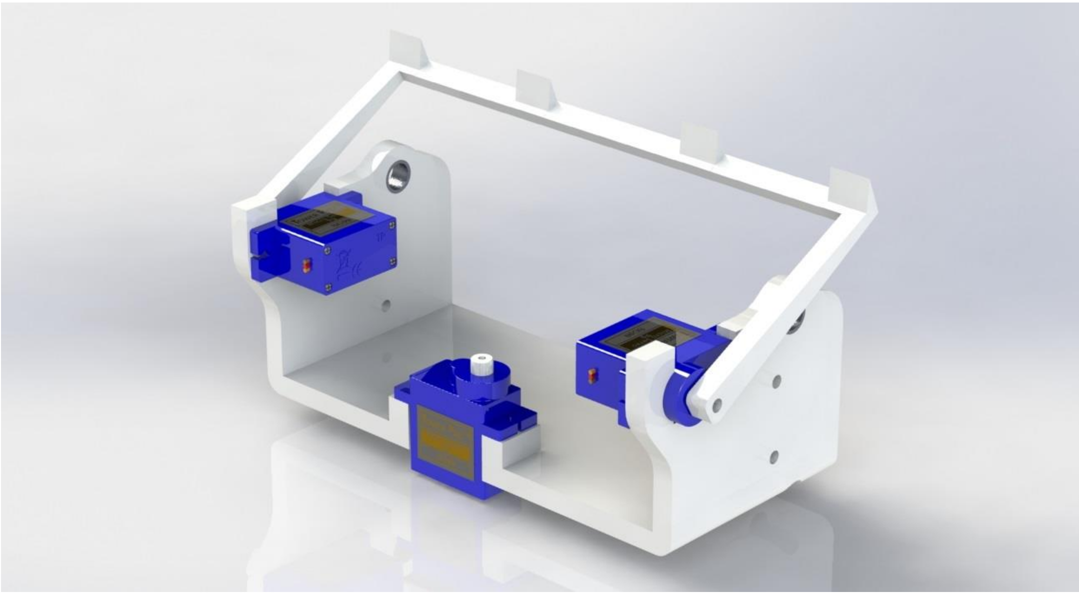


Figure 5.2.2a: Rendering of the rover's frame, including the three servo motors and the actuating arm.

As shown in Figure 5.2.2a, the main frame of the rover will be printed as a single part out of nylon, onto which other components are fastened. The three servos will be attached using M2 screws threaded through the shoulder of the servos. The actuating arm will be printed with Nylon as well, and attached to the servos with servo horns and screws. The decision to use Nylon was made due to the high impact strength and durability of nylon compared to other FDM printing filaments available. The specific filament that will be used is Taulman3D Alloy 910, which is the strongest and toughest material offered by the company. This nylon has a tensile strength of 55 MPa and a stiffness (Young's Modulus) of 503 MPa.

The dimensions of the frame are shown in Table 5.2.2.

Length	120 mm
Width	74 mm
Height	55.7 mm

Table 5.2.2: Dimensions of the rover frame.

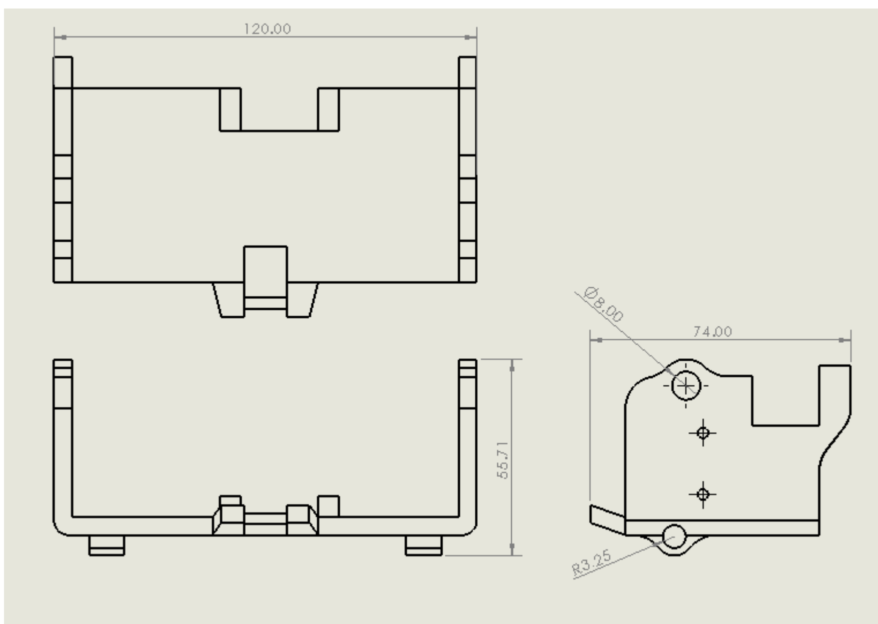


Figure 5.2.2b: Drawing of the frame.

With a print setting of four perimeters and a 30% infill, the estimated mass of the frame is 45 grams.

5.2.3 Motor and motor driver

The main drive motors for the rover are a pair of DFRobot Geared DC motor with encoder, with a nominal no-load speed of 160 rpm at 6 V. The motor was selected based on the criteria of size, torque, speed and feedback capability. This motor includes an encoder which will be used to keep track of the number of rotations of the shaft, and therefore the total travel distance. The two motors are driven by a L298N dual H-bridge motor driver, which allows reversible driving and PWM-based speed control. The driver is rated for up to 35 V and 2 Amps which is well above our driving power requirements.

The motor will be attached to the side wall of the frame with two M3 screws and nuts, and the driver will be attached to the base of the frame with four M3 screws and nuts. Figures 5.2.3a and 5.2.3b show the CAD model of the motor and the drawing respectively.

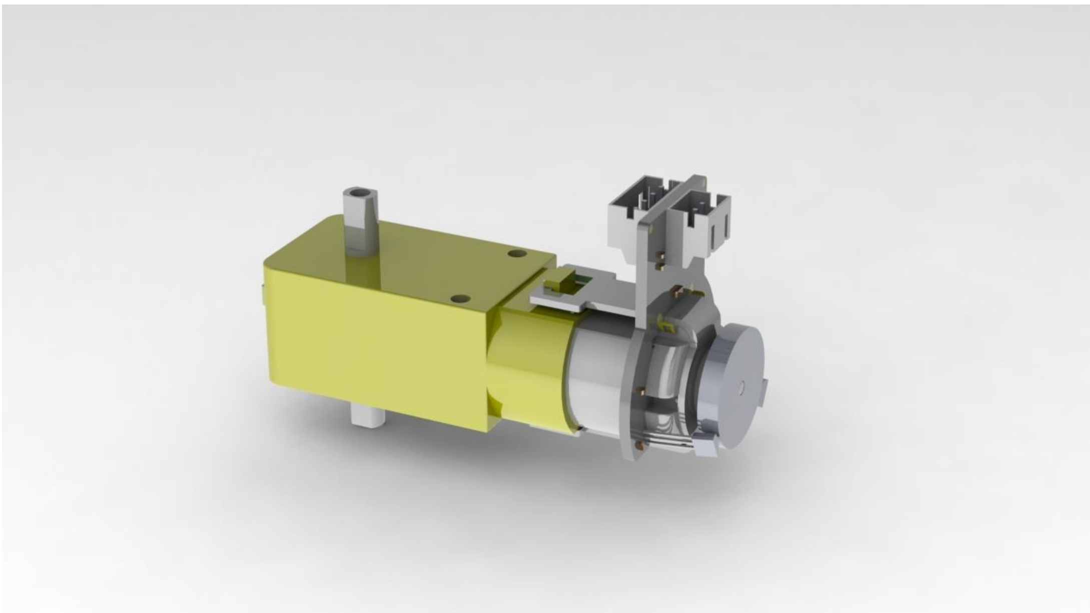


Figure 5.2.3a: CAD rendering of the DFRobot DC gear motor with encoder.

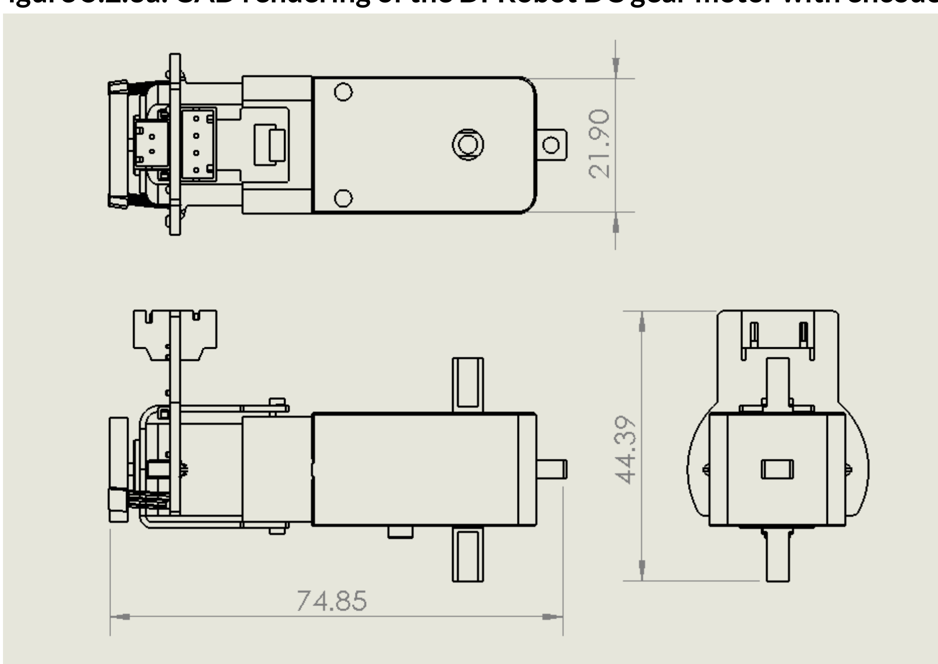


Figure 5.2.3b: Drawing of the DFRobot DC gear motor with encoder.

5.2.4 Microcontroller

An Arduino Nano will be used to control all functions of the rover, including radio communication, obstacle detection and avoidance, and driving of the motor and servos. The choice was based on the size and the number of I/O pins available on the board. The Nano offers a compact size and provides adequate hardware pins to communicate with the transceiver on the SPI protocol and with the accelerometer on the I^2C protocol simultaneously, while also controlling the motors and receiving feedback from the encoder and the ultrasonic sensors. Figure 5.2.4 shows a rendering of an Arduino Nano with header pins soldered on.

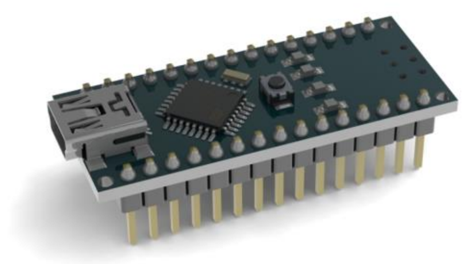


Figure 5.2.4: Arduino Nano.

5.2.5 Obstacle avoidance

Choosing a sensor for the rover was limited to three types: an Ultrasonic sensor, an Infrared sensor, and a Load Cell sensor. Ultrasonic sensors consist of an emitter and a receiver. The emitter transmits sound waves which reflect off any blocking obstacle and comes back to the sensor to its receiver. Infrared sensors are similar to ultrasonic, but instead utilize an infrared laser rather than sound. When the sound or light comes back to its respective sensor, a signal will be received and it will be known that there is an obstacle in the path of the rover. Load cell sensors, however, work via physical contact between it and any obstacles. This type of sensor is both larger and more expensive than the other two. As a result, it performed poorly in its decision matrix in deciding which to use. Infrared sensors are have adequate sizing and pricing, but its accuracy is lacking in that during the daytime (i.e. when the launch will take place), infrared sensors tend to pick up more than what is desired. Similarly, ultrasonic sensors can still operate even if dirt or other contaminants adhere to the sensor. With all of these in mind, a decision matrix was created (Table 5.2.5) and it was determined that the ultrasonic sensor would be the best to use.

	Ultrasonic Sensor	Infrared Sensor	Load Cell Sensor
Size (30%)	10	10	0
Accuracy (30%)	8	6	8
Price (20%)	10	9	0
Ease of Application (20%)	10	10	10
Total	9.4	8.6	4.4

Table 5.2.5: Decision matrix for obstacle sensor selection.

The HC-SR04 ultrasonic sensor will be mounted on a servo using a 3D printed adapter, as shown in Figure 5.2.5. After deployment, the sensor will constantly scan the area in front of the rover by having the servo turn side to side. This will allow the microcontroller to create a live picture of solid obstacles in front of it and use an algorithm to drive away from them. The algorithm will be designed such that the rover will not have to drive in reverse to avoid obstacles. This is possible since the rover can make sharp turns by keeping one wheel stationary. The microcontroller will keep track of all turns and drive sufficiently far so that the rover will end up at least 10 ft from the launch vehicle. Any major turns will reset the distance counter and the rover will start over.

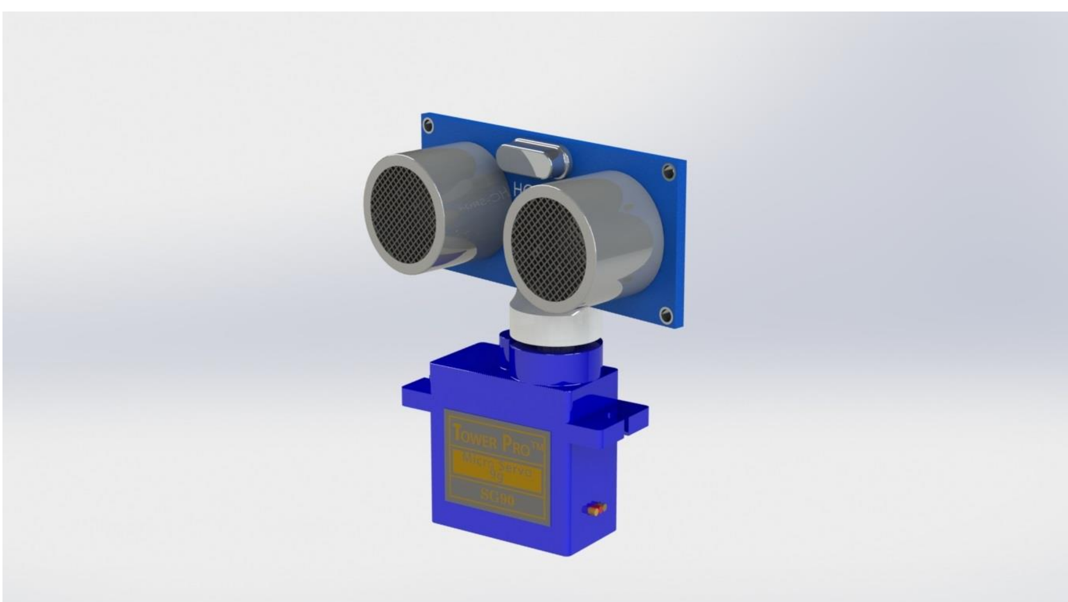


Figure 5.2.5: Rendering of an HC-SR04 Ultrasonic sensor mounted on a SG90 micro servo with an adapter.

5.2.6 Orientation Detection - Accelerometer

There is a possibility that the rover will be deployed in a sub-optimal orientation, such as upside down or face-down. To mitigate this problem, it is necessary for the rover to know its orientation when it starts up. This is best achieved using an accelerometer, since the values don't drift as easily as on a gyroscope, and the rover only needs a rough approximation to determine which one of the four possible discrete orientations it has been deployed in. Table 5.2.6 shows the decision matrix that was used to select the best accelerometer for this purpose.

	MPU-6050	MPU-9250	BNO055	FXOS8700 + FXAS21002	LSM9DS1	L3GD20H
Fusion Calculations (30%)	10	0	10	0	0	0
Zero Rate (15%)	3	6	8	10	1	7
Price (30%)	5	7	3	7	7	10
Power (15%)	5	6	4	7	7	4
Total	5.7	3.9	5.7	4.65	3.3	4.65

Table 5.2.6: Decision matrix for accelerometer selection.

The MPU 6050 chip is an accelerometer and gyroscope module. This module is available as a breakout board called a GY-521 and it easily interfaces with an arduino through the I^2C protocol. Open source libraries are also available for this board, which makes it easy to read and process the data. Only the accelerometer function will be used by the rover.

If the rover detects that it has deployed in an off-nominal orientation, the controller will deploy the actuating arm forward and retract it back. It will then re-check the orientation to make sure the rover has righted itself, and then start scanning with the ultrasonic sensor before beginning to drive. The rover will also occasionally check the orientation during the drive and digging operations to make sure it hasn't flipped by accident.

5.2.7 Ground Communication

The rover will communicate with the ground station through an Adafruit RFM95W 433 MHz transceiver which is connected to the Arduino Nano through the SPI protocol. The Adafruit RFM95W was selected over the previous choice of HC-12 after ground tests revealed that the range of HC-12 transceivers was not as advertised. After deployment, the ground station will command the rover to begin operations, and the rover will send a confirmation. When the rover is done collecting the soil sample, it will send a signal to the ground indicating a successful completion of its mission.

The rover will utilize an omnidirectional, low gain helical antenna due to size constraints, and the ground system will use a high gain directional Moxon antenna to achieve a reliable connection.

5.2.8 Power

The rover will receive power from a MJX X101 7.4V 1200 mAh Lithium-polymer battery. The battery will be located at the base of the frame and affixed to the body using 3D printed constraining parts which will be fastened to the frame using M3 screws and nuts. For redundancy, a 3M industrial double sided tape will be used to attach the battery to the frame first. This will make sure that the battery will be firmly secured to the frame through the duration of the flight and during ground operation. The battery will be taped bright orange for visibility and will be clearly marked as a fire hazard. One of the leads from the battery will be routed through a safety switch located at the bottom of the rover's frame. This switch can be used as an emergency stop and will cut power to all electronics on the rover.

5.2.9 Part and mass list

Section	Part	Quantity	Individual Mass (g)	Total Mass (g)
Rover	Frame	1	45	45
	Wheels	2	20	40
	Geared DC motor	2	32	64
	Battery	1	70	70
	Arduino	1	10	10
	Motor driver	1	20	20
	Transceiver	1	6	6
	Servo	3	9	27
	Hook/flipper	1	8	8
	Accelerometer	1	4	4
	Threaded adapter	2	8	16

	Rear Wheel	1	8	8
	M3 Screws	16	7	112
	Total			**430**
Deployment System	Bottom bulkhead	1	65	65
	Top bulkhead	1	52	52
	Eyebolt + nut	1	30	30
	Stepper motor	1	155	155
	Pusher	1	35	35
	Threaded rod	1	116	116
	Smooth rod	1	64	64
	Adapter	1	13	13
	M4 screws and nuts	3	12	36
	Total			**566**
Total mass				**996**

Table 5.2.9: List of parts and masses of all components of the payload and the retention and deployment system.

The rover has a total mass of 430 grams and the retention and deployment system weighs 566 grams. This results in a total payload system mass of 996 grams.

5.3 Payload Interface with Vehicle

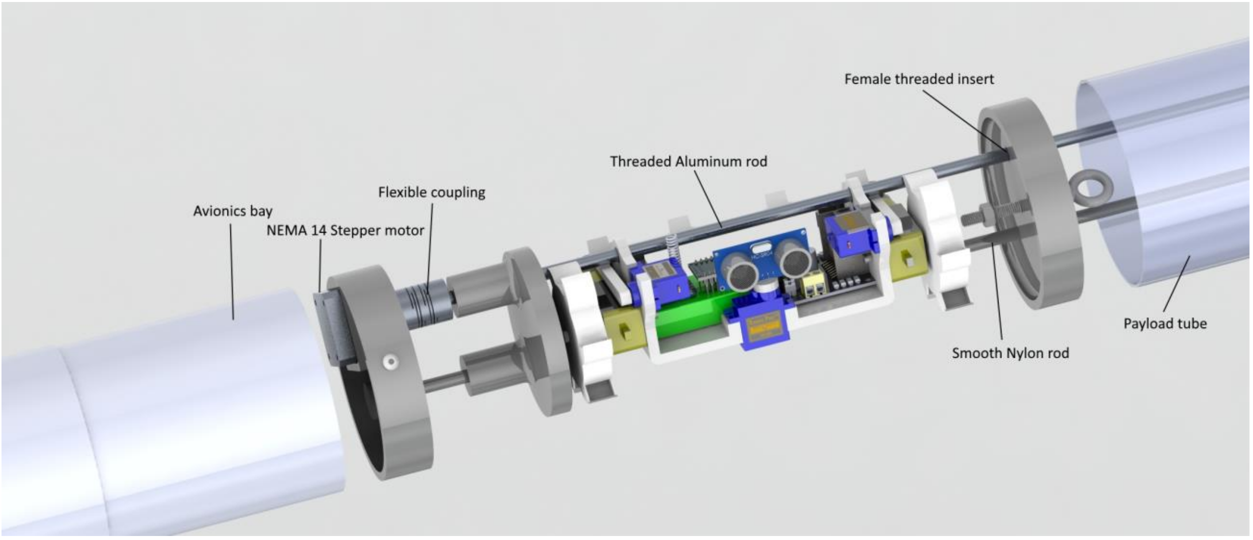


Figure 5.3a: CAD render showing the payload retention and deployment mechanism.

The rover will be housed in the payload section of the launch vehicle below the nose cone, and will be secured in between two bulkheads during flight, as shown in Figure 5.3a. The two bulkheads are 3D printed with nylon for strength and impact resistance. The aft bulkhead sits ahead of the avionics bay and is attached to the payload tube with three M4 round headed hex screws and nuts. It houses a NEMA 14 stepper motor which will be used to turn an aluminum threaded rod to deploy the rover. The aft bulkhead also contains an attachment point for a smooth nylon rod which will be used for stability and to keep the rover from moving during flight. In front of the aft bulkhead is a pusher part, also printed with nylon and having a female threaded insert, whose function is to make sure the rover exits the vehicle completely. Without this part, the rover would lose contact with the threaded rod while part of it is still inside the payload tube.

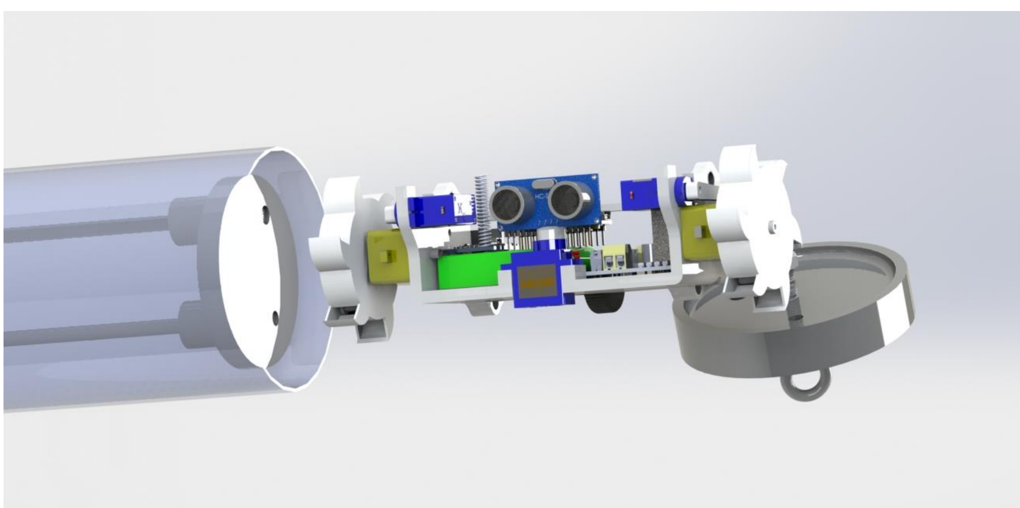


Figure 5.3b: Rendering of the expected rover deployment process

As seen in Figures 5.3a and 5.3b, the forward bulkhead also contains a threaded insert, which allows it to be ejected along with the rover. The drogue parachute is placed on top of the forward bulkhead and attached to the eyebolt. This means that during descent under the drogue parachute, the weight of the vehicle will be supported by the threaded rod connected to the aft bulkhead through the stepper motor. The eyebolt will be kept close to the location of the threaded insert in the forward bulkhead to minimize any bending or shearing stresses. Figure 5.3c shows the bottom portion of the payload retention and deployment assembly with key parts labeled.

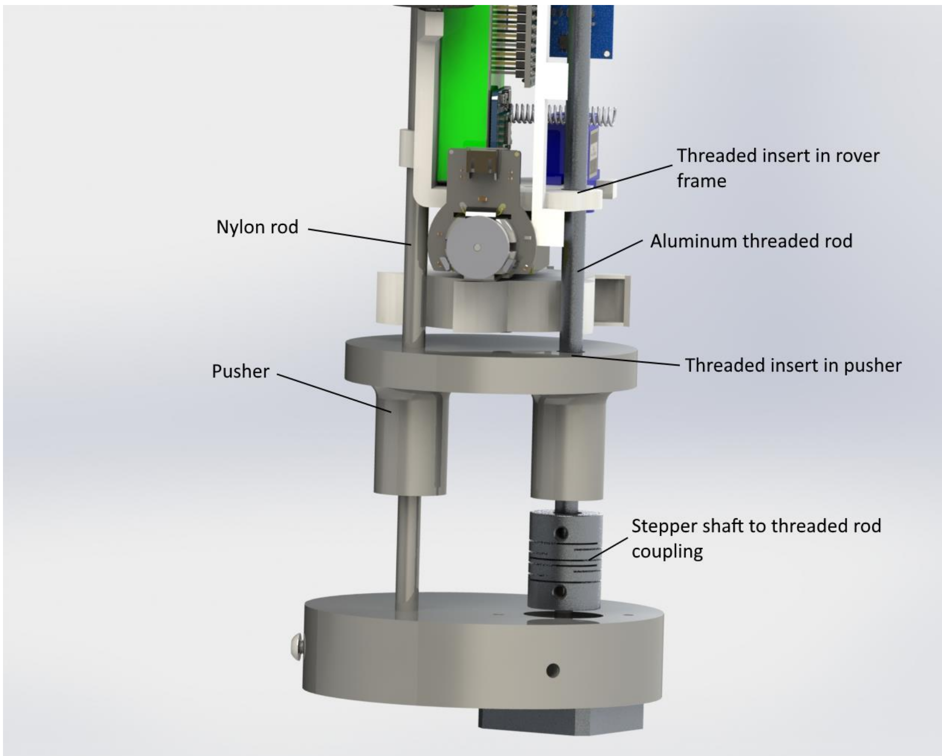


Figure 5.3c: Illustration of the bottom part of the payload retention system

To make sure that the retention system can support this load, the material for the rod was selected to be 6061 aluminum alloy. The rod will be ¼ " (6.35mm) in diameter, and with a yield strength of 55 MPa, it can support a maximum load of 1,741 N or roughly 17 times the weight of the rocket. This allows an ample safety margin against the possibility of the rod failing in yield during flight.

The aluminum rod is connected to the stepper motor through a threaded coupling. While the loads from a drogue deployment are much smaller than that from the main parachute, it is still critical to validate this coupler for the expected tensile loads before this design is flown, so one of the planned tests includes a tensile loading test of the entire assembly.

Active retention of the payload is achieved by keeping it constrained between two bulkheads and attached to the two rods at all times during flight. The stepper motor is powered on before flight and is kept powered through the payload deployment. This provides holding torque that prevents any of the parts of the retention system from unscrewing themselves. In the unlikely event of a complete loss of power to the stepper motor, the four different contact points for the female threads to the rod is expected to provide enough friction to overcome any torque-induced rotation, especially since the threaded insert in the forward bulkhead would experience bending forces, and the fact that the rod will have fine threads (28 threads per inch). This failure mode will also be tested as part of the overall tensile test of the payload retention system.

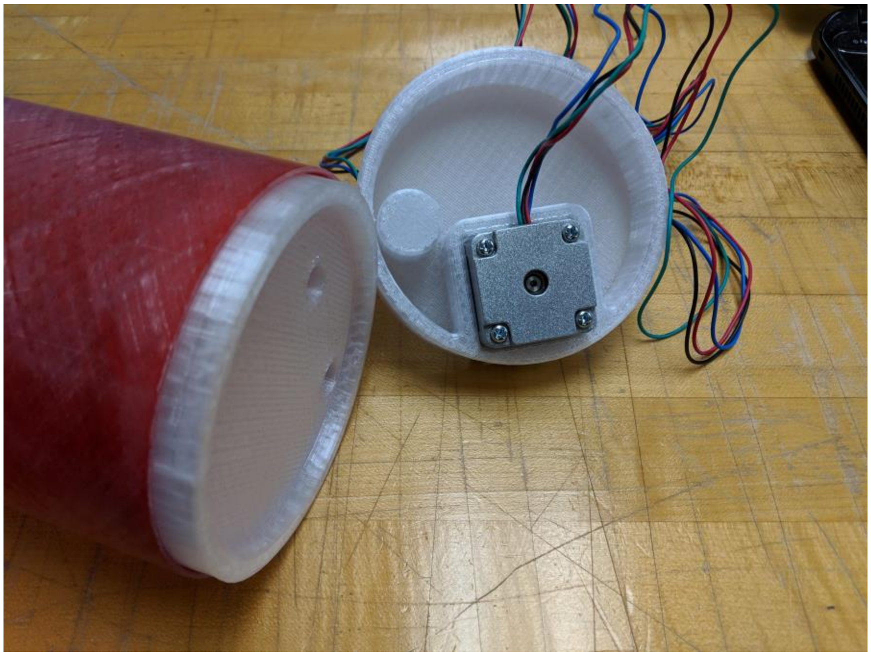


Figure 5.3d: Preliminary fit and tolerance testing of the 3D printed fore and aft bulkheads. The dimensions of the bulkheads will be adjusted to allow a smooth deployment without excessive friction or any binding of the fore bulkhead.

5.4 Electronic Components, Block Diagram and Schematic

Component	Details	Quantity
Battery	7.4V 1200mAh LiPo battery pack	1
Microcontroller	Arduino Nano R3	1
Motor controller	L298N dual H-bridge driver	1
Motor	DFRobot TT motor with encoder	2
Obstacle Sensor	HC-SR04 Ultrasonic sensor	1
Radio Transceiver	Adafruit RFM95W 433 MHz	1
Gyroscope for orientation	GY-521 breakout for the MPU 6050 module	1

Servo for flipping rover	TowerPro SG90 270° micro servo	2
Servo for obstacle sensor	TowerPro SG90 micro servo	1
Safety switch	Two pin toggle switch	1

Table 5.4: A list of electronic components that will be used in the rover.

Table 5.4 lists all the electronic components that the rover will contain. Individual components and their functions have been explained in Section 5.2 above. The block diagram shown in Figure 5.4a shows how all components and subsystems interact with each other.

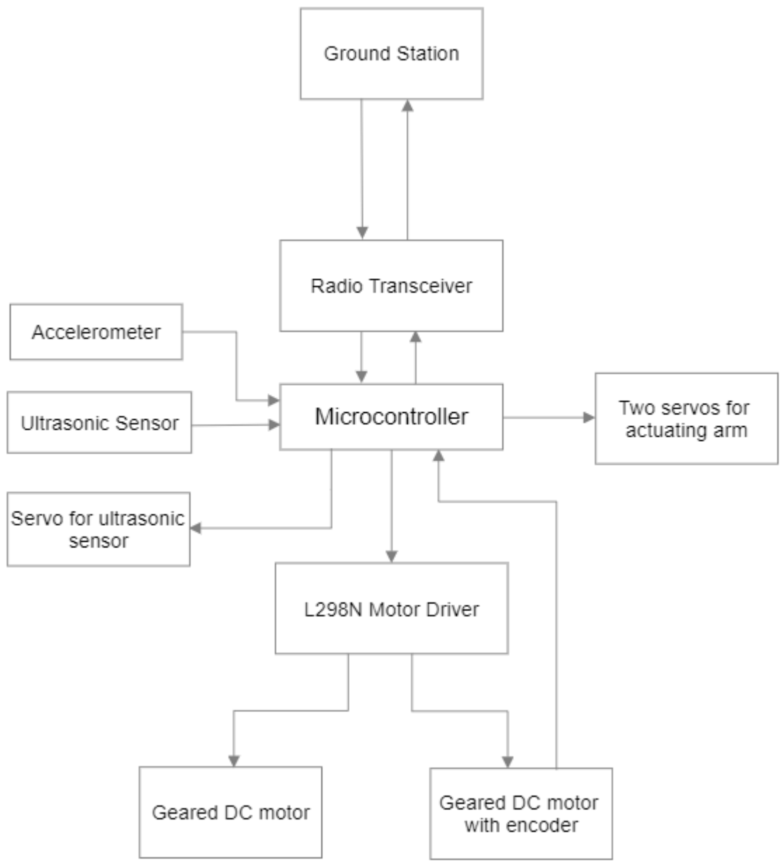


Figure 5.4a: Block diagram of the rover's subsystems.

The microcontroller takes inputs from the accelerometer for orientation information, and the ultrasonic sensor for obstacle detection. It maintains a two-way communication with the transceiver, which in turn is connected to the ground system. The microcontroller also sends output to the actuating arm servos for either flipping the rover upright, or to begin soil collection. To drive the rover, it sends output to the motor driver using PWM and digital signals for speed and direction control, respectively. The encoder in one of the motors then sends input signals to the microcontroller as the rover moves to keep track of the distance travelled.

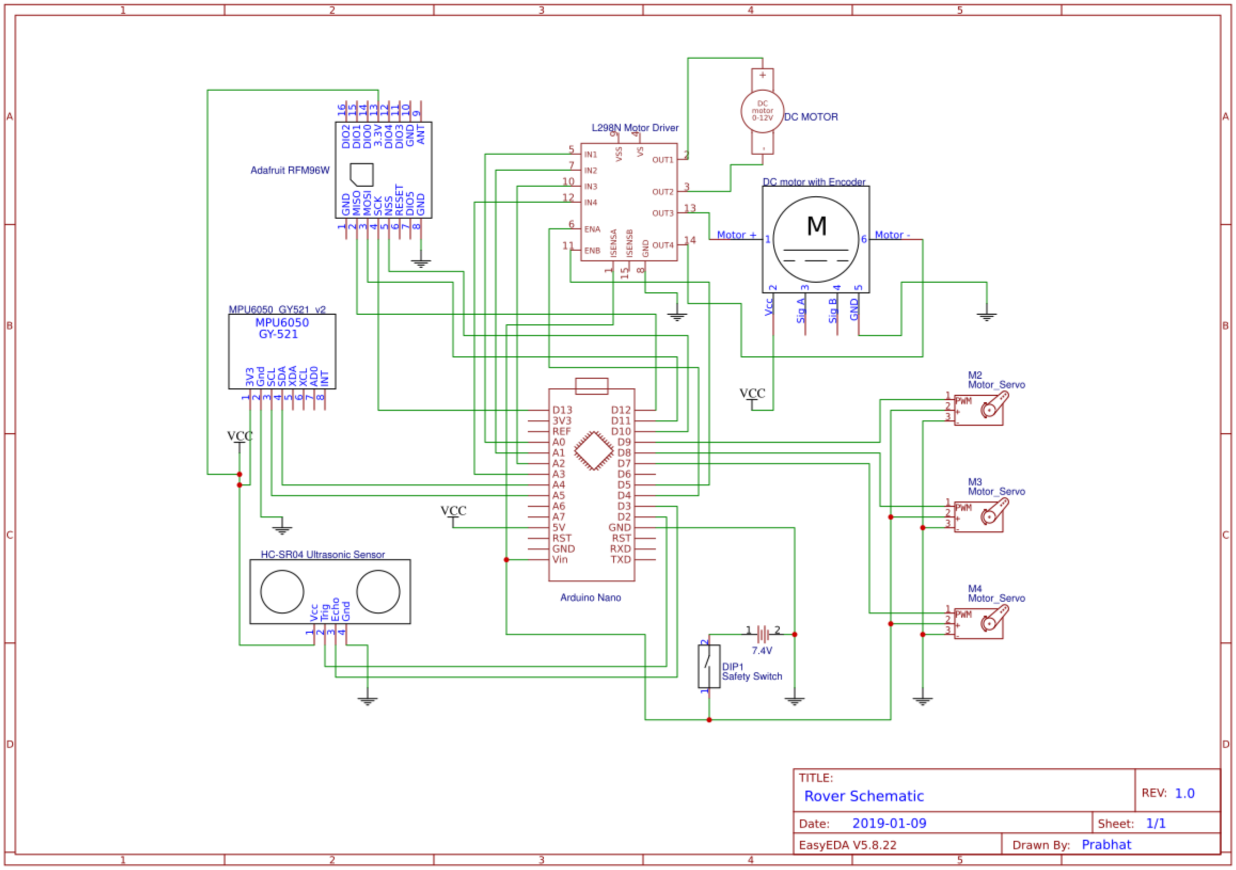


Figure 5.4b: Schematic showing all electronic components of the rover.

Figure 5.4b shows the complete schematic of the rover's electronic system. It can be seen that the battery is directly connected to the safety switch before any other components are connected. The safety switch will be a simple one way, two-terminal switch located on the outside of the frame protruding slightly from the bottom surface. There are two voltage levels in the circuit: the full battery voltage (7.4 V nominal) ranging from 8.4 V to 7 V depending on the state of charge of the battery, and the 5 volt supply from the arduino. The motor driver and the three servos are connected directly to the battery voltage, while the transceiver, accelerometer and the ultrasonic sensor are powered by the 5 V output (VCC) from the arduino. Finally, all ground terminals are connected together.

6 Project Plan

6.1 Requirements Verification

6.1.1 Rules Based Requirements

Requirement	Verification

General requirements	
Students on the team will do 100% of the project, including design, construction, written reports, presentations, and flight preparation with the exception of assembling the motors and handling black powder or any variant of ejection charges, or preparing and installing electric matches (to be done by the team's mentor).	Demonstration will be used to verify that students on the team will do 100% of the project by recording all members involved any given task.
The team will provide and maintain a project plan to include, but not limited to the following items: project milestones, budget and community support, checklists, personnel assignments, STEM engagement events, and risks and mitigations.	This is demonstrated with the Gantt charts below, the recorded work of the systems team, the personnel hazard analysis, and the failure modes and effect analysis.
Foreign National (FN) team members must be identified by the Preliminary Design Review (PDR) and may or may not have access to certain activities during launch week due to security restrictions. In addition, FN's may be separated from their team during certain activities.	All Foreign National Team Members have filled out the appropriate paperwork and have been identified in the PDR
The team must identify all team members attending launch week activities by the Critical Design Review (CDR). Team members will include: ● Students actively engaged in the project throughout the entire year. ● One mentor ● No more than two adult educators.	All PRT members actively engaged in team activities starting September 2018 through the CDR, our mentor Duane Wilkey, our advisor Matthew Barry, and up to one other adult educator will be recorded as being a part of the Pitt Rocketry Team from 2018-2019 by the CDR.
The team will engage a minimum of 200 participants in educational, hands-on science, technology, engineering, and mathematics (STEM) activities, as defined in the STEM Engagement Activity Report, by FRR. To satisfy this requirement, all events must occur between project acceptance and the FRR due date and the STEM Engagement Activity Report must be submitted via email within two weeks of the completion of the event.	As described in our proposal, through collaboration with Pitt's Society of Physics Students, PRT will present to 2-3 schools about the technical information regarding rocketry as well as opportunities in STEM fields. The details of each presentation are currently being developed and the meeting of this requirement will continue to be demonstrated through each report.
The team will establish a social media presence to inform the public about team activities.	Our team's Instagram account can be found at https://www.instagram.com/pittrocketryteam

	/ where we post about team activities.
Teams will email all deliverables to the NASA project management team by the deadline specified in the handbook for each milestone. In the event that a deliverable is too large to attach to an email, inclusion of a link to download the file will be sufficient.	All team deliverables have been sent to NASA by the deadline as requested.
All deliverables must be in PDF format.	All deliverables are in PDF format
In every report, teams will provide a table of contents including major sections and their respective sub-sections.	Table of Contents provided as seen in beginning of report
In every report, the team will include the page number at the bottom of the page.	See bottom of page
The team will provide any computer equipment necessary to perform a video teleconference with the review panel. This includes, but is not limited to, a computer system, video camera, speaker telephone, and a sufficient Internet connection. Cellular phones should be used for speakerphone capability only as a last resort.	PRT will reserve all necessary space and computer equipment to teleconference with the review panel prior to each design review.
All teams will be required to use the launch pads provided by Student Launch's launch services provider. No custom pads will be permitted on the launch field. Eight foot 1010 rails and 12 foot 1515 rails will be provided. The launch rails will be canted 5 to 10 degrees away from the crowd on launch day. The exact cant will depend on launch day wind conditions.	PRT will not create a custom pad for the launch. The launch vehicle will be compatible with the launch pad provided by the Student Launch's launch services provider.
Each team must identify a "mentor." A mentor is defined as an adult who is included as a team member, who will be supporting the team (or multiple teams) throughout the project year, and may or may not be affiliated with the school, institution, or organization. The mentor must maintain a current certification, and be in good standing, through the National Association of Rocketry (NAR) or Tripoli Rocketry Association (TRA) for the motor impulse of the launch vehicle and must have	The PDR demonstrates that Duane Wilkey, a level 3 certified NAR member is our team's mentor. Duane possesses evidence of the necessary requirements to assist our team as the designated mentor.

flown and successfully recovered (using electronic, staged recovery) a minimum of 2 flights in this or a higher impulse class, prior to PDR. The mentor is designated as the individual owner of the rocket for liability purposes and must travel with the team to launch week. One travel stipend will be provided per mentor regardless of the number of teams he or she supports. The stipend will only be provided if the team passes FRR and the team and mentor attend launch week in April.	
Vehicle requirements	
The vehicle will deliver the payload to an apogee altitude between 4,000 and 5,500 feet above ground level (AGL). Teams flying below 3,500 feet or above 6,000 feet on Launch Day will be disqualified and receive zero altitude points towards their overall project score.	The mechanical subteam will design the rocket to reach an apogee of 4,750 feet, and the result will be demonstrated by the readout of the altimeters during test flights and on launch day.
Teams shall identify their target altitude goal at the PDR milestone. The declared target altitude will be used to determine the team's altitude score during Launch Week.	Our target altitude is 4,750 feet.
The vehicle will carry one commercially available, barometric altimeter for recording the official altitude used in determining the Altitude Award winner. The Altitude Award will be given to the team with the smallest difference between their measured apogee and their official target altitude on launch day.	The avionics bay contains two Stratologger Altimeters that will be used to record the altitude.
Each altimeter will be armed by a dedicated mechanical arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad. Each altimeter will have a dedicated power supply.	Inspection will show that our recovery system has been designed including these specifications.
Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	Our choice in arming switches will adhere to this guideline.
The launch vehicle will be designed to be recoverable and reusable. Reusable is defined	There will be no expendable components on the vehicle. The vehicle will be re-armable

as being able to launch again on the same day without repairs or modifications.	following a launch.
The launch vehicle will have a maximum of four (4) independent sections. An independent section is defined as a section that is either tethered to the main vehicle or is recovered separately from the main vehicle using its own parachute.	The PRT-01 will consist of three sections: A nose cone, the body tube (which contains the avionics bay) , and the booster section
Coupler/airframe shoulders which are located at in-flight separation points will be at least 1 body diameter in length.	The lengths of the airframe shoulders were measured and compared to that of their respective diameter and it was found that the lengths are indeed at least their diameter. These values can be seen in figures 3.2.1a-e.
Nose cone shoulders which are located at in-flight separation points will be at least ½ body diameter in length.	The length and diameter of the nose cone shoulder were measured and it was proved that the length was at least ½ its body diameter. These values can be seen in figure 3.2.1a.
The launch vehicle will be limited to a single stage.	The only motor used will be a single stage refuelable motor.
The launch vehicle will be capable of being prepared for flight at the launch site within 2 hours of the time the Federal Aviation Administration flight waiver opens.	The rocket is capable of going through preflight preparations within two hours.
The launch vehicle will be capable of remaining in launch-ready configuration on the pad for a minimum of 2 hours without losing the functionality of any critical on-board components.	All sensitive (namely electronic) components will be left idle in flight configuration for a minimum of two hours to ensure that the launch vehicle is capable of remaining in launch-ready configuration on the pad for a minimum of 2 hours without losing the functionality of any critical on-board components.
The launch vehicle will be capable of being launched by a standard 12-volt direct current firing system. The firing system will be provided by the NASA-designated launch services provider	The motor in use is able to be launched with a standard 12-volt DC firing system. This will be verified by inspection.
The launch vehicle will require no external circuitry or special ground support equipment to initiate launch (other than what is provided by the launch services provider).	The motor in use will require no external circuitry or special ground support to be launched. This will be verified by inspection.

The launch vehicle will use a commercially available solid motor propulsion system using ammonium perchlorate composite propellant (APCP) which is approved and certified by the National Association of Rocketry (NAR), Tripoli Rocketry Association (TRA), and/or the Canadian Association of Rocketry (CAR).	The motor used is a commercially available APCP fueled motor. APCP purchased will be certified by the NAR or TRA.
Final motor choices will be declared by the Critical Design Review (CDR) milestone. Any motor change after CDR must be approved by the NASA Range Safety Officer (RSO) and will only be approved if the change is for the sole purpose of increasing the safety margin. A penalty against the team's overall score will be incurred when a motor change is made after the CDR milestone, regardless of the reason.	Motor has already been chosen through research. Further tests and research will ensure that the correct motor is chosen before the CDR milestone. If motor needs to be changed after this, the NASA RSO will be notified for approval.
Pressure vessels on the vehicle will be approved by the RSO and will meet the following criteria: ● The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) will be 4:1 with supporting design documentation included in all milestone reviews. ● Each pressure vessel will include a pressure relief valve that sees the full pressure of the tank and is capable of withstanding the maximum pressure and flow rate of the tank. ● Full pedigree of the tank will be described, including the application for which the tank was designed, and the history of the tank, including the number of pressure cycles put on the tank, by whom, and when.	The final rocket design does not utilize a pressure vessel.
The total impulse provided by a College or University launch vehicle will not exceed 5,120 Newton-seconds (L-class).	Motors with impulses greater than 5,120 Ns will not be considered for our rocket.
The launch vehicle will have a minimum static stability margin of 2.0 at the point of rail exit. Rail exit is defined at the point where the forward rail button loses contact with the rail.	Masses within the launch vehicle and fin surface area will be adjusted as necessary throughout design process to ensure stability margin is greater than 2.0
The launch vehicle will accelerate to a	The motor will be chosen to ensure that the

minimum velocity of 52 fps at rail exit.	launch vehicle will accelerate to a velocity greater than 52 fps at the point of rail exit.
All teams will successfully launch and recover a subscale model of their rocket prior to CDR. Subscales are not required to be high power rockets.	A subscale rocket has been manufactured and launched prior to the CDR deadline.
The subscale model should resemble and perform as similarly as possible to the full-scale model, however, the full-scale will not be used as the subscale model.	The subscale rocket has same design as the full-scale model but smaller to ensure the subscale performs as similarly as possible to the full scale rocket. The subscale rocket is not full-scale size.
The subscale model will carry an altimeter capable of recording the model's apogee altitude.	Two Perfectflite StratoLoggerCF altimeters were used on the subscale model.
The subscale rocket must be a newly constructed rocket, designed and built specifically for this year's project.	Our team has completed construction of the subscale rocket that recently launched on Saturday January 5th .
Proof of a successful flight shall be supplied in the CDR report. Altimeter data output may be used to meet this requirement.	This altimeter output from the subscale flight is included in the CDR.
An FRR Addendum will be required for any team completing a Payload Demonstration Flight or NASA required Vehicle Demonstration Re-flight after the submission of the FRR Report.	If PRT requires a NASA required Vehicle Demonstration Re-Flight or a Payload Demonstration flight, an FRR addendum will be submitted to NASA after the FRR report.
Teams required to complete a Vehicle Demonstration Re-Flight and failing to submit the FRR Addendum by the deadline will not be permitted to fly the vehicle at launch week.	PRT will complete a Vehicle Demonstration Re-Flight if necessary, in a timely manner to ensure FRR Addendum is submitted by the correct deadline.
Teams who successfully complete a Vehicle Demonstration Flight but fail to qualify the payload by satisfactorily completing the Payload Demonstration Flight requirement will not be permitted to fly the payload at launch week.	The Pitt Rocketry Team will complete all tasks by their deadlines.
Teams who complete a Payload Demonstration Flight which is not fully successful may petition the NASA RSO for permission to fly the payload at launch week. Permission will not be	The Pitt Rocketry Team will complete all tasks by their deadlines.

granted if the RSO or the Review Panel have any safety concerns.	
Any structural protuberance on the rocket will be located aft of the burnout center of gravity.	All structural protuberances such as fins will be located aft of the center of gravity after burnout.
The team's name and launch day contact information shall be in or on the rocket airframe as well as in or on any section of the vehicle that separates during flight and is not tethered to the main airframe. This information shall be included in a manner that allows the information to be retrieved without the need to open or separate the vehicle.	This information will be listed on the fins of the rocket, verifiable by inspection. Additionally, it will be listed on the top of the rover.
Vehicle demonstration flight	
All teams will successfully launch and recover their full-scale rocket prior to FRR in its final flight configuration. The rocket flown must be the same rocket to be flown on launch day. The purpose of the Vehicle Demonstration Flight is to validate the launch vehicle's stability, structural integrity, recovery systems, and the team's ability to prepare the launch vehicle for flight. A successful flight is defined as a launch in which all hardware is functioning properly (i.e. drogue chute at apogee, main chute at the intended lower altitude, functioning tracking devices, etc.).	Our final, full-scale design of the rocket will be tested prior to FRR in its final flight configuration. This will be done at local launchings in an audience and supervision of trained and accredited rocket specialists, as well as general rocket hobbyists.
The vehicle and recovery system will have functioned as designed	The vehicle and recovery system has been thoroughly tested and proven to operate as designed
The full-scale rocket must be a newly constructed rocket, designed and built specifically for this year's project.	The full scale rocket is a newly constructed rocket built and designed by PRT for the NASA 2019 Student Launch competition.
The payload does not have to be flown during the full-scale Vehicle Demonstration Flight. The following requirements still apply: • If the payload is not flown, mass simulators will be used to simulate the payload mass. • The mass simulators will be located in the same approximate location on the	If payload is unable to be flown on Vehicle Demonstration flight, a mass will be added to simulate the mass of the payload and located in the same area as the payload.

rocket as the missing payload mass.	
If the payload changes the external surfaces of the rocket (such as with camera housings or external probes) or manages the total energy of the vehicle, those systems will be active during the full-scale Vehicle Demonstration Flight.	The PRT payload is not designed to change the external surface of the rocket, but if payload design changes to affect the external rocket surface the external systems will be active during the full-scale Vehicle Demonstration Flight
Teams shall fly the launch day motor for the Vehicle Demonstration Flight. The RSO may approve use of an alternative motor if the home launch field cannot support the full impulse of the launch day motor or in other extenuating circumstances.	The launch day motor will be used during the Vehicle Demonstration Flight. If launch field cannot support full impulse of launch day motor on Vehicle Demonstration Flight, an alternative motor will be used with the approval of the RSO.
The vehicle must be flown in its fully ballasted configuration during the full-scale test flight. Fully ballasted refers to the same amount of ballast that will be flown during the launch day flight. Additional ballast may not be added without a re-flight of the full-scale launch vehicle.	A check will be performed to verify that the vehicle flown during the full-scale test flight is in its fully ballasted configuration.
After successfully completing the full-scale demonstration flight, the launch vehicle or any of its components will not be modified without the concurrence of the NASA Range Safety Officer (RSO).	Following the successful completion of the full-scale demonstration flight, the launch vehicle and its components will not be modified without the concurrence of the NASA Range Safety Officer.
Proof of a successful flight shall be supplied in the FRR report. Altimeter data output is required to meet this requirement.	The recovery altimeters will collect and store flight data in their on-board loggers. The flight data will be recovered following the flight for use in the FRR report.
Vehicle Demonstration flights must be completed by the FRR submission deadline. If the Student Launch office determines that a Vehicle Demonstration Re-flight is necessary, then an extension may be granted. This extension is only valid for re-flights, not first-time flights. Teams completing a required re-flight must submit an FRR Addendum by the FRR Addendum deadline.	The team will ensure that the Vehicle Demonstration flights are completed by the FRR submission deadline. If the Student Launch office determines that a Vehicle Demonstration Re-flight is necessary and an extension is granted, the team will submit an FRR Addendum by the FRR Addendum deadline.
Payload Demonstration Flight	
The payload must be fully retained throughout	The retention mechanism will keep the rover

the entirety of the flight, all retention mechanisms must function as designed, and the retention mechanism must not sustain damage requiring repair	constrained between two bulkheads and supported by two rigid rods throughout the duration of the flight. The deployment stepper motor will be powered and apply a holding torque so any of the components don't move inadvertently.
The payload flown must be the final, active version.	The final version of the rover will be ready before the payload demonstration flight.
If the above criteria is met during the original Vehicle Demonstration Flight, occurring prior to the FRR deadline and the information is included in the FRR package, the additional flight and FRR Addendum are not required.	If the payload is ready by the Vehicle Demonstration Flight, it will be flown during that flight.
Payload Demonstration Flights must be completed by the FRR Addendum deadline. No extensions will be granted	The team will ensure that the Payload Demonstration Flights are completed by the FRR Addendum deadline.
Vehicle Prohibitions	
The launch vehicle will not utilize forward canards. Camera housings will be exempted, provided the team can show that the housing(s) causes minimal aerodynamic effect on the rocket's stability.	The vehicle will be designed to not contain any forward canards or camera housings. This is verifiable by inspection.
The launch vehicle will not utilize forward firing motors.	The vehicle design will not utilize forward firing motors. This is verifiable by inspection.
The launch vehicle will not utilize motors that expel titanium sponges (Sparky, Skidmark, MetalStorm, etc.)	The vehicle design will not utilize that expel titanium sponges. This is verifiable by inspection.
The launch vehicle will not utilize hybrid motors.	The vehicle design will not utilize hybrid motors. This is verifiable by inspection.
The launch vehicle will not utilize a cluster of motors.	The vehicle design will not utilize a cluster of motors. This is verifiable by inspection.
The launch vehicle will not utilize friction fitting for motors.	The vehicle design will not utilize friction fitting for motors. This is verifiable by inspection.
The launch vehicle will not exceed Mach 1 at any point during flight.	The motor utilized and the overall final design of our rocket will be incapable of producing enough thrust force to achieve Mach 1.

Vehicle ballast will not exceed 10% of the total unballasted weight of the rocket as it would sit on the pad (i.e. a rocket with and unballasted weight of 40 lbs. on the pad may contain a maximum of 4 lbs. of ballast).	The ballasted weight of our rocket design will be checked prior to launch and made sure not to exceed 10% of the unballasted weight. This will be done by calculating and summing the individual weights of the parts, then comparing it to the weight expected to be used for unballasting purposes.
Transmissions from onboard transmitters will not exceed 250 mW of power	The transceivers and antenna used will be incapable of transmitting a signal of 250 mW of power.
Excessive and/or dense metal will not be utilized in the construction of the vehicle. Use of lightweight metal will be permitted but limited to the amount necessary to ensure structural integrity of the airframe under the expected operating stresses.	Only the desired and appropriate amount of metal needed for our design will be used. Likewise, dense metal will not be used.
Recovery System Requirements	
The launch vehicle will stage the deployment of its recovery devices, where a drogue parachute is deployed at apogee and a main parachute is deployed at a lower altitude. Tumble or streamer recovery from apogee to main parachute deployment is also permissible, provided that kinetic energy during drogue-stage descent is reasonable, as deemed by the RSO.	The recovery system is designed to stage the deployment of the drogue and main parachutes, with the main to be deployed at a lower altitude.
Each team must perform a successful ground ejection test for both the drogue and main parachutes. This must be done prior to the initial subscale and full-scale launches.	Tests for the recovery systems were properly tested before the launches.
At landing, each independent section of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf.	Appropriate parachute sizes to reduce the kinetic energy of the rocket below 75 ft-lbf have been calculated and will be used in the recovery system.
The recovery system electrical circuits will be completely independent of any payload electrical circuits.	The recovery system electrical circuits have been designed to be completely independent of all other electrical circuits on the vehicle. The recovery system will be tested and verified independent of the rest of the vehicle.
All recovery electronics will be powered by	The recovery system design includes only

commercially available batteries.	commercially available batteries.
The main parachute shall be deployed no lower than 500 feet.	The altimeters will be tested to confirm that they can precisely deploy the main parachute at a height greater than 500 feet. Analysis of the flight logs will verify that this requirement is satisfied.
The apogee event may contain a delay of no more than 2 seconds.	The altimeters have been tested to confirm that they can precisely deploy the drogue within 2 seconds of reaching the apogee. Analysis of the flight logs verify that this requirement is satisfied.
The recovery system will contain redundant, commercially available altimeters. The term “altimeters” includes both simple altimeters and more sophisticated flight computers.	The design of the recovery system includes two Perfectflite StratologgerCF altimeters, both able to activate the charges for the parachutes.
Motor ejection is not a permissible form of primary or secondary deployment.	The motor will not be ejected during flight.
Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.	Removable shear pins are used in the design of the parachute compartments.
Recovery area will be limited to a 2,500 ft. radius from the launch pads.	The recovery area will be limited to a 2,500 ft. radius from the launch pad based on the subscale design and simulations, as well as initial testing of the rocket.
Descent time will be limited to 90 seconds (apogee to touch down).	The descent time will be limited to 90 seconds. This will be done so based on simulations, the subscale design, and mathematical calculations.
An electronic tracking device will be installed in the launch vehicle and will transmit the position of the tethered vehicle or any independent section to a ground receiver	All sections of the rocket are to be tethered to each other, allowing the GPS system in the avionics bay and the GPS system on the releasable payload to satisfy this requirement. The tethers will be chosen to withstand the tensile forces that may be imposed on them during flight.
Any rocket section or payload component, which lands untethered to the launch vehicle, will contain an active electronic tracking device.	The rocket and payload will be the only two separated components.

The electronic tracking device(s) will be fully functional during the official flight on launch day.	The GPS units will be powered and sending data at a constant frequency during launch and recovery. The batteries chosen for the rocket will have enough power to sustain this.
The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	The recovery system electronics will be in a separate compartment from all other on-board electronics.
The recovery system altimeters will be physically located in a separate compartment within the vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.	The recovery system electronics will be in a separate compartment from all other on-board electronics.
The recovery system electronics will be shielded from all onboard transmitting devices to avoid inadvertent excitation of the recovery system electronics.	The compartment housing the recovery system electronics will be protected with radio frequency shielding.
The recovery system electronics will be shielded from all onboard devices which may generate magnetic waves (such as generators, solenoid valves, and Tesla coils) to avoid inadvertent excitation of the recovery system.	Any device that may create enough magnetic waves to affect the recovery system will be surrounded with a high permeability metal to prevent the waves from reaching the recovery electronics compartment. As of the current design, it is highly unlikely that this would be necessary, but proper magnetic protection can be verified through appropriate testing.
The recovery system electronics will be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	Proper testing can verify that the recovery system will be unaffected by other onboard devices.
Payload Experiment Requirements	
Each team will choose one experiment option from the following list. ● Option 1: Deployable Rover/Soil Sample Recovery ● Option 2: Deployable UAV/Beacon Delivery	The team will build a deployable rover that will recover a soil sample after landing
An additional experiment (limit of 1) is allowed, and may be flown, but will not contribute to scoring.	The team will be flying any additional experiments so there is no verification plan in place.
If the team chooses to fly an additional	The team will not by flying any additional

experiment, they will provide the appropriate documentation in all design reports so the experiment may be reviewed for flight safety.	experiments so there is no verification plan in place.
Deployable Rover / Soil Sample Recovery Requirements	
Teams will design a custom rover that will deploy from the internal structure of the launch vehicle.	The custom designed rover as explained in section 5.1 will be housed in the payload section of the launch vehicle.
The rover will be retained within the vehicle utilizing a fail-safe active retention system. The retention system will be robust enough to retain the rover if atypical flight forces are experienced.	The rover will be threaded into an aluminum rod and supported on both sides by solid surfaces. For active retention, the deployment stepper will be powered on during the flight and set to hold torque to avoid any movement. The retention system will be tested to verify that the payload will be retained in the event of power loss of the stepper motor.
At landing, and under the supervision of the Remote Deployment Officer, the team will remotely activate a trigger to deploy the rover from the rocket.	The avionics bay will be able to receive a remote signal to deploy the rover.
After deployment, the rover will autonomously move at least 10 ft. (in any direction) from the launch vehicle. Once the rover has reached its final destination, it will recover a soil sample.	The rover will use motors with an encoder to keep track of the distance travelled, and will navigate using active obstacle avoidance. In the event of sharp turns, the distance counter will reset and the rover will travel greater than 10 ft before beginning soil collection
The soil sample will be a minimum of 10 milliliters (mL).	The wheels that collect soil are designed to have an internal volume of 22 ml each, and each wheel is expected to collect up to half of its internal volume worth of soil. This will be verified through testing.
The soil sample will be contained in an onboard container or compartment. The container or compartment will be closed or sealed to protect the sample after collection.	The soil sample will be contained within the interior of the wheels which are covered. The soil sample is accessible by removing the side of the wheel
Teams will ensure the rover's batteries are sufficiently protected from impact with the ground.	The rover's battery will be constrained with 3D printed parts on the the impact-resistant nylon frame, and will be attached with an industrial double sided tape for redundancy.
The batteries powering the rover will be brightly colored, clearly marked as a fire	The batteries on the rover will be covered with a high visibility tape and marked as a fire

hazard, and easily distinguishable from other rover parts.	hazard on all sides.
Safety Requirements	
Each team will use a launch and safety checklist. The final checklists will be included in the FRR report and used during the Launch Readiness Review (LRR) and any launch day operations.	The team will develop a launch and safety checklist to be included in the FRR report and used in the Launch Readiness Review and any launch day operations.
Each team must identify a student safety officer who will be responsible for the following requirements: • Monitor team activities with an emphasis on Safety during: Design of vehicle and payload, Construction of vehicle and payload, Assembly of vehicle and payload, Ground testing of vehicle and payload, Subscale launch test(s), Full-scale launch test(s), Launch day, Recovery activities, STEM Engagement Activities • Implement procedures developed by the team for construction, assembly, launch, and recovery activities. • Manage and maintain current revisions of the team's hazard analyses, failure modes analyses, procedures, and MSDS/chemical inventory data. • Assist in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	Thomas Sullivan Harrington has been identified as the student safety officer and will perform the listed requirements.
During test flights, teams will abide by the rules and guidance of the local rocketry club's RSO. The allowance of certain vehicle configurations and/or payloads at the NASA Student Launch does not give explicit or implicit authority for teams to fly those vehicle configurations and/or payloads at other club launches. Teams should communicate their intentions to the local club's President or Prefect and RSO before attending any NAR or TRA launch.	The team will not launch the vehicle designed for the NASA Student Launch at any NAR or TRA launch unless allowed by the local President or Prefect and RSO. If the team wishes to launch the vehicle at any NAR or TRA launch, a member will contact the President or Prefect and RSO for permission.
Teams will abide by all rules set forth by the FAA.	The team has read the rules set forth by the FAA, and has and will ask any necessary

	questions to ensure that the rules are fully understood.

Table 6.1.1.

6.1.2 Team Derived Requirements

Requirement	**Verification**
Vehicle	
Avionics bay must be easily accessible.	Bulkhead at access point will be removable such that the avionics bay can be removed from launch vehicle.
Electronics in the avionics bay must be protected from water which could permanently damage flight-critical hardware.	Build and test a protective enclosure to go inside the bay. Check the amount of water that permeates the enclosure upon submersion and impact with a body of water.
Outside of airframe is must be smooth.	Airframe will be sanded and a clear coat will be added on top of sticker used to identify rocket.
Fins must be properly spaced and attached.	Create and utilize a jig for fin attachment.
Recovery	
Shear pins must break during recovery stage of flight at parachute deployment.	Simulation and testing will confirm that the black powder is able to appropriately break the shear pins.
Parachute must not get tangled to ensure the recovery system operates successfully.	Parachute ejection tests and simulations of parachute placement will verify that the parachute does not tangle during recovery. Swivels will be used to minimize parachute and shock cord entanglement.
Payload	
Rover must be rigid when held in its enclosure.	Extensive testing will confirm that the rover stays secured under various loads. A mechanical rig will be built for this purpose.
Rover egress must not be hindered by any launch vehicle components such as bulkheads or shock cords.	Simulate landings and test rover deployments. Adjust stepper motor power and speed until reliable deployment is confirmed.

Wheel scoops must work in a variety of common soil conditions.	Test and refine scoop design to make it work in different soil conditions.
Wheel scoop valves must open to allow maximum soil containment for a given interval volume.	Build various prototypes with different valves and hinge angles to choose the best design.
Obstacle avoidance system must work reliably.	Test the rover with simulated obstacles.
Team performance	
Avionics tests are to be performed by team members not involved in their creation to remove bias from test results.	The avionics lead will assign tests being conducted to the appropriate team members to meet this requirement.

Table 6.1.2.

6.2 Testing

After determining which tests are required, component and system test procedures are being developed. For the avionics team, these procedures are carried out by team members unassociated with the design of the relevant component or system. This improves the validity and rigor of these tests by helping to remove bias from data collection and interpretation. Figure 6.2 shows a diagram included in the test procedure for the altimeters. For the mechanical and payload teams, testing procedures will be developed and carried out by members working on that system, as they have the most knowledge of the system and what aspects of it need to be tested. These tests will be reviewed by other members of the team and our mentor to ensure their validity.

Test	**Black powder charge effectiveness**
Objective	Ensure that the black powder properly sections the rocket
Success criteria	The quantity of black powder used is able to reliably separate the rocket sections
Methodology	After calculating an amount of black powder that should be able to section the rocket, test whether that amount of black powder is sufficient to perform its task.
Reason for necessity	Failure to separate rocket sections would prevent parachutes from deploying and result in a catastrophic hazard
Potential outcomes	- Increase amount of black powder used - Use fixed charges instead of free floating charges - Change shear pins
Testing plan	Assemble rocket with black powder charges routed to the outside. Place the rocket on the ground. Using extra long leads, trigger the

	charges from a safe distance. Observe whether the quantity of black powder used was sufficient to section the rocket.
Results	The test confirmed that 1.5 grams was appropriate for sectioning the subscale.

Test	**Black powder charge security**
Objective	Ensure that the black powder does not prematurely ignite
Success criteria	The altimeters do not output current to the charges before apogee or 550 feet descending
Methodology	Determine whether the StratologgerCFs output voltage to the charges before it is supposed to in a flight by simulating a flight with reducing and increasing pressure around an altimeter.
Reason for necessity	Premature parachute deployment is a dangerous hazard and must be prevented
Potential outcomes	- Switch altimeter boards - Design a protective circuit to filter altimeter output
Testing plan	Plug the altimeter into a battery and safety switch as it would be in the avionics bay. Attach its charge outputs to 25 watt resistors probed by oscilloscopes. Monitor the graphs produced for spikes in voltage over simulated flights. The flights were simulated by using a bike pump to suck air out of a plastic container with the altimeter inside.
Results	The test confirmed that the altimeters function properly, outputting the maximum voltage of the battery at the appropriate times without producing any outputs before they are scheduled.

Test	**Shock cord durability**
Objective	Ensure shock cords are reliable and able to be reflown
Success criteria	Shock cords do not break or sustain tearing or damage
Methodology	Visual inspections preflight and postflight of all shock cords and recovery systems
Reason for necessity	Team safety, range safety, and reusability of the rocket rely on proper functioning of the shock cords
Potential outcomes	- Shock cords need to be strengthened - Redundant cords need to be implemented

Testing plan	Before and after every flight, shock cords and recovery systems will be thoroughly inspected and compared to preflight conditions in the case of postflight inspection.

Test	**Wire connection**
Objective	Ensure that all wires are secure
Success criteria	It is reasonably certain that no wires in the avionics and recovery systems will be disconnected during flight or recovery
Methodology	Approximate forces that wires may experience by tugging on the assembled avionics system to determine if part of the system is poorly designed or built
Reason for necessity	A poorly assembled avionics system may unexpectedly lose functional capacity during launch if it experiences too much force. Preventing this increases the safety the launch, allowing for proper recovery.
Potential outcomes	- Disassemble and reassemble avionics system - Choose new connectors
Testing plan	Tug on all wires after the avionics have been assembled as if for launch.
Results	Partial completion. Only the recovery system's connections have been tested. A JST adaptor will be connected between the battery and the altimeters to add slack to the wire and reduce the likelihood of the landing removing the wires from the altimeters.

Test	**Avionics Bay Systems Test**
Objective	Ensure that none of the components interfere with each other when operated from the same controller.
Success criteria	All systems perform their desired action under similar flight conditions.
Methodology	Operate the avionics bay under simulated flight conditions.
Reason for necessity	All mission critical and recovery critical components depend on the proper functioning of the avionics bay.
Potential outcomes	- Different components are required - Different component placement is required - Different component orientation is required
Testing plan	Simulate flight conditions, and test for proper function from every component in the avionics bay.

Test	**Recovery RF Shielding Tests**
Objective	Ensure quality and effectiveness of RF shielding applied to the Recovery system.
Success criteria	Recovery system receives no interference from the flight computer.
Methodology	Simulate conditions inside of the avionics bay, and test different RF shielding in these conditions.
Reason for necessity	Ensuring that the recovery system works is critical to safety and reusability.
Potential outcomes	- More RF shielding is needed - A better RF shielding material is needed
Testing plan	Apply RF shielding to necessary components. Test under simulated flight conditions, while specifically observing systems that are susceptible to RF interference.

Test	**Recovery Altimeter Precision Verification**
Objective	Investigate precision of the barometric altimeters used to deploy the parachutes.
Success criteria	Signals are sent to the drogue charges at apogee and main parachute at specifically set heights.
Methodology	Simulate altitude change by manipulating atmospheric pressure.
Reason for necessity	The electronics of the recovery system are necessary to precisely deploy the parachutes for a swift and safe recovery.
Potential outcomes	- Main parachute may need to be triggered at a slightly greater height depending on the error of the device.
Testing plan	Use test rig in Figure 6.2

Test	**Computer connection**
Objective	Ensure that the ground computer is connected to the avionics system and payload
Success criteria	The ground system can receive telemetry from the flight computer and send data to the flight computer to command rover release.
Methodology	Add a feature to the ground and flight computers to visibly

	demonstrate connection.
Reason for necessity	This test will need to be performed for a full scale launch to make sure that the GPS is usable
Potential outcomes	- Diagnose the problem and try connecting again - Modify communication methods - Change communication methods
Testing plan	Simulate flight conditions while ensuring constant two-way communication. Increment the distance between the ground and flight computer and record results.

Test	**Device power**
Objective	Ensure that all electronic devices will stay powered for the duration of the flight and potential delay on the launch pad
Success criteria	The flight computer, altimeters, payload, and ground computer can stay powered for at least three hours
Methodology	Calculate the approximate energy capacity needed for batteries on the rocket and payload and then test their ability to power their devices for at least three hours. Calculations will be based on measured efficiency characteristics of on-board voltage regulator and rated peak current-draws of all on-board electronics.
Reason for necessity	The avionics must be powered to function. Their failure could cause a failed recovery. A recovery system failure presents a team and range safety hazard.
Potential outcomes	- Reselect batteries - Find ways to reduce power consumption
Testing plan	Set up the full avionics system. Simulate a 2 hour wait on the launch pad by providing no input, then simulate the flight on the altimeters by reducing and then increasing the pressure around them. Release and activate the payload. Wait the additional time needed to recover the landed rocket and payload. Check the power levels of the batteries.

Test	**Transceiver Effectiveness**
Objective	Ensure the transceiver pair is able to operate within the range specified by the rocket flight profile.
Success criteria	Both the flight computer and the ground station are able to send and receive valid data consistently at a number of ranges.

Methodology	Simulate transmitting and receiving data in a landed situation.
Reason for necessity	Maintaining a link between the ground computer and the flight computer is mission critical.
Potential outcomes	- Refine transceiver / avionics circuitry - Use different antennas on the ground computer and/or flight computer - Choose a different transceiver module
Testing plan	The ground computer and the flight computer will be enabled and perform mission related tasks. The flight computer will be obstructed by the same material found on the rocket body, and moved away from the ground computer. The signal strength with respect to range will be logged and graphed.

Test	**GPS Tracking**
Objective	Ensure the flight computer is able to maintain a GPS signal fix in various simulated mission scenarios
Success criteria	The flight computer consistently receives GPS data in every test case
Methodology	Simulate environmental factors and potential scenarios which could impede the GPS module antenna from acquiring a fix. This includes testing the module with other flight hardware that may cause interference.
Reason for necessity	GPS telemetry is necessary to track the flight and eventually recover and refly the rocket.
Potential outcomes	- Reposition GPS module on the avionics bay - Select a new GPS module
Testing plan	Test the GPS module in scenarios where the antenna is obstructed by materials found in the avionics bay. Test the GPS antenna in multiple different orientations that could occur during landing.

Test	**Flight Preparation Practice**
Objective	To ensure that our team can get the rocket ready for launch quickly and efficiently
Success criteria	The rocket is flight ready from storage within one and a half hours
Methodology	Create an assembly procedure. Multiple “dress rehearsals” will be practiced to ensure the rocket can be reliably assembled within our set time parameters.

Reason for necessity	The rocket must be capable of being prepared for flight at the launch site within the two hour launch window specified by the Federal Aviation Administration flight waiver.
Potential outcomes	- Assembly procedures need to be modified - Assembly procedures need to be practiced.
Testing plan	Assemble the rocket and simulate preparing all systems for launch within the launch window. Disassemble the rocket and rehearse multiple times for practice and consistency.

Test	**Flight Delay Readiness**
Objective	To ensure that the vehicle can stay prepared for launch for up to two hours and still function properly
Success criteria	Vehicle remains flight ready and can have a successful launch two hours after being prepared
Methodology	Simulate pre-flight scenario on the avionics bay.
Reason for necessity	The rocket could potentially spend 2 hours on the pad before launch
Potential outcomes	- Power consumption needs to be reduced - Larger batteries are required
Testing plan	Observe voltage levels on all avionics components in flight configuration.

Test	**Payload : Orientation Detection and Correction Mechanism**
Objective	To ensure that the rover can properly detect if it's upside down and correct itself.
Success criteria	The rover successfully flips itself over after being deployed in an abnormal orientation
Methodology	Deploy the rover in different orientations to allow it to adjust and re-align to the proper orientation.
Reason for necessity	The rover needs to be in the proper position (on its wheels) throughout its entire journey in order for it to move.
Potential outcomes	- Different actuating arm must be used - More accurate sensor must be used
Testing plan	Observe rover in different positions and see how well it can adjust itself.

Test	**Deployment Assembly Function**
Objective	To ensure that the rover can exit the rocket without any issues.
Success criteria	The rover is able to properly function after it successfully exits the rocket.
Methodology	Test the rover exiting the payload section by building prototypes of the retention and deployment mechanism
Reason for necessity	The rover needs to be able to exit the rocket without any parts failing in order for it to drive forward and collect soil samples.
Potential outcomes	- Stepper motor torque might need to be adjusted - The bulkheads must be designed to allow free movement
Testing plan	Build the prototypes for the rover retention system and use either the completed rover or a simulated mass and perform the deployment

Test	**Payload Retention System at Expected Loads**
Objective	To ensure that the rover is properly retained in the payload section of the rocket and can withstand all loads present on it throughout the launch process and rover removal.
Success criteria	The rover is able to function properly after all loads act on it during its mission.
Methodology	Build a tension testing rig and apply expected flight forces on the retention system, and simulate a loss of power for the stepper motor
Reason for necessity	The rover needs to be retained in the payload section during the launch process such that no parts will fail throughout its mission.
Potential outcomes	- Different threaded rod must be used - Different coupling must be used - Stepper motor shaft must be modified
Testing plan	Apply known tensile loads on the forward bulkhead with and without the stepper motor at the aft bulkhead being powered

Test	**Obstacle Detection and Avoidance**
Objective	To ensure that the rover can properly detect any obstacles in its path and that it can adequately avoid them.
Success criteria	The rover detects and avoids all obstacles in its presence.

Methodology	Place various sized obstacles in different locations around the roverand see how well it detects and avoids them.
Reason for necessity	The rover needs to be able to move around its path without the risk of being blocked or stopped by any obstacle in its way.
Potential outcomes	- Different sensor must be used - Location of the sensor must be changed - Algorithm must be improved
Testing plan	Observe the rover moving in an area with many obstacles present in the area as well.

Test	**Soil Collection**
Objective	To ensure that the rover can collect the required amount of soil sample
Success criteria	The rover collects at least 10 ml of soil in the two wheels combined
Methodology	Test run the rover in various soil environments and examine the performance of the wheels
Reason for necessity	The rover needs to collect 10 ml of soil to be successful in the competition
Potential outcomes	- Valve design must be improved - Wheel must be made wider - Time allocated for digging must be increased - Wheel speed must be increased or decreased - Anchor arm design must be improved
Testing plan	Take the rover to various places with soil and record results

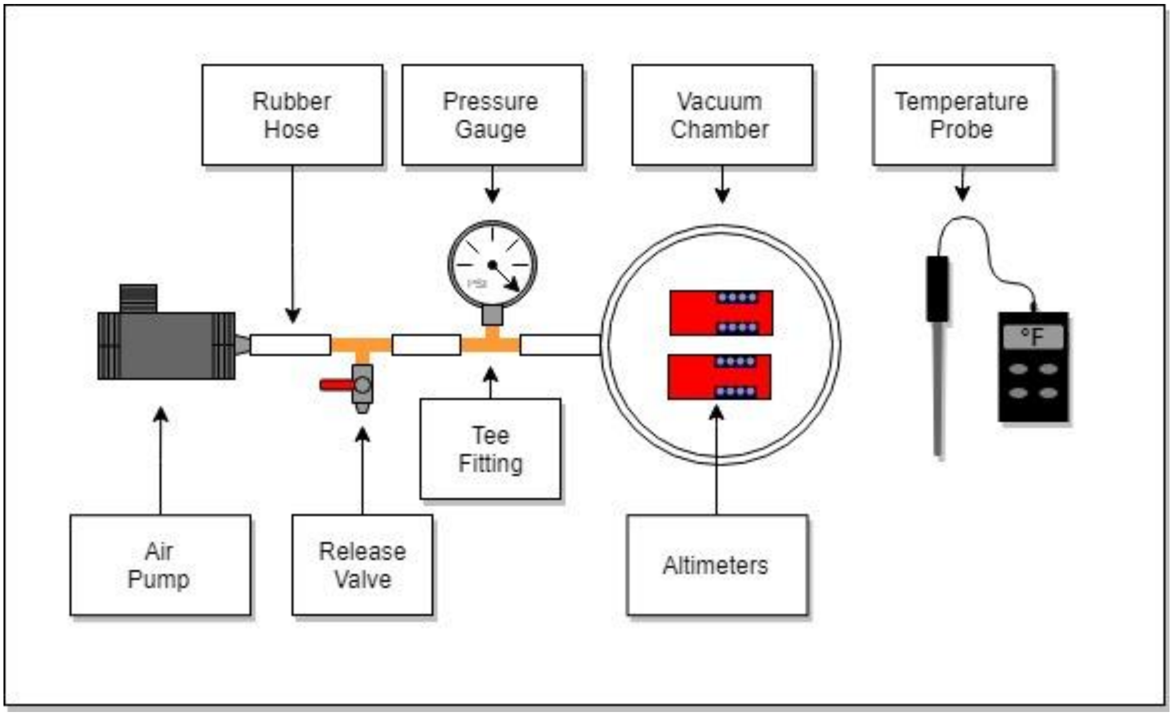


Figure 6.2: A diagram of the test procedure for the altimeters.

6.3 Budgeting and Funding

Our team has procured multiple sources of funding from within our university and is working towards the acquisition of additional funds from other sources. Our first donor is the mechanical engineering department at the Swanson School of Engineering, which has granted us $5,000. The Swanson School of Engineering itself will also be providing a further $5,000 for our team. One of the makerspaces located on campus has also supplied us with some materials that can be used in the production of our rocket. Other opportunities on campus that we are pursuing include fundraisers and a grant provided by the Student Government Board, which is a student run organization that can allocate funds to student groups on campus through an application process. Both of these sources can be used to supplement any travel or manufacturing expenses.

We are also making an effort to contact and establish relationships with local companies in order to secure funding, materials, sponsorship, and mentorship. Pittsburgh features a thriving community of engineering firms and our main focus will be on those that have good ongoing relationships with the faculty and students here at the University of Pittsburgh. Our research into this is being conducted mainly with the University of Pittsburgh Alumni offices, as well as with other clubs regarding what companies are likely to sponsor Pitt engineering teams. Lastly, we have considered collecting dues from team members in order to create an emergency fund that will only be used should a major incident occur. These funds will be transferred into the budget of future teams should it not be necessary for the current team.

Whilst our current funding brings us close to our funding goal, we are anticipating for unexpected costs throughout the process therefore we will continue to establish funding even after our goal has been reached.

Description	Vendor	Cost
PR75-2G-W	Wildman Rocketry	$157.99
StratoLoggerCF (x2)	Perfect Flite	$98.92
Arduino Mega	Arduino	$33.00
GPS Breakout (x2)	Adafruit	$79.90
Antenna SREI038-S9P (x2)	Mouser	$20.22
Antenna Adaptor	Amazon	$9.98
HC-12 Communication unit	Amazon	$21.00
Fire Starters	Apogee Rockets	$12.83
Gyroscope (IMU BNO055)	Adafruit	$34.95
Epoxy	Apogee Rockets	$60.04
Shear Pins	Apogee Rockets	$39.73
Darkstarr Jr Kit	Darkstarr	$125.99
Main Chute (72”)	Rocketman	$235.07
Drogue Chute (18”)	Rocketman	$59.33
Ball Bearings for Subscale	Apogee Rockets	$4.86
Swivels for Subscale	Apogee Rockets	$8.74
Large Ejection Canister Subscale	Apogee Rockets	$18.00
Rivets for Subscale	Apogee Rockets	$3.35
Fiberglass for Subscale	Eplastics	$69.43
Shaft Collars	McMaster-Carr	$15.51
Motor Adapter	Apogee Rockets	$31.01
Ejection Canisters	Apogee Rockets	$18.00
HC-SR04 Ultrasonic Sensor	Amazon	$5.25

Rail Button	Apogee Rockets	$3.35
433Mhz SI4463 Wireless HC-12 Transceiver (x3)	HiLetgo	$23.97
RFM96W LoRa Radio Trver Breakout (x4)	Adafruit	$79.80
Edge-Launch SMA Connector for 1.6mm / 0x062 (x4)	Adafruit	$10.00
uFL SMT Antenna Connector (x4)	Adafruit	$3.00
Perma-Proto Half-sized Breadboard PCB	Adafruit	$12.50
Motor Retainer	AeroPack	$55.56
Thrust Plate		$44.49
NylonX Spool	Matter Hackers	$58.00
Rocket Body	Wildman Darkstarr	$339.00
Hardened Steel Nozzle for Lulzbot TAZ 6 printer	Matter Hackers	$22.50
Kevlar	Wildman Rocketry	$18.75
Subscale Chute Protectors	Wildman Rocketry	$13.90
Subscale Motor Retainer	Wildman Rocketry	$25.00
Subscale Main Chute (30”)	Apogee Rockets	$14.75
Subscale Drogue Chute (18”)	Apogee Rockets	$7.49
HC-12 for Subscale	HiLetgo	$15.98
TOTAL		$1,911.14

Table 12: Vehicle budget.

Payload Budget

Description	Vendor	Cost
DC Gear motor with encoder	DFRobot	$26.80
L298N motor driver	Amazon	$5.99

Arduino Nano clone	Amazon	$13.86
7.4V LiPo Battery	Amazon	$23.98
SG90 Servo Motor	Amazon	$10.99
NEMA 14 Stepper Motor	Amazon	$14.99
A4988 Stepper Driver	Amazon	$5.99
Flexible Coupling	Amazon	$6.96
M8 T-nuts	Amazon	$7.49
M8 Nylon threaded rod	McMaster-Carr	$17.57
⅜ " extruded Nylon rod	Gamut	$8.29
Wheels	On-hand	$0
TOTAL		$142.91

Table 13: Payload budget.

Business and Travel

Description	Cost
Advertisement	$100
Non-Competition Travel	$1000
Competition Travel	$5,000
Lodging	$2,000
Outreach	$500
Emergency Fund	$1000
TOTAL	$9600

Table 14: Business and travel budget.

Total Expenses

Description	Cost
Vehicle	$1911.14
Payload	$142.91
Business and Travel	$9600
TOTAL	$11,654.05

Table 15: Total expenses for competition.

6.4 Timeline

The updated timelines for the subteams responsible for design and fabrication are shown in Figures 6.4a-d. We believe that these are more realistic projections for the design process than what was originally shown in our proposal. So far, all subteams are either on track with their timeline or slightly ahead.

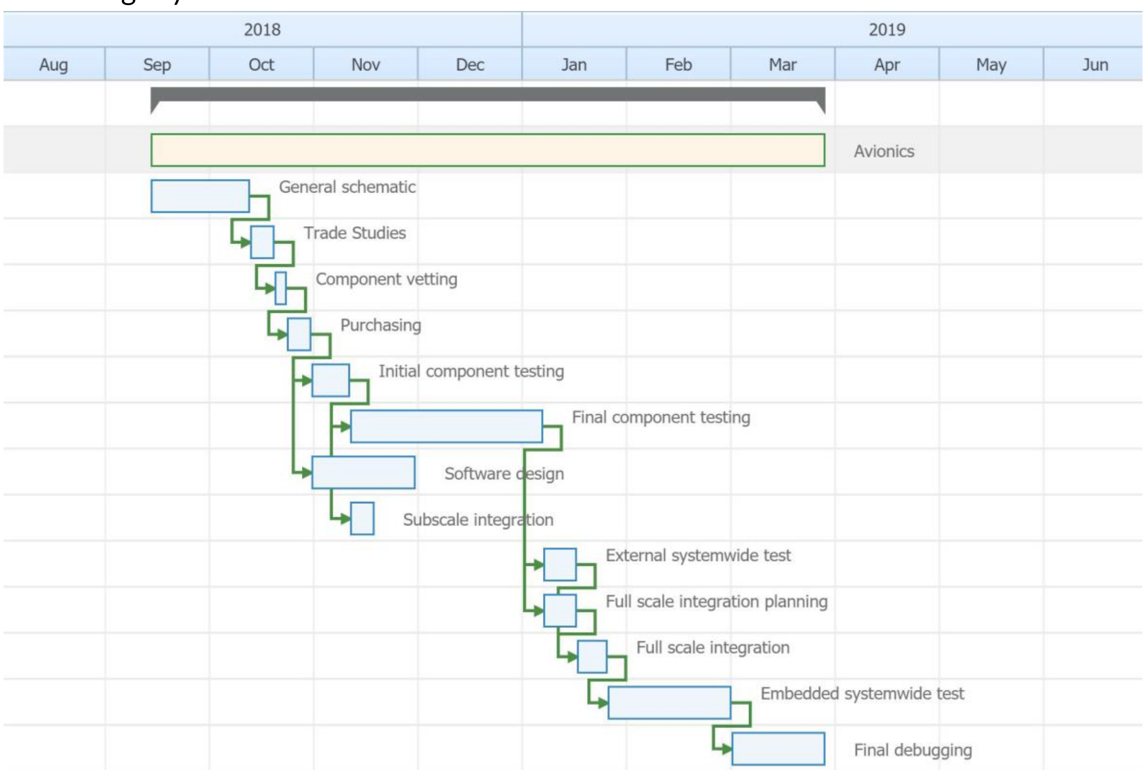


Figure 6.4a: The avionics subteam's timeline.

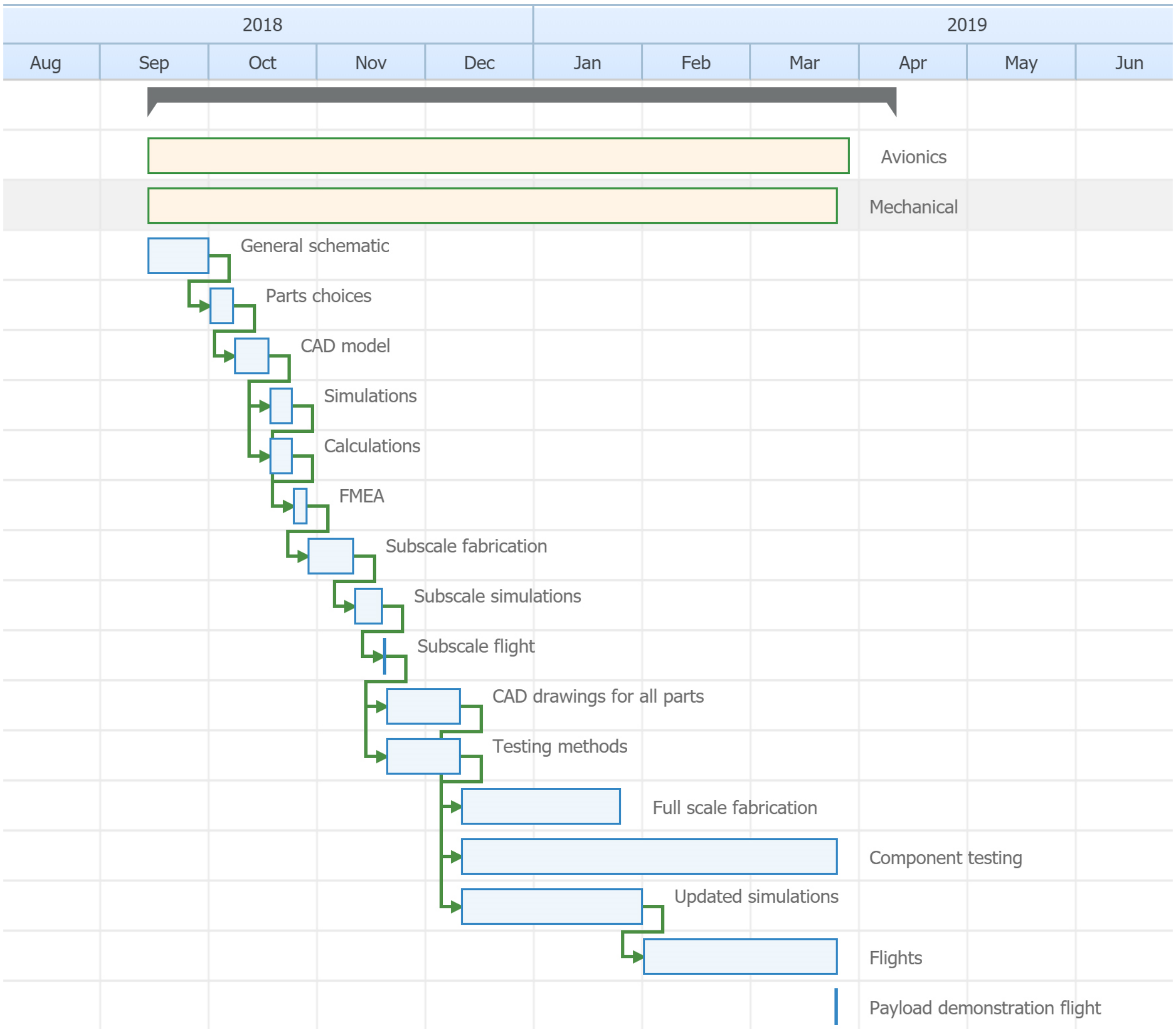


Figure 6.4b: The mechanical subteam's timeline.

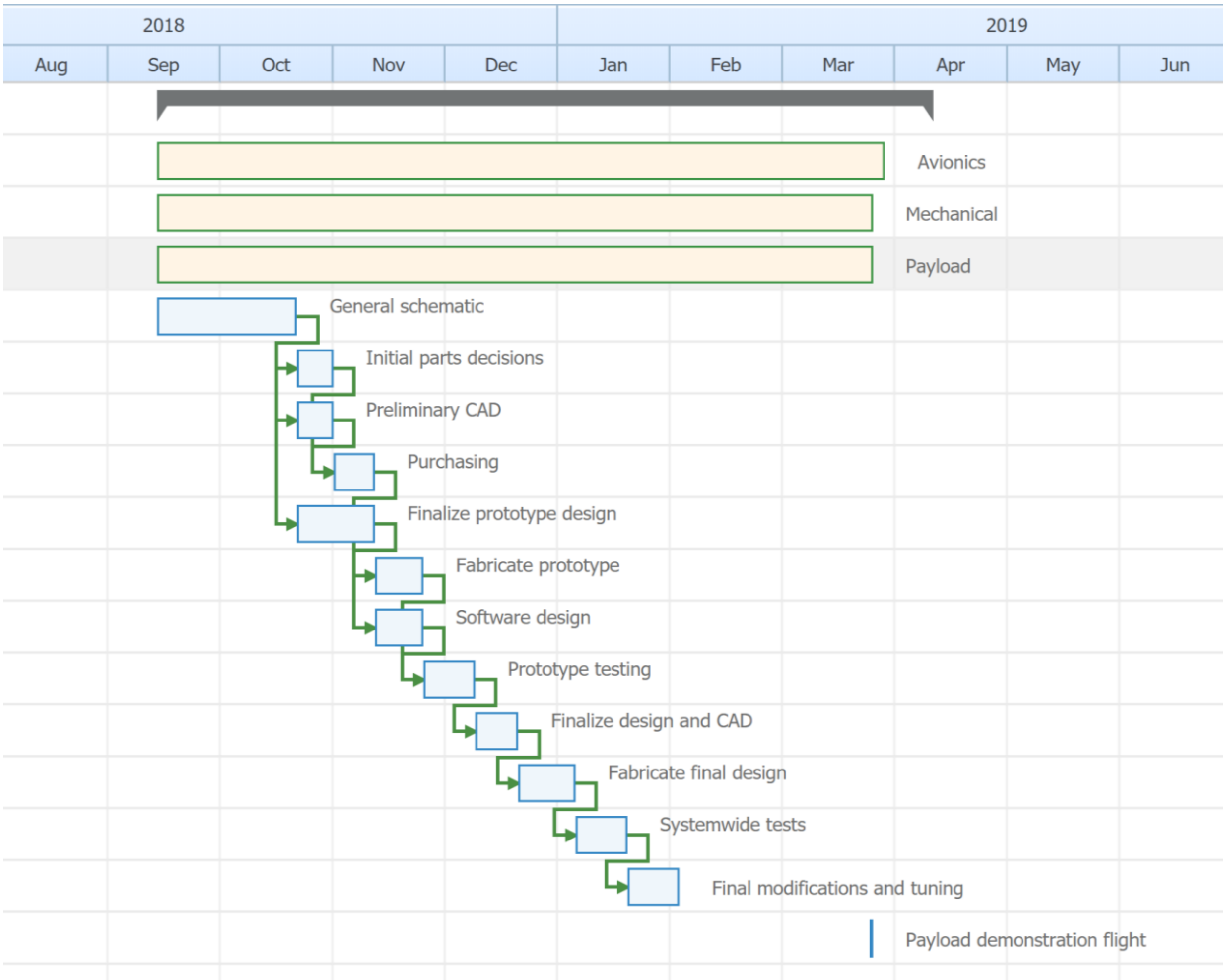


Figure 6.4c: The payload subteam's timeline.

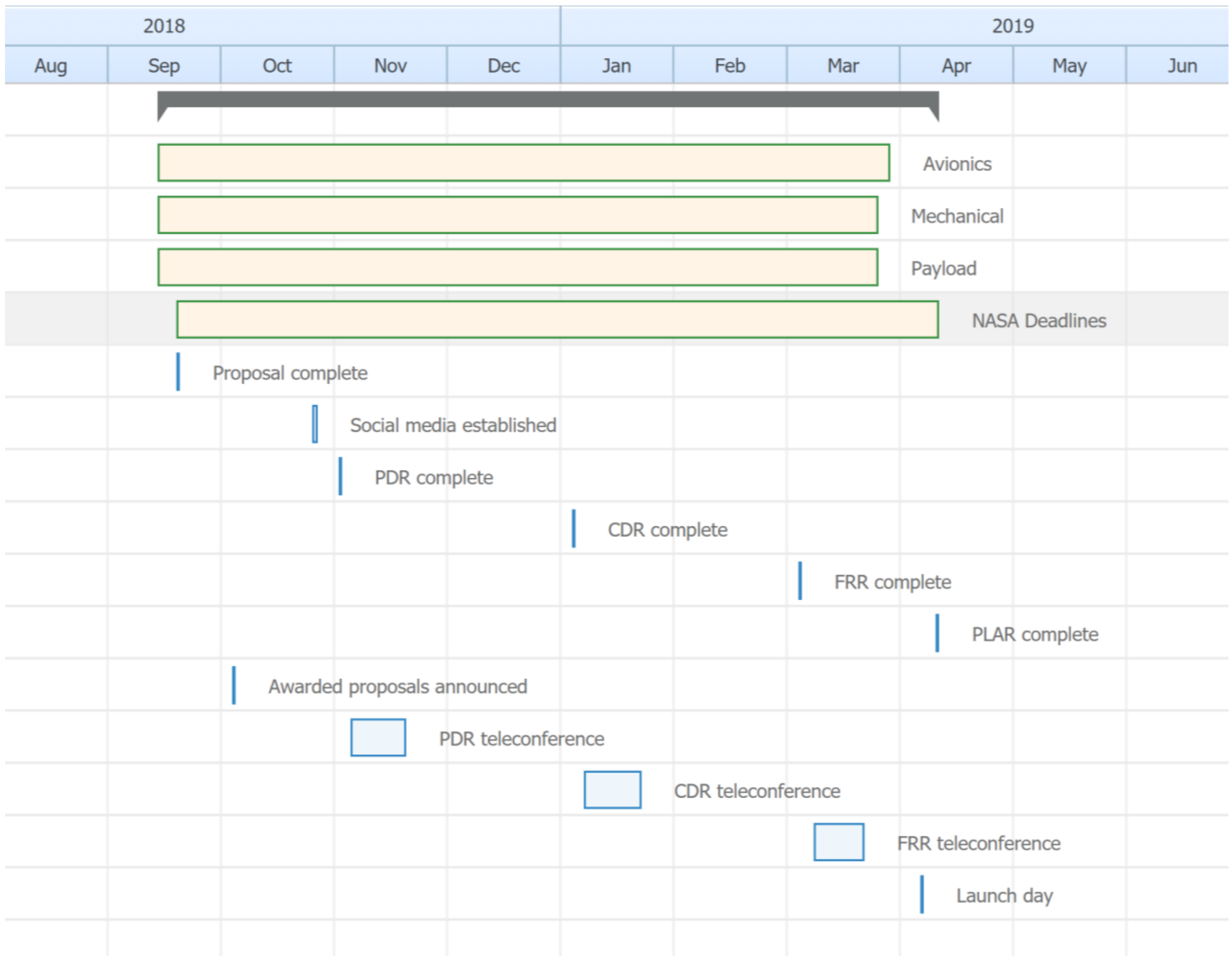


Figure 6.4d: NASA deadlines.

In addition to the general project plans shown by these charts, we were sponsored by Workzone starting the week of November 4th and have been using their schedule planning tool to actively manage these events. We hope to fully integrate it into our planning for future events so that it can replace the current GANTT charts and consolidate our scheduling.

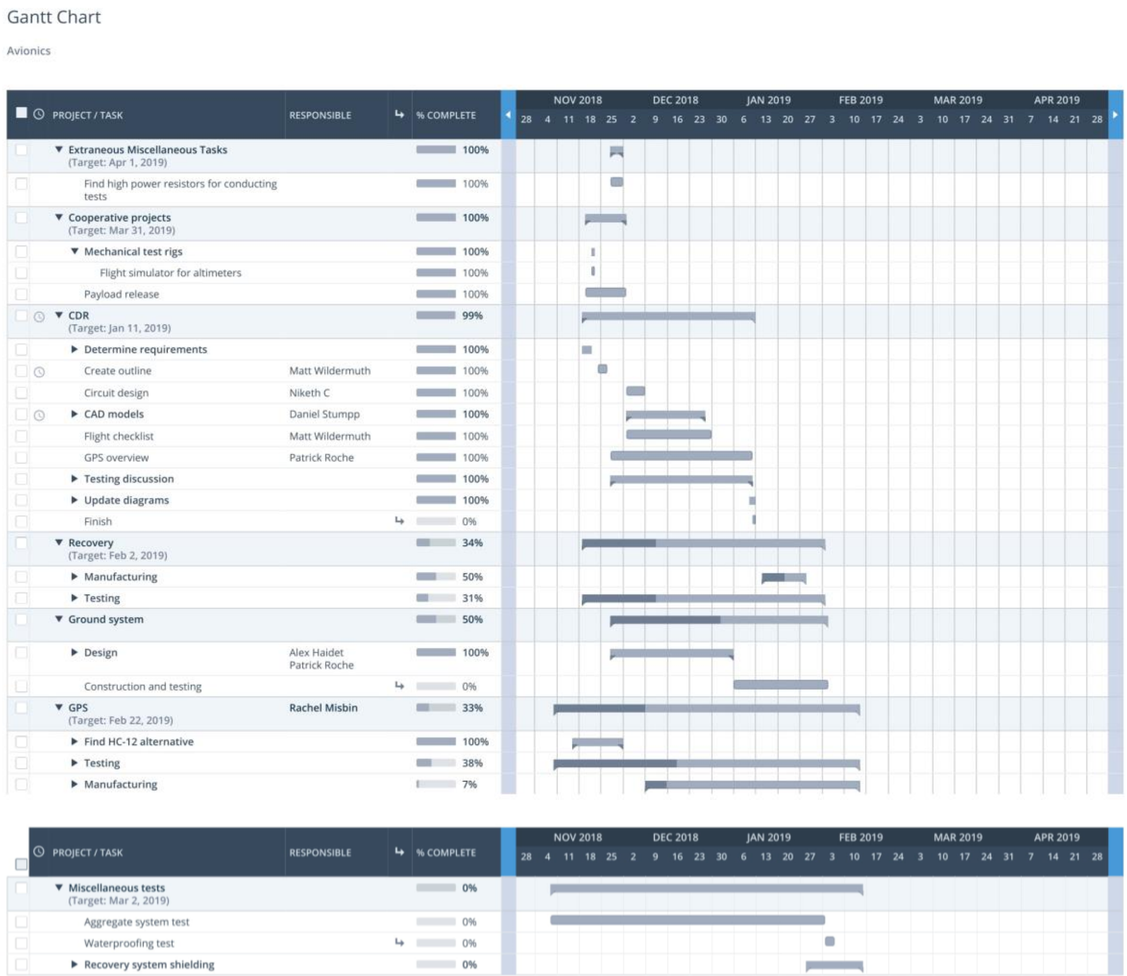


Figure 6.4e: The avionics subteam's Workzone activity.

Figure 6.4f: The mechanical subteam's Workzone activity.

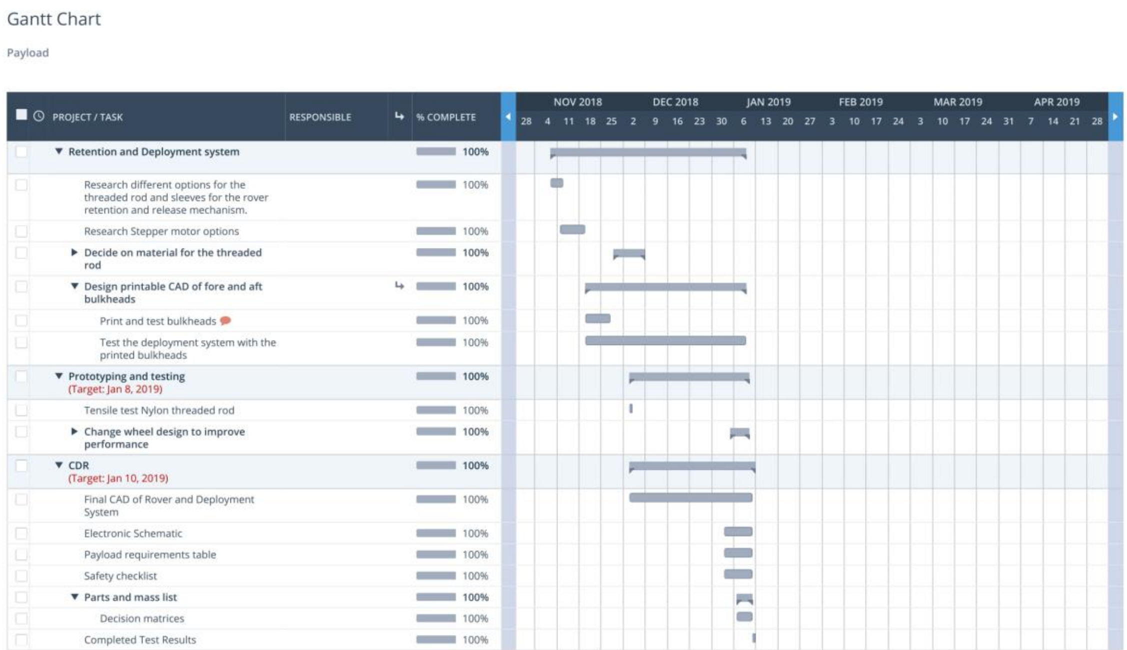


Figure 6.4g: The payload subteam's Workzone activity.

The STEM education outreach timeline has been solidified and dates have been scheduled, as shown in Table 6.4e.

School	Event Date	Expected Attendees
Bethel Park High School	January 30, 2019	120
Norwin Senior High	February 13, 2019	120
New Brighton Elementary	March 21, 2019	100

Table 6.4e: STEM education outreach timeline.

7 Appendix

7.1 References

Glenn Safety Manual – Chapter 1A from NASA - Glenn Research Center https://www.grc.nasa.gov/wp-content/uploads/sites/82/chapter_01a.pdf

7.2 Decision Matrix Criteria

All decision matrices follow the following score classifications unless otherwise stated.

Score (S)	Qualitative descriptor
$S \leq 2$	Very Poor
$2 < S \leq 4$	Poor
$4 < S \leq 6$	Fair
$6 < S \leq 8$	Good
$S > 8$	Excellent

7.3 Subscale Flight Data

Time (s)	Altitude (ft)	Temp	Time (s)	Altitude (ft)	Temp	Time (s)	Altitude (ft)	Temp
0	0	55.4F	1.6	250	55.5F	3.25	978	55.5F
0.05	1	55.4F	1.65	272	55.5F	3.3	999	55.5F
0.1	0	55.4F	1.7	292	55.5F	3.35	1020	55.5F
0.15	1	55.4F	1.75	315	55.5F	3.4	1039	55.5F
0.2	0	55.4F	1.8	336	55.5F	3.45	1058	55.5F
0.25	-1	55.5F	1.85	355	55.5F	3.5	1077	55.5F
0.3	1	55.4F	1.9	379	55.5F	3.55	1098	55.5F
0.35	0	55.5F	1.95	401	55.5F	3.6	1117	55.5F
0.4	2	55.4F	2	426	55.5F	3.65	1136	55.5F
0.45	7	55.4F	2.05	450	55.5F	3.7	1155	55.5F
0.5	10	55.4F	2.1	474	55.5F	3.75	1170	55.5F
0.55	5	55.5F	2.15	494	55.5F	3.8	1187	55.5F
0.6	10	55.5F	2.2	515	55.5F	3.85	1203	55.5F
0.65	15	55.5F	2.25	540	55.5F	3.9	1220	55.5F
0.7	20	55.5F	2.3	564	55.5F	3.95	1236	55.5F
0.75	22	55.5F	2.35	587	55.5F	4	1254	55.5F
0.8	28	55.4F	2.4	610	55.5F	4.05	1273	55.5F
0.85	34	55.5F	2.45	634	55.5F	4.1	1289	55.5F
0.9	44	55.5F	2.5	659	55.5F	4.15	1308	55.5F
0.95	55	55.5F	2.55	682	55.5F	4.2	1323	55.5F
1	66	55.5F	2.6	707	55.5F	4.25	1341	55.5F
1.05	75	55.5F	2.65	730	55.5F	4.3	1359	55.6F
1.1	87	55.5F	2.7	751	55.5F	4.35	1377	55.6F
1.15	100	55.5F	2.75	774	55.5F	4.4	1393	55.6F
1.2	113	55.5F	2.8	794	55.5F	4.45	1411	55.5F
1.25	127	55.5F	2.85	813	55.5F	4.5	1425	55.6F
1.3	144	55.5F	2.9	835	55.5F	4.55	1442	55.6F
1.35	158	55.5F	2.95	855	55.5F	4.6	1460	55.6F
1.4	173	55.5F	3	874	55.5F	4.65	1475	55.6F
1.45	190	55.5F	3.05	897	55.5F	4.7	1494	55.6F

1.5	210	55.5F		3.1	918	55.5F		4.75	1509	55.6F
1.55	230	55.5F		3.15	937	55.5F		4.8	1524	55.6F

Time (s)	**Altitude (ft)**	**Temp**		**Time (s)**	**Altitude (ft)**	**Temp**		**Time (s)**	**Altitude (ft)**	**Temp**
4.9	1555	55.6F		6.55	1979	55.6F		8.2	2289	55.7F
4.95	1568	55.6F		6.6	1988	55.6F		8.25	2297	55.7F
5	1582	55.6F		6.65	1997	55.7F		8.3	2305	55.7F
5.05	1594	55.6F		6.7	2008	55.7F		8.35	2312	55.7F
5.1	1607	55.6F		6.75	2018	55.7F		8.4	2319	55.7F
5.15	1620	55.6F		6.8	2027	55.7F		8.45	2326	55.7F
5.2	1634	55.6F		6.85	2038	55.7F		8.5	2332	55.7F
5.25	1648	55.6F		6.9	2049	55.7F		8.55	2339	55.7F
5.3	1664	55.6F		6.95	2058	55.7F		8.6	2345	55.7F
5.35	1676	55.6F		7	2069	55.7F		8.65	2350	55.7F
5.4	1690	55.6F		7.05	2079	55.7F		8.7	2358	55.7F
5.45	1704	55.6F		7.1	2089	55.7F		8.75	2362	55.7F
5.5	1719	55.6F		7.15	2099	55.7F		8.8	2370	55.7F
5.55	1731	55.6F		7.2	2109	55.7F		8.85	2374	55.7F
5.6	1745	55.6F		7.25	2119	55.7F		8.9	2380	55.7F
5.65	1759	55.6F		7.3	2129	55.7F		8.95	2387	55.7F
5.7	1772	55.6F		7.35	2138	55.7F		9	2394	55.7F
5.75	1784	55.6F		7.4	2148	55.7F		9.05	2398	55.7F
5.8	1798	55.6F		7.45	2158	55.7F		9.1	2404	55.7F
5.85	1811	55.6F		7.5	2167	55.7F		9.15	2410	55.7F
5.9	1824	55.6F		7.55	2175	55.7F		9.2	2417	55.7F
5.95	1837	55.6F		7.6	2185	55.7F		9.25	2422	55.7F
6	1849	55.6F		7.65	2193	55.7F		9.3	2428	55.7F
6.05	1863	55.6F		7.7	2203	55.7F		9.35	2433	55.7F
6.1	1875	55.6F		7.75	2211	55.7F		9.4	2441	55.7F
6.15	1888	55.6F		7.8	2220	55.7F		9.45	2445	55.7F
6.2	1899	55.6F		7.85	2230	55.7F		9.5	2451	55.7F
6.25	1913	55.6F		7.9	2238	55.7F		9.55	2456	55.7F

6.3	1925	55.6F		7.95	2246	55.7F		9.6	2462	55.7F
6.35	1938	55.6F		8	2256	55.7F		9.65	2467	55.7F
6.4	1947	55.6F		8.05	2263	55.7F		9.7	2472	55.7F
6.45	1957	55.6F		8.1	2273	55.7F		9.75	2477	55.7F

Time (s)	Altitude (ft)	Temp		Time (s)	Altitude (ft)	Temp		Time (s)	Altitude (ft)	Temp
9.85	2487	55.7F		11.5	2601	55.7F		13.15	2614	55.8F
9.9	2491	55.7F		11.55	2604	55.7F		13.2	2613	55.8F
9.95	2495	55.7F		11.6	2607	55.7F		13.25	2614	55.8F
10	2500	55.7F		11.65	2607	55.7F		13.3	2611	55.8F
10.05	2505	55.7F		11.7	2609	55.7F		13.35	2611	55.8F
10.1	2510	55.7F		11.75	2610	55.7F		13.4	2612	55.8F
10.15	2515	55.7F		11.8	2613	55.7F		13.45	2610	55.8F
10.2	2518	55.7F		11.85	2612	55.7F		13.5	2608	55.8F
10.25	2522	55.7F		11.9	2613	55.7F		13.55	2607	55.8F
10.3	2526	55.7F		11.95	2616	55.7F		13.6	2606	55.8F
10.35	2530	55.7F		12	2617	55.7F		13.65	2607	55.8F
10.4	2534	55.7F		12.05	2617	55.7F		13.7	2604	55.8F
10.45	2536	55.7F		12.1	2617	55.7F		13.75	2602	55.8F
10.5	2540	55.7F		12.15	2618	55.7F		13.8	2601	55.8F
10.55	2544	55.7F		12.2	2618	55.7F		13.85	2599	55.8F
10.6	2547	55.7F		12.25	2618	55.7F		13.9	2598	55.8F
10.65	2552	55.7F		12.3	2618	55.7F		13.95	2595	55.8F
10.7	2556	55.7F		12.35	2618	55.8F		14	2594	55.8F
10.75	2560	55.7F		12.4	2620	55.7F		14.05	2593	55.8F
10.8	2563	55.7F		12.45	2619	55.8F		14.1	2590	55.8F
10.85	2565	55.7F		12.5	2618	55.7F		14.15	2586	55.8F
10.9	2569	55.7F		12.55	2618	55.8F		14.2	2586	55.8F
10.95	2572	55.7F		12.6	2620	55.8F		14.25	2585	55.8F
11	2575	55.7F		12.65	2620	55.7F		14.3	2582	55.8F
11.05	2579	55.7F		12.7	2619	55.8F		14.35	2580	55.8F
11.1	2581	55.7F		12.75	2617	55.8F		14.4	2576	55.8F

11.15	2585	55.7F		12.8	2617	55.8F		14.45	2575	55.8F
11.2	2588	55.7F		12.85	2619	55.8F		14.5	2572	55.8F
11.25	2589	55.7F		12.9	2617	55.8F		14.55	2569	55.8F
11.3	2593	55.7F		12.95	2617	55.8F		14.6	2567	55.8F
11.35	2595	55.7F		13	2617	55.8F		14.65	2564	55.8F
11.4	2596	55.7F		13.05	2616	55.8F		14.7	2561	55.8F

Time (s)	**Altitude (ft)**	**Temp**		**Time (s)**	**Altitude (ft)**	**Temp**		**Time (s)**	**Altitude (ft)**	**Temp**
14.8	2555	55.8F		16.45	2421	55.8F		18.1	2188	55.9F
14.85	2552	55.8F		16.5	2415	55.8F		18.15	2181	55.9F
14.9	2548	55.8F		16.55	2410	55.8F		18.2	2173	55.9F
14.95	2544	55.8F		16.6	2406	55.8F		18.25	2163	55.8F
15	2541	55.8F		16.65	2399	55.8F		18.3	2157	55.9F
15.05	2537	55.8F		16.7	2395	55.8F		18.35	2150	55.8F
15.1	2533	55.8F		16.75	2387	55.8F		18.4	2145	55.8F
15.15	2529	55.8F		16.8	2380	55.8F		18.45	2138	55.8F
15.2	2526	55.8F		16.85	2373	55.8F		18.5	2127	55.9F
15.25	2522	55.8F		16.9	2366	55.8F		18.55	2118	55.8F
15.3	2519	55.8F		16.95	2359	55.8F		18.6	2112	55.8F
15.35	2513	55.8F		17	2353	55.8F		18.65	2103	55.9F
15.4	2511	55.8F		17.05	2345	55.8F		18.7	2097	55.9F
15.45	2507	55.8F		17.1	2337	55.8F		18.75	2083	55.9F
15.5	2502	55.8F		17.15	2329	55.8F		18.8	2075	55.9F
15.55	2499	55.8F		17.2	2322	55.8F		18.85	2065	55.9F
15.6	2494	55.8F		17.25	2315	55.8F		18.9	2058	55.9F
15.65	2490	55.8F		17.3	2308	55.8F		18.95	2052	55.9F
15.7	2486	55.8F		17.35	2299	55.8F		19	2041	55.9F
15.75	2482	55.8F		17.4	2290	55.8F		19.05	2034	55.9F
15.8	2479	55.8F		17.45	2283	55.8F		19.1	2028	55.9F
15.85	2475	55.8F		17.5	2277	55.9F		19.15	2021	55.9F
15.9	2472	55.8F		17.55	2270	55.8F		19.2	2013	55.9F
15.95	2469	55.8F		17.6	2258	55.9F		19.25	2003	55.9F

16	2464	55.8F		17.65	2256	55.8F		19.3	1992	55.8F
16.05	2461	55.8F		17.7	2246	55.8F		19.35	1986	55.9F
16.1	2454	55.8F		17.75	2237	55.8F		19.4	1976	55.8F
16.15	2451	55.8F		17.8	2232	55.8F		19.45	1968	55.9F
16.2	2444	55.8F		17.85	2226	55.8F		19.5	1961	55.9F
16.25	2442	55.8F		17.9	2219	55.9F		19.55	1944	55.9F
16.3	2439	55.8F		17.95	2208	55.8F		19.6	1939	55.9F
16.35	2432	55.8F		18	2202	55.9F		19.65	1920	55.9F

Time (s)	Altitude (ft)	Temp		Time (s)	Altitude (ft)	Temp		Time (s)	Altitude (ft)	Temp
19.75	1902	55.9F		21.4	1551	55.9F		23.05	1136	55.9F
19.8	1890	55.9F		21.45	1542	55.9F		23.1	1125	55.9F
19.85	1879	55.9F		21.5	1527	55.9F		23.15	1117	55.9F
19.9	1868	55.9F		21.55	1514	55.9F		23.2	1108	55.9F
19.95	1857	55.9F		21.6	1505	55.9F		23.25	1089	55.9F
20	1847	55.9F		21.65	1493	55.9F		23.3	1078	55.9F
20.05	1835	55.9F		21.7	1478	55.9F		23.35	1061	55.9F
20.1	1826	55.9F		21.75	1466	55.9F		23.4	1052	55.9F
20.15	1815	55.9F		21.8	1447	55.9F		23.45	1041	55.9F
20.2	1800	55.9F		21.85	1435	55.9F		23.5	1020	55.9F
20.25	1787	55.9F		21.9	1425	55.9F		23.55	1007	55.9F
20.3	1778	55.9F		21.95	1411	55.9F		23.6	981	55.9F
20.35	1770	55.9F		22	1398	55.9F		23.65	972	55.9F
20.4	1759	55.9F		22.05	1383	55.9F		23.7	959	55.9F
20.45	1751	55.9F		22.1	1374	55.9F		23.75	944	55.9F
20.5	1741	55.9F		22.15	1361	55.9F		23.8	934	55.9F
20.55	1732	55.9F		22.2	1350	55.9F		23.85	918	55.9F
20.6	1725	55.9F		22.25	1335	55.9F		23.9	906	55.9F
20.65	1715	55.9F		22.3	1324	55.9F		23.95	893	55.9F
20.7	1703	55.9F		22.35	1307	55.9F		24	881	55.9F
20.75	1691	55.9F		22.4	1300	55.9F		24.05	867	55.9F
20.8	1685	55.9F		22.45	1287	55.9F		24.1	853	55.9F

20.85	1671	55.9F	22.5	1277	55.9F	24.15	837	55.9F
20.9	1660	55.9F	22.55	1265	55.9F	24.2	825	55.9F
20.95	1649	55.9F	22.6	1259	55.9F	24.25	813	55.9F
21	1633	55.9F	22.65	1245	55.9F	24.3	797	55.9F
21.05	1625	55.9F	22.7	1232	55.9F	24.35	783	55.9F
21.1	1612	55.9F	22.75	1218	55.9F	24.4	767	55.9F
21.15	1605	55.9F	22.8	1201	55.9F	24.45	755	55.9F
21.2	1594	55.9F	22.85	1188	55.9F	24.5	737	55.9F
21.25	1583	55.9F	22.9	1178	55.9F	24.55	729	55.9F
21.3	1571	55.9F	22.95	1169	55.9F	24.6	708	55.9F

Time (s)	**Altitude (ft)**	**Temp**	**Time (s)**	**Altitude (ft)**	**Temp**	**Time (s)**	**Altitude (ft)**	**Temp**
24.7	691	55.9F	26.35	230	56.4F	28	2	56.7F
24.75	390	55.9F	26.4	216	56.4F	28.05	3	56.6F
24.8	627	55.9F	26.45	199	56.4F	28.1	3	56.6F
24.85	648	55.9F	26.5	184	56.4F	28.15	3	56.6F
24.9	588	55.9F	26.55	181	56.4F	28.2	3	56.7F
24.95	604	56.0F	26.6	183	56.4F	28.25	3	56.7F
25	616	56.0F	26.65	159	56.4F	28.3	2	56.7F
25.05	635	56.0F	26.7	149	56.4F	28.35	4	56.7F
25.1	560	56.0F	26.75	125	56.4F	28.4	3	56.7F
25.15	534	56.1F	26.8	112	56.4F	28.45	3	56.7F
25.2	528	56.1F	26.85	115	56.4F	28.5	3	56.7F
25.25	541	56.1F	26.9	134	56.5F	28.55	3	56.7F
25.3	523	56.1F	26.95	83	56.5F	28.6	4	56.7F
25.35	494	56.1F	27	73	56.5F	28.65	3	56.7F
25.4	471	56.1F	27.05	46	56.5F	28.7	2	56.7F
25.45	470	56.2F	27.1	30	56.5F	28.75	3	56.7F
25.5	459	56.2F	27.15	32	56.5F	28.8	3	56.7F
25.55	461	56.2F	27.2	46	56.5F	28.85	2	56.7F
25.6	428	56.2F	27.25	3	56.5F	28.9	6	56.7F
25.65	392	56.3F	27.3	3	56.5F	28.95	16	56.6F

25.7	373	56.3F		27.35	-1441	56.5F		29	24	56.6F
25.75	365	56.3F		27.4	-1024	56.6F		29.05	-1	32.0F
25.8	367	56.3F		27.45	-318	56.6F				
25.85	360	56.3F		27.5	-92	56.6F				
25.9	343	56.3F		27.55	-10	56.6F				
25.95	321	56.3F		27.6	2	56.6F				
26	306	56.3F		27.65	2	56.6F				
26.05	300	56.4F		27.7	1	56.6F				
26.1	287	56.4F		27.75	1	56.6F				
26.15	273	56.4F		27.8	2	56.6F				
26.2	262	56.4F		27.85	3	56.6F				
26.25	253	56.4F		27.9	4	56.6F				