Preliminary Design Review

NASA University Student Launch Initiative

November 2nd, 2020

Worcester Polytechnic Institute

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Worcester, MA 01609

Contents

1 Figures

2 Tables

3 Acronyms

1. 3D: Three Dimensional

2. AARD: Advanced Retention Release Device
3. ABS: Acrylonitrile Butadiene Styrene (FDM Filament)
4. AGL: Above Ground Level
5. AIAA: American Institution of Aeronautics and Astronautics
6. APCP: Ammonium Perchlorate (Composite Solid Fuel)
7. BRC: Bridgeton Area Rocket Club
8. COVID-19: Coronavirus Disease 2019
9. CMASS: Central Massachusetts Space Modeling Society
10. CNC: Computer Numerical Control
11. CRMRC: Champlain Region Model Rocket Club
12. CTI: Cesaroni Technology Incorporated
13. DOF: Degrees of Freedom
14. EBI: Ensign-Bickford Industries, Inc.
15. E-Match: Electric Match
16. EnP: Electronics and Programming
17. FAA: Federal Aviation Administration
18. FDM: Fused Deposition Modeling (3D Printing Technology)
19. GPS: Global Positioning System
20. HPR: High Power Rocketry
21. HPRC: High Power Rocketry Club
22. IDE: Integrated Development Environment (For Software Development)
23. IMU: Inertial Measurement Unit
24. LED: Light Emitting Diode
25. LiPo: Lithium Polymer (Battery)
26. LoRa: Long Range (Wireless Protocol)
27. LWHPR: Lake Winnipesaukee High Power Rocketry
28. MQP: Major Qualifying Project (Senior Project)
29. MSFC: Martial Space Flight Center
30. NAR: National Association of Rocketry
31. NASA: National Aeronautics and Space Administration
32. NFPA: National Fire Protection Association
33. PC: Polycarbonate (FDM Filament)
34. PLA: Polylactic Acid (FDM Filament)
35. PLAR: Post Launch Assessment Review

36. PPE: Personal Protective Equipment
37. PWM: Pulse Width Modulation
38. RSO: Range Safety Officer
39. SGA: Student Government Association
40. SLI: Student Launch Initiative
41. STEM: Science, Technology, Engineering, and Mathematics
42. STM: ST Microelectronics
43. TPU: Thermoplastic Polyurethane
44. TRA: Tripoli Rocketry Association
45. UAV: Unmanned Arial Vehicle
46. URRG: Upstate Research Rocketry Group
47. USLI: University Student Launch Initiative
48. WPI: Worcester Polytechnic Institute

4 Summary of PDR

4.1 Team Summary

4.1.1 Team Information

Worcester Polytechnic Institute High Power Rocketry Club (WPI HPRC)
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4.1.2 Team Mentor

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4.1.3 Hours Spent on PDR

WPI HPRC has spent a total of around 1980 hours collectively on the PDR milestone. This includes all meetings, individual member work as well as time spent directly on the PDR documentation and presentation. Assuming the average member contributes 5 hours per week, the average officer 10 hours per week, and the executives contributing 20 hours/week, we were able to estimate this amount of time. The calculated number was obtained by assuming there were forty general members, seven officers and three executives each working for the six weeks between the proposal and PDR submissions.

4.2 Launch Vehicle Summary

WPI's launch vehicle will have an outer diameter of 6.17 in, a length of 108 in, and a total wet mass of 49 lb. The vehicle will utilize a Cesaroni Technologies Incorporated (CTI) L1395 as its primary motor, with the CTI L2375 serving as a backup motor. The launch vehicle is expected to reach an apogee of 4550 ft, which will serve as our official target altitude. The recovery system will consist of a dual bay, dual deployment recovery layout, ejecting a 32" drogue parachute at apogee, and a 120" main parachute at 600 ft during descent. The ejection charges and altimeters will be redundant, with delays of 2 seconds between primary and backup ejection charge.

4.3 Payload Summary

WPI's payload will be ejected from the airframe at apogee and remain tethered to the launch vehicle. At 1000 feet, it will detach descending under its own parachute which will be released upon landing. After, the payload will self-right and level itself to within the five-degree tolerance. After these processes are complete, the payload will take a panoramic photo to be transmitted back to the ground station.

5 Changes Since Proposal

5.1 Rocket Criteria

The launch vehicle has shortened slightly to 108 in, with a small change in wet mass from 49.2 to 49 lb. The fins have also had dimensions adjusted, as described in Section 6.3.3.

Due to changes in the payload design, the recovery system design was altered significantly, from a single bay dual deployment system to a more standard dual bay dual deployment system. With this, a new subsystem - the recovery bay - was added between the middle and upper airframe section, which will house the recovery electronics. After analysis of the descent parameters of the vehicle, the drogue and main parachutes have decreased in size to 32 in and 120 in respectively, though both will be purchased from the same manufacturers as listed in proposal.

The fin can design has replaced large, single part rings with modular parts that are easier to manufacture, including custom brackets for attaching to the fins and to the airframe. The 3D printed center support for the fins has been removed, in favor of updating the fin core material to plywood.

The airbrake system has been iterated upon, with an updated constraining system for the fins, and the removal of an additional unnecessary plate from the design by inverting its orientation in the vehicle.

The avionics system has replaced the planned GPS module, a u-blox NEO-M8M with a newer version, the NEO-M9N, and has added a magnetometer to the onboard sensors.

5.2 Payload criteria

The payload self-righting and stabilization systems are using new actuators, with the self-righting system using an entirely different driving scheme. The stabilization system now uses a different foot design as well as different materials.

The payload camera system has changed to now use a singular 360-degree camera. The photography sub team has now become the photography and retention subteam. This sub team's responsibilities will now include payload retention and how said design will be integrated withing the launch vehicle.

5.3 Project plan

At the time of submission for Proposal, the team did not know if they would be able to launch their subscale rocket due to WPI's strict travel policy during the COVID-19 pandemic. Prior to the submission of proposal, the officer board appealed to WPI and recently have been granted permission for two students to attend the last CMASS launch of the season on November 21st. The members who will be attending the launch will be following safety precautions including, but not limited to riding in separate vehicles, always wearing masks, sanitizing everything that is touched by both members, and maintaining 6 feet apart from others and each other as much as possible. The members going will also adhere to the CMASS COVID-19 guidelines put in place by the CMASS club.

This has resulted in an accelerated construction schedule due to the earlier than anticipated launch date. This shift in schedule can be seen in the Gantt Chart in Section 9.5. Although this subscale launch is earlier than expected, this is likely the only opportunity the team will get to launch their subscale rocket. WPI's decision to allow HPRC to launch can be revoked at any moment based on how well the surrounding area, including WPI itself, is handling the COVID-19 pandemic. It is unlikely that the team will be able to find another allowable method of launching the subscale and appeal to WPI before the CDR deadline and therefore then be disqualified from the competition.

6 Rocket Design

6.1 Launch Vehicle Summary

WPI's launch vehicle consists of 6 major sections, each with uniquely defined tasks and requirements. The lower airframe contains the fin can and motor retention system, responsible for securing the fins and motor during flight. The avionics bay, between the lower and middle airframes, houses the avionics system and the airbrakes, the former of which will collect transmit, and analyze data used to control the airbrakes, which will actively control the vehicle's apogee in flight. The middle airframe will contain the main parachute, and the recovery bay, situated between the middle and upper airframes, will contain the electronics and recovery hardware necessary for parachutes to be deployed. The upper airframe will contain the drogue parachute, as well as the payload, and will attach to the nosecone.

WPI's launch vehicle's Cg and Cp will be located 64.9 in and 83.8 in aft of the nosecone tip, respectively. The vehicle will land in 3 independent sections, excluding the payload vehicle, with a maximum KE at landing of 64 ft-lbf, and a maximum descent time of 86.1 seconds.

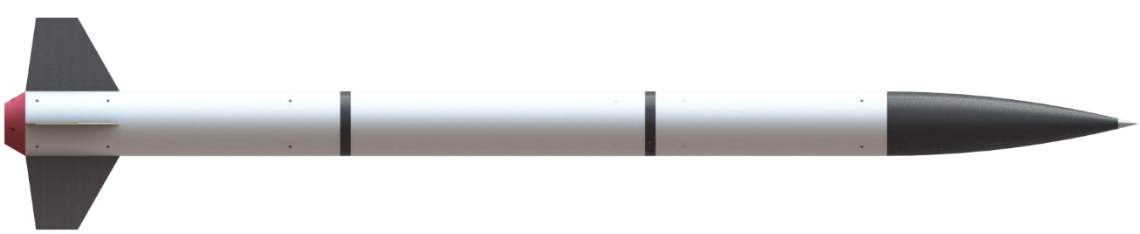


Figure 6.1 Launch Vehicle Side View

Figure 6.2 Launch Vehicle Section View

6.2 Mission Performance Predictions

6.2.1 Target Altitude

WPI HPRC's target apogee is set to be 4550 ft. For a flight, the expected unguided apogee is designed to overshoot this target, and our actual apogee will be reduced through the use of the airbrake system described in Section 6.6.1. A detailed analysis of rocket mass, aerodynamics, and the effects of the airbrake system was undertaken to arrive at this target apogee.

6.2.2 Flight Profile Simulations

The unguided ascent of the rocket was simulated using OpenRocket using multiple simulation setups with differing wind speeds and launch rod angles to determine an acceptable range of apogees from which the airbrakes could adjust to reach the target apogee. The major components of the launch vehicle were designed and evaluated in OpenRocket, and more detailed subsystems were designed and evaluated using SOLIDWORKS.

Using both these tools, a mass budget was created. In this budget, parts were assigned a mass, quantity, and mass margin, which describes the additional percentage of the part's mass added to account for possible future design changes. For student developed parts, the mass margin is 10% with parts involving composites and epoxy layup having mass margin of 20%. Purchased components have a mass margin of 0%, as their mass is a known quantity. In cases where specific geometry or parts were still unknown, a best guess was used with a more substantial 30% mass margin. The full mass budget is available in Appendix 10.2, with a breakdown of section totals available in Table 6.1. This table contains the wet (on pad) mass of the sections; depending on the stage of flight, sections may have different masses, which will be noted as they occur.

Section	Mass (lb)
Lower Airframe	18.19
Middle Airframe	6.38
Upper Airframe	14.61
Avionics Bay	4.21
Recovery Bay	3.76
Airbrakes	1.49
Total	48.64

Table 6.1 Wet Mass Budget Totals

As described, multiple simulations with varying conditions were run in OpenRocket to create a range of possible apogees. The parameters adjusted were the wind speed and the launch rod angle, with the values for each simulation, as well as the calculate apogees, shown in Table 6.2.

Simulation	Wind Speed (mph)	Launch Rod Angle (deg)	Apogee (ft)
Best Case	3	5	5079
Standard Case	8	7.5	4912
Worst Case	20	10	4404

Table 6.2 OpenRocket Simulation Parameters and Results

Parameters were determined based on NASA's requirements for launch rod angle, and a qualitative analysis of historical wind data at our expected launch site, St. Albans VT, in April [1]. As can be seen, there is a significant difference between the expected apogees of the best and standard cases, which are both relatively near to 5000 ft, and the worst-case scenario which is near to 4400 ft. While such a large difference is concerning for estimating an apogee, the weather conditions of the worst-case scenario are quite unlikely, and if they were to occur it is likely the wind itself or a related weather condition would prevent launches from occurring safely. For this reason, the apogee of our standard case, 4912 ft, will be used as a starting estimate for our eventual apogee target. From here, we determine what effect the airbrakes can have on the vehicle's flight.

OpenRocket outputs the total drag coefficient of the launch vehicle for the case with no airbrakes. For the airbrakes, we must determine the additional drag coefficient for the case of the airbrakes being fully extended. The airbrakes will be activated at motor burnout, so by finding the drag coefficient using an aerodynamic model and SOLIDWORKS Flow Simulation with and without airbrakes deployed, we can find the difference between the two cases to determine the additional drag coefficient provided by the fully extended airbrakes. The CFD analysis outputs the force on the body, so to find the drag coefficient for each case we must use Equation 1.

$$C_d = \frac{D}{\frac{1}{2}\rho v^2 A_{ref}}$$

Equation 1 Drag Coefficient

Conveniently, OpenRocket outputs many of the parameters needed for this equation. The parameters used in the simulation are found in Table 6.3, and are taken as the average of the values between motor burnout and apogee

Parameter	Value
Pressure	87000 Pa
Temperature	282.25 K
Velocity	285 ft/s
Reference Area	29.9 in^2

Table 6.3 CFD Simulation Parameters

The results of the CFD simulations are described in Table 6.4.

Simulation	Drag Force (N)	Drag Coefficient
Zero Extension	33.85	0.434
Fully Extended	50.84	0.653

Table 6.4 CFD Results

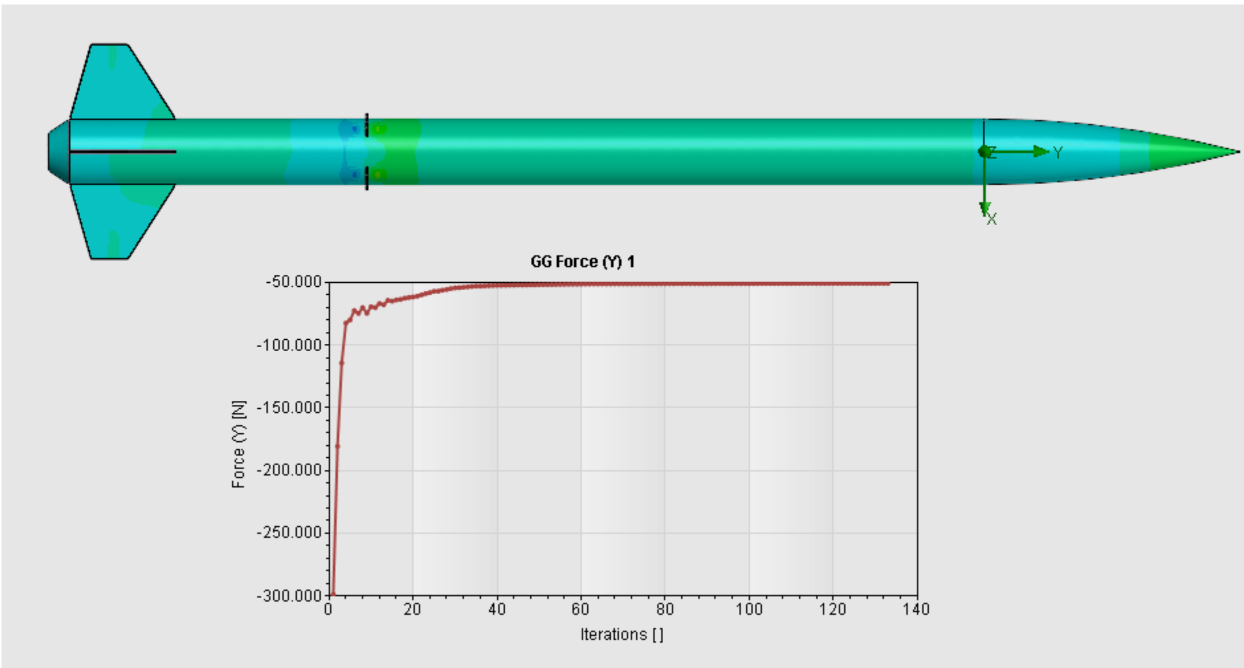


Figure 6.3 CFD Goal Convergence Plot

From this data we can calculate the added drag coefficient provided by the fully extended airbrakes to be 0.218 for a total vehicle drag coefficient of 0.688 from a zero extension drag coefficient of .47. With this data, we can use a modified version of the team's 3 DOF parachute descent simulator, described further in Section 6.2.4, to estimate the new apogee. Our initial conditions for the simulation become those at burnout, as determined by OpenRocket. The maximum altitude reached with the zero extension drag coefficient is 5032 ft, with the fully extended apogee being 4630 ft. The results of these calculations indicate the airbrake system is able to reduce the apogee by roughly 400 ft.

Based on the expected apogee from OpenRocket, and the expected maximum apogee decrease due to the airbrake system, our target apogee is set to 4550 ft, allowing margin for any additional mass that is added to the rocket, as well as sources of parasitic drag that OpenRocket cannot calculate.

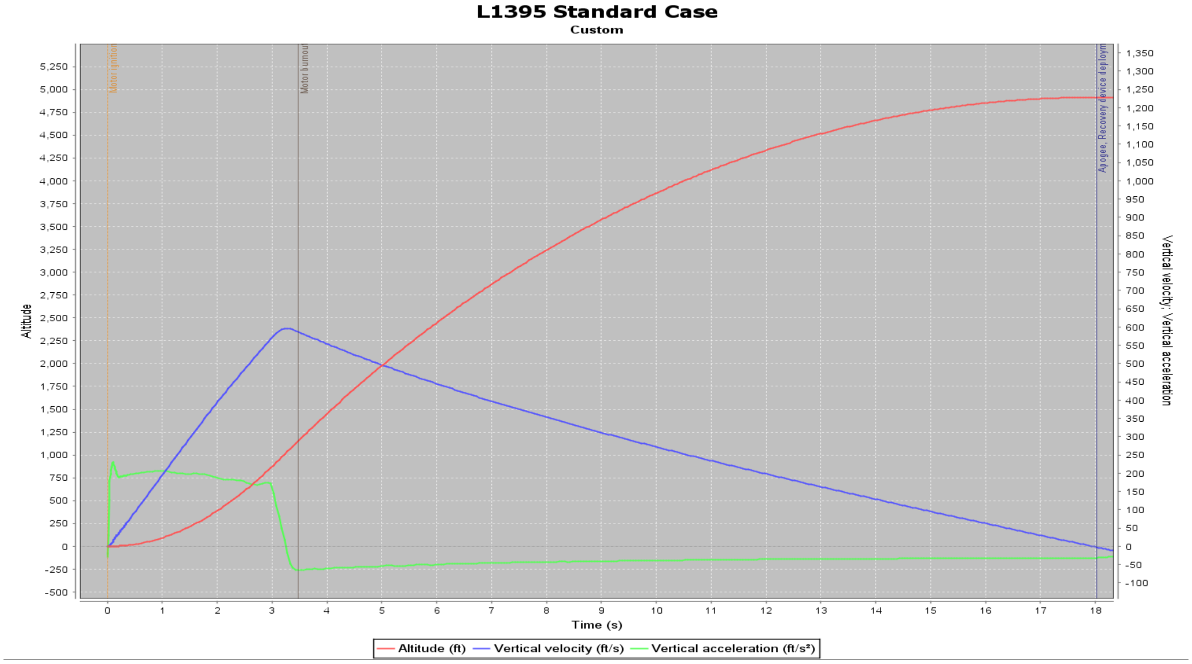


Figure 6.4 Open Rocket Results - Zero Extension

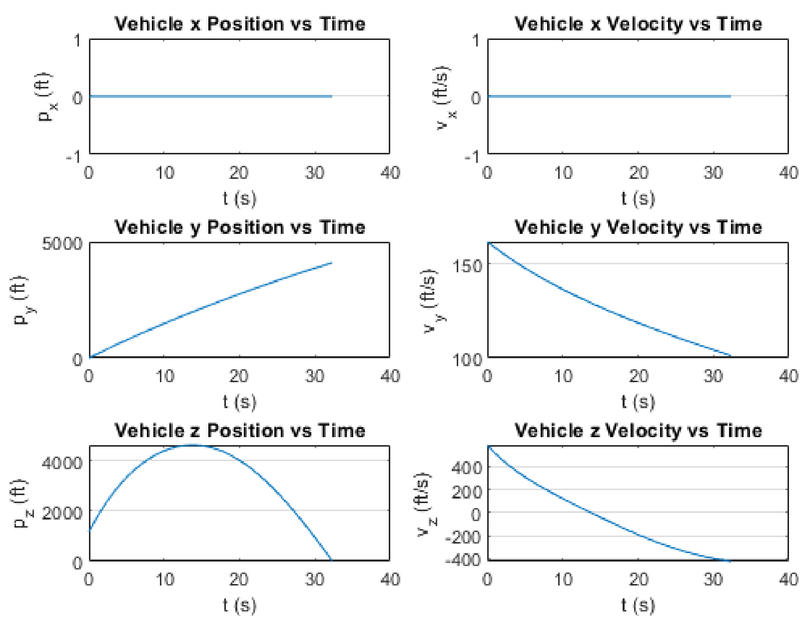


Figure 6.5 3 DOF Simulator Results - Full Extension

6.2.3 Stability

The stability margin, Cp, and Cg locations for the launch vehicle are outlined in Figure 6.6.

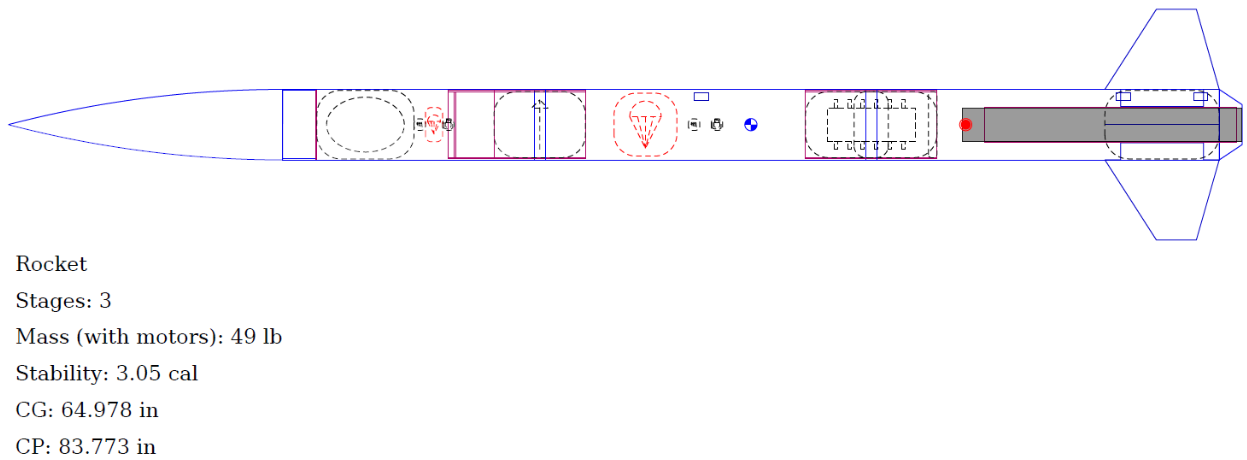


Figure 6.6 Vehicle Stability Parameters

The vehicle's static stability margin throughout the launch is shown in Figure 6.7. The vehicle leaves the rod with a static stability margin of roughly 3.05 cal, and at motor burnout has a margin of roughly 3.75 cal. These results are within the desired range of 2-6 cal for avoiding an under or over stable launch.

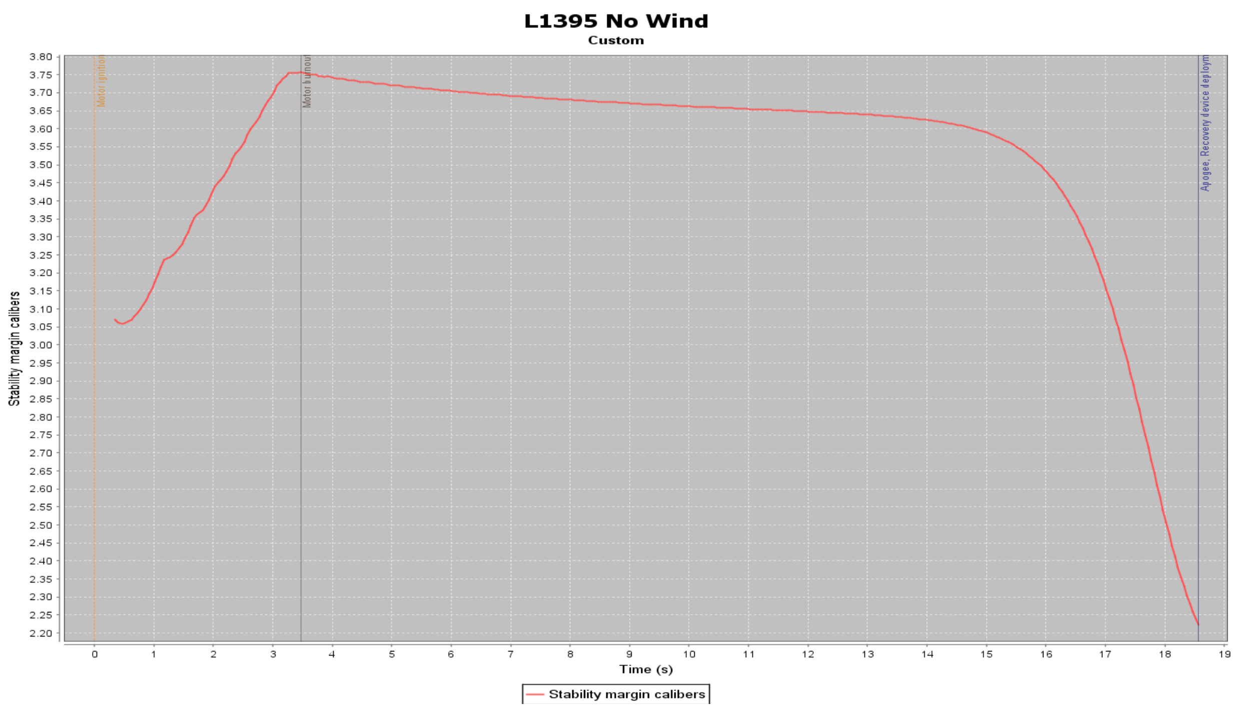


Figure 6.7 Vehicle Stability during Launch

6.2.4 Vehicle Descent

Due to the requirements of this year's competition, the vehicle must eject its payload during the descent phase and be recovered separately. While OpenRocket does support staging, staged components must be external components, which prevents the mass of the payload from being ejected in the flight simulations. As this ejection causes a significant change in mass, the descent of the vehicle is simulated using an improved version of the 3 DOF descent simulator developed and verified during the 2019-2020 competition year.

The improved version can now take in matrices containing variables such as stage event triggers, vehicle parameters, and environmental variables and automatically switches between each stage at the specified points in flight. The full code can be found in Appendix 10.1.

The initial conditions for the vehicle place it directly above the launch site at our target apogee of 4550 ft, with zero velocity in any direction. The vehicle mass is equal to the burnout weight of 43.53 lb. The drogue chute deploys immediately, and the rocket descends at a rate of approximately 95 ft/s, decreasing slightly as the air density increases. At 1000ft, the vehicle mass drops by 5.5lb as the payload drops away from the vehicle. This brings the descent velocity to roughly 89 ft/s, until 600 ft, when the main parachute is deployed. The vehicle mass remains constant, and it descends at a rate of 13.6 ft/s until reaching the ground.

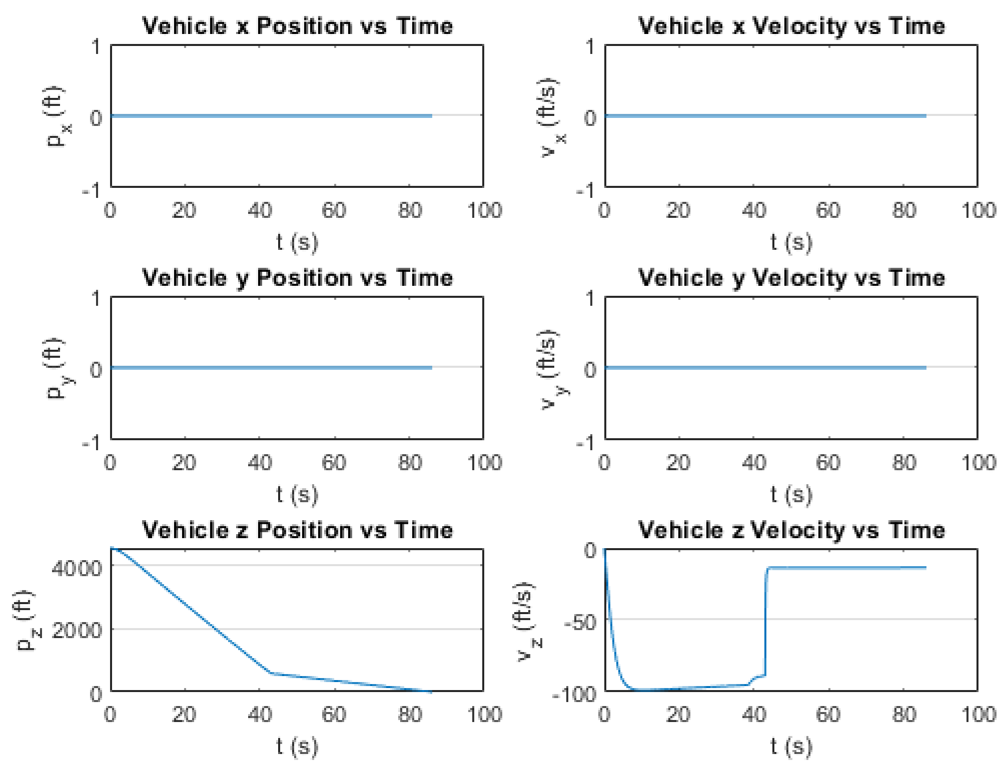


Figure 6.8 Simulated Vehicle Descent Profile

With a landing velocity of 13.6 ft/s for the launch vehicle and 17.2 ft/s for the payload, the vehicle sections have kinetic energies and descent times as described in Table 6.5.

Section	Section Mass (lb)	Kinetic Energy (ft-lbf)	Descent Time (sec)
Lower Section	22.3	64.0	86.1 s
Recovery Bay	3.8	10.9	
Upper Section	14.6	42.0	
Payload	5.5	25.2	67.7

Table 6.5 Descent Parameters

Though OpenRocket, values will be slightly different due to the aforementioned issues of weight and apogee, they are presented in Figure 6.9 and Table 6.6 as validation for the results of the team's custom descent simulator. The calculated main descent velocity increases to 14.5 ft/s, and the launch vehicle descent time decreases slightly as a result of the extra weight of payload, offset by the increased apogee. The additional payload weight is not included in the kinetic energy total for the upper section.

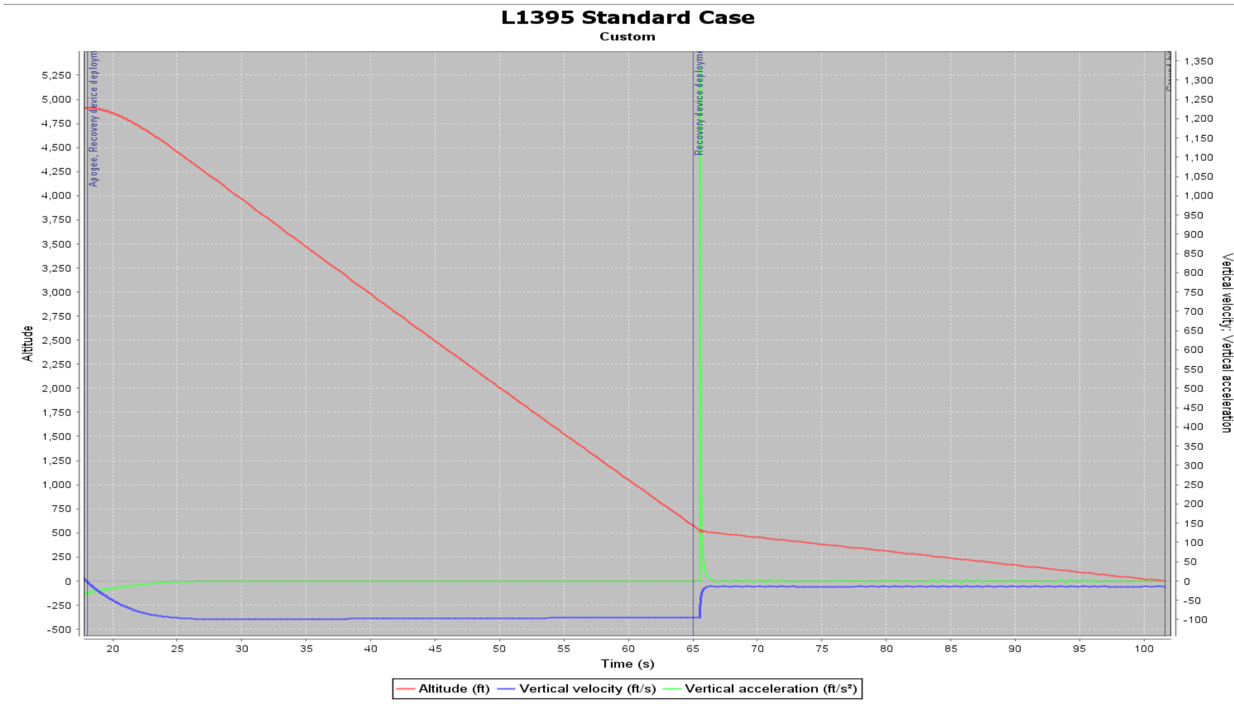


Figure 6.9 OpenRocket Descent Profile

Section	Section Mass (lb)	Kinetic Energy (ft-lbf)	Descent Time (sec)
Lower Section	22.3	72.9	85
Recovery Bay	3.8	12.4	
Upper Section	14.6	47.7	

Table 6.6 OpenRocket Descent Parameters

While these parameters are not equal to the results from the custom simulator, they differ in the expected direction and with a reasonable magnitude given the known differences between the simulations. Further references to these parameters will use the values calculated in Table 6.6, as these are the most accurate representation of the launch vehicle's descent.

With the calculated descent time, it is possible to determine the expected drift of each section in different wind conditions assuming the section travels at a constant velocity with the wind, as listed in Table 6.7.

Section	0 mph (ft)	5 mph (ft)	10 mph (ft)	15 mph (ft)	20 mph (ft)
Launch Vehicle	0	631.5	1263.1	1894.6	2526.2
Payload	0	496.2	993.0	1488.7	1985.0

Table 6.7 Section Drift

During the descent, it is also useful to know the forces placed on the parachutes as they open. Particularly for the large main parachute, the opening shock load can be significant as a result of the high descent rate under drogue and large parachute area. An infinite mass deployment scenario, in which the parachute inflates fully before beginning to slow down the vehicle, could place well over 1000 lbf of load into the vehicle. While this number is relatively simple to calculate, it is not representative of the actual (finite mass) loading scenario, where the vehicle will slow down as the parachute opens, resulting in the maximum opening shock load occurring before the parachute is fully deployed.

To calculate a better estimate of the opening load on the vehicle, we must be able to predict the parachute inflation time, and the area of the parachute during the inflation. Parachute Recovery Systems: Design Manual, by T.W. Knacke offers some experimentally determined equations relating these values to parachute dimensions, velocity, and a non-dimensional parameter, the canopy fill constant, n.

$$t_f = \frac{n * D}{v}$$

where n = Canopy Fill Constant (dependent of parachute type)

D = Parachute Diameter

v= velocity at opening

Figure 6.10 Parachute Inflation Time

$$A^{parac\ hute}(t) = A * \left(1 - \frac{A_p}{A} * \frac{t}{t_f}^3 + \frac{A_p}{A}\right)^2$$

where A = Parachute Area

A_p = Area at beginning of inflation (in our case the packed area)

t_f = Inflation time

Figure 6.11 Parachute Inflation Area

The canopy fill constant is generally determined experimentally, though tables of suggested values are available. No values could be found for the annular parachute style of the launch vehicles main parachute, so the minimum general n value of 4 was used in its place. This results in an opening time of 0.45 seconds, a reasonable, and importantly conservative assumption especially given the inclusion of a reefing system, which can often double the canopy fill constant of any given parachute.

At main parachute deployment, with the velocities and masses described previously the maximum acceleration experienced by the vehicle is calculated as 13.4 G.

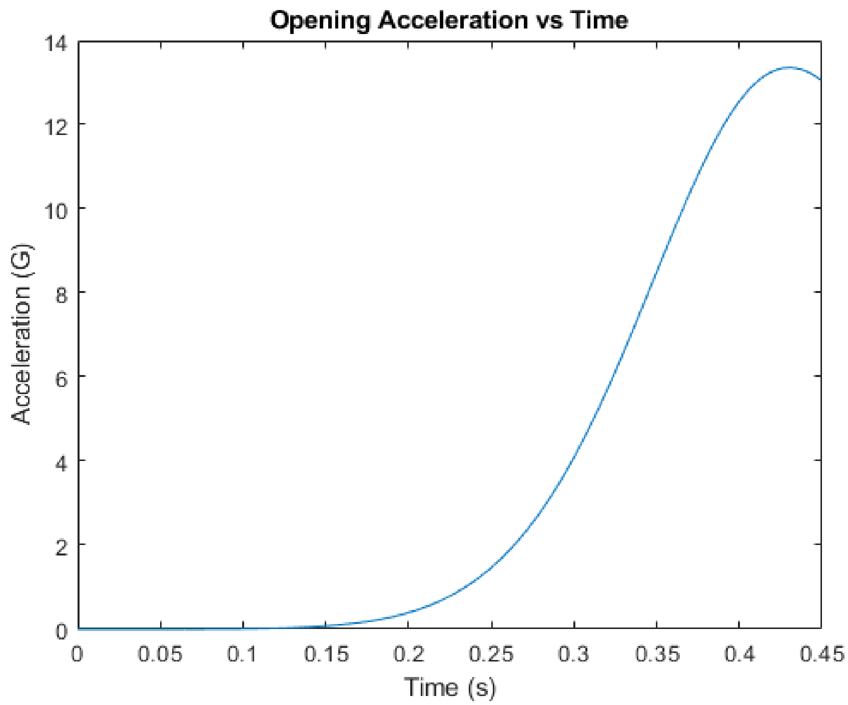


Figure 6.12 Main Parachute Opening Acceleration

For each section, resulting forces are calculated using the section's mass and the maximum acceleration. These values are used in structural analysis of the parts to verify their ability to withstand the recovery loads. These values represent the loads on the attachment hardware at closest to the main parachute, where the total load of 508 lb is felt.

Section	Opening Shock Load (lb)
Lower Section	298.4
Recovery Bay	246.2
Upper Section	195.6

Table 6.8 Vehicle Opening Shock Loads

6.3 Aerostructures

6.3.1 Airframe

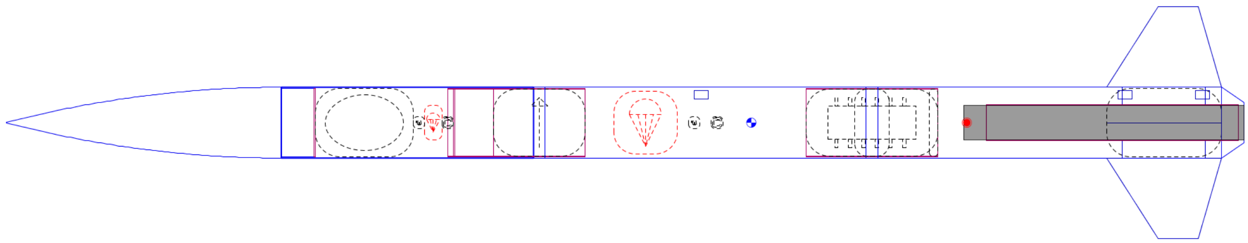


Figure 6.13 OpenRocket Design

The launch vehicle was designed using Open Rocket version 15.03. The airframe of the launch vehicle will use a 6" inner diameter airframe made of G12 filament wound fiberglass from Madcow Rocketry. The diameter of the airframe will allow for ample room to house the electronics bay, the payload retention, as well as the recovery system. The spatial capacity for these components will ensure a smoother deployment of the payload and the recovery system. Furthermore, fiberglass is the ideal choice for the airframe, for its low cost and considerable strength. The material is sturdy and resilient and is transparent to radio waves and has high heat resistance.

The airframe of the vehicle will be separated into three sections: upper airframe, middle airframe, and lower airframe. The lower airframe is 29.95" in length, specifically such that the motor retention hardware will interface with the avionics bay. The middle airframe is 28 in, and the upper airframe is 22 in. The upper airframe of the launch vehicle will house the payload and the drogue parachute. The middle airframe will house the main parachute. The lower airframe will hold the fin can and motor retention system. The fiberglass tubes are sold at lengths of 30 and 60 inches, therefore keeping the airframe lengths below 30 in, and the total length below 90 in, eases the fabrication process and lowers the cost of the airframe. Fiberglass couplers will be used between the airframe sections, as they are designed to fit the airframe. An added benefit of using couplers made of the same material as the airframe is that the thermal expansion of each piece will be the same, preventing sections from binding or becoming loose in temperature swings

The airframe will be made of fiberglass for its strength and lower cost compared to carbon fiber. The airframe must be able to withstand the compression between the motor's thrust and the nose cone and withstand lift forces from the side when the launch vehicle is tilted. It also must be able to withstand extreme temperatures. Blue Tube 2.0 and carbon fiber were considered as materials for the airframe. Although it has higher strength and higher heat resistance, due to the higher cost of carbon fiber, and its mitigation of radio signals, it was avoided. Although Blue Tube 2.0 is considerably lighter and cheaper, it was not chosen since it often warps and zippers more easily than other materials. This causes problems related to dimensional inaccuracies during assembly. Additionally, as the team will attach most subassemblies with bolts rather than the more typical epoxy, Blue Tube 2.0 does not withstand shearing loads well, and there is concern about failure of the tube during flight.

6.3.2 Nosecone

No changes have been made to our nose cone design since proposal. The nose cone, which will be purchased from Madcow Rocketry is tangent ogive in shape (4:1 with a shape parameter of 1), 24 in long, and has a shoulder length of 6 in. In addition to being more affordable than materials like carbon fiber, a fiberglass nose cone is stronger than plastic and more flexible than carbon fiber. These properties provide the durability needed to maintain structural integrity in the event of a nose-down landing. The aluminum tip further aids durability and brings the center of gravity toward the front to increase the overall stability of the rocket.

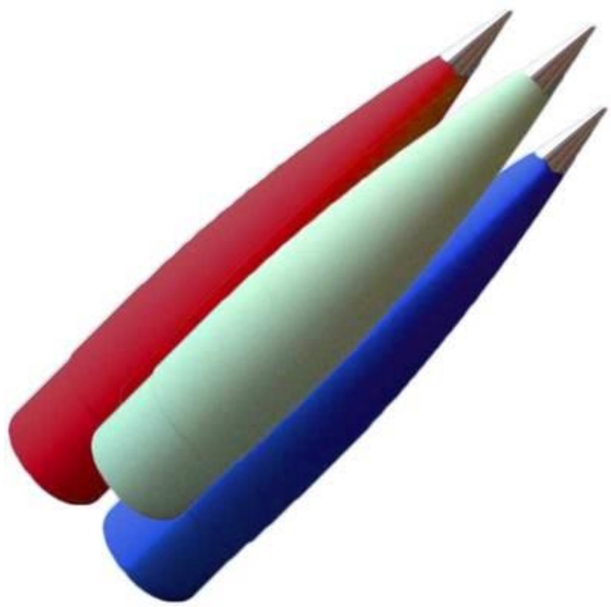


Figure 6.14 Madcow 6" Fiberglass Nosecone

Alternative shapes considered for our nosecone were elliptical, conical, and Von Karman. Von Karman nosecones are most optimal in transonic speeds, though the vehicle will not be reaching these speeds, remaining in the subsonic region. Elliptical and conical nosecones are not commonly available, so sourcing the components would be difficult. Purchasing the nosecone is our best option due to difficulties that would come with manufacturing our own, as well as restrictions on group work due to COVID-19. This limits the team to either a large 5.5:1 Von-Karman Nosecone, or a 5:1 or 4:1 ogive nosecone. Because the rocket remains subsonic, the drag effects are dominated by skin friction, so the smallest 4:1 Ogive nosecone will have the lowest drag, as well as being the lightest, which will help to increase our apogee.

6.3.3 Fins

The fins on our rocket provide stability during flight. This is accomplished by increasing the surface area at the aft end of the rocket with fins, which in effect lowers the center of pressure on the rocket. The distance between the center of pressure and the center of gravity on our rocket is what determines the vehicle's stability. If the rocket is not stable enough, the flight will be unpredictable and dangerous to those nearby. However, if the rocket is too stable it may turn into the wind.

There are many other considerations and wants we must consider when constructing our fins. One of these is the weight of the fins. In order to save weight where we can, we want the fins to be as lightweight as possible. Another factor to consider is manufacturability of materials and shapes. The fins should also not be fixed to the rocket. This will make transport and construction at the launch site simpler.

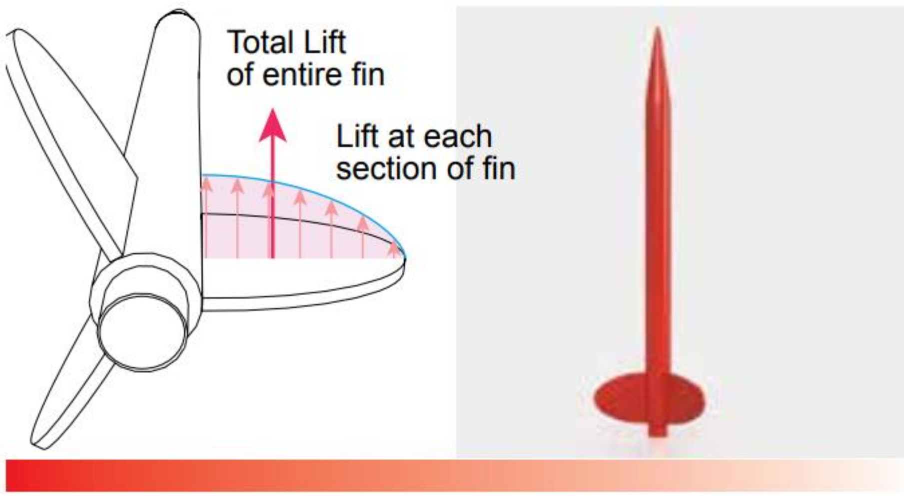


Figure 6.15 Elliptical Fin Lift Distribution [2]

There were several design options that were being considered for our fins on the rocket this year. Two fin shapes were being considered when we began our initial design, trapezoidal and elliptical. Elliptical shaped fins produce the least amount of induced drag cause by vortices at the tip of the fin because their lift profile tapers out towards the tip of the fin as shown in Figure 6.15. However, it is difficult to manufacture due to its curved edges. A trapezoidal shaped fin provides many similar benefits to elliptical while being significantly easier to build. Tapered fins also provide most of the strength and stiffness at the root of the fin, essential for preventing fin flutter or damage upon landing. The fins will be made of a composite layup, with the composite of choice being carbon fiber fabric. Carbon fiber possesses the best stiffness to weight ratio, and since stiffness is directly related to flutter resistance, it is the ideal option. There were also three core materials being considered for the fins: wood, fiberglass, and foam. Although foam is lightweight, it is much weaker than the other options, and is susceptible for crushing. The fin can design includes a small contact surface area, so there is concern that the foam would crush and damage the carbon fiber. Wood and fiberglass both provide good strength to the fins while also being easy to manufacture. Fiberglass is stiffer, however also heavier than wood. Since the core material is not meant to take bending loads, the lighter wood should be sufficient for the fin core.

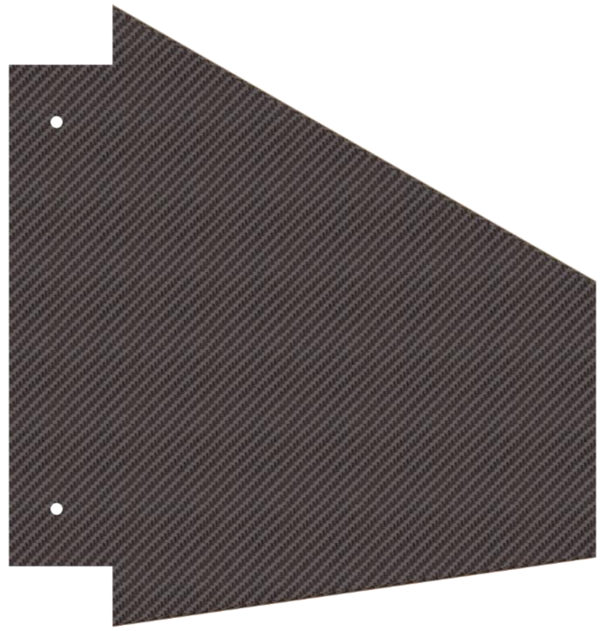


Figure 6.16 Fin Profile

Our current fin design for our launch vehicle includes four fins, mounted to the fin can at the bottom of the rocket. The fins are bolted into the fin can using two bolts attached to the fin brackets described further in section 6.4.1. The shape of the fins will be trapezoidal, with a root chord of 10 in, a tip chord of 3.5 in, and a height of 7 in. In addition, the fins have a sweep length of 4.5 in. A trapezoidal shape was chosen due to its ease of manufacturability, and the fact it provides many benefits of an elliptical fin. An elliptical fin would be ideal; however, it would be much more difficult to build. The material of our fins will also be a birch wood core with an exterior overlay of carbon fiber. The carbon fiber will provide a large amount of strength to the fins while being extremely lightweight. The wood core also provides the strength needed, while being easy to work with and acquire. The fin will also have a rounded leading and trailing edge, accomplished using sanding, to help reduce the drag of the fins.

6.3.4 Tailcone

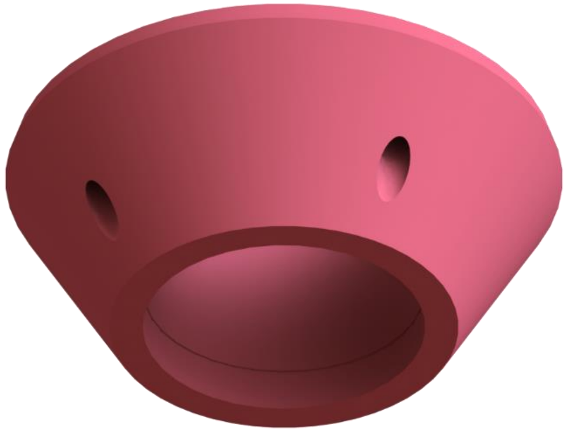


Figure 6.17 Tailcone

The purpose of the tail cone is to reduce the base drag at the end of the rocket. It must be durable enough to withstand impact and protect the engine casing, but also light. Our design is very similar to the proposal. The shape of the cone will have straight sides with a maximum diameter of 6.17 inches that tappers down to 3.86 inches. The hole left for the motor casing will be 3.14 inches. The tail cone will be printed using MatterHackers NylonX filament. This allows for easy manufacturing, a light weight, and durability. The tail cone will be mounted through the bottom of the fin ring on threaded standoffs to allow room for fin can mounting hardware. It will also surround the motor casing.

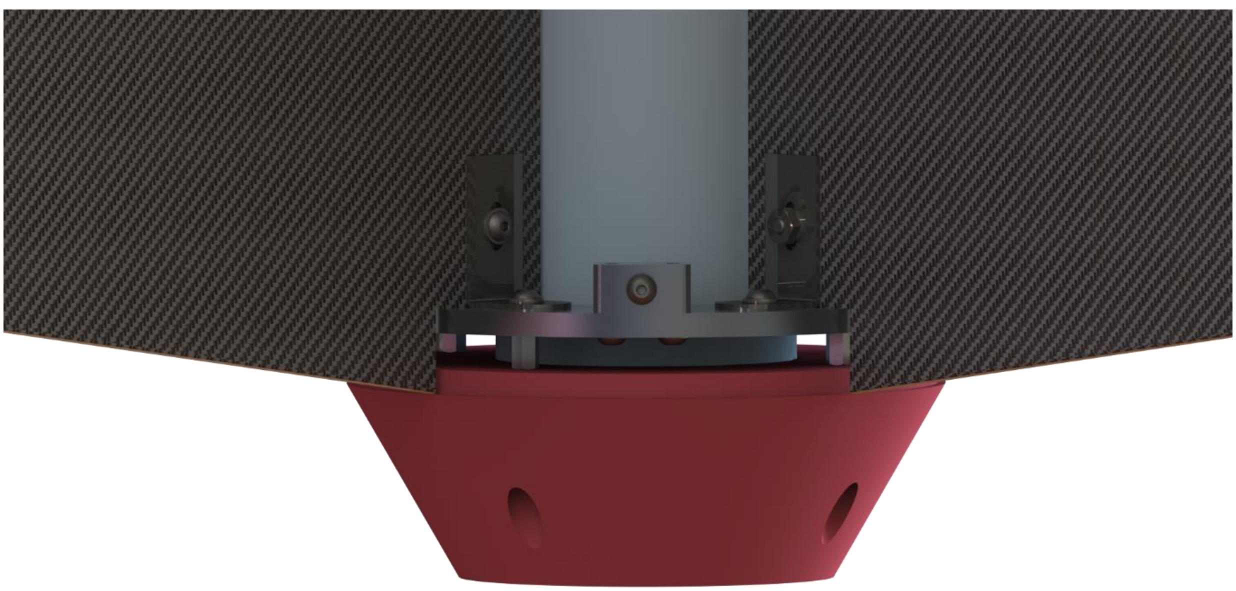


Figure 6.18 Tailcone Attachment

Other considerations for the material were carbon fiber or machined aluminum. These methods are expensive and harder to manufacture. Machined aluminum is also very heavy.

Another design consideration was how to attach the tail cone to the rocket. The options were to put the screws through the bottom of the tail cone, or through the sides of the body. It was decided to put the bolts through the bottom, since putting them through the side would be harder to remove, could get in the way of other components, and less aerodynamic. It was also considered to use 8 bolts instead of 4, but it was decided to use 4 because it is lighter and adequate to hold the tail cone which is not load bearing.

6.4 Propulsion Integration

The primary function of the fin can is to secure the fins in place and act as a centering apparatus for the motor casing and motor tube. Each fin must be directly connected to the fin rings and will need to remain attached throughout all stages of the launch. When designing the fin can assembly, the shear and normal stresses sustained by the fins and thrust from the motor were carefully considered, as well as the weight of the combined assembly. Designs were highly focused on manufacturability and modularity. Manufacturability was a main concern because it would determine the amount of time and material needed to produce the fin can components. We prioritized modularity as well to help save time and money from a manufacturing standpoint, improve ease of assembly and disassembly, and replacement of components if necessary. Rather than overly complex, single-body designs with greater potential for manufacturing error, our team focused on designing a system composed of multiple simpler parts that improve ease of manufacturing, testing, and construction.

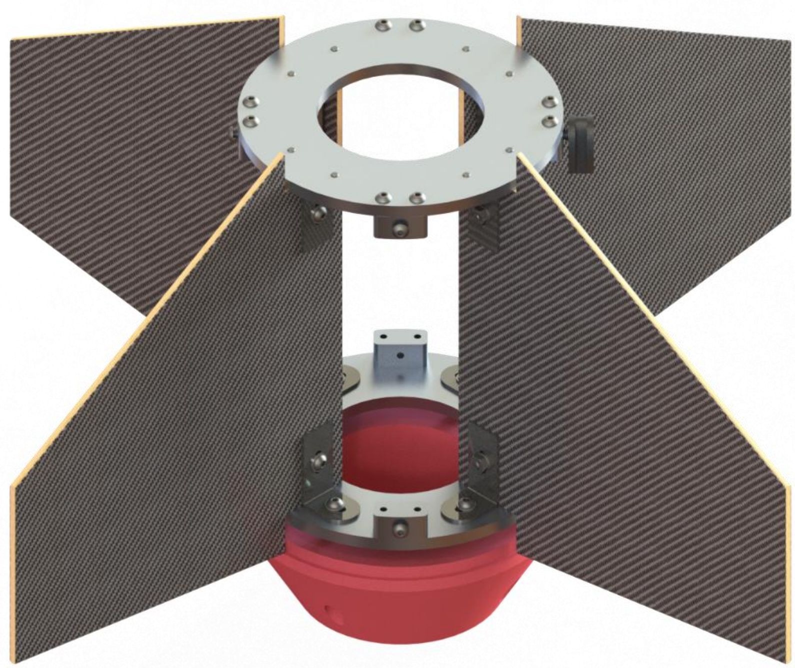


Figure 6.19 Fin Can Assembly

The fin can is designed with the intent to ensure the fins are held in place. The system consists of two centering rings, fin brackets, fins, radial brackets, and tailcone as shown in Figure 6.19. The fins will be secured by right-angle brackets that fasten to the centering rings. Each centering ring will be attached to the airframe by four radial brackets.

Figure 6.20 Alternative Fin Can Design

Several different assembly options were considered, including a three-part design consisting of a 3D printed mid-section fin bracket between two centering rings with tabs as shown in Figure 6.20. The alternative designs were discarded due to machining complexity and excessive weight. The final design chosen capitalizes on the advantages of modularity and manufacturability that came with designing the simpler centering rings and right-angle brackets.

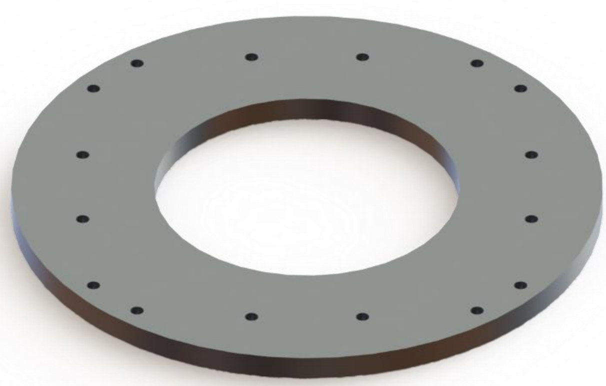


Figure 6.21 Centering Ring

The purpose of the centering ring is act as an intermediary component upon which the fin brackets and radial brackets are secured and help center the motor. There are two centering rings, one above and one below the fin tabs extending inside the lower airframe. Each ring will have an outer diameter of 6 inches, an inner diameter of 3.1 inches, be a ¼-inch thick, and be manufactured out of 6061-T6 aluminum using a waterjet. Both centering rings will have sixteen #8 holes for securing the radial and fin brackets.

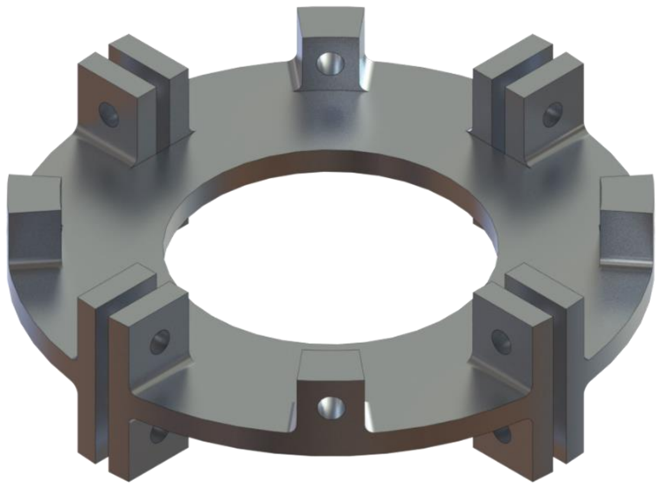


Figure 6.22 Initial Centering Ring Design

Our initial centering ring design proposal had tabs extending above and below the ring intended to secure the fins directly to the rings and bolt the ring to the lower airframe body tube as shown in Figure 6.22. However, the tabs in this design greatly increased the manufacturing complexity due to more difficult machinery requirements and limited team experience. Furthermore, the initial design had four 3D printed mid-section components between the two centering rings with the intention of combatting shear stresses on the fins. This idea was discarded since it would add unnecessary weight and the team could not simulate 3D printed parts since they are printed in horizontal layers and are susceptible to shearing and other orthotropic material properties.

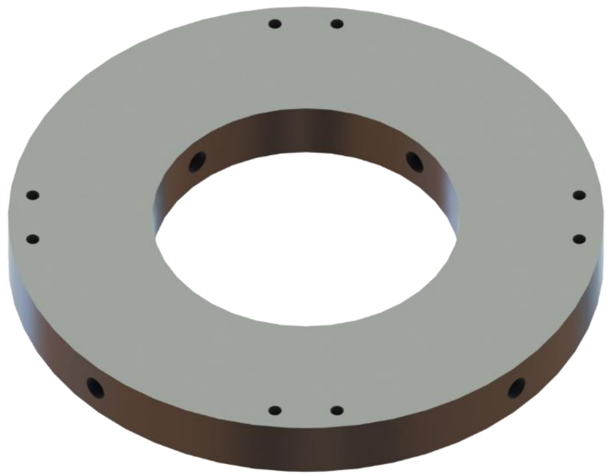


Figure 6.23 Alternative Centering Ring Design

One of the alternative designs we considered was a thick centering ring with four threaded radial holes to bolt the rings to the airframe that would use right angle brackets to attach the fins to the centering rings as shown in Figure 6.23. Excess weight in this design lowers the overall rocket's center of gravity, negatively impacting stability. In the final iteration of our design, we addressed the issue of weight by reducing ring thickness to ¼ inch and added individual radial brackets that attach the rings to the airframe. Our current design is lightweight, modular, fulfills necessary functions, and easier to manufacture compared to our initial design.

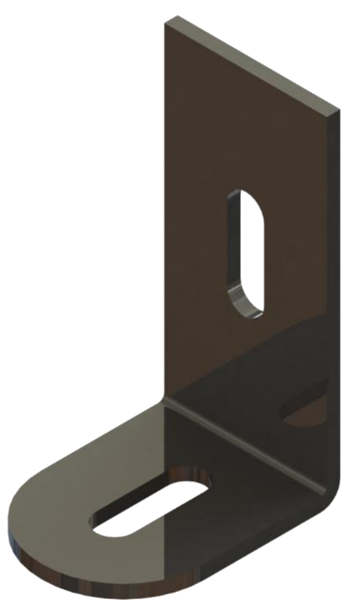


Figure 6.24 Fin Bracket

The fin brackets secure the fins in place and are attached to the centering rings. Manufactured out of plain carbon steel sheet metal, these custom right angle brackets will attach to the centering rings and fins using #8 bolts. The brackets are 1.5 inches tall and 1 inch wide, with 0.5-inch-long slots on both flanges as shown in Figure 6.24. The slots allow greater tolerance for hole alignment and bolt placement. Because the fin bracket must attach to two other components, single holes would greatly increase the difficulty of assembly due to the issue of hole alignment. Various designs for the fin brackets were explored. Design parameters for the fin bracket included appropriate dimensions, two slotted holes for bolts, suitable shape, good amount of surface area contact with the fins, and tolerances for manufacturing error. The team initially considered purchasing COTS right angle brackets but did not find a product that suited our exact needs. While buying COTS brackets in bulk is more convenient then manufacturing our own brackets in-house, we elected to create a custom design for the fin brackets because it allowed us to optimize the bracket geometry and retain a reasonable level of design manufacturability, with only a single bend in the part.

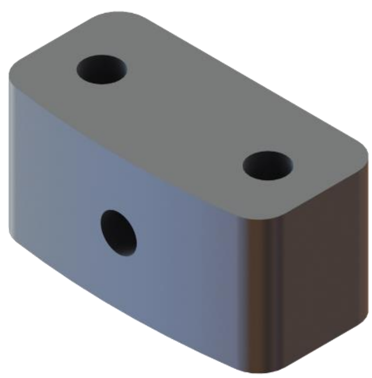


Figure 6.25 Radial Bracket

The main function of the radial brackets is to attach the two centering rings and the thrust plate to the airframe. The radial brackets will be CNC machined out of 6061-T6 aluminum on a mill. Four radial brackets will be fastened onto the centering rings and the thrust plate by two #8 screws each. Six out of eight of the radial brackets will be secured by #8 bolts to the airframe. Two radial brackets will have ¼ inch, threaded holes for the two rail buttons. The rail buttons will slide into the launch rail to maintain a vertical trajectory during the initial launch stage. The brackets were designed to be a relatively simple, rectangular component that is easily manufactured. The sides that will contact the cylindrical airframe will be rounded so the full surface of this face will be in contact with the airframe, preventing stress concentrations. There will be two configurations of this design due to the varying diameters of the airframe body tube and the coupler tube.

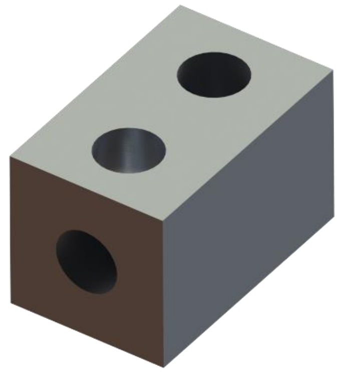


Figure 6.26 Alternative Radial Bracket Design

Multiple alternative designs were considered for the radial brackets. Unlike the fin brackets, typical angle brackets would not be suitable for the purpose due to the need for threads on the radial holes; once in the airframe, the inside of the fin can could not be accessed to tighten nuts onto angle brackets. The initial design for the centering rings and thrust plate involved tabs extending from the parts with threaded holes used to attach the parts to the airframe. It was later decided to design separate, modular radial brackets that would be far easier to machine than a single part with tabs extending from it. Thick centering rings with holes drilled into the rounded sides were considered but was dismissed in favor of a lighter, easier to manufacture design. Another design that was strongly considered was a block smaller in width and greater in length than the final design as shown in Figure 6.26. In this alternative design, two holes along the bottom face of the bracket would be drilled and tapped underneath the radial hole for airframe attachment. The primary issue with that design was hole interference and insufficient thread engagement. Resolving this issue would have required additional manufacturing processes to ensure enough threads would be cut, such as using a bottoming tap. Hole interference could have been avoided in this design by increasing the part's length, but this would have added more material and weight. Altogether, the current design was chosen so hole interference could be avoided while using less material, thereby eliminating some weight, and maintaining ease of manufacturing.

6.4.1 Motor Retention Design

The motor retention system is composed of the thrust plate, motor tube, and fin rings. Located at the top of the motor casing, the thrust plate design is optimized to transfer thrust to the airframe. The top-side retention prevents the motor from being ejected during the recovery stage of our descent from apogee. In addition to the two fin rings, the thrust plate acts as a third centering point for the motor tube and motor. The assembly assures that the thrust plate will transfer the energy from the motor into the airframe. The primary design goal for this system was to create a design which would remain relatively strong, while minimizing the amount of material used. To help maintain reasonable stability, we focused on a lightweight design that will minimize the amount of weight near the bottom of the rocket.

Figure 6.27 Motor Retention System

The purpose of the thrust plate assembly is to evenly transfer the thrust exerted by the motor to the airframe. The thrust plate will be CNC machined out of 6061-T6 aluminum on a mill. It is attached to the airframe via four radial brackets and to the motor casing by a single 3/8-inch countersunk screw, which both centers and retains the motor in the rocket. The solid bottom face of the thrust plate helps isolate the motor system from the electronics bay. The intent of this feature is to block heat emitted by the motor, which could be potentially dangerous to the rocket's operation due to proximity to the sensitive electronics. Minimizing material and creating a design with a minimum safety factor of 3 was carefully considered in the overall design process. An alternative design that was considered included a thrust plate manufactured as a thick single piece with four radial bolt holes along the outer perimeter design as shown in Figure 6.28. While easier to manufacture, the single piece design was ultimately discarded due to the excessive amount of material it would have required to support all the bearing stresses in the radial bolt holes. The final design chosen successfully distributes the stress evenly throughout the plate and avoids stress points that increase the possibility of bending and warpage.

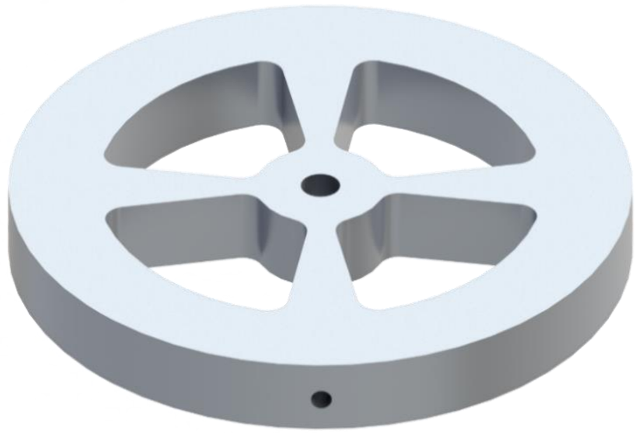


Figure 6.28 Alternative Thrust Plate Design

Initial FEA simulations and hand calculations were run on the thrust plate designs to determine point of high stresses, material displacement, and safety factor which ultimately led us to our current design. A simplified study setup involved fixing the faces of the thrust plate that are to be in contact with the radial brackets and applying a force at the central hole where the thrust plate is to be attached was completed. In 5 iterations of a h-adaptive study, the model did not converge. A factor of safety distribution showing only areas below a factor of safety of 3 can be seen in Figure 6.29. This result reveals the cause of the simulation's failure to converge, namely singularities at the edges of the fixed faces, and at the central hole.

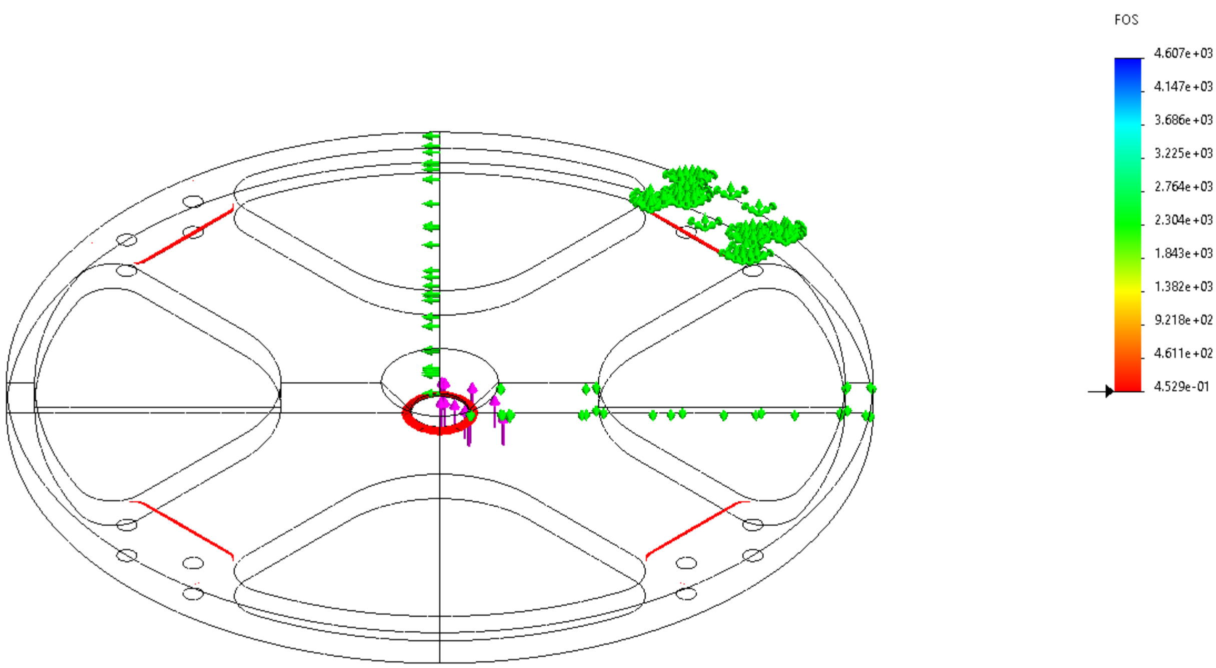


Figure 6.29 Thrust Plate Safety Factor Plot (F.S. < 3)

Based on the simulation setup and the plot results, the low safety factors can be dismissed as an artifact of the simulation process itself rather than an actual physical effect. In the real world, the radial bracket would deform slightly to spread the stress more evenly across the part surfaces than can be replicated with a fixed face. This assumption is further supported by the displacement plot in Figure 6.30, showing the part will only deform by .182 mm at maximum, a negligible displacement which suggests the thrust plate will not plastically deform or fail during the launch.

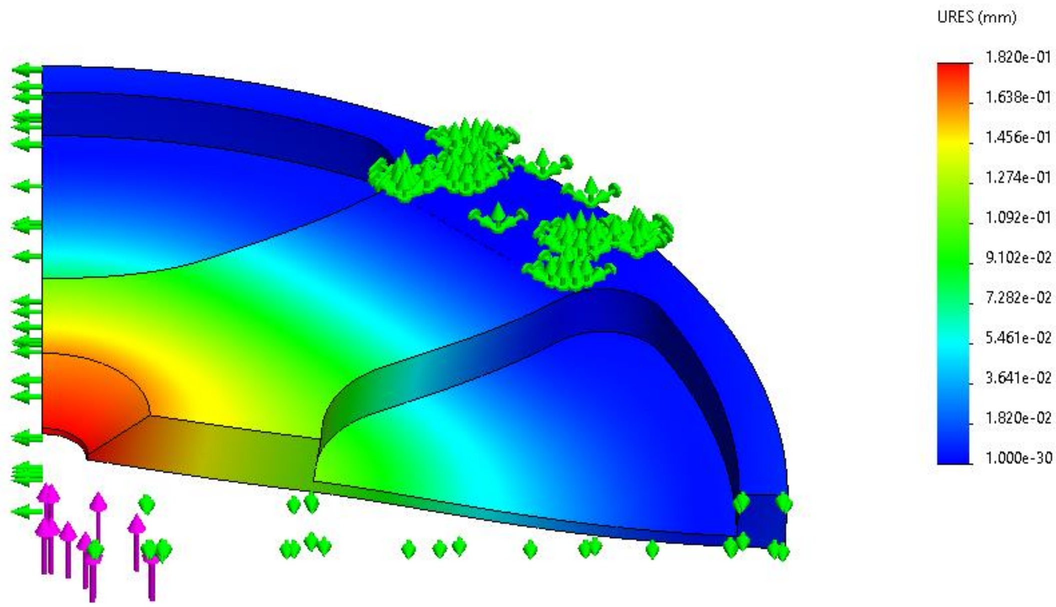


Figure 6.30 Thrust Plate Displacement Plot

Furthermore, analysis of the radial brackets was undertaken to ensure they would support the loads of the motor under thrust. Using the methods outlined in reference [3], safety factors for bolt tear out and bearing failure in the radial bolt hole were determined, along with a factor of safety for shear failure of the bolt itself. With a hole diameter to edge distance ratio of 1.5, these parts are within the applicable range for these equations to apply.

For bolt tear out, where the material shears around the bolt hole, we calculate the shear area conservatively as the area formed by 2 planes from each edge of the bolt hole to the material edge. This is a conservative estimate because the actual shear planes will be at some angle to the bolt hole, resulting in a larger shear area. With the motor loads, and a #8-32 bolt hole, the calculated factor of safety for tear out is 32.76.

Tear Out Failure								
Bolt Size	**#2-56**	**#4-40**	**#6-32**	**#8-32**	**#10-32**	**#12-24**	**1/4-20**	**3/8-16**
Nominal Diameter (in)	0.086	0.112	0.138	0.164	0.19	0.216	0.25	0.375
e/D	2.9	2.2	1.8	1.5	1.3	1.2	1.0	0.7
As (in^2)	0.16	0.15	0.14	0.13	0.12	0.11	0.09	0.05
Pult (lb)	4036.50	3783.00	3529.50	3276.00	3022.50	2769.00	2437.50	1218.75
FS	40.37	37.83	35.30	32.76	30.23	27.69	24.38	12.19

Figure 6.31 Radial Bracket Tear Out Failure Analysis

Bearing failure occurs when the material around the bolt deforms significantly, but the material does not shear. We calculate the bearing stress area as 25% of the total bolt hole surface area, again a conservative estimate. For the motor loads and an #8-32, the bearing failure factor of safety is 72.13.

Bearing Failure								
Bolt Size	**#2-56**	**#4-40**	**#6-32**	**#8-32**	**#10-32**	**#12-24**	**1/4-20**	**3/8-16**
Nominal Diameter (in)	0.086	0.112	0.138	0.164	0.19	0.216	0.25	0.375
Abr (in^2)	0.1	0.1	0.1	0.1	0.1	0.2	0.2	0.3
Pbru (lb)	3782.5	4926.0	6069.6	7213.1	8356.6	9500.2	10995.6	16493.4
FS	37.82	49.26	60.70	72.13	83.57	95.00	109.96	164.93

Figure 6.32 Radial Bracket Bearing Failure Analysis

Finally, for the shearing of the bolt, we calculate the shear area as the cross sectional area of the bolt, and estimate the shear strength of the bolt to be 60% of its tensile strength, in this case 87000 psi for alloy steel. With the motor loads and an #8-32 bolt, the calculated bolt shear safety factor is 11.0.

Bolt Shear Failure							Bolt τ_y (psi)	87000
Bolt Size	**#2-56**	**#4-40**	**#6-32**	**#8-32**	**#10-32**	**#12-24**	**1/4-20**	**3/8-16**
Minor Diameter (in)	0.0648	0.0822	0.1008	0.1268	0.1404	0.1664	0.1905	0.3005
As (in^2)	0.003	0.005	0.008	0.013	0.015	0.022	0.029	0.071
Py (lb)	286.9	461.7	694.3	1098.6	1346.9	1892.0	2479.7	6170.2
FS	2.9	4.6	6.9	11.0	13.5	18.9	24.8	61.7

Figure 6.33 Radial Bracket Bolt Shear Failure Analysis

As a final verification, we must prove that the airframe will be able to support the loads placed upon it. Without access to hardware for testing, and due to the difficulty of performing simulations of composite materials, particularly those where the manufacturing process is not known precisely, we look to the section of 7.1.3 of reference [4]. The Bulkhead Tensile Loading test was conducted in the exact same airframe as the team's airframe, with 4 #6 stainless steel screws. In the test, the airframe was shown to withstand up to 1517 lb, well above our maximum thrust of 400.48 lb. Additionally, the team will be using #8 alloy steel screws, which are stronger, and have a larger surface area to distribute loads to the airframe.

These analyses prove that the motor retention system is more than capable of handling the flight loads placed upon it, as the largest load is expected to occur at the motor's maximum thrust. These results also verify the radial brackets for use in other areas of the launch vehicle, including the fin can, avionics bay, and nosecone, where loads will be smaller.

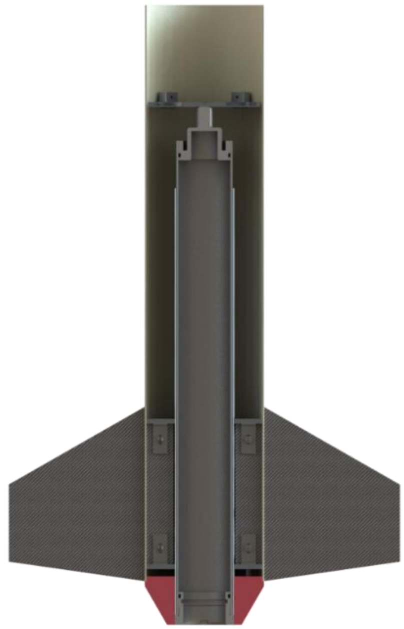


Figure 6.34 Cross Section of Lower Airframe Assembly

The fin can and motor retention systems integrate propulsion elements with critical structural components in the lower airframe. The primary criteria for these systems include securing the fins in place, centering the motor, transferring thrust to the airframe, and mounting for the tailcone. Component designs focused on modularity, manufacturability, and weight reduction.

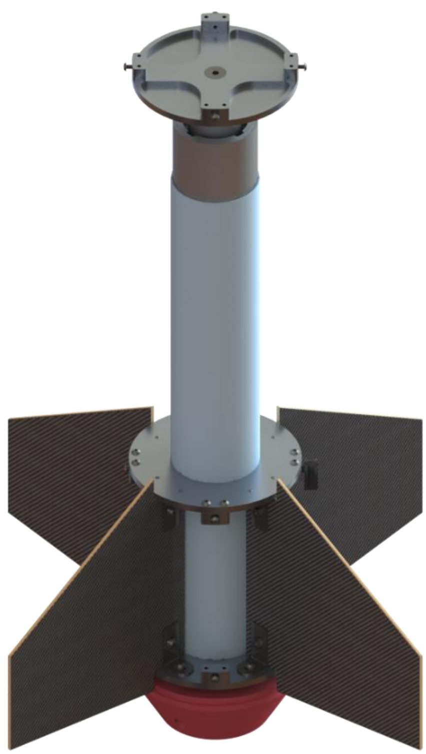


Figure 6.35 Lower Airframe Assembly

6.4.2 Projected Motors

The primary motor chosen for the launch vehicle is the L1395-BS, a class L motor manufactured by Cesaroni Technology. It has a peak thrust of 1800 N, total impulse of 4895.40 Ns, diameter of 2.95 in, length of 24.45 in, and E-Match igniter. The best- and worst-case launch conditions for the launch vehicle's flight were simulated using OpenRocket. The simulation results confirmed that the L1395-BS motor will bring us within range of our target apogee in either scenario. The thrust curve of this motor is exhibited in Figure 6.36.

Designation	L1395-BS
Average Thrust	1418.86 N
Peak Thrust	1800 N
Total Impulse	4895.40 Ns
Total Weight	4323g
Class	91% L
Diameter	2.95 in
Length	24.45 in
Delays	Plugged Seconds
Igniter	E-Match
Letter	L
Manufacturer	CTI
Name	L1395
Propellant	APCP
Propellant Weight	2364.9 g
Thrust Duration	3.45s
Type	Reload

Table 6.9 L1395 Motor Specifications

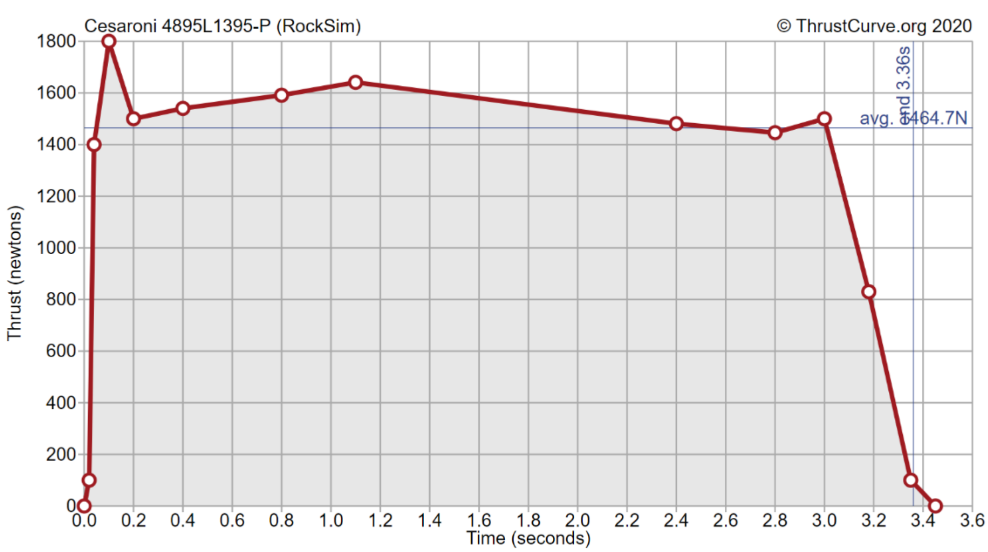


Figure 6.36 L1395 Thrust Over Time Graph

The secondary motor selected for the launch vehicle is the L2375-P, another L class motor manufactured by Cesaroni Technology. It has a peak thrust of 2608.3 N, a total impulse of 4905.17 Ns, diameter of 2.95 in, length of 24.45 in, and E-match igniter. The average thrust is of the L2375-P is higher than the L1395 and the impulse is of a similar value. Because the dimensions of the primary and backup are identical, we will be able to easily switch motors if the launch vehicle is determined to have a greater weight than the values used in simulations. The launch vehicle flight was simulated in OpenRocket with the L2375-P. The motor carries the rocket to a higher than desired apogee, but this is by design as the motor is intended to serve as a backup should the weight or drag of the rocket decrease our apogee past an acceptable level. The thrust curve for this motor is exhibited in Figure 6.37.

Designation	4864L2375-P
Average Thrust	2324.7 N
Peak Thrust	2608.3 N
Total Impulse	4905.2 Ns
Total Weight	4161 g
Class	92% L
Diameter	2.95 in
Length	24.45 in
Delays	Plugged Seconds
Igniter	E-Match
Letter	L
Manufacturer	CTI
Name	L2375
Propellant	APCP
Propellant Weight	2322 g
Thrust Duration	2.11 S
Type	Reload

Table 6.10 L2375 Motor Specifications

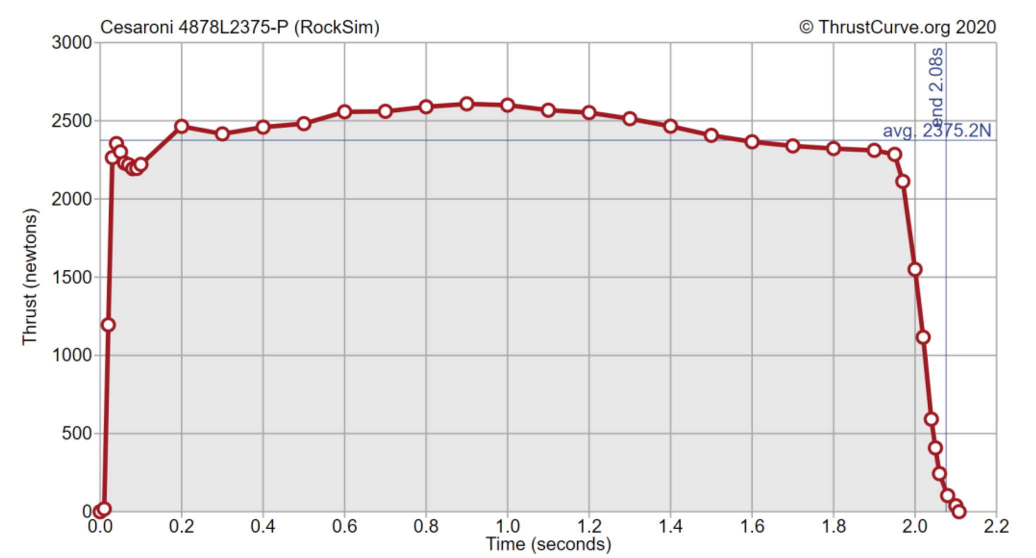


Figure 6.37 L2375 Thrust Over Time Graph

6.5 Recovery

6.5.1 Recovery Bay

The original recovery electronics bay design was integrated into the avionics bay. In order to better accommodate payload deployment, the recovery electronics were moved to a separate recovery bay in the coupler between the upper and middle airframes. The recovery system was also changed to a dual bay design, with the drogue parachute deployed from the upper airframe and the main parachute deployed from the middle airframe. The twist-lock mechanism that was used for the avionics bay would no longer work with a dual bay design since parachute deployment occurs from both sides of the recovery bay, so we decided to redesign the recovery bay entirely, keeping in mind that it should still be able to take loads produced by parachute deployment and the electronic components should be easily accessible.

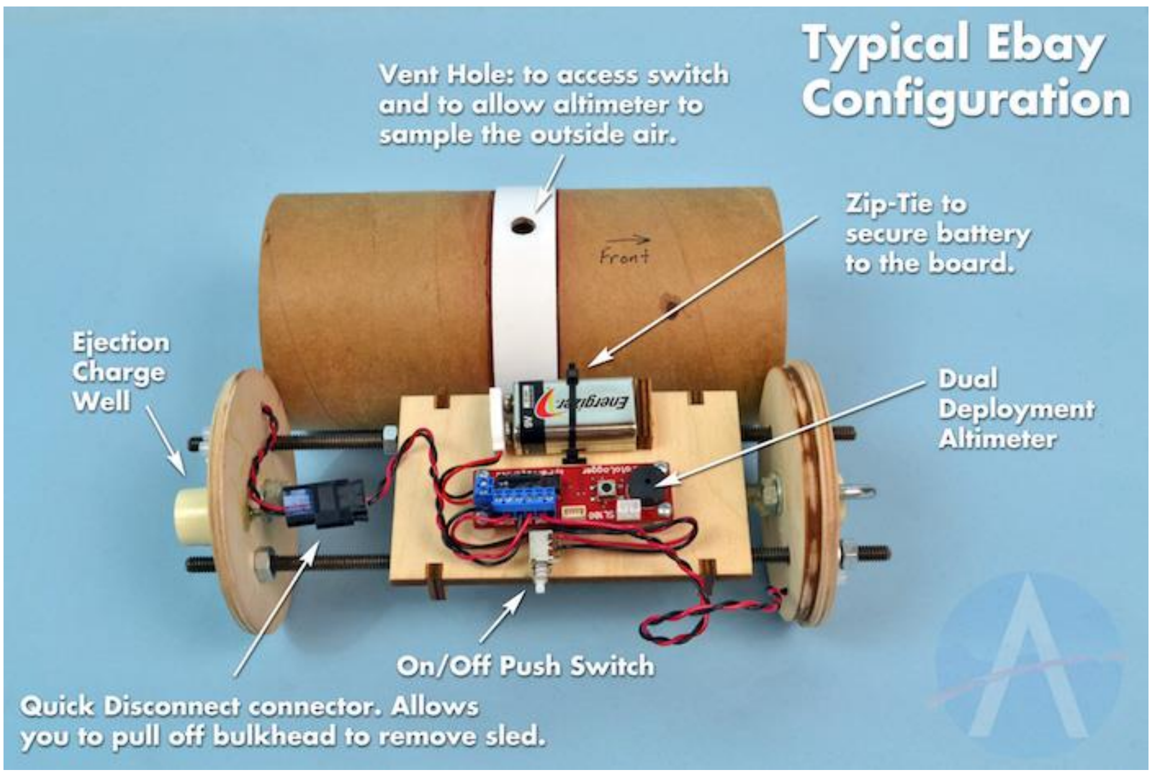


Figure 6.38 Typical Ebay Configuration

During design brainstorming, a dual threaded-rod spine was considered to connect the two bulkheads. This design is typical of most high powered rockets, as shown in Figure 6.38, and would replace the central spine from the twist-lock design and provide extra stability, but it made accessing the electronics sled more complex than necessary. We decided to keep the central spine and fix it to both bulkheads with a single bolt.

Eyebolts were also considered as a replacement for the U-bolts, as they could transfer the parachute deployment loads directly to the central spine. Eyebolts were decided against due to their swiveling capabilities. A swiveling connection adds a potential for detachment and entanglement of the parachute shroud lines, which could lead to failure of the entire recovery system.

While considering the U-bolts' placements, the application of forces was considered. Typically, the bulkhead serves to transfer the loads from the attachment points of the threaded rods or spine. This requires that the bulkhead be made of a material strong enough to handle these loads, typically thick wood or G10/FR4 fiberglass. On most of the outer area bulkhead though, there are no loads, so the bulkhead become unnecessarily heavy. An adapter to be machined out of 6061-T6 aluminum was designed to attach between the spine and the U-bolts.

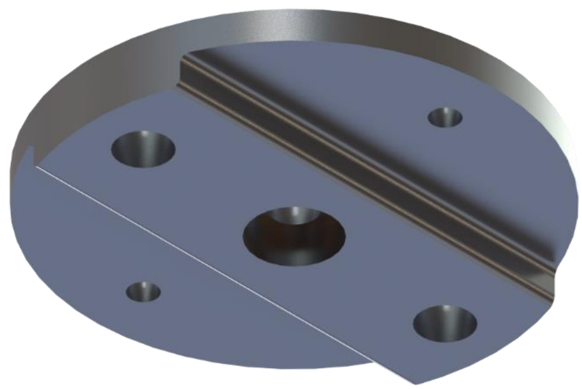


Figure 6.39 Recovery Bay Adapter

The adapter centralizes the deployment loads applied to the U-bolts so that it moves directly into the spine, lessening the loads applied on the bulkheads. Since the bulkheads no longer take the majority of the deployment loads, we also redesigned them to be made out of 1/8 in fiberglass plates in order to decrease the weight in the recovery bay. We considered both fiberglass and carbon fiber bulkheads but decided on fiberglass since it is strong and light enough for our purposes, as well as cost-effective.

Figure 6.40 Recovery Bay Adapter Assembly

Since all the loads are going through the adapter and central spine, the bulkheads can be attached using thumb screws, which will be easily removable at the launch site, and the recovery bay may be slid out of its coupler for easy access to the electronics. Figure 6.40 shows the top section of the recovery bay with the bulkhead sectioned away.

Figure 6.41 Recovery Bay Spine

The central spine is hex-shaped, with circular sections turned into the ends, so that the electronics sled can easily fit over it and lock rotation. The electronics sled, which will be 3D printed, will hold all the primary and backup recovery electronics including the StratoLogger altimeters, Lithium Polymer batteries, and rotary switches. The altimeters will be wired to the primary and backup black powder ejection wells on both the forward and aft bulkheads. The switches will also be accessible externally through the switch band in between the middle and upper airframes.

Figure 6.42 Recovery Bay Assembly

6.5.2 Recovery Electronics

The recovery bay will house the primary and backup StratoLogger CF altimeters, which will be wired to external switches. Every component in the recovery bay is redundant in order to prevent any failure during the recovery events for parachute deployment. When switched on, the altimeters will confirm proper orientation of the launch vehicle. The StratoLogger CF is accurate and cost-effective, while also being simple enough for our purposes of deploying the drogue and main parachutes in a dual event recovery. Although it does not have an accelerometer, we deemed it unnecessary for parachute deployment. The electronics layout is outlined in Figure 6.43.

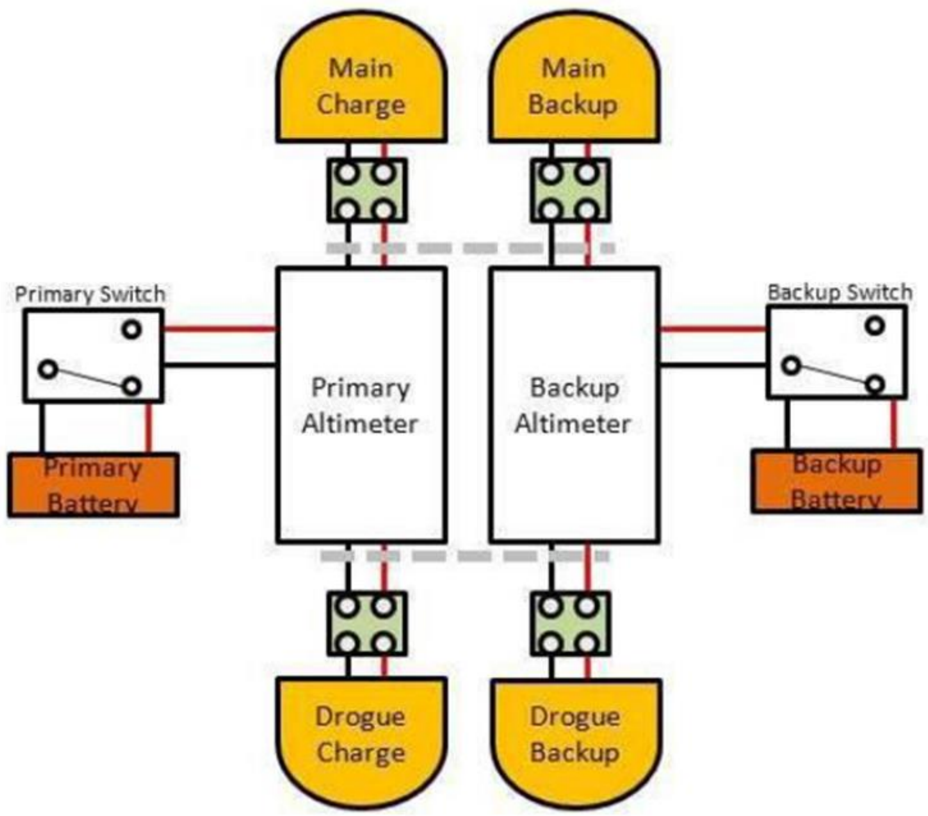


Figure 6.43 Redundant Electronics Layout

The switches will be rotary switches, accessible through holes in the switch band on the coupler connecting the middle and upper airframes. The rotary switch is a 110/220V selector power switch that will be used to switch the outputs of the altimeters to safe or armed. It can be easily turned by a flathead screwdriver, with clear indication which position it is in to ensure safety.

Two lithium polymer (LiPo) batteries will be used, similar to previous years. LiPo batteries are very compact, and much better at handling launch forces than alternative battery options. They are also much more resistant to the effects of cold temperatures and will be used during our test launches done in the northern winter months.

6.5.3 Parachute Selection

The recovery system will be a dual event system. The drogue parachute, used to slow down the initial descent of the launch vehicle, is located in the upper airframe above the recovery bay and will be released at apogee. The main parachute will fully slow down the descent of the launch vehicle so that the landing kinetic energy does not exceed 75 ft-lbf per independent section of the vehicle. It will be located in the middle airframe, directly under the recovery bay. The main chute should be deployed at some altitude after the apogee event, not lower than 500 ft, but the launch vehicle should still be able to land within the 90-second descent time limit.

Using the sizing guide by the University of Idaho [5], we can use weight and drag to solve for the radii of the two parachutes.

$$r = \sqrt{\frac{2mg}{\pi C_d A_p v^2}}$$

Using the following equation, we can use the mass of the heaviest independent section to calculate the weight with gravity as the only acceleration.

$$W = ma = mg$$

Using this weight, we can calculate the drag force with the drag coefficient of the parachute (C_d), the area of the parachute (A_p), the density of air (ρ), and the relative fluid velocity (v).

$$D = \frac{1}{2} C_d A_p \rho v^2$$

Using these equations and verified using OpenRocket flight simulations, we were able to size and select the materials for the two parachutes. Depending on the masses of the independent sections of the launch vehicle, they may change in future design reviews. The drogue parachute will have a diameter of 32 in and a drag coefficient of 0.75 and will be purchased from Spherachutes. The main parachute will have a diameter of 120 in and drag coefficient of 2.20 and will be purchased from Rocketman Enterprises. The main parachute was chosen as a high Cd type chute so that a single main could return the vehicle safely, while being able to fit in the launch vehicle, which would not be true of a correctly sized parachute with a lower Cd. The main parachute will be deployed at 600 ft to ensure that each independent section of the launch vehicle lands with a kinetic energy lower than 75 ft-lbf, as well as minimizing the descent time. Additionally, the main parachute will incorporate a reefing ring from Rocketman Enterprises, to help slow the opening of the parachute and reduce the shock loading on the vehicle. The reefing ring is known to work on any Rocketman parachutes, so they addition of the ring is unlikely to cause issues with opening. Both parachutes will have canopies made of ripstop nylon and will be attached to the independent airframe sections using 1 in tubular nylon shock cord with a total length of 300 in per section.

6.5.4 Parachute Retention & Release

The retention and release system for the parachute has gone through multiple changes since the design proposal, as it has switched from a single bay design to a dual bay design. Thus, the ARRD/Tender Descender configuration has also switched to using redundant black powder charges for deployment, as the former design is more useful for a single bay design. The new recovery system design is a dual event dual bay system, with the drogue parachute housed in the upper airframe under payload, the main parachute housed in the middle airframe, and the recovery bay housed in the coupler directly between the two parachutes as shown in Figure 6.44, along with locations of energetics in the vehicle.

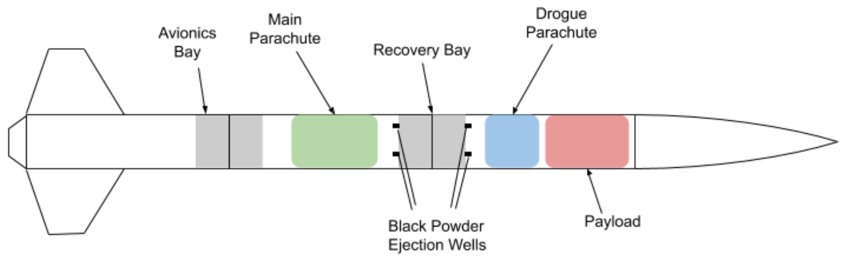


Figure 6.44 Recovery System Configuration

One alternative design was to use CO2 ejection instead of black powder charges for parachute deployment. If CO2 ejection were used, pressurized cartages would eject the parachute once the altimeter sent out the signal to do so. However, CO2 ejection is heavier, more expensive and it is complicated to design redundant systems that would release the main parachute properly. It would take less time and money to perfect the black powder charges, especially as it has been used in previous years.

At apogee, the altimeter will signal for the black powder charges to ignite and deploy the drogue parachute and payload. The shear pins under the forward bulkhead will break, ejecting the drogue parachute from the upper airframe, along with the payload deployment bag. During the apogee event, the main parachute and recovery bay will remain housed within the middle airframe.

Figure 6.45 Recovery Bay Piston

To protect the payload from ejection gases, as piston ejection system will be used, as shown in Figure 6.45. A section of coupler with an epoxied bulkhead will be loaded between the top of the recovery bay and the drogue and payload. When the ejection charges go off, the volume under the bulkhead will be pressurized. The piston will be prevented from moving up in the airframe by the drogue parachute and payload, which are pressed against the nosecone bulkhead. The piston, along with drogue and payload, will be free to fall out of the upper airframe after separation. Shock cord (not shown in the figure) will be looped from the recovery bay through a quick link on the piston, which will attach to the rest of the upper section.

Once the launch vehicle descends to 600 ft, the altimeters will signal the black powder charges to ignite and eject the main parachute, breaking the shear pins between the aft bulkhead and middle airframe. During this second recovery event, the main parachute will deploy, and the recovery bay will also be ejected from the airframe, still attached to the middle and upper airframes with shock cord. Both recovery events are shown in Figure 6.46.

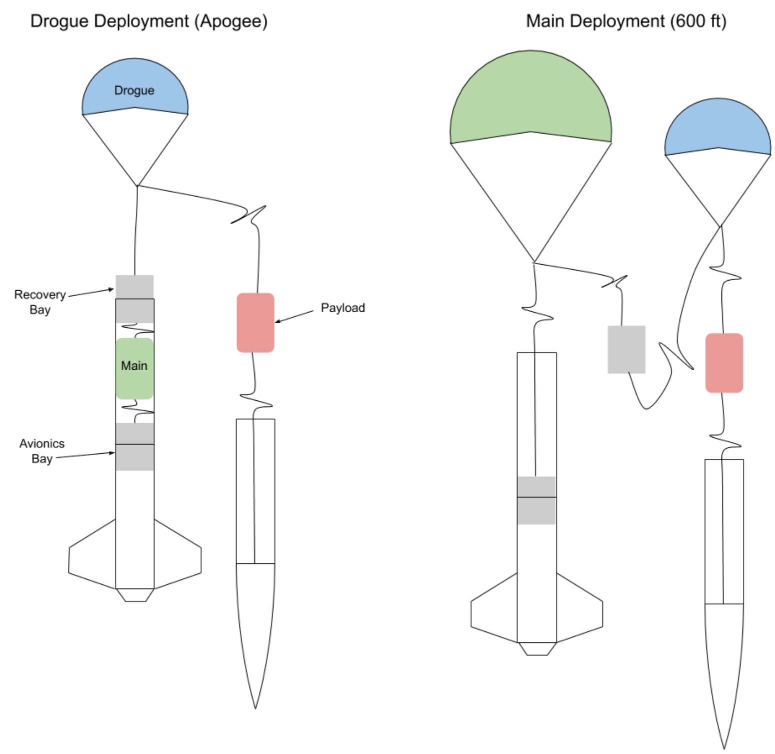


Figure 6.46 Parachute deployment events

6.6 Mechanical Systems

6.6.1 Airbrake Design Overview

The airbrake system is designed to be deployed out of the lower airframe of the rocket, such that the force due to drag can be controlled during the rocket's accent. A controlled accent is essential in guiding the rocket to the target apogee, especially when flying in varying conditions. The airbrakes consist of four fins connected to a guide plate via pins. The guide plate was designed to have three sets of rails per fin in order to increase the stability of the fin's actuation. With the guide plate facing down and the fins on top, an actuator plate fits the middle pins found in each fin using four equiangular spiral rails. The use of the equiangular spiral ensures an equal amount of torque on each pin, thus allowing for the design to spin. The actuator plate is free to spin around its center within the equiangular curves and is attached to a ball bearing in its center. On top of the actuator plate is the gear system, which consists of a larger drive gear that is bolted to the actuator plate, a servo gear that is meshed with the drive gear, and a servo that is attached to the servo gear and used to spin the gear system. Above the gears is a motor plate that helps keep the servo body in a fixed place relative to the actuator plate. The servo is bolted to the motor plate. When properly lined up, the guide plate is bolted to the motor plate, such that the bolts are equidistant between each fin. To connect the airbrakes to the avionics bay, a spine is attached to the center of the guide and motor plates. This allows for the guide and motor plates to remain stationary relative to the actuator plate. Spine rings are bolted around the centers of both the guide and motor plates and attached via pins to the spine. In the case of the actuator plate the spine goes through the center of the ball-bearing, which allows for the actuator plate to spin around the spine with limited friction. When put together, the airbrake system will allow for a reciprocating motion, as the fins are deployed out the side of the rocket to reach the necessary drag.

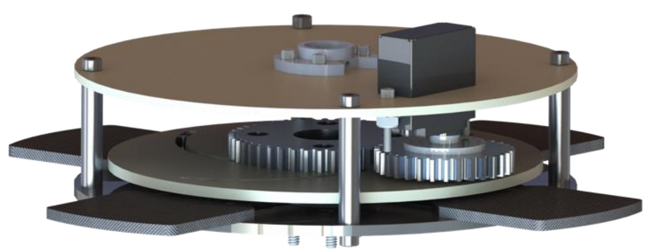


Figure 6.47 Airbrake System Assembly

6.6.2 Airbrake Gear System

The Airbrake gear system's main design requirement was the effective actuation of the airbrakes via a servo. This would be done via the turning of the actuation plate with the spiral slot cutouts. While this system's requirements were simple in scope, there were many design desires which included a modular system which could interface multiple gear ratios and compactness. Further we wanted our gear system to install onto the spine as a singular assembly. These requirements as well as desires drove our design choices. The power transfer system, consisting of two gears, contains a actuator gear seen in Figure 6.50, which is a relatively larger spur gear that connects directly to the actuator plate, and the servo gear Figure 6.49, a relatively smaller gear which connects to a servo mounted above the airbrakes in the figure shown.

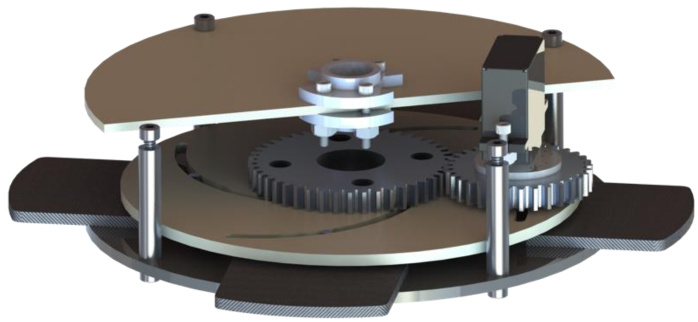


Figure 6.48 Airbrake Gear System Assembly

Our servo of choice is the Hi-Tec 7985 MG servo, we choose this motor for two main reasons. We have experience working with Hi-Tec Servos from prior years. Furthermore, the resolution of the 7985 MG Servo was one of the highest commercially available combined with its high torque rating of up to 10-inch pounds, which we felt was sufficient for effective actuation. These two main factors combined with a mass of only 62 grams led to our decision of the 7985 MG servo. While designing the gearing system, there were three main choices we had for the gearing: Bevel, Worm, or Spur Gearing. While the Bevel Gearing system would provide higher efficiency, it requires a more complex mounting solution while only having a limited gear ratio. On top of this a marginally higher efficiency over the spur gearing was not enough to warrant a switch. The worm gear solution provides compactness; however it is highly inefficient as well as slow at transferring power relative to either spur or bevel gearing due the presence of relatively high amounts of sliding friction. Lastly the spur gearing, our solution of choice, provided mounting and operation simplicity combined with high efficiency, as well as a wide range of gearing ratios that could be used. We have also utilized the spur gearing system in the past in similar applications, and spur gearing provided the simplest and most efficient solution to our design desire of the entire system mounting as a singular piece onto the spine.

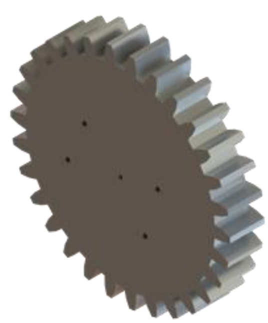


Figure 6.49 Drive Gear

Our current gearing system provides an effective torque of 16 in/lbs. on the actuator plate. Both the spur gears will be purchased from a custom manufacturer of gears. In terms of operation, the gearing system's operation logic is as follows: the servo spins in the appropriate direction to open the airbrakes when a positive voltage is applied to the servo, and it follows that when a negative voltage is applied over the servo's terminals, the airbrakes close. The last main component of the gear system is the motor brace plate, it is a plate with a cutout that holds the servo in place. This component was a result of our design process, after deciding on the spur gearing system, the brace plate was a necessary accommodation for the servo to effectively actuate motion through to the airbrakes.

Figure 6.50 Actuator Gear

6.6.3 Avionics Bay

The airbrakes and electronics are fit into the avionics bay. The main structural component is the middle cylindrical spine. Thus, the airbrake guide plate and motor plate are fixed in place by the bolted spine ring. Additionally, the connection ring is bolted on top of the motor retention system.

The spine was modified from a hexagonal cross section to cylindrical, making airbrake integration easier. Hexagonal holes are also complicated to manufacture. The spine lock was reduced in height to minimize weight, and the 3D printed ABS connection ring was fitted for the lock's specifications. Both bulkheads are made from G10, thus having a reduced thickness as well. The T-shaped Aluminum spine provides structural support to the system. The twist lock mechanism is designed to provide easy access to the electronics. We utilized radial brackets to fix the coupler in place. The U-bolt holds the shock chord during separation, it is also used as a handle to make locking more convenient

Figure 6.51 Avionics Bay Spine

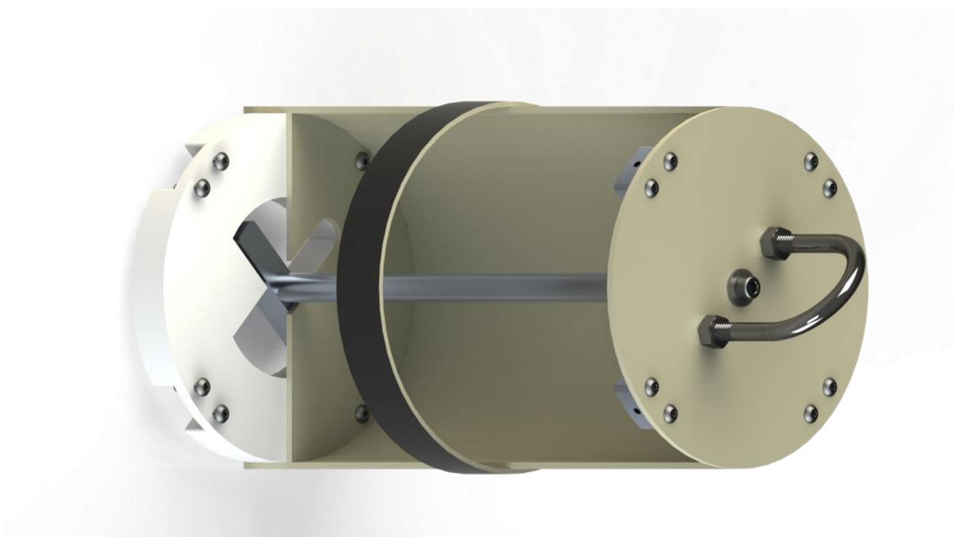


Figure 6.52 Sectioned Avionics Bay View.

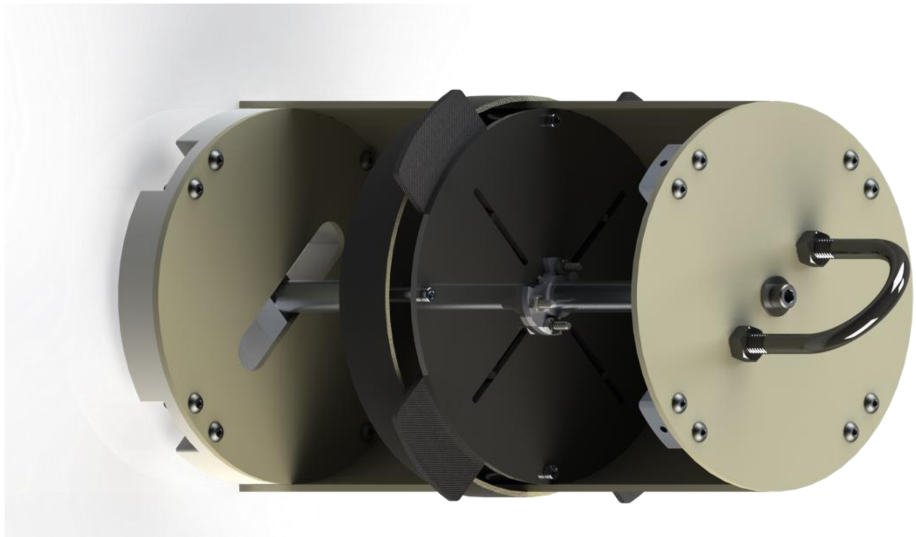


Figure 6.53 Avionics Bay with Airbrakes

6.6.4 Fin Design

The fins for the air brake system need to be able to be deployed and withstand the force of the air as the rocket is traveling upwards to control the apogee of the rocket. The fins need to have a low friction coefficient to keep the fins sliding freely and easily so the system has more control. We also want to keep the air brake system as light as possible. This is how we ended up using carbon fiber, due to its high strength, low weight, and low friction properties. We chose a rounded trapezoidal shape for the fins, such that we minimize the material needed inside the system but maximize the area that is producing the active drag. This allows us to keep the weight of the fins down while maintaining surface area of active drag. The fins have two guide pins and one actuator pin embedded in the fin to guide the fin out of the system and to produce the active drag. We added two guide pins that fit into slots in the guide plate to reduce friction from our previous design. Our previous design involved rails on the guide plate that kept the fins in line while being deployed. The guide pins are a better design because we will have much less friction when deploying the fins. Our guide pins and actuator pins are 1/8-inch diameter and made from 6061 aluminum. These pins will be fit into the carbon fiber airbrakes fins with epoxy. Another option we considered using was a bucking inside the holes inside of the fin for the guide pins. Friction fit is going to be better than the bucking because the bucking will add unnecessary weight without the additional strength.

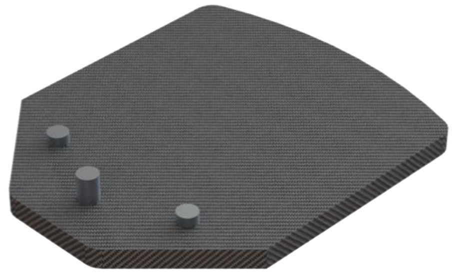


Figure 6.54 Airbrake Fin Assembly

6.6.5 Actuation System

The goal of the actuation system is to be able to deploy and retract the airbrake fins at any time during ascent to control our apogee. The actuation system is driven by the airbrakes gear system mentioned previously in section 6.6.2. The actuation system utilizes an actuator plate with 4 equiangular slots to drive the fin pins when the plate is rotated by the airbrake gear system. These slots as seen in section 6.6.5.1 allow for us to apply equally torque on the system at any time keeping deployment and retraction smooth. The actuation system also utilizes a guide plate with 1 slot for each of the fin pins to move the fins and 2 guide slots for each fin so the guide pins can keep the fins on track reducing the risk of fin deployment errors. The guide plate can be seen below in section 6.6.5.2.

This actuation system will give us the ability to easily control the surface area of the fins exposed outside the body of the rocket by changing the angle the servo in the airbrake gear system is spun. This ability to control the exposed surface area will allow for us to control the drag produced by the airbrakes at any time during ascent allowing for us to adjust to simulation and launch time data increasing our ability to accurately hit our target apogee.

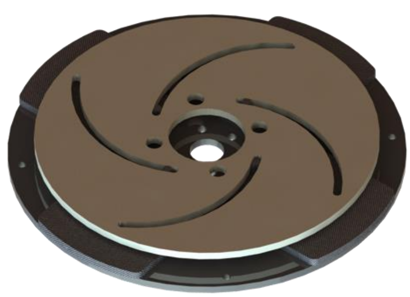


Figure 6.55 Actuation System While Closed

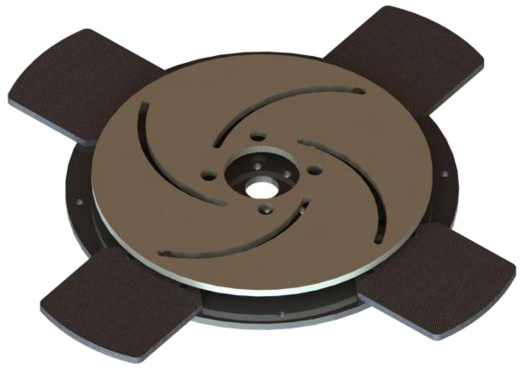


Figure 6.56 Actuation System While Open

6.6.5.1 Actuator Plate

The airbrakes actuator plate's main purpose is to drive the fin pins in the airbrake actuation system. This is accomplished by using 4 equiangular slots as seen in Figure 6.57: 1 slot per fin pin. As the actuator plate is spun by the attached gear system the fin pins are pushed either inwards our outwards of the center of the plate allowing for the pins to be driven in the guide plate slots deploying or retracting the fins from the rocket body respectively. The actuator plate will be water jet cut out of g10 fiber glass to allow for rigidity of the component.

Figure 6.57 Actuator Plate

6.6.5.2 Guide Plate

The guide plate's main purpose in the air brake system is to guide the fins smoothly out of the rockets body tube. The selected design incorporates 3 slots for each of the 4 fins: 1 slot for the fin pin and 1 slot on either side of the fin pin slot for the guide pins. A previous design included rails in place of guide pins to control the fins movement. The use of guide pins was ultimately selected to reduce friction. The guide plate will be laser cut out of Delrin.

Figure 6.58 Guide Plate

6.7 Avionics

6.7.1 Controller

In order to process data from the sensors and control the airbrake system on the rocket, a microcontroller is needed. The microcontroller that the team decided to use is the Teensy 3.2, with the Teensyduino add-on to program in the Arduino environment. The final subsystem will be a single board solution that integrates the Teensy microchip as one of the components.

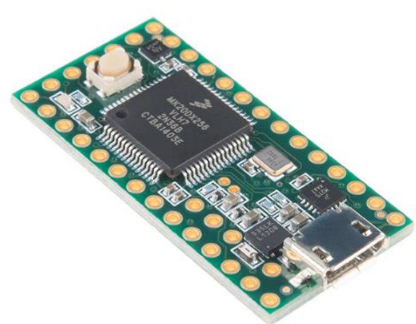


Figure 6.59 Teensy 3.2 microcontroller from Sparkfun

The other option the team was considering was the STM32F3Discovery, with a STM32F303VCT6 microcontroller.

Figure 6.60 STM32F3Discovery from STMicroelectronics

Both the STM microcontroller and the Teensy microcontroller use a 32-bit system with a Cortex-M4 core and have 256 kB of memory, but the Teensy has more onboard RAM than the STM. It can temporarily store up to 64 kB of information whereas the STM can temporarily store only up to 48 kB. One benefit of the STM over the Teensy is that it contains a built-in accelerometer and gyroscope. However, the Teensy is more compact and user friendly since its programed in the Arduino environment, which is why the decision to choose the Teensy with an external accelerometer was made.

6.7.2 Sensors and Datalogging

An important role of the avionics system is to collect and log data from multiple sensors on the rocket. These sensors will track the rocket's acceleration, altitude, and orientation, and this data will be used to control the airbrakes. We would like to have multiple methods of determining the rocket's position and orientation for more accuracy using sensor fusion. This ensures that there is at least one position measurement in the event of a failure. The selected sensors are discussed below.

MPU-6050 Accelerometer. This is a triple axis MEMS accelerometer and gyroscope that can measure up to 16 g accelerations. Most other three axis accelerometers have a maximum of 4 or 8 g of acceleration, so we decided with this 16g accelerometer to stay well within the range of acceleration. The accelerometer will be used to track acceleration, and the data can be integrated to measure position and velocity. This accelerometer can measure the accelerations we expect the rocket to produce, and we are familiar with using it.

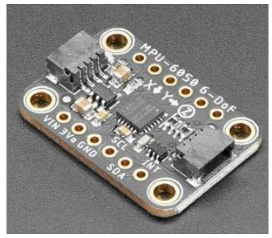


Figure 6.61 MPU-6050 Accelerometer board from Adafruit MPU-6050

MPL3115A2 Barometer. This is a pressure sensor with a 1.5 pascal resolution, corresponding to an altitude change of 0.3m. The barometer will help keep track of altitude accurately, without the need for integrating acceleration which will lead to less error. This accelerometer was a good combination of accuracy vs price.

Figure 6.62 MPL3115A2 Barometer from Adafruit

MLX90393 Magnetometer. This is a triple axis magnetometer with a resolution of 0.161 μT. This will be used to determine the rocket's orientation relative to earth's magnetic field. This will be used in conjunction with the gyroscope in the MPU to accurately track orientation.

Figure 6.63 MLX90393 Magnetometer board from SparkFun

We decided on our sensors based on the accuracy of their readings and because they can measure in all three axes. These sensors are proven to be reliable and precise.

Another option we considered was the BNO055 9 degree of freedom (DOF) sensor, which combines an accelerometer, gyro, and magnetometer on one board. We decided against this due to the limited range of the accelerometer and because we already have and are familiar with the MPU6050. A second option we decided against was the LIS331HH. This accelerometer had a greater range, but it did not have an integrated gyroscope. Ultimately, having an integrated gyroscope was considered more necessary than having a greater range.

A second barometer we considered was the BMP280. This sensor has a greater resolution and more accurate temperature sensor than the MPL3115A2, however it does not have a built-in altitude calculator. We decided against the BMP280 because the difference in its resolution and temperature readings were not great enough to validate the trade-off of the built-in altitude calculator.

We will also have a storage device to record the sensor data onboard the rocket. We have not yet decided on the specific storage device, but we will likely use either a flash memory chip or a MicroSD card.

6.7.3 Telemetry and Inertial Sensing

The avionics bay will provide live telemetry and tracking data to the ground station throughout the duration of the flight and continuing after the rocket has landed. This will allow for the team to see important position, velocity, and sensor data while the launch vehicle is in flight. Position tracking for the rocket will be accomplished using a NEO-M9N GPS. The team will utilize a LoRa RFM-95W radio transceiver to transmit GPS and sensor data to the ground station.

The team will be using long range radio (LoRa) for communications between the rocket and ground station. LoRa was chosen for communication because it allows for signals to be sent and received over much greater ranges than alternatives such as Wi-Fi or Bluetooth. Two different frequency LoRa transceivers had been under consideration: 900 MHz and 915MHz. LoRa transceivers are also available in the 433 MHz range, but this is not a viable option for our project because United States regulations do not permit data transmission on this frequency for an extended period of time like the duration of our flight. The difference in radio frequencies represents a tradeoff between speed of transmission and range. Lower frequencies can generally operate at a greater range but offer reduced transmission rates compared to higher frequencies. It was decided that both the 900 MHz and 915 MHz models have sufficiently high transmission rates to support our data transmission needs, so the team elected to use the 900 MHz model to optimize for greater range capability. Sensing and GPS data will also be stored onboard the rocket on a MicroSD or IC flash chip to be recovered after the flight in the event that telemetry data is not transmitted as expected.

Figure 6.64 Adafruit LoRa RFM-95W 900 MHz

The avionics bay with also feature a GPS module to collect position data for the rocket. The GPS will serve as the primary instrument for measuring the position and velocity of the rocket during flight. Additionally, the rocket will continue to transmit GPS data after the rocket has landed to assist with location and recovery of the rocket in the event that line of sight is lost. Several GPS options were considered including the MTK3339 and NEO M8M. The NEO-M8M model offers a few key advantages over the MTK3339 such as a greater update rate and slightly improved accuracy of position and velocity data. The downside however is that this chip is not manufactured on a breakout board which would make it more difficult to work with, particularly during testing. Since submitting the proposal, the team discovered the NEO-M9N which is an updated model of the NEO-M8M. This GPS is an ideal choice for our purposes because it has similar advantages to the NEO-M8M with a higher update speed and better accuracy, while it is also available on a breakout board manufactured by SparkFun. The NEO-M9N has a max update rate of 25MHz and is accurate to within 1.5m for horizontal position and 0.05m/s for velocity. For testing purposes, the team will be using the breakout board option with a u.FL connector in order to attach an external 10mm GNSS Antenna to collect satellite data. In the final subsystem design, the standalone GPS chip will be integrated into the custom avionics board, and the external antenna will likely be connected to this custom board using a u.FL connector.

Figure 6.65 NEO-M9N, u.FL GPS

Figure 6.66 10mm GNSS Antenna

6.7.4 Dynamics and Control

One of the primary functions of the avionics system is to control the servo which operates the airbrakes which are described in Section 6.6.1. To reach the team's target altitude precisely, it is necessary to implement a control law.

Before designing the control law, it is necessary to develop a dynamical simulator of the vehicle. This will allow the algorithms to be tested in a safe, simulated, environment before being implemented on the actual vehicle. For this purpose, the team is considering using either MATLAB or Simulink (a MATLAB extension). Because MATLAB is a text-based programing language, this would require the team to write the simulator from scratch but may give more fine control over the systems functionality. On the other hand, Simulink uses graphical programing which is more visual. Additionally, tools such as the aerospace blockset can simplify the work. The team is also considering the fidelity of the model.

Primarily between choosing a 3 degree of freedom model or a 6 DOF model. Using a 6 DOF model will be more accurate but is significantly more complex to design than a 3 DOF model.

The team is looking at several different control algorithm options for the airbrake system. One of the most basic options is to use a Proportional, Integral, Derivative (PID) controller. This involves computing an error function, representing the difference between the current state and the desired stat. Being multiplied by specified constants, the error function, its integral, and its derivative are added to compute a control input. Selecting the PID constants is relatively straight forward and can be done either through trial and error or one of a number of tuning methods. While the simplicity of PID control makes it relatively easy to design, it also means that it is generally only useful for single input single output (SISO) systems. Because the rocket cannot be described by a single state, this will likely make it more difficult to implement the algorithm.

$$u = K_p e(t) + K_i \int_0^t e(t)dt + K_d \frac{de(t)}{dt}$$

Equation 2 The general transfer function of a PID controller

Another option is to use full state feedback. This involves designing a k matrix such that multiplying the state by it produces an output vector. Unlike PID, full state feedback is ideal for multiple input multiple output (MIMO) systems. The k matrix can be designed using manual pole placement or through more advanced methods such as a linear quadratic regulator (LQR), which creates a k matrix that optimizes the control costs and the state.

$$\vec{u} = -k\vec{x}$$

Equation 3 The general equation for full state feedback

Once the team selects a control algorithm, it will first be designed and validated using the dynamical model of the vehicle. The final algorithm will be implemented on the avionics board within the main flight code.

6.8 Subscale

The subscale launch vehicle for WPI HPRC will be used primarily as a test bed for a basic data collection system, and to validate the avionics simulator of a rocket's flight. The COVID-19 pandemic has precluded launching subscale rockets of similar sizes to past years, so this rocket will fly as a model rocket, on a single use G79W-10 motor from Aerotech.

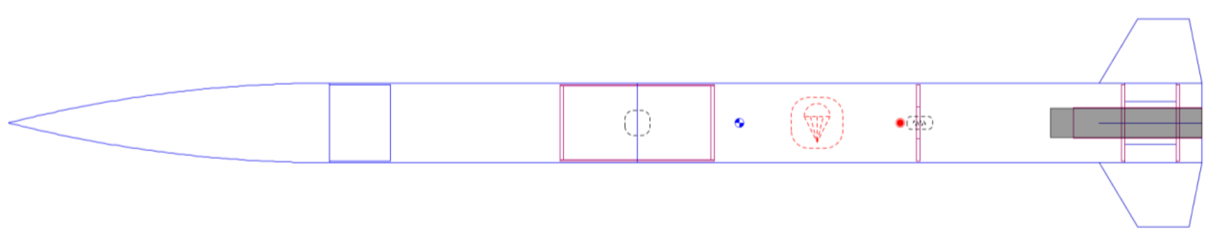


Figure 6.67 Subscale Vehicle Design

The subscale vehicle will be 46.5 in long, with a diameter of 3.1 in. The body tubes will be made out of kraft paper, and the centering rings, bulkheads, and fins will be made from 1/8" thick plywood. The nosecone will be ogive, with a length of 12.5 in, and will be made of plastic.

Despite the small size of the vehicle, the subscale will contain 2 electronic systems, specifically a single StratoLogger CF for primary ejection (with motor ejection serving as a backup) and a data logging system using breakout boards for the Teensy 3.2, MPU 6050 6DOF IMU, and the MPL 3115A2 pressure sensor. Both systems will collect data on the vehicle's flight, which will be used to verify the results of the team-created flight simulator. The simulator will be used to tune the airbrake control system, so it is essential that the results it produces are accurate.

To collect as much data as possible, and to verify that the simulator is accurate not just for a standard rocket, but for when the airbrakes are deployed as well, the team will fly the rocket 3 times. Onboard the rocket will be an interchangeable, static airbrake system. Before launch, 4 panels will be bolted into the vehicle, and will extend varying amounts out of the side of the airframe depending on the desired airbrake configurations.

Figure 6.68 Static Airbrake System

The 3 configurations that will fly will be the zero extension configuration, where the panels extend to be flush with the airframe, the full extension configuration, where the panels extend out to a proportional distance from the airframe, based on based on the full scale extension and diameter, and a half extension configuration, where the panels extend half the distance from the airframe to the fully extended configuration.

When the simulator is completed, we can input the data from each launch, and compare it to a launch simulated with the same conditions. This will allow us to verify that our simulator provides an accurate representation of the flight of a rocket. Additionally, we avoid complexity in scaling the subscale aerodynamically, since as long as the subscale and full scale are not significantly different, the results of the subscale flight will allow us to verify that the simulator functions, while not being directly comparable to the full scale vehicle.

7 Payload Design

7.1 Payload Overview

This year's payload has been designed to be ejected from the airframe at apogee, remain tethered descending with the rocket until 1000ft when it will detach and open its parachute descending on its own. Upon landing the payload will release its parachute and self-right itself and stabilize itself within 5 degrees of level. From there it will take and transmit a 360 photo to the ground station. In order to perform these functions, the mechanisms are split into Retention, Self-righting, Stabilization, Photography, and Electronics/Programming. The payload measures 6in OD by 8in long and weighs 4.2lbs.

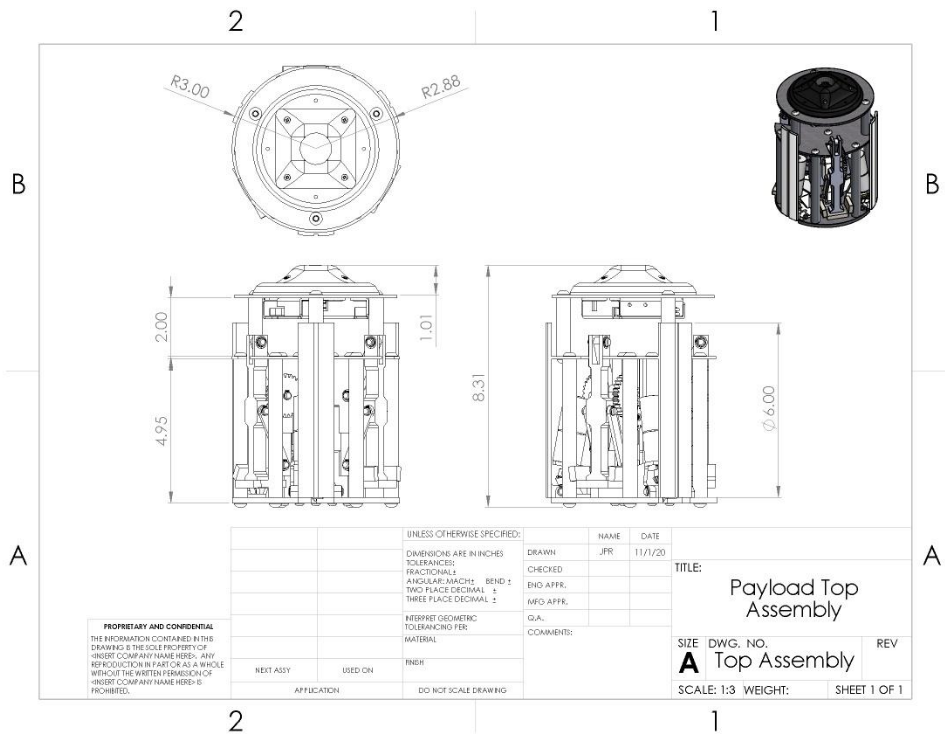


Figure 7.1 Payload Top Dimension Drawing

Figure 7.2 Payload Upon Landing

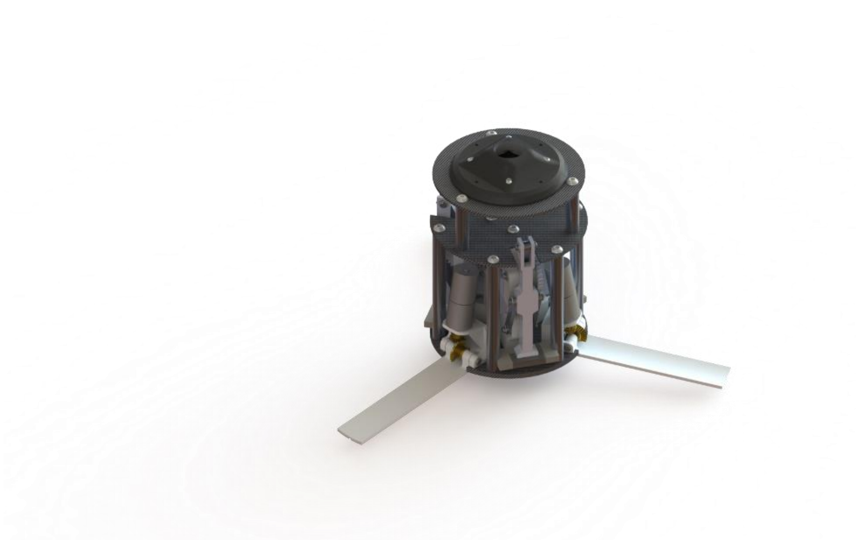


Figure 7.3 Payload After Self Righting

Figure 7.4 Payload Stabilizing

7.2 Retention and Deployment

7.2.1 Design Goals

The goal of the payload retention and deployment systems are to retain the payload during flight, deploy the payload at the designated altitude, and release the parachute lines upon safely landing. This system is comprised of three sections. The first section, alignment pins, deals with the retention of the payload during flight. The second, the Tender Descender, deals with the payload's detachment from the rocket's main body. The third section is the Rotary latch which is responsible for releasing the parachute connection after landing.

7.2.2 Alignment Pins

For retention of the payload inside the upper airframe, the payload will be retained by 3 aluminum standoffs. These standoffs will slide into three equidistant guide holes in the topmost carbon fiber plate of the payload. This system restricts the payload from rotation, translation, and overall vibration within the upper airframe. When the separation charge inside the rocket is triggered, the payload is pulled out under the drogue deployment. The payload will slide on the aluminum standoffs and then out of the bottom of the upper airframe with the rest of the retention systems.

7.2.3 Tender Descender

After the payload vehicle has exited the upper airframe, the payload will remain connected to the rocket until the desired payload release altitude. At 1000ft AGL, the payload will separate from the main rocket bodies and begin it's decent. The payload will utilize a Rocketman 60in (5ft) parabolic parachute to descend giving it an 18.9ft/s decent rate at a landing energy of 27.81 lbft. This parachute will be deployed at 900ft AGL via two Jolly Logic Chute Releases. Jolly Logics retain the parachute from deploying until a predetermined altitude. We have decided to use Jolly Logics because of their reliable and easy use in previous launches. We are using two Jolly Logics for increased redundancy. To control when the payload separates, we will have a Fruity Chutes Tender Descender Aluminum L2

Recovery Tether as shown in Figure 7.5. This aluminum and stainless-steel mechanism allows for two quick links to be retained until a black powder charge and electric match allows for separation. The Tender Descender uses a 0.25-gram charge to eject the link retainer assembly holding the two-quick links to the aluminum housing as shown in Figure 2.

Figure 7.5 Fruity Chutes Tender Descender.

Figure 7.6 Fruity Chutes Tender Descender upon separation.

Our Tender Descender will be suspended between the rocket airframe's and the payload via shock cord. Running electrical cables from the payload or the rocket's airframe pieces to the Tender Descender has a high risk of snagging under deployment. The decided solution was to mount the altimeter and the power supply onto the Tender Descender itself so there are less degrees of freedom for cables to get caught in as shown in Figure 7.7.

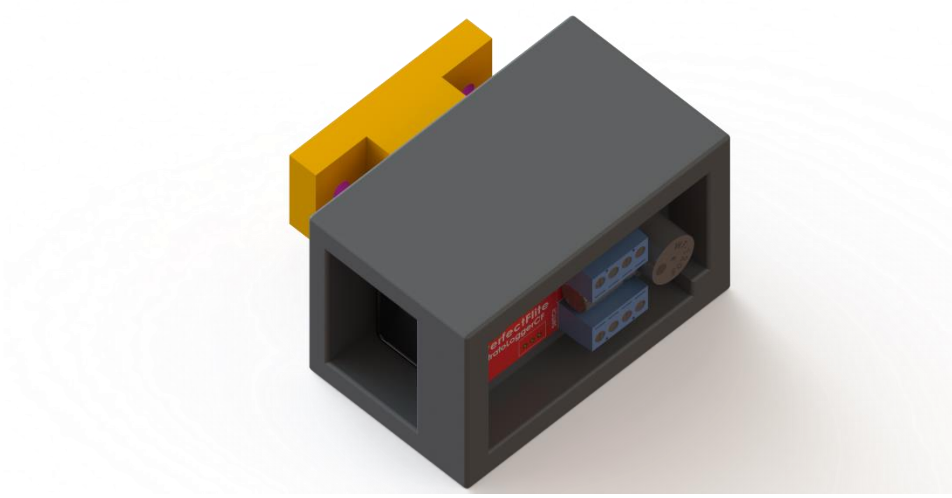


Figure 7.7 Tender Descender and Altimeter Package.

The altimeter we decided to use was the PerfectFlite StratoLogger CF Altimeter, as shown in Figure 7.8, which is a barometer altimeter system. The StratoLogger CF has proved reliable to the team in previous years of competition.

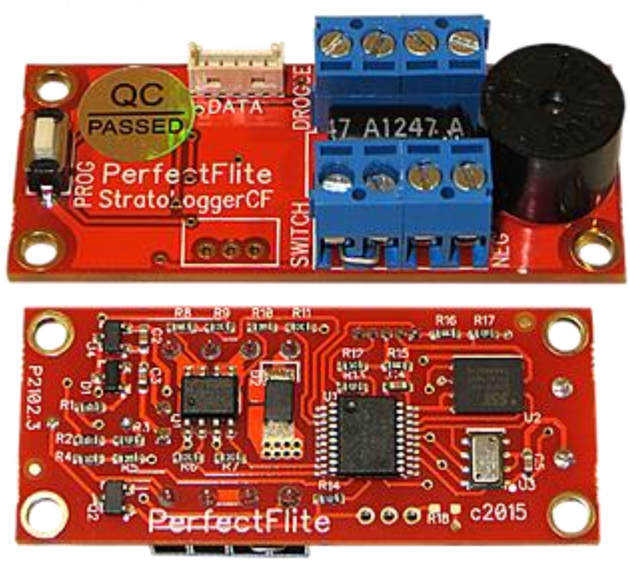


Figure 7.8 PerfectFlite StratoLogger CF Altimeter.

The StratoLogger CF is significantly cheaper than similarly capable altimeters that can be seen in Figure 7.9. The benefit of the StratoLogger CF to the Tender Descender application is the small form factor and low power requirements. Compared to other altimeters, the StratoLogger CF is compact in size measuring 2”x0.84”x0.5”. The StratoLogger CF is also attractive for its simplicity where a more complex altimeter such as the Featherweight Raven4 provides more opportunities for user programming errors.

Display columns: reset | ☑ Dimensions | ☑ Sensors | ☑ Interfaces | ☑ Flight reporting | ☑ Pyro control | ☑ Battery/Voltage | ☑ Features | ☐ Show old models

Model Rocket Altimeters: Comparison Guide

Compiled by: http://RocketsEtc.com/ Help

Basic information					Dimensions & Weight					Sensors		Interfaces			
Vendor	Model	Price	Kit	Max Alt	Fit	L	W	H	Wt	Baro	Accel	Audio	LED	LCD	PC
PerfectFlite	FireFly	$ 25	$ 25	100,000	18	1.1"	0.68"	0.31"	3g	✓	-	-	✓	-	-
Eggtimer	Quark (DIY)	$ 20	$ 25	30,000	24	1.85"	0.75"	0.4"	5g	✓	-	✓	-	-	✓
PerfectFlite	APRA	$ 30	$ 30	100,000	18	2.75"	0.55"	0.62"	9g	✓	-	✓	-	-	-
Estes	Estes Altimeter	$ 40	$ 40	10,000	24	2.17"	0.71"	0.71"	12g	✓	-	-	-	✓	-
Eggtimer	Eggtimer (DIY)	$ 35	$ 40	30,000	29	3.9"	1"	?"	20g	✓	-	✓	✓	-	✓
Eggtimer	Quantum WiFi (DIY)	$ 40	$ 40	60,000	24	2.5"	0.9"	0.5"	15g	✓	-	✓	-	-	✓
MissileWorks	RRC2+	$ 45	$ 45	40,000	24	2.28"	0.925"	0.5"	10g	✓	-	✓	-	-	-
Jolly Logic	AltimeterOne	$ 50	$ 50	30,000	24	1.93"	0.71"	0.57"	10g	✓	-	-	-	✓	-
Altus Metrum	MicroPeak	$ 30	$ 60	100,000	18	0.7"	0.55"	0.26"	2g	✓	-	-	✓	-	✓
Jolly Logic	AltimeterTwo	$ 70	$ 70	30,000	24	1.93"	0.71"	0.57"	10g	✓	✓	-	-	✓	-
Eggtimer	Proton (DIY)	$ 70	$ 70	60,000	38	3.25"	1.25"	0.5"	20g	✓	✓	✓	-	-	✓
Adrel	ALT-BMP	$ 65	$ 77	30,000	13	0.76"	0.32"	0.16"	1g	✓	-	-	-	-	✓
PerfectFlite	StratoLoggerCF	$ 55	$ 78	100,000	24	2"	0.84"	0.5"	11g	✓	-	✓	-	-	✓
PerfectFlite	Pnut	$ 55	$ 78	100,000	18	2.5"	0.59"	0.45"	7g	✓	-	✓	-	-	✓
Altus Metrum	EasyMini	$ 80	$ 80	100,000	24	1.5"	0.8"	0.56"	7g	✓	-	✓	-	-	✓
MissileWorks	RRC3 Sport	$ 70	$ 95	40,000	24	3.92"	0.925"	0.5"	17g	✓	-	✓	-	-	✓
Jolly Logic	AltimeterThree	$ 100	$ 100	30,000	24	1.93"	0.71"	0.57"	10g	✓	✓	-	-	✓	✓
MissileWorks	RRC3 Xtreme	$ 80	$ 105	100,000	24	3.92"	0.925"	0.5"	17g	✓	-	✓	-	-	✓
Entacore	AIM USB	$ 115	$ 115	40,000	?	2.76"	1"	0.6"	?g	✓	-	✓	-	-	✓
Jolly Logic	Chute Release	$ 130	$ 130	30,000	42	2.1"	1.2"	0.4"	17g	✓	-	-	-	-	-
Eggtimer	TRS (DIY)	$ 90	$ 140	30,000	38	3.9"	1.3"	?"	25g	✓	-	✓	-	✓	✓
Featherweight	Raven 3	$ 155	$ 155	100,000	?	1.8"	0.8"	0.55"	7g	✓	✓	✓	-	-	✓
Marsa Systems	Marsa33LHD	$ 200	$ 200	?	38	3.7"	1.3"	0.5"	?g	✓	✓	✓	-	✓	✓
Altus Metrum	TeleGPS	$ 215	$ 215	100,000	29	1.5"	1"	0.37"	13g	-	-	-	-	-	✓
AED Electronics	RDAS Tiny	$ 300	$ 300	40,000	38	3.15"	1.1"	?"	?g	✓	✓	✓	-	-	✓
Altus Metrum	EasyMega	$ 300	$ 300	100,000	38	2.25"	1.25"	0.56"	14g	✓	✓	✓	-	-	✓
Altus Metrum	TeleMetrum	$ 300	$ 400	100,000	29	2.75"	1"	0.56"	18g	✓	✓	✓	-	-	✓
Entacore	AIM XTRA	$ 325	$ 420	100,000	?	4.315"	1.18"	0.6"	?g	✓	✓	✓	-	-	✓
Altus Metrum	TeleMega	$ 400	$ 500	100,000	38	3.25"	1.25"	0.56"	25g	✓	✓	✓	-	-	✓

	Flight reporting			Pyro control		Battery / Voltage			Features			More info		
Model	Post-flight	Record	Mem	Outputs	Config	Holder	Min V	Max V	GPS	Radio	Tele	Competition	OS	Datasheet
FireFly	Click!	Max Alt	1	-	-	CR1025	3	3	-	-	-			Link
Quark (DIY)	Max Alt	Max Alt	1	2	6 Presets	-	3.7	16	-	-	-	-	Win only	Link
APRA	Click!	Max Alt	1	-	-	A23	4	16	-	-	-	✓		Link
Estes Altimeter	Max Alt	Max Alt	10	-	-	A11	6	6	-	-	-			Link
Eggtimer (DIY)	Max Alt	Click!	32	2	Software	-	4.5	30	-	-	✓	-	Win only	Link
Quantum WiFi (DIY)	Click!	Click!	15	2	Software/WiFi	-	6	16	-	-	✓	-	Win,Mac,Linux,Android,iOS	Link
RRC2+	Max Alt	Max Alt	1	2	8 presets	-	3.5	10	-	-	-	✓		Link
AltimeterOne	Max Alt	Max Alt	100	-	-	Built-in	-	-	-	-	-	✓		Link
MicroPeak	Max Alt	Click!	1	-	-	CR1025	3	3	-	-	-	✓	Win,Mac,Linux	Link
AltimeterTwo	Click!	Click!	100	-	-	Built-in	-	-	-	-	-	✓		Link
Proton (DIY)	Click!	Click!	14	6	Browser/WiFi	-	4.5	16	-	-	✓	-	Win,Mac,Linux,Android,iOS	Link
ALT-BMP	Max Alt	Click!	1	-	-	-	3.6	6	-	-	-	✓	Win only	Link
StratoLoggerCF	Click!	Click!	16	2	Software	-	4	16	-	-	✓		Win,Mac	Link
Pnut	Click!	Click!	31	-	-	Built-in	3.7	3.7	-	-	-	✓	Win,Mac	Link
EasyMini	Max Alt	Click!	1	2	Software	-	3.7	5	-	-	-	-	Win,Mac,Linux	Link
RRC3 Sport	Max Alt	Click!	15	3	Software	-	3.5	10	-	-	✓		Win only	Link
AltimeterThree	Click!	Click!	100	-	-	Built-in	-	-	-	-	-		Android, iOS	Link
RRC3 Xtreme	Max Alt	Click!	15	3	Software	-	3.5	10	-	-	✓		Win only	Link
AIM USB	Max Alt	Click!	1	2	Software	-	3.7	12	-	-	-	-	Win only	Link
Chute Release	-	-	-	1	9 presets	Built-in	-	-	-	-	-			Link
TRS (DIY)	Click!	Click!	32	2	Software	-	4.5	30	✓	✓	-	-	Win only	Link
Raven 3	Max Alt	Click!	5	4	Software	-	3.8	20	-	-	-		Win only	Link
Marsa33LHD	Click!	Click!	2	4	Software	-	7	12	-	-	-		Win only	Link
TeleGPS	Click!	Click!	54	-	-	-	3.7	5	✓	✓	-	-	Win,Mac,Linux	Link
RDAS Tiny	Max Alt	Click!	1	4	Software	-	9	15	-	-	-		Win only	Link
EasyMega	Max Alt	Click!	4	6	Software	-	3.7	5	-	-	-	-	Win,Mac,Linux	Link
TeleMetrum	Click!	Click!	8	2	Software	-	3.7	5	✓	✓	-	-	Win,Mac,Linux	Link
AIM XTRA	Click!	Click!	6	4	Software	-	3.5	8.4	✓	✓	-	-	Win only	Link
TeleMega	Click!	Click!	4	6	Software	-	3.7	5	✓	✓	-	-	Win,Mac,Linux	Link

Figure 7.9 Altimeter Comparison Chart from RocketsEtc.com

The team also looked for altimeters which had an internal battery but there are currently none on the market which offer internal batteries and wire leads for electric matches. Therefore, we knew that we had to also connect a power source to the altimeter and Tender Descender assembly. The StratoLogger CF's low power requirements enable us to use a standard 9V consumer battery. Although we are planning to use Energizer 9V Lithium Batteries which are rated to -40 degrees Celsius, we will likely switch to a hobby LIPO battery such as the Turnigy nanotech 370mah 3S 25~40C LiPo Pack. The StratoLogger CF and 9V battery will be in a 3D printed and PLA housing that is screwed into the back of the Tender Descender. (Figure 3)

7.2.4 Rotary Latch

The goal of retention is to hold the shock cord to the payload while also being able to release the parachute once the payload hits the ground. To achieve this goal, we will use a pre-built electronic rotary latch as shown in Figure 7.10. We chose the R4-EM-R21-161 model, having dimensions of 2.72"(69mm) x .712" (18.2mm) x 2.56" (65mm) from Southco because it has ¼-40 standard threaded through holes for simple installation, its high load capacity, and the latch is electronically actuated.

Figure 7.10 R4-EM-R21-161.

We chose this design because it reduces the complexity of our overall system by having all the electronics and motors in one box. It also reduces the wiring complexity since all the wires come from one place. This system will work by having a stainless-steel ring connected to the shock cord, and that ring will be held down by the rotary latch. Upon deployment of the payload's parachute, the deployment forces will be transmitted through the shock cord and into the ring that the shock cord ties to. This ring then transmits the force to the lever arm of the rotary latch where the force is dissipated in the payload structure. When the electronics register that the payload has landed, the latch will be signaled to let the parachute-shock cord go (by opening the latch). This design is straight forward and efficient system to use. Other designs possibilities are to make our own rotary latch with a worm gear and servo. This would allow us to have more control over how and where the mechanism is implemented but it would take more time and space to implement. Buying a prebuilt system ensures that the latch meets our design requirements without the need for rigorous testing.

The other option was a three ringed retaining system that sky diver's use on their parachutes.

Stage 1: Stage 2: Stage 3:

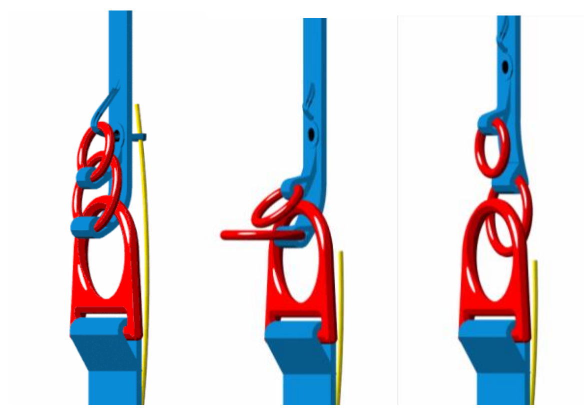


Figure 7.11 Three stages of three ring mechanism.

This system works by having the three rings set up so that they form two class two leavers. When the yellow cable is pulled, the mechanism can separate as seen in Figure 7.11. The benefit of this is that the retention force is significantly less than directly retaining the shock cord. This system has never been used by our team and has a more complicated set up and installation requirements than the pre-built switch.

7.3 Self-Righting

7.3.1 Design Goals

The goal of the self-righting system is to bring the payload to an upright position once it has landed. This system does not need to stabilize the payload to within 5 degrees of its vertical axis as this is the purpose of the stabilization system described in Section 7.6. In order to accomplish our goal, it was decided that a mechanism was needed that would be capable of acting as a lever arm which, by pushing off the ground, would bring the payload into the upright position. This lever arm took the form of multiple 'petals' which would be located equidistantly along the outside circumference of the payload. Originally, we wanted six petals in total, but this design proved too difficult to work with when considering the size of the driving mechanisms behind the petals. Having six petals overcomplicated the entire design and as such the petal count was reduced by half, to three petals. For the mechanism driving the petals, the design needed to be small due to space limitations and produce enough torque to lift the payload. The required torque and its calculations can be found in Required Torque. It was also desirable for this driving mechanism to be able to independently deploy and control the petals such that the payload could intelligently deploy when on uneven terrain. Another consideration that heavily influenced the design process was the necessity to create a system which had low complexity and was easy to manufacture, as this would simplify production, assembly, and operation of the system. The design should also keep the payload's center of mass as low as possible to lower the required torque to upright the payload, allowing our methods of producing this torque to be less limited by higher torque requirements. Additionally, the driving mechanisms should be able to produce much of the torque needed to lift the payload, independent of each other, such that any single petal can lift the payload on its own, or at least produce a significant amount of the torque necessary to do so. With these design considerations in mind a decision matrix was developed to help decide the most capable design, seen in Figure 7.12.

7.3.2 Leading Design

Design	Complexity	Force	Cost	Size	Score
Gearmotor	3	5	3	4	77
Torsion Spring	5	1	5	5	51
Gas Springs	3	3	2	1	44
Linear Actuators	3	3	1	2	43
Multigear Drive	1	4	2	2	34
Weight	40	30	5	25	100

Figure 7.12 Decision matrix for the self righting system.

The mechanisms listed above (Figure 7.12) were evaluated using a decision matrix after some brainstorming and preliminary investigations. Each design was evaluated on categories deemed important by the payload subteam, weighted in accordance with the needs of the teams. Each category ranges from 1-5 with 5 being the best. The weights can be seen at the bottom of the above table. Reducing complexity and part count was a major design goal and therefore it received a 40% weight on the final score. Force is the expected force produced by the design. Cost is the expected cost of all components involved in the design. Size is the amount of volume taken up by the system in the payload. The score is calculated by dividing the sub score in each category by 5 and then multiplying by the category weight to arrive at a proportional score out of 100.

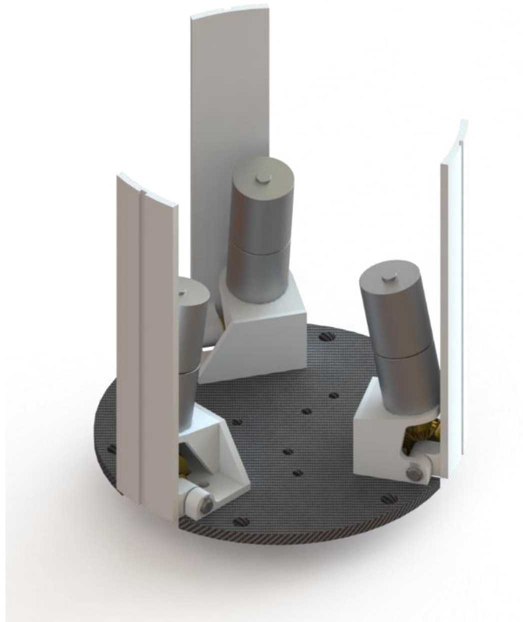


Figure 7.13 Self-righting system in its stowed configuration.

The basic operation of the selected design (pictures in Figure 7.13 and Figure 7.14), that of the gearmotor, is an electric motor with a built-in gearbox driving the righting petals through a bevel gear to rotate the torque. The righting petals are three sections of the payload wall that rotate against the ground to push the body of the payload upright.

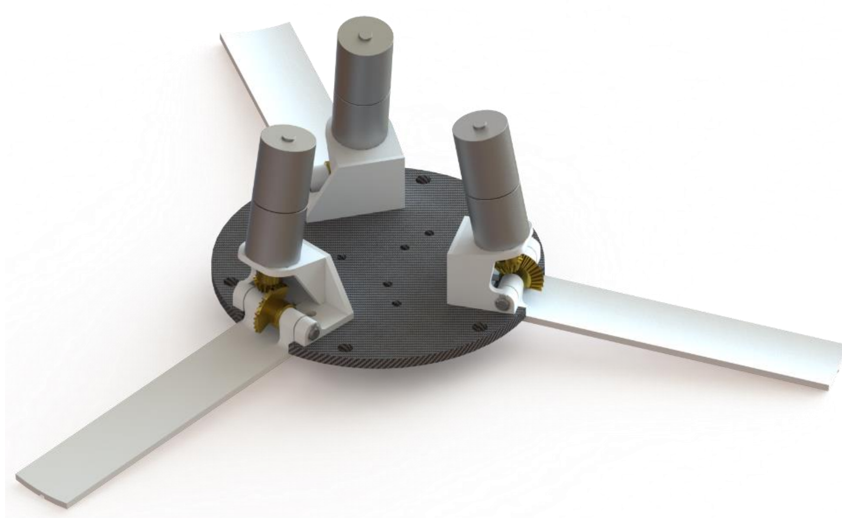


Figure 7.14 Self-righting system with all 3 petals fully deployed.

7.3.3 Component Design

The self-righting mechanism uses three "petals" that are driven by motors in order to right the payload. The petals have an axis of rotation at the bottom of the payload, so that they can be used as a lever arm to move the payload to a relatively upright position. Originally, the design used six petals that were individually driven, each having the necessary torque to right the payload by itself. While some driving mechanisms considered were able to provide enough torque to right the payload on one petal, the payload was too small to fit six of these mechanisms within it. Therefore, a three-petal self-righting system was chosen, with the assumption being made that two petals will be in contact with the ground at any time. This greatly simplifies the torque requirements and complexity of the driving system for these petals. The petals are slightly curved to fit around the outer diameter of the payload and connect to a hinge at the payload baseplate. For deployment of the payload, each petal has a channel down its centerline that acts as a groove for a pin to slide through, ensuring a smooth release from the body tube of the rocket.

The driving system for the petals that was decided upon was a high torque metal gearmotor to provide the necessary torque to right the payload, and a bevel gear system to redirect this torque into the petal. Several motors were considered, ranging from diameters of 20mm to 37mm. Originally a 37mm motor was considered due to the large torque required to right the payload, however, there were packing issues with integrating this motor into the payload body with the stabilization system. Because of this, a different motor was considered, the 25mm diameter ActoBotics 19 RPM Econ Gear Motor. This motor provides similar torque to the 37mm motor considered with a significantly smaller form factor and weight. With a 499:1 gear ratio, this motor has a stall torque of 19Nm, which allows the motor to be ran at significantly below stall torque. Specifically, the motor being run at 25% of stall current provides enough torque for each petal to be able to lift the payload independently. Because in almost all cases, two petals will be in contact with the ground, this system provides and additional 2x factor of safety.

The gearmotors are not able to be mounted parallel to the petals without exiting the perimeter of the payload. In order to integrate the self-righting system into the payload to fit with the stabilization system, the motors will be mounted almost vertically, at a slight angle towards the center of the payload. The mount for the motors is integrated into the petal hinge for simplicity, and this part is planned to be 3D printed due to the somewhat complex shape of the mount. Due to the angled mounting orientation, two bevel gears will be used to transmit the torque to the petals to rotate them outwards. One bevel gear will be mounted to the shaft of the motor, and the other to the hinge shaft of the petal. Because the petals only need to rotate approximately 95 degrees to right the payload, the vertical bevel gear is cut to only have teeth on the portion that is needed to rotate the petal. This is done to prevent the gear from interfering with the baseplate of the payload.

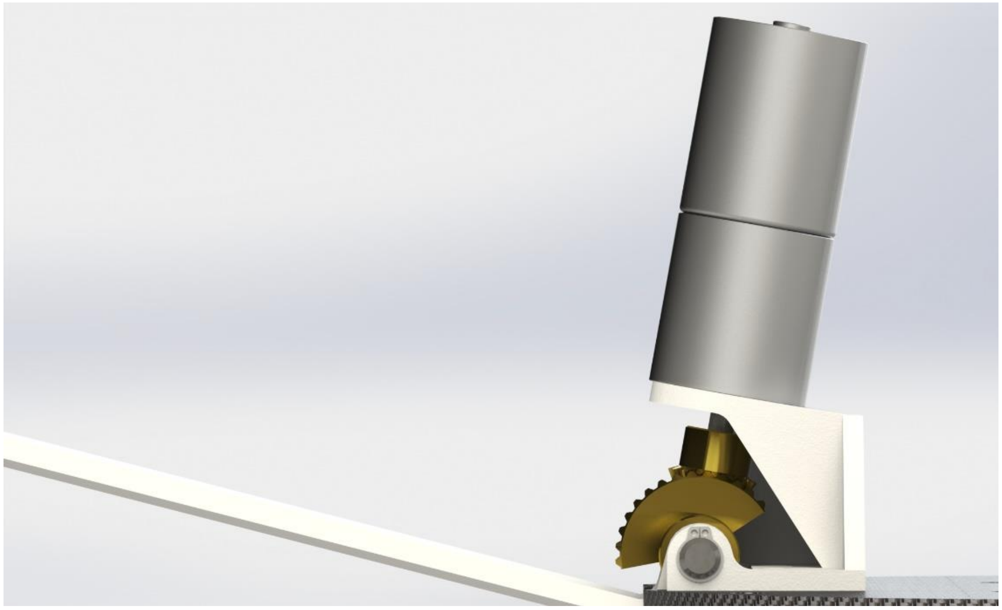


Figure 7.15 Motor mount angle and bevel gear design

7.3.4 Key Dimensions

The petals for the self righting system are currently made from curved sections with an inner radius of 2.875" and an outer radius of 3", giving them a thickness of 1/8". They are 6" tall with ¼" through holes 0.2" above the bottom to allow for the hinge shaft to be mounted. There is a ½" gap between the two hinge mounting points to allow for the bevel gear to be mounted to the shaft. Additionally, the petals have a groove in them which aids in retention and deployment.

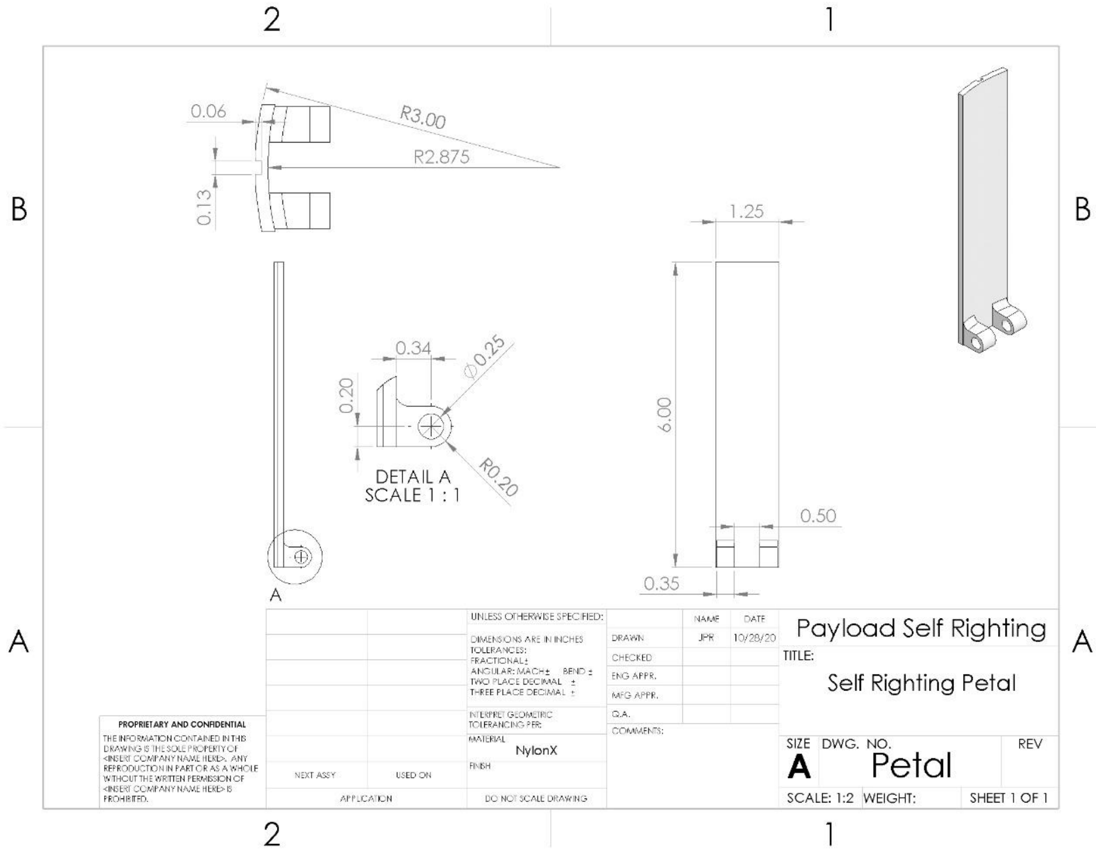


Figure 7.16 Dimensioned drawing of the self righting petal.

The bevel gear on the output shaft of the self-righting system, which drives the petal, has been cut down so it does not interfere with the bottom plate of the payload in the stowed configuration. The cut can be defined in two ways—the angle formed between the two flats created by the cut measures 202.5°, or the cut leaves only 14 teeth remaining on the bevel gear.

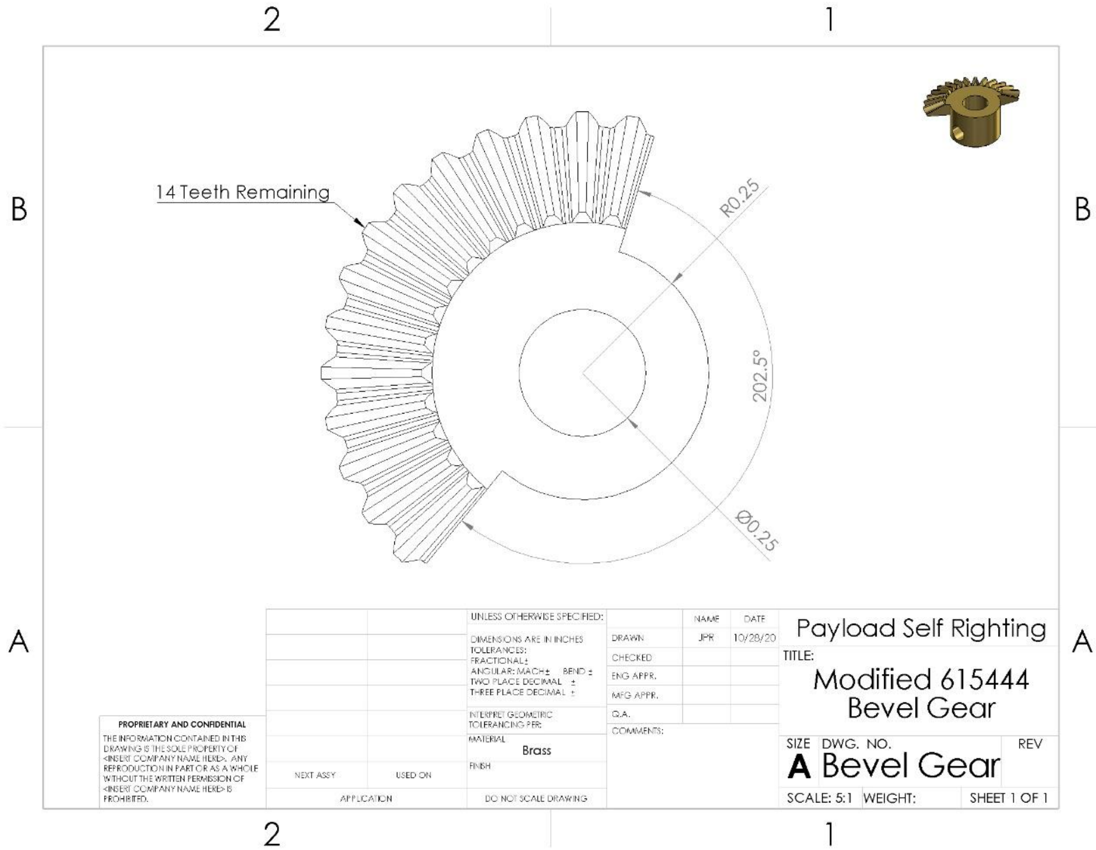


Figure 7.17 Dimensioned drawing of the cut down bevel gear.

The hinge which facilitates rotation of the petals has been combined with the mount for the driving motor. This means there is a bit of complex geometry in the combined part, as seen in the drawing below. The main dimensions, however, are the angle between the motor and vertical (10°), the mounting spacing (0.75"), the cutout for the petal (1.25" wide), and the clearance hole for the shaft at 0.27". There is also a clearance cutout for the bevel gear, which measures ¼" tall by 0.19" wide. The spacing and dimensions of the holes for the motor mount are defined by the motor and its mounting pattern.

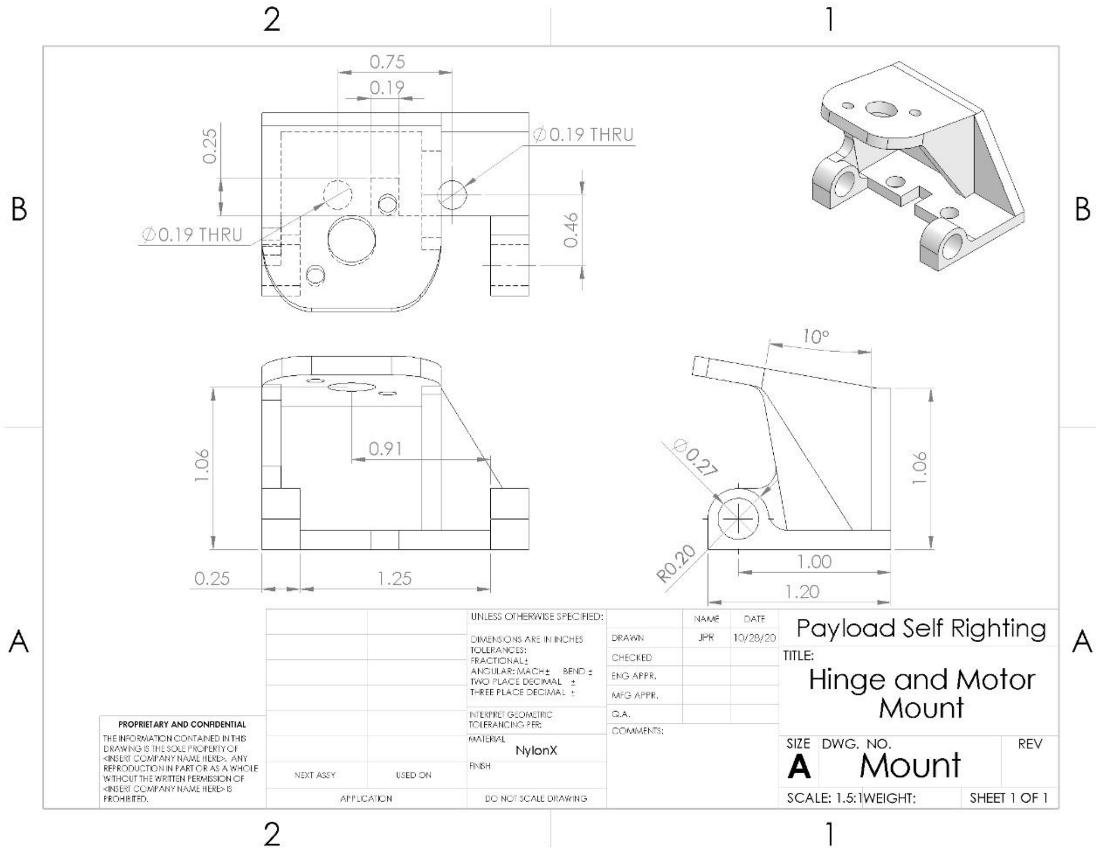


Figure 7.18 Dimensioned drawing of the integrated hinge and motor mount.

7.3.5 Required Torque

The amount of torque required to right the payload from its side can be estimated with the following equation, where r is the radius of the payload, h_{cm} is the height of the center of mass, w is the weight of the payload, and θ is the angle between the weight vector and the vector from the center of mass to the hinge.

$$\tau_{right} = \sqrt{r^2 + h_{cm}^2} \cdot w \cdot \sin(\theta)$$

This will give the worst-case scenario torque needed when θ is set to π/2 radians, or 90 degrees. This is unlikely to physically happen, as the payload would have to be on a slope with the top facing downwards, but it is still possible. If the payload lands on flat ground and falls on its side, sin(θ) is equal to the height of the center of mass over the lever arm length. This gives the following equation.

$$\tau_{right,flat} = \sqrt{r^2 + h_{cm}^2} \cdot w \cdot \frac{h_{cm}}{\sqrt{r^2 + h_{cm}^2}} \rightarrow \tau_{right} = w \cdot h_{cm}$$

Assuming a payload mass of 6lbs and the current radius of 3in, we obtain the following plot of the two functions as they vary with the height of the center of mass (θ is π/2 for the first equation).

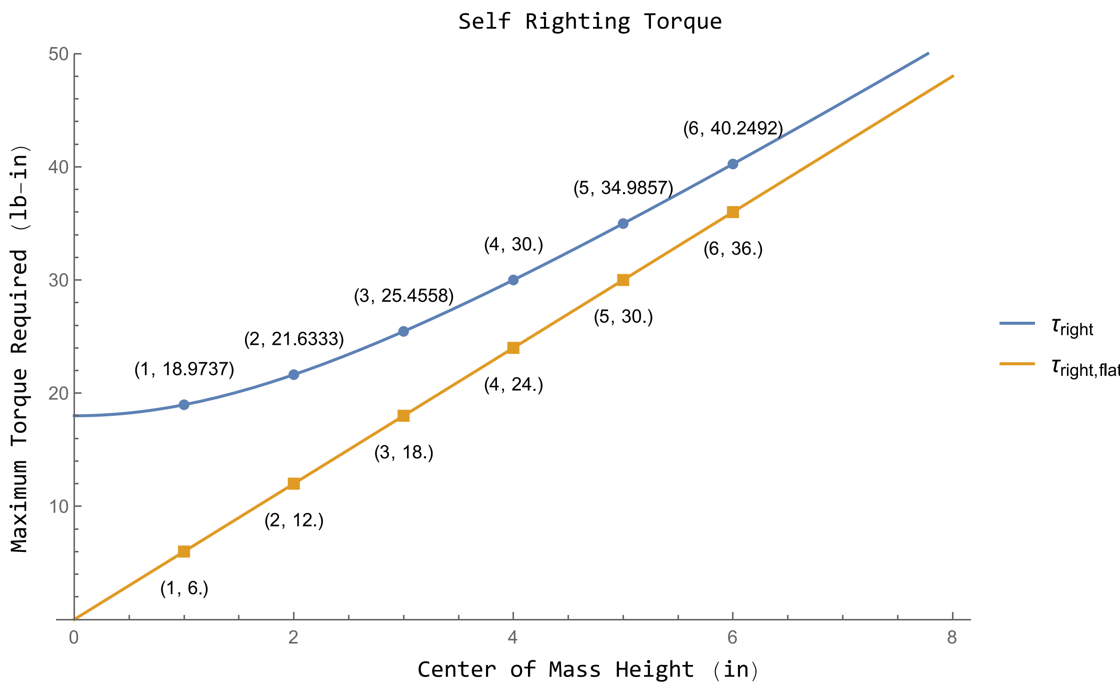


Figure 7.19 Plot of the two torque curves.

7.3.6 Previous Designs

Torsional springs in the form of metallic strips were considered as the main driving force on the petals. These strips, in combination with servo regulated releasing wires attached to the petals, would allow for a simple and easy to manufacture design that would provide a more passive method of self-righting. This design would also allow the releasing wire to individually control when the petals deployed and their rate of deployment which is desirable for the control of the self-righting process as it allows the petals to be intelligently lowered in uneven terrain. However, the metallic strips were not able to produce the required torque to bring the payload to an upright position without exceeding the dimensions of the petal. They would also need to be cut from an entire sheet of material which made its expected torque unpredictable, as such this design did not undergo further consideration.

As a replacement for the torsional springs, compressed gas cylinders were considered. These gas springs would be used as the main driving force on the petals instead of the metallic strips. The same servo regulated releasing wires would be used to control petal deployment as in the torsional spring design. The gas springs would be mounted in the lower section of the payload as shown in Figure 7.20. This change from torsional to gas springs was originally chosen for its potential to better produce the required lifting torque. However, no combination of mounting angle, barrel diameter, barrel extension, and extension force could be made to satisfy the requirements of this system—to bring the payload to an upright position. Additionally, the size of the entire design, including the servo regulated releasing wires (not shown), was no longer practical considering the amount of space available in the lower section of the payload. This design would have also added significant complexity to the manufacturing and assembly process, having to account for the gas cylinders, servos, and releasing wire. As such, alternate designs were explored.

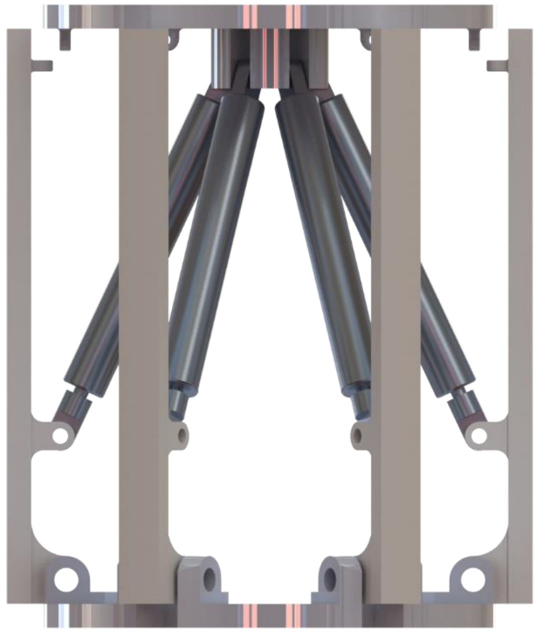


Figure 7.20 Gas spring design concept.

Linear actuators were chosen as an option to replace the previous gas springs and servo driven release wires. These electric actuators would be mounted like the gas springs, with their shaft end attached to the lower section of the petals and the other end attached to the payload mid-plate as shown in Figure 7.21. This design was chosen due to the ability of linear actuators to produce higher forces than gas springs as well as being a simpler design overall which would allow for ease of manufacturing, assembly, and operation. The actuators could also be driven independently of each other as in previous designs which was one of our main design wants. However, the linear actuators being considered for this design became quite expensive and all the achievable mounting angles of the actuator did not produce the required torque to lift the payload, as such, other avenues of producing the required lifting torque were considered.

Figure 7.21 Linear actuator design concept.

A multi-gear drive powered by an electric motor was chosen to replace the linear actuators. A motor mounted above the baseplate of the payload would drive a gearbox attached to the hinge of the petal as shown in Figure 7.22. This design produced the required amount of torque through gear ratios and a small driving motor, while also being less expensive than the previous designs. Also, with each petal having its own driving motor and gearbox the petals could all be driven independently. However, in order to produce the required gear ratios a complex gear train was necessary. This gearbox could be prone to misalignment during flight thus causing total failure of the self-righting system which was too large of a risk. Also, having this gearbox for each petal would further complicate the manufacturing process of this system due to its complexity and add too many potential points of failure. This design was able to be simplified however, producing our leading design using a larger motor and direct gear drive on the petal hinges.

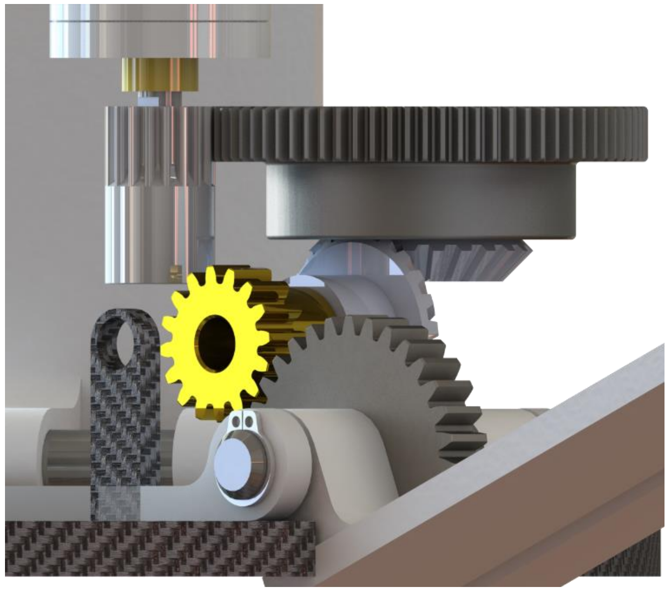


Figure 7.22 Multi-gear drivetrain design concept.

7.4 Electrical System

The electrical system is necessary to allow the payload to stabilize after landing, self-level within specifications, and transmit a captured panoramic photo back to the team. To successfully complete these tasks, electronics will manage the autonomous control sequence using a state machine to determine when to perform tasks such as self-righting, stabilizing, and transmitting the image. To determine each system state, sensors will be used to provide real-time data corresponding to the lander position and orientation in 3D space. This system is also responsible for providing signaling commands and proper power to the lander actuators, specifically the servo motors in the stabilization system and the brushed DC motors in the self-righting system.

The currently selected system requires that electrical input power be provided to seven actuators, one of which is the parachute rotary release latch referred to in Section 7.2.4, while the others are purposed in the subsystems used to level the lander. These devices will each have their own dedicated power and signal connections appropriate for their specifications and safe operation. The servomotors will each be powered using a Henge BEC (Battery Eliminator Circuit) operating at 7.4V to provide maximum power and allows a 4A continuous current which the servo 3A stall rating should not exceed. The DC motors will be powered using two Pololu MC33926 dual motor drivers for a total of four DC motor controllers, three for the self-righting system and the fourth for the rotary latch which along with the self-righting motors can draw up to 3A at 12V under our maximum expected torque loads. Powering the entire payload lander system is a Turnigy Nano-tech 12V 2.5Ah LiPo battery selected primarily for its physical size and capacity.

The Arduino Nano shown in Figure 7.23 will be the main processor responsible for communication with all desired peripherals and storing the lander operation program. It has a maximum speed of 16MHz and more than enough DIO (digital input/output) pins for interfacing with the selected sensors and commanding the actuators. Another processor consideration was the Teensy microcontroller which was selected by the avionics team for the launch vehicle air brake system. Though this device is much faster at up to 400MHz and with more DIO, the Arduino Nano was chosen due to its more available libraries written for communicating with the peripherals and proven reliability from use in past years.

Figure 7.23 Arduino Nano breadboard compatible breakout

The systems sensors include a GPS, a barometer, an IMU, and rotary encoders, which all provide data necessary for state transitions in the lander's software. The GPS is included in the photography system described below and records the latitude and longitude location of the payload over time. The Adafruit BMP388 barometer takes atmospheric pressure readings and interprets them as altitude from ground level. The IMU MPU9255 contains an accelerometer, a gyroscope, and a magnetometer which together provide inertial navigation data. Since the components all operate in three axes, this will allow for accurate estimation of the lander position and orientation in 3D space which is needed for state transitions and verification of completing the mission. The system will also include rotary encoders which are necessary for motor positional feedback. These will either be already built into the motors or selected and mounted on a drive shaft. The team will test to confirm the estimated power draw of actuator components are operating under their designated loads. Indicator LEDs will be used to indicate power, component status, connectivity or error codes which saves time for on-field debugging. Like the launch vehicle avionics system, the lander will possess a LoRa transceiver module for sending telemetry data to a ground station receiver in order to verify the performance of the system throughout its mission, determine location through GPS data, and possibly send the image should the photography system fail to do so. This module will be capable of 1 Watt or about 30dbm transmission power and will not transmit until the lander is on the ground ready to begin the autonomous sequence.

The potagraphy system of the payload detailed in Section 7.5 acts as a sensor peripheral to the main controller board which will communicate data to the main processor over Serial. This system consists of a Raspberry Pi Zero interfacing with the panoramic camera and connecting to a shield breakout board consisting of a GPS, Bluetooth, and GSM (Global System for Mobile communications) chip which the team intends to use for transmitting the captured photo to the base station receiver. By using this board, the image can simply be texted over MMS (Multimedia Messaging Service) with a subscription to a moblie carrier to access this network through a SIM (Subscriber Identity Module) card. The motivation behind pursuing this transmission scheme is to ensure that the lander will have the widest-possible area of connectivity given that the team may not have the ability to select a certain launch location due to travel restrictions and must be open to any possible site. This board pictured in Figure 7.24 was selected for its compact size allowing it to be easily mounted on top of the Raspberry Pi Zero.

Figure 7.24 Raspberry Pi GSM shield board

The current team development plan is to have each peripheral breakout board tested individually, then once sufficient functionality is proven for each individual component, testing will begin using them all together. A wiring diagram of these peripherals when tested together on a breadboard can be seen in Figure 7.25 which includes additionally tested components not selected to be part of the final lander electronics. After component functionality has been verified, the team will then finalize wiring diagrams complete with data communication and power rails which will assist us in designing a custom PCB. Combining the components into a single PCB will significantly minimize the form factor of the electronic system and increase the robustness of the design given less loose wires and connection to make in lander. This PCB will likely be designed using KiCAD and most likely be manufactured by JLCPCB.

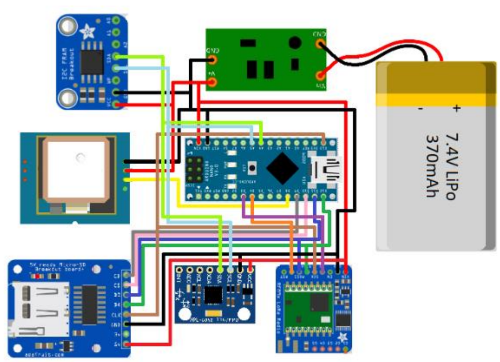


Figure 7.25 Lander peripherals all wired to the main processor for testing

The previously mentioned Arduino Nano selected as the main processor runs Arduino C/C++ making it the primary language to be used in developing the lander software, the program of which will be organized in the form of a robot state machine. This method was selected because throughout the different phases of the mission, the lander must perform unique and sometimes multiple tasks such as commanding actuators using feedback from sensors sampled in real time. Additionally, the use of event checkers to monitor sensor data will allow simple and reliable transition between states once certain events occur. There may be conditions which must also be true in parallel with an event occurring for the state to switch to the next state. These can be referred to as guard conditions and the team will make use of these methods to ensure proper transitions between states. There may be, however, a potential for these states to not properly transition correctly if sensor data is incorrect or delayed. Therefore, timeout events may also use timers started at the time of power-on to serve as a final backup to transition states to prevent conditions such as program infinite looping which could cause the lander to remain "stuck" in a certain state. Figure 7.26 shows a rough outline of the state diagram which will be programmed onto the lander.

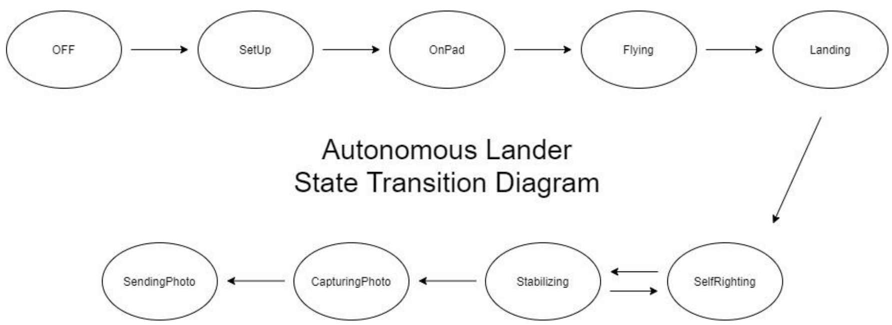


Figure 7.26 Simple lander state transition diagram

7.5 Photography System

The leading camera design is a single lens fisheye 360-degree camera. We decided to use this method for its robustness but also simplicity. Rather than using heavy computation to stitch together multiple photos from multiple camera sensors, or a complicated rotating camera setup, we have opted for processing one image from a wide-angle fisheye lens. The camera we chose was a PICAM360 as shown in Figure 7.27.

Figure 7.27 PICAM360

The PICAM 360 is an eight-megapixel USB (Universal Serial Bus) operated camera with a 360-degree horizontal field of view and a 235-degree vertical field of view. When placed at the top of our payload (above the ground height between 8.2 inches and 10.2 inches, depending on the state of the self-righting mechanism), we will be able to obtain an image that originates between 15.7 and 19.6 inches from ground level at the base of the payload. This is a result of the 27.5-degree field of view below the horizontal we can obtain with this camera setup as seen in Figure 7.28.

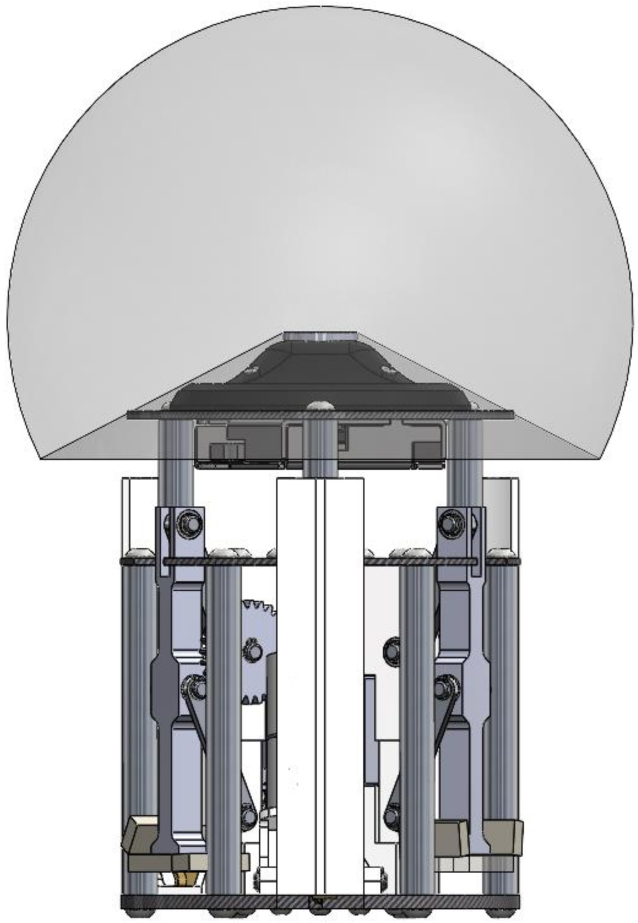


Figure 7.28 Camera FOV

This camera and lens combo will allow us to take one image in a classic fisheye view. The resultant image is a round image with the vertical being in the center and the ground or surrounding objects lining the outside of the image Figure 7.29.

Figure 7.29 Picture from testing with PICAM360

If desired, this image can later be unstitched into an equirectangular, or classic panorama, format as seen in Figure 7.30.

Figure 7.30 Panorama Example

This camera will be operated by an onboard Raspberry Pi Model 3B+ (shown in Figure 7.31) or, ideally a Raspberry Pi Zero W (seen in Figure 7.32), if computationally compatible. The Pi Zero W is 0.79”(20mm) by .75”(19mm) smaller than the Pi 3B+ and is thus desirable for its smaller size requirements inside the vehicle. The Pi Zero W has less computational power, however, so we are still testing to see if it is a viable replacement for the larger Pi 3B+ will be used.

Figure 7.31 Raspberry Pi Model 3B+

Figure 7.32 Raspberry Pi Zero

The Raspberry Pi will serve as the camera controller, image storage, image processing, and transmission origin in a small and lightweight form factor computer. This setup was also chosen for its cost effectiveness. Both the Raspberry Pi's and PICAM360 are much cheaper than the alternative camera setup, which are all hundreds of dollars. The camera will be mounted in a 3D printed PLA casing, to protect it from impact or vibration while landing. It will be painted matte black to reduce optical interference as shown in our CAD model in Figure 7.33.

Figure 7.33 Camera Housing

Our overall design was chosen over a multitude of different options. Our original idea was a single camera that would rotate about a central axis and take pictures on intervals. This idea was ruled out for its complicated mechanical and software obstacles. The camera's rotation and stitching of the images would add unnecessary complications when compared to the other options. An additional idea was to have an array of cameras arranged around a central point to all take pictures that would later be stitched together. This would solve the complication of rotating the physical camera, but the images would still need to be stitched to achieve a true 360-degree singular image.

7.6 Stabilization

7.6.1 Leading Design Overview

The stabilization system is designed to bring a self-righted payload to within five degrees of vertical so that a panoramic picture may be taken. This system accomplishes the objective with a four-bar linkage to adjust the payload's attitude, and a foam composite foot to provide traction across a wide variety of surface conditions. The design chosen provides the best tradeoffs between mechanical complexity, reliability, and manufacturability to create the most effective design from the ones considered.

7.6.2 Base

For the base structure of the stabilization system, we have chosen a roughly rectangular box 3D printed from NylonX filament. The box provides structure for the payload bay through the holes on the top and bottom of the piece, which are threaded using nut inserts. The base also has mounting points for the motor, utilizing nut inserts and residing within the large cutout to the rear, and the crank and follower links, which are mounted concentric to the two larger holes in the front using retaining rings. Three bases will be placed in a 120-degree, equally spaced pattern around the bottom of the baseplate of the payload. The bases will be mounted by inserting threaded inserts into the base and bolting them to the payload's structural plates.

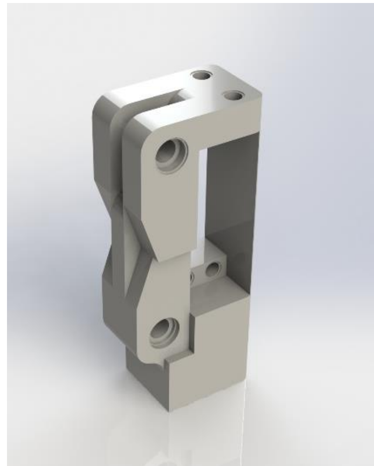


Figure 7.34 Payload Stabilization Base Isometric View

The primary difference between this and alternate designs is the material choice and the manufacturing processes chosen. Given the geometry required of the base, the alternate design of a DELRIN base manufactured in a mill would be unfeasible due to the geometric complexity of the part, as shown in the part drawing below. The variable material properties of a 3D printed part were judged to have a negligible impact on the design. The 3D printed part also has the advantage of being lighter than the milled part due to the material properties of the NylonX filament.

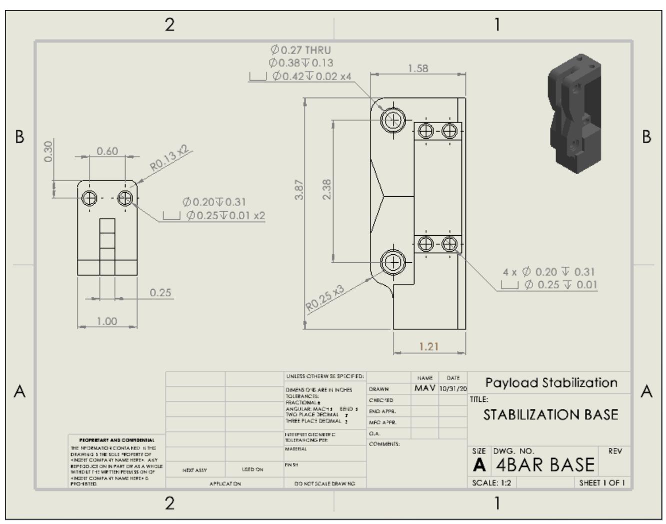


Figure 7.35 Important dimensions for the stabilization base

7.6.3 Lift Mechanism

The lift mechanism will consist of a parallel four bar linkage, with a single crank located at the top of the base and two followers located below. This design was weighed against other options in Figure 7.36 but was ultimately chosen for its relationship between reliability, its weight, and its extension characteristics. The crank will be driven by a dual mode servo that is covered in more detail below.

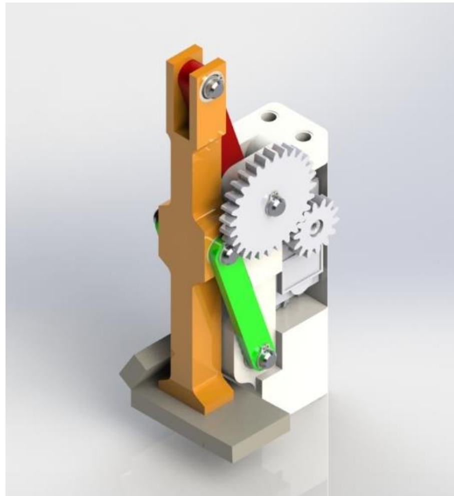


Figure 7.36 The lift mechanism with key components highlighted. The crank is highlighted in red, the coupler in orange, the followers in green, and the base in white.

The primary dimensions for the linkage are the link lengths; the ground link (a pseudo-link located on the base) and the distance between the crank and follower joints on the coupler are both 2.375 in, and the length of both the crank and the followers is 2 in. The coupler extends another 2.375 in downwards for a total length of 4.75 in. A diagram of these dimensions can be found in the figure below.

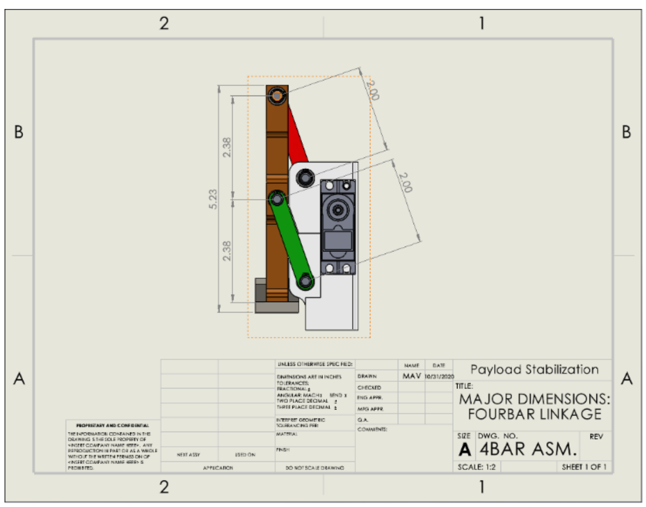


Figure 7.37 The key dimensions of the four bar linkage. All dimensions are in inches.

The torque requirements for the lift mechanism were determined with a Motion Study in SolidWorks 2020, shown in the figure below. Each link was created in Aluminum 6061-T6, the heaviest material considered for the links. It was also assumed the full weight of the payload, estimated to be 5 pounds, would be acting vertically against the motion of the motor. It was discovered that link weight has little effect on the torque required to run a motor located at the crank joint at a constant and arbitrary speed of 3 RPM. The maximum torque required of the motor was 10 in-lbf, which occurred when the crank and follower links were roughly horizontal.

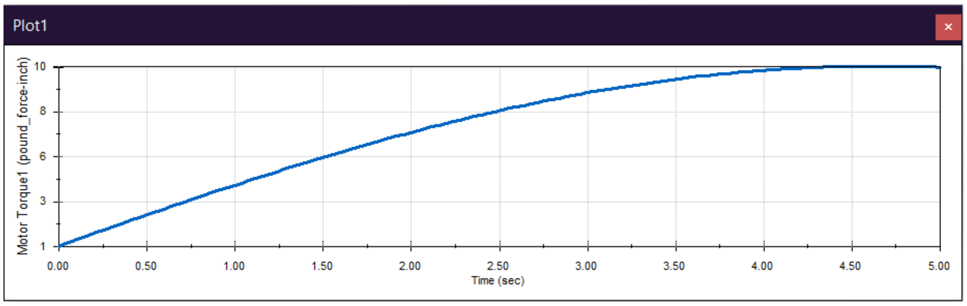


Figure 7.38 A measure of the torque in in-lbf required to rotate the crank at a constant speed of 3 RPM. In the motion study, the linkage started in the stowed configuration, then folded outwards to simulate the stabilization process.

With this information, an appropriately sized motor can be chosen. After some deliberation, the goBILDA 2000-0025-0002 dual-mode servo was chosen, primarily for its torque-to-size ratio. A geartrain was designed to double the torque output at the crank shaft through a 14-tooth to 28-tooth power transmission. This reduced the torque requirement placed on the servo, which in turn reduces the current draw of the motor.

For the bottom of the 4-bar mechanism, it was determined that the piece in contact with the ground should be comprised of a different material than the coupler, as we determined a foot element design should attempt to maximize the surface area in contact with the ground for maximum traction. As a rigid plastic, DELRIN is not well-suited to this task. To address this challenge, a modular foot will be attached to the coupler on the 4-bar.

7.6.4 Foot

The leading design for the foot of the stabilization system involves a rigid modular folding design containing a composite foam material. This design will ensure maximum contact with the ground after landing while also being able to fold into a compact storage configuration during ascent. The foot itself is a rectangular prism with dimensions 1.5 in. x 1 in. x 0.5 in. in its folded configuration. This is accomplished by folds stretching from front to back at 0.5 in from each end of the prism seen in Figure 7.39 The foot will be comprised of a composite foam material, with one polyurethane layer found in between two polyethylene layers. This pattern will optimize the strength of the composite without sacrificing flexibility at the contact point and at the connection between the foot and the traction bottom, discussed below. This also minimizes the risk of the foot shearing due to a lack of flexibility. This composite material is designed to maximize material strength, ensure a balance of rigidity and compressibility, maximize material flexibility, and minimize vibrations that would increase shear and tensile stress on the four-bar linkage pins. The summation of these material attributes allows for better and more reliable performance when stabilizing the lander. Also taking into consideration the varying ground terrains the lander may encounter, our design includes a removable traction bottom 3D printed from a flexible filament, NinjaTech Cheetah. This 3D printed layer will be attached to the bottom of the foot to provide better traction and increase the vibrational damping on the coupler. The traction bottoms may vary in design based on which launch site is selected, as the two sites considered have very different terrain.

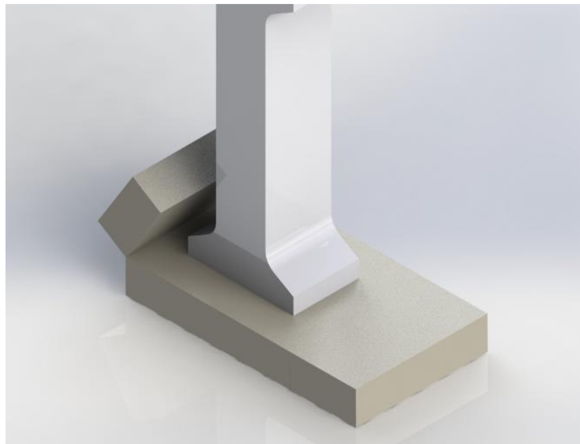


Figure 7.39 An isometric view of the stabilization foot. Note that the left side of the foot is folded, while the right side is unfolded. Neither the composite structure nor the traction bottoms are shown in this render.

7.6.5 Alternate Designs

7.6.5.1 Base

Given the geometry of the base and the CNC tools available to the team, it was determined that milling would be an impractical way of manufacturing the base. It was then decided that that the base would be printed from NylonX, a strong 3D printing filament that the team has experience printing in. With 3D printing, the required geometry of the base was now manufacturable with similar strength to that of DELRIN.

Base			
	Weight Scale	3D Printed NylonX	Milled DELRIN
Weight	4	5	4
Manufacturability	3	5	1
Stiffness/Support	2	4	5
Space	1	4	4
Totals		65	47

Table 7.1 Decision Matrix comparing base designs

7.6.6 Lift Mechanism, Motor

A few different mechanisms were considered for the lift mechanism. Table 7.2 and Table 7.3 below give an overview of how the leading design was decided upon. The first alternate design was relatively simple: an L16 Linear Actuator from Actuonix would drive a custom foot vertically to control the attitude of the lander. While this was the most promising alternate design due to the team already possessing a few of these motors, it was ultimately scrapped due to the lack of reliability of the plastic connector piece on the end of the servo. This piece, which would have to support the full weight of the payload, tended to break when exposed to high loads. Also, the linear actuator would not increase the size of the base of the payload during extension, which would make the payload less stable the farther above the ground it was, which could prove problematic in the field.

Another alternative lift mechanism considered was the arrested four bar linkage, which attempted to increase the extension of a parallel linkage. The mechanism would be similar in dimensions and design to the parallel four bar, but the coupler-follower pin joint would be replaced with a slot. During the first phase of operation, shown in Figure 7.40 below, the follower joint would be arrested at the bottom of the slot by a set of extension springs, which would result in motion identical to a parallel four bar linkage. A mechanical stop would prevent the follower from moving past the horizontal. If more extension were needed, the crank could continue moving, which would extend the spring and move the coupler such that the follower joint would travel up the slot. This is shown in Figure 7.41. While this design provided greater extension than the parallel four bar linkage, the complexities of controlling this linkage and generating a motion profile for it led it to be rejected.

Figure 7.40 The arrested four bar linkage in its parallel phase of deployment

Figure 7.41 The arrested four bar linkage in its uneven phase of deployment

Lift Mechanism				
	Weight Scale	Parallel Four bar Linkage	Linear Actuator	Arrested Four bar Linkage
Extension	5	3	4	5
Weight	4	5	4	2
Reliability	3	2	4	3
Ease of Construction	2	4	3	1
Mech. Complexity	1	4	3	1
Totals		51	44	45

Table 7.2 Decision Matrix comparing lift mechanism designs

A few different motors, whose qualitative characteristics are listed in Table 7.3 below, were considered. The primary concerns when choosing a motor were torque output and size; the motor used to drive the linkage must be able to fit within the small area allotted to the stabilization system while providing a torque output uncommon for its size. Weight also greatly factored into this selection, since the payload requires three stabilization modules to perform its mission. The Hitec brand motors considered were able to provide the proper amount of torque at an acceptable current level only if a 1:4 power transmission connected the motor to the crank, whereas the goBILDA motor could use a smaller 1:2 motor, significantly reducing the overall weight of the system. The Hitec motors were also longer than the goBILDA, which adversely affected the packing arrangement of the stabilization modules. These factors all led to the Hitec brand motors being rejected for the design.

Motor				
	Weight Scale	goBILDA 2000-0025-0002	Hitec HSR2645-CR	Hitec HSR1425-CR
Torque Output	5	5	3	2
Size	4	4	3	3
Weight	3	4	2	5
Power	2	3	3	3
Current Draw	1	2	3	5
Totals		64	42	47

Table 7.3 Decision Matrix comparing motor selections

7.6.6.1 Foot

Other concepts of designs that were weighed during the initial design phase include a stiffer foot design that would be connected to the coupler using a ball joint supported by a spring tension system. Table 7.4 below give an overview of how the leading design was decided upon. The foot would still be the same size constraints placed on it by the base, but would be more rigid, in that it would be a solid material, and would have a fluted corrugated contact surface, which would not be able to mold to its surroundings as well as a composite foam material.

Additional design concepts that were introduced at the beginning of the design phase include a pressure vessel attached to bottom of a rigid foot design. The rigid foot would be attached like how the leading design is being fixed to the coupler. The pressure vessel would be under-inflated, giving it the compliance needed to conform to variable surroundings. This design was considered, but not chosen due to many aspects including the variation in air pressure during the flight of the vehicle, making it difficult to regulate the pressure of the vessel.

Foot Design				
	Weight Scale	Modular Composite	Rigid Ball Joint	Pressure Vessel
Size (surface area and storage)	5	5	3	2
Flexibility/Compressibility	4	4	1	4
Terrain Adaptability	3	5	1	4
Strength	2	4	4	2
Vibration minimization	1	5	2	2
Totals		69	34	40

Table 7.4 Decision Matrix comparing foot designs

8 Safety

WPI HPRC is dedicated to creating and maintaining a safe environment for team members and others at all times. Safety is the primary consideration in all team activities including design, construction, testing, launch, and other events. The team fosters a safety-first atmosphere where each member understands their own personal responsibility with respect to best safety practices. The team safety officer, Michael Beskid, is responsible for educating all team members about safety, overseeing safe practices in all HPRC activities, and observing strict adherence to the NAR Safety Code and local laws. The safety officer is also responsible for all items detailed in section 5.3 of the 2021 NASA Student Launch Handbook.

The following sections contain vital information which will serve as a basis for making decisions with respect to predetermined safety guidelines. Each section entails an analysis of hazards that may be encountered, accompanied by mitigation techniques for each in order to reduce risk. The Project Risks section outlines potential threats to the successful completion of the project with respect to time, budget, resources, and similar logistical concerns. Careful consideration of possible dangers to team members, bystanders, and others is then detailed in the Personnel Hazard Analysis. The Failure Modes and Effects Analyses follows, identifying potential hazards associated with the proposed rocket and payload design and technical failures. Finally, the Environmental Concerns section considers the possible hazards to the team and to the mission posed by the environment, as well as the adverse effects that team activities may cause to the environment.

Due to the unfortunate circumstances of the COVID-19 pandemic, there are more risks involved with the project for those working in person. These risks have been mitigated by the team policy of only meeting in person for the purposes of construction of the launch vehicle and payload. The risks of working together in person are outlined below and given the appropriate classification as well as proper mitigation. In addition, the normal project risks are overall skewed as well due to the conditions the COVID-19 pandemic have provided the team with, specifically seen in our school's travel restrictions and other rules.

8.1 Project Risks Overview

This section provides a detailed analysis of risks that could affect the successful completion of the project as a whole. More specifically, project risks include those which may impact the budget, timetable, or logistics throughout the scope of the project and competition. If not mitigated, these risks may result in delays, reduction in design quality, or at worse the inability to complete the project and withdrawal from the competition. Each of these risks are categorized according to both their probability and severity in order to assess the potential impact on the project. A thorough understanding of such project risks is critical in order to develop a mitigation plan to minimize risk and give the project the best chance to succeed.

Project Risk Probability Definitions	
Rating	Description
A	The risk is probable if it is not mitigated.
B	The risk may occur if it is not mitigated.
C	The risk is unlikely to happen if it is not mitigated.
D	The risk is highly unlikely to happen if it is not mitigated.

Table 5 Project Risk Probability Definitions

Project Risk Severity Definitions	
Rating	Description
I	Irrecoverable failure.
II	Significant loss of money, time, or major design overhaul.
III	Minor loss of money, time, or minor design overhaul.
IV	Negligible effect to design, timeline, and budget.

Table 6 Project Risk Severity Definitions

Project Risk Probability	Severity			
	I - Irrecoverable	II - Significant	III - Minor	IV – Negligible
A – Probable	AI	AII	AIII	AIV
B – May Occur	BI	BII	BIII	BIV
C - Unlikely	CI	CII	CIII	CIV
D – Highly Unlikely	DI	DII	DIII	DIV

Table 7 Project Risk Assessment Matrix

Project Risks Overview					
Risk	**Probability/ Severity**	**Schedule Impact**	**Budget Impact**	**Design Impact**	**Mitigation**
COVID-19	DI	A few or in the unlikely case of multiple team members contracting COVID-19 would possibly require WPI stepping in and causing our club to go virtual only for an undetermined amount of time. This could cause construction to cease and our launch vehicle and payload to not be ready for launch, resulting in a disqualificati	Little impact on budget unless in the unlikely event WPI revokes club funding.	The design quality may be negatively impacted depending on how many members cannot focus on the design anymore due to their condition.	WPI HPRC members will follow all WPI COVID-19 guidelines. Students in person will get tested at least once a week and produce a negative test result to be able to work with others. Members will work in their subteams or a sign-up and rotation system will be implemented for bigger group projects. Sanitizing

		on from competition.			and PPE gear will be provide by WPI and HPRC and are expected to be used frequently and thoroughly.
Subscale Launch Cancellation	CI	It is likely that the team will be removed from competition due to inability to launch because of WPI's travel restrictions and no other later launch dates for local clubs.	No impact	No impact	There is little mitigation due to this decision being up to WPI and or CMASS. WPI's decision to continue to support our going is dependent on COVID-19 cases in the area. CMASS' decision will be based on this as well as weather.
Destruction of Full Scale	DI	Possible disqualification from the competition. The team will have to reorganize the schedule to compensate to build a	The budget would have to be increased to compensate for the construction of a new launch vehicle. The team may not be	The design would need to be altered to prevent another full-scale destruction.	Test all aspects of the full-scale launch vehicle individually to ensure they work correctly. After, test the

		new full-scale rocket.	able to afford to construct a new launch vehicle.		components together. Analyze and test all electronics within the launch vehicle. Do not expose the rocket to any hazardous environments.
Full Scale launch fail	DII	If no damage was done to the rocket, minor time delays to reschedule the launch. Two to three-week delays to reorder parts and rebuild the rocket. Additional time to edit the design.	The budget could be affected significantly (up to 2000$), depending on the number of repairs that need to be done.	The design will be altered to avoid future launch fail.	Analyze results of a subscale launch and simulations to ensure that the rocket will not fail at launch. Follow all the instructions given by the RSO and all NAR regulations.
Destruction of payload in testing.	DII	Two to three-week delays to reorder parts and rebuild the payload.	The budget could be affected significantly (up to 500$), depending on the number of repairs that	Significant design changes will be made to ensure that the payload does not fail again.	Use of simulations and separate testing of the UAV and the retention system before test launches.

			need to be done.		
Damage to construction material	CIII	Small to hefty schedule impact depending on damaged material.	May need to buy more material.	May need to use different methods or materials for construction.	WPI HPRC members will use construction material carefully and sparingly.
Sub-scale launch fail	DI	It is likely that the team will be removed from competition due to WPI's travel restrictions and no other later launch dates for local clubs. The sub-scale launch will have to be rescheduled, causing minor delays. One-two-week delays to reorder parts and rebuild the sub-scale. Additional time to edit the design.	The budget will be affected in a minor to significant way depending on the cause of launch to fail.	The design will be altered to avoid future launch fail.	WPI HPRC members will use simulations to ensure that the sub-scale rocket will not fail at launch. Follow all the instructions given by the RSO and all NAR regulations.

Unexpected expenses (higher than expected shipping, parts, etc.)	CIII	Little schedule impact unless a shortage of funds results in an incomplete order of needed parts or the faster shipping cannot be afforded for a necessary part.	Budget may have to be supplemented and more money would have to be raised to offset any additional costs.	May impact supplies able to order due to looking for cheaper options to offset the more expensive ones.	WPI HPRC's Treasurer, will keep a detailed budget and account for shipping when budgeting.
Parts damaged or delayed during shipping	CIII	Time to complete testing and construction would be increased as new parts may need to be order or the one in hand modified.	May need to use extra funds from budget to pay for parts damaged or order new ones.	May need to use different parts to replace those lost or damaged.	WPI HPRC will order parts from reputable companies the team has worked with before.
Parts damaged or delayed in route to launch	DI	Possible disqualification from competition due to failure to launch. Likely unable to recover in time to make another launch due	May need to use extra funds from budget to pay for parts damaged or order new ones.	May need to use different parts to replace those damaged.	WPI HPRC members will pack the launch vehicle and payload very carefully.

		to WPI travel restrictions. If granted a new launch date, the schedule would shift a few weeks to accommodate re-construction.			
Injury	CIII	Delays may occur due to ensuring the injured member's safety and determining the cause of the injure and ways of mitigating it.	No impact.	No impact.	WPI HPRC members will follow all safety procedures, consult the MSDS sheets, listen to the RSO, and follow the NAR requirements.

Table 8 Project Risk Overview

8.2 Personnel Hazard Analysis

There are inherent dangerous involved in the construction, testing, and launch of high power rockets. As such, the personal safety of our team members and bystanders is of paramount importance. WPI HPRC aims to minimize the risk of personal injury by careful analyzing potential hazards and implementing a plan for hazard mitigation. This section provides an analysis of such hazards that may be encountered in high power rocketry and classifies them according to the likelihood and severity of each. Failure to mitigate these risks could result in minor injuries requiring simple first aid, more severe injuries, or even permanent injury or death. For this reason, it is imperative that the team is diligent about following all mitigation guidelines to minimize these hazards and create a safe environment for all personnel.

Personnel Hazard Probability Definitions	
Rating	**Description**
A	The hazard is probable if it is not mitigated.
B	The hazard may occur if it is not mitigated.
C	The hazard is unlikely to happen if it is not mitigated.
D	The hazard is highly unlikely to happen if it is not mitigated.

Table 9 Personnel Hazard Probability Definitions

Personnel Hazard Severity Definitions	
Rating	**Description**
I	Significant chance of death or permanent injury.
II	Possibility of major injuries requiring hospitalization or permanent minor disability.
III	Chance of injury requiring hospitalization or period of minor disability.
IV	May cause minor injury which may require first aid.

Table 10 Personnel Hazard Severity Definitions

Personnel Hazard Probability	Severity			
	I - Irrecoverable	**II - Significant**	**III - Minor**	**IV – Negligible**
A – Probable	AI	AII	AIII	AIV
B – May Occur	BI	BII	BIII	BIV
C - Unlikely	CI	CII	CIII	CIV
D – Highly Unlikely	DI	DII	DIII	DIV

Table 11 Personnel Hazard Assessment Matrix

Personnel Hazard Analysis						
Section	**Hazard**	**Cause**	**Effect**	**Probability/Severity**	**Mitigation & Controls**	**Verification**
Construction	Power Tool Injury	Improper training or human error during the use of power tools	Injuries include, but are not limited to cuts, scrapes, and even amputation or crushing.	DII	HPRC members will receive proper training and will have access to instructions on how to operate each tool. Members will also wear proper PPE specific to each tool. If an injury does occur, a member will be given proper medical attention.	Safety officer, leads and/or the lab safety monitor is present during the use of potentially dangerous tools to ensure proper usage and PPE.
	Hand Tool Injury	Improper training or human error during the use of tools	Injuries include, but are not limited to cuts, scrapes, even amputati	CIII	HPRC members will receive proper training and will have access to instructions on how	Safety officer, leads and/or the lab safety monitor is present during the use of potentially

			on or crushing.		to operate each tool. Members will also wear proper PPE specific to each tool. If an injury does occur, a member will be given proper medical attention.	dangerous tools to ensure proper usage and PPE.
	Caught in a machine	Loose items of clothing/jewelry/hair/gloves getting pulled into a machine	Partial or complete destruction of an item pulled in; injuries as severe as amputation.	DII	Members will not be allowed to use machines while wearing loose items of clothing/jewelry/gloves or having long hair that are not contained.	Safety officer, leads and/or the lab safety monitor will be present during the machining process to ensure members aren't wearing loose items.
	Fire	Human error, short circuit amongst any other event that could	Burns, inhalation of toxic fumes, and in extreme cases, death.	DII	Members will only work in facilities with proper fire safety	Safety officer, leads and/or the lab safety monitor will be present to ensure

		cause a fire to start.			systems installed.	proper use of machines and will inspect the area for clear indications of emergency exits
	Electric Shock	Member coming in contact with an exposed wire.	Burns, and in extreme cases, death from electrocution.	DII	Members will inspect all wires before working with them and not deal with live wires often, if at all.	HPRC members will perform an analysis of wires.
	Debris from machine	Improper securing of the material/object that is being machined.	Injuries include, but are not limited to eye injuries, cuts, crush injuries.	CIII	Members will be properly trained to use the machines and will wear proper PPE specific to each machine.	Safety officer, leads and/or the lab safety monitor is present during machining to ensure proper usage and PPE.
Chemical	Exposure to epoxy	Improper PPE worn during construction.	Eye and skin irritation; prolonged and reputativ	BIV	During work with epoxy, members will wear proper	MSDS sheet for epoxy will be consulted and members

			e skin contact can cause chemical burns.		PPE including safety goggles, gloves, and clothes that protect the skin from encountering the material.	will be wearing proper PPE.
	Exposure to carbon fiber/ fiberglass dust and debris	Sanding, using a Dremel tool, machining carbon fiber/ fiberglass.	Eye, skin and respiratory tract irritation.	CII	During work with carbon fiber/ fiberglass members will wear proper PPE including safety goggles, gloves, long pants and long sleeve shirt, as well as a mask to protect their lungs.	MSDS sheet for each material will be consulted to make sure members are wearing proper PPE.
	Exposure to black powder	Loading charges for stage separations or any other	Serious eye irritation, an allergic skin	CIII	Only people who are trained in working with black	Safety officer will ensure that unauthorized members

		contact with black powder.	reaction; can cause damage to organs through prolonged and repetitive exposure.		powder will be allowed to handle it. They will wear proper PPE. Clothing that has black powder on it will be washed in special conditions.	do not work with black powder. MSDS sheet for black powder will be consulted to make sure members are wearing proper PPE
	Fire	Chemical reaction, explosion or any other event in which a chemical catches fire.	Burns, inhalation of toxic fumes, death.	DII	Members will only work in facilities with proper fire safety systems installed.	Safety officer, leads and/or the lab safety monitor will be present to ensure proper use of chemicals and will inspect the area for clear indications of emergency exits. Chemicals that are in use will be kept track of to inform firefighters

						in case of a fire.
	Exposure to LiPo	LiPo battery leakage.	Chemical burns if contacts skin or eyes.	DIII	The battery will not be dismantled and will be checked for leaking before use.	WPI HPRC members will provide analysis of the battery.
	Exposure to APCP	Motor damage.	Eye irritation, skin irritation.	DIII	Only a few select HPRC members handle the motor and will wear proper PPE while doing so.	MSDS sheet for APCP will be consulted to make sure members are wearing proper PPE.
Launch	Injuries due to recovery system failure	Parachute or altimeter failure	The rocket/ parts of the rocket go in freefall and injure personnel and spectators in the area causing bruising and possible death	DI	HPRC members will pack the parachutes correctly, ensure the altimeter will be calibrated correctly, and that the amount of black powder in separation chares are weighed	HPRC Recovery subteam lead, along with others will oversee this process.

					on an electronic scale for accuracy.	
	Injuries due to the motor ejection from launch vehicle	Motor installed and secured improperly.	Motor and other parts of the rocket go in freefall and injure personnel and spectators in the area causing burns and possible death.	DI	The motor will be installed by a certified mentor	Safety officer will ensure that the motor is installed by a certified mentor. Prior to the launch, the rocket will be inspected following a checklist.
	Injuries from premature ignition of separation charges	Improper installation of igniters, stray voltage.	Severe burns.	DI	The battery will be switched off during installation of the igniters, black powder in separation charges will be weighted on an electronic scale.	Safety officer will ensure that all safety procedures are followed during the installation of the charges.

	Injuries due to a premature motor ignition	Improper storage of the motor, damage of the motor or early ignition.	Severe burns.	DI	Motor and igniters will be bought from official suppliers, properly installed by a certified mentor and ignited by the RSO.	Safety officer will ensure that installation of the motor and ignition are done by certified personnel.
	Injuries due to unpredictable flight path	Wind, faulty parachute, or instability in thrust.	If the rocket goes in unexpected areas, it could injure personnel or spectators.	DI	The rocket will not be launched during strong winds, the rocket design will be tested through simulations to make sure that it is stable during flight.	Weather conditions will be assessed, the rocket will be launched only if the RSO considers the weather safe. Multiple simulations will be run to ensure that the rocket is stable.

Table 12 Personnel Hazard Analysis

8.3 Failure Modes and Effects Analyses (FMEA)

Our proposed rocket and payload constitute a complex system with many parts, and as such there is potential for the failure of any component or system to jeopardize the chance of a successful flight. The failure modes and effects analyses below identify potential risks to the mission from a technical perspective, and classify such risks based upon the probability and severity of each. In order to give the mission the highest chance of success, mitigation techniques will be implemented such as including redundant backups of critical systems, performing simulations and testing, and checking components for quality. Failure to mitigate these hazards may result in damage to the rocket or payload, the inability to complete all team objectives, or at worse the complete loss of the mission. For these reasons, the team has completed a thorough analysis of potential failure modes and effects and will implement all proposed mitigation techniques to minimize risk to the mission.

FMEA Probability Definitions	
Rating	**Description**
A	The failure is probable if it is not mitigated.
B	The failure may occur if it is not mitigated.
C	The failure is unlikely to happen if it is not mitigated.
D	The failure is highly unlikely to happen if it is not mitigated.

Table 13 FMEA Probability Definitions

FMEA Severity Definitions	
Rating	**Description**
I	Complete loss of the item or system.
II	Significant damage to the item or system. Item requires major repairs or replacement before it can be used again.
III	Damage to the item or system which requires minor repairs or replacement before it can be used again.
IV	Damage is negligible.

Table 14 FMEA Probability Definitions

FMEA Probability	Severity			
	I - Irrecoverable	II - Significant	III - Minor	IV – Negligible
A – Probable	AI	AII	AIII	AIV
B – May Occur	BI	BII	BIII	BIV
C - Unlikely	CI	CII	CIII	CIV
D – Highly Unlikely	DI	DII	DIII	DIV

Table 15 FMEA Assessment Matrix

8.3.1 Launch Vehicle FMEA

Launch Vehicle FMEA					
Hazard	**Cause**	**Effect**	**Probability /Severity**	**Mitigation & Controls**	**Verification**
Vehicle does not separate at apogee	Insufficient ejection charge, altimeter failure	The rocket would descend at a dangerous terminal velocity. If the main parachute deploys at this speed, the airframe will most likely be severely damaged and the payload cannot safely deploy.	CI	Calculate appropriate ejection charge sizing, and ensure the correct quantities of black powder are used	Testing of the recovery system
Drogue parachute does not inflate	The parachute may not be packed properly, or it might be too tight of a fit in the airframe.	The rocket would descend more rapidly than anticipated velocity. If the main parachute deploys at this speed, the airframe and vehicle will most likely	CII	The drogue parachute will be properly sized and have a redundant system to deploy it.	Testing of the recovery system including subscale and full scale testing

		sustain minor damage and the payload cannot safely deploy.			
Parachute detaches from launch vehicle	Improper installation of the recovery system	This would result in the probable destruction of the rocket and payload upon ground impact as well as failure to complete the payload mission criteria. It could also injure personnel on the ground due to debris upon impact or impact near a person.	DII	Proper installation of the recovery system and select correct sizes of hardware to handle ejection forces.	Testing of recovery system including subscale and full scale testing
Main parachute does not deploy	The parachute may not be packed properly, or it might be too tight of a fit in the airframe.	If the drogue parachute deploys, the rocket would still fall at a high speed, leading to damage. The significance of the damage being less than if the drogue did not open.	CII	The main parachute will be properly sized and also have multiple systems to deploy it.	Testing of the recovery system including subscale and full scale testing

Melted or damaged parachute	The parachute bay is not properly sealed, or the parachutes are not packed correctly.	This could prevent the parachutes from slowing the rocket's descent rate, resulting in the possible loss of the rocket and payload.	DII	Proper protection and packing of the parachutes.	Testing of recovery system including subscale and full scale testing
Shock cord tangles	Parachutes are not packed properly	Could decrease the parachutes' effectiveness, resulting in the loss of the rocket and payload upon ground impact.	CII	Properly pack the parachutes	Testing of recovery system including subscale and full scale testing
Electronics bay is not secured properly	Electronic bay does not fit tightly into the airframe	Potential electronics and recovery failure	DII	Manufacture the electronics bay to fit accurately in the airframe	Subscale and full scale testing
Motor ejected from launch vehicle	The motor is secured improperly.	The motor could possibly go into freefall during flight. If it is still ignited, it may harm personnel in the vicinity or destroy the launch vehicle. It could also create free falling debris	DI	The motor will be installed by a certified mentor. The motor retention system will also be inspected prior to launching the rocket.	Subscale and full scale testing

		that could cause harm.			
Fins break off during ascent	Large aerodynamic forces or poor fin design	Rocket cannot be relaunched, damage to airframe or internal components	DII	Mount fins properly onto the airframe	Material testing of the fins and full scale testing
Rail buttons fail during launch	Unexpected forces, damage to attachment components	Rocket does not achieve sufficient stability, possible danger to personnel at large distance	DII	Calculate expected loads on rail buttons & attachment hardware, conduct qualitative “hang” test	Full scale testing
Launch rail/tower fails	Poorly maintained equipment, improper setup	Rocket does not safely exit rod, damage to vehicle, danger to personnel at a large distance	DI	Launch tower will be setup and maintained by a responsible person at the launch club, and inspected by the safety officer prior to launch	Full scale testing
Airframe separates during ascent	Improper connection of airframe sections; large aerodynamic forces cause the airframe to separate	Rocket cannot be relaunched, damage to airframe or internal components	DI	Couplers are tight enough within the airframe to keep the airframe sections attached during ascent	Complete analysis of coupler and material strength testing
Altimeter failure	Loss of power, low battery,	Incorrect altitude readings and	DI	There will be a backup altimeter	Altimeter testing included in

	disconnected wires, destruction by black powder charge, or burnt by charge detonation	altitude deployment; can result in potential loss of rocket and payload		with a second power source in case the main altimeter fails. There will also be a set of backup black powder charges connected to the backup altimeter. Both altimeters will also be tested before launch.	subscale and full scale testing
Altimeter switch failure	Switch comes loose or disarms during launch or component failure	Incorrect altitude readings and altitude deployment; can result in potential loss of rocket and payload	DI	Test switches before launch	Altimeter testing included in subscale and full scale testing
Recovery electronics bay failure	Loss of power, disconnected wires, destruction by black powder charge, or burnt by	Altimeter or recovery system failure	DII	Test the electronic bay and altimeter before launch	Subscale and full scale testing

	charge detonation				
Descent too fast	Parachute is too small	Potential damage or loss of rocket and payload	DII	Properly size parachute; test recovery system before launch	Subscale and Full scale testing and testing of recovery system
Motor Misfire	Damaged motor or damage to ignitor prior to launch.	Significant to unrepairable damage to the rocket and possibility of harm to personnel	DI	The motor is only handled by a certified team mentor. If there is a misfire, the team will wait at least 60 seconds before approaching the launch vehicle and will follow the instructions of the RSO.	Subscale and Full scale testing
Premature motor ignition	Damaged motor or accidental early ignition.	Possibility to harm personnel in vicinity during ignition.	DII	The motor will be replaced. It will be properly installed by a certified mentor and inspected by the RSO.	Subscale and Full scale testing
Motor fails to ignite	Ground support equipment failure, faulty or damaged motor	Launch vehicle cannot launch. Could possibly result in disqualificatio n of team	DIII	The ground support equipment will be maintained by	Full scale testing

				responsible persons from the launch site club. The motor will be stored according to specified guidelines.	
Premature ejection charge detonation	Inadvertent arming, recovery electronics failure	Minor damage to vehicle and harm to personnel in vicinity	DII	Arming switches will be locking, and detailed instructions will be kept and followed pertaining to the arming process.	Full scale testing
Shock cord is severed	Faulty shock cord, weak cord from repeated testing, destruction by black powder charge, or burnt by charge detonation	The parachutes would detach from the rocket, leading to the loss of the rocket and payload.	DI	The shock cord will be properly sized to handle ejection loads. It will also be inspected before the parachutes are packed. A Nomex blanket will protect the shock cord from fire damage and the black powder charges will be measured carefully.	Testing of recovery system including subscale and full scale testing

Fins do not keep the rocket stable	Damaged fins, improper fin sizing	Predicted apogee is not reached, vehicle sustains minor damage.	CII	Use OpenRocket simulations to make sure the fin design will keep the rocket stable	Subscale and full scale testing
Fins break off during landing	High impact during landing; point stresses on fins	Rocket cannot be relaunched	CII	Avoid fin designs with weak points and test fins with forces of final descent velocity	Material testing of the fins, and full scale testing
Descent too slow	Parachute is too large	Landing outside of max drift zone	CIII	Properly size parachute; test recovery system before launch	Subscale as well as Full scale testing and testing of recovery system
Pressure not equalized inside airframe	Vent holes are too small	Altimeters do not register accurate altitude	DII	The vent holes will be drilled according to recommend ations determined by external testing	Inspection and subscale and full scale testing
Airbrakes fail to deploy or deploy incorrectly	Electrical or software failure, mechanical parts become stuck	Vehicle overshoots expected apogee	BIV	The airbrake system will be tested prior to launch using simulated flight data, and hardware in the loop testing.	Testing of full scale vehicle

				Mechanical actuation will be attempted with expected loads	
Airbrakes deploy asymmetrically	Driving plate or fin pins fail in one section but not others	Vehicle experiences unexpected loads and flight forces, causing an unpredictable trajectory or damage to other components	DII	Conduct analysis of part mechanical strength. Airbrake system is designed to force all fins to deploy evenly when there is no damage to parts	Testing of full scale vehicle
Electronic Systems ignite	High temperatures, short circuits, physical damage	Significant damage to vehicle, danger to personnel in vicinity due to energetics or harmful gases	DII	Temperature monitored during launches, components tested independently, electronics protected from damage	Full scale testing
Avionics systems fail	Damaged components, faulty power system	Vehicle overshoots expected apogee, flight data is not recorded. GPS positions are not transmitted, causing	CIII	Test avionics systems before launch, verify functionality	Full scale testing

		possible loss of vehicle			
Payload comes loose in payload bay	Damaged components, improperly designed retention system	Minor damage to vehicle, alteration of flight path	CIII	Perform analysis of payload retention system under expected flight loads, and test strength prior to launch	Payload demonstration flight

Table 16 Launch Vehicle FMEA

8.3.2 Payload FMEA

Payload FMEA					
Hazard	**Cause**	**Effect**	**Probability /Severity**	**Mitigation & Controls**	**Verification**
Payload retention failure	Severe damage to the upper airframe and retention pins	Payload deploys prior to apogee	DI	Inspection of upper airframe and retention pins prior to flight	WPI HPRC will create a payload inspection checklist
Retention system becomes insecure	Damage to retention pins	Payload rattles within upper airframe and causes damage to itself	DII	Inspection of upper airframe and retention pins prior to flight	WPI HPRC will create a payload inspection checklist
Payload Ejection failure	Incomplete separation of upper airframe	Entire launch vehicle tumbles until main deployment	DI	Inspection of black powder charges and wiring	WPI HPRC will create a rocket inspection checklist
Payload becomes damaged during ejection process	Excessive forces on shock cord during deployment	Payload is damaged	DII	Inspection of shock cord	WPI HPRC will create a rocket inspection checklist

Battery catches fire	Overheating of the internals of the payload during launch or outside temperature, faulty battery, incorrect wiring leading to an ignition, ignition within rocket that impacts the security of the payload	The rocket catches on fire and burns during launch, the rocket becomes ballistic and could hurt the environment or people in the crowd, the drone is destroyed and unable to complete its mission	DI	WPI HPRC will design the lander to be well ventilated to prevent overheating.	The lander will be run at acceptable levels to not overexert the battery's
Failure of tender descender	Improper wiring of pyro charge or improper programming of altimeter	Payload remains tethered to the rocket for the full descent	DIII	All wiring and pyro charges will be inspected prior to integration and launch	WPI HPRC will create a payload inspection checklist
Failure of Jolly Logic chute releases	Improper programming and actuation	Freefall of lander and potential loss of lander	DI	Jolly Logics will be inspected prior to launch to look for any catching and battery's will be charged	WPI HPRC will create a payload inspection checklist

Table 17 Payload FMEA

8.4 Environmental Concerns

Beyond the hazards identified above, environmental concerns must also be considered to ensure the safe and successful completion of the project. Various environmental factors which may negatively impact our mission were considered. These effects include both risks to the safety of our team members and risks to the successful flight and operation of the rocket and payload. Furthermore, it was considered how the rocket and team activities may have adverse effects on the environment. The possible risks identified have been classified based upon the probability and severity of each. A plan for mitigation accompanies each hazard identified. Failure to mitigate environmental hazards could result in unsafe conditions for team members, damage or malfunction of the rocket or payload, negative environmental impact, or at worse loss of the mission. The team will utilize this information to implement safe practices and minimize the risks to the project resulting from environmental-factors.

Environmental Conditions Probability Definitions	
Rating	**Description**
A	The condition is probable if it is not mitigated.
B	The condition may occur if it is not mitigated.
C	The condition is unlikely to happen if it is not mitigated.
D	The condition is highly unlikely to happen if it is not mitigated.

Table 18 Environmental Conditions Probability Definitions

Environmental Conditions Severity Definitions	
Rating	**Description**
I	The condition may cause death or permanent disability to personnel or loss of the system.
II	The condition may cause major injuries or significant damage to the system.
III	The condition may cause injury or minor damage to the system.
IV	The condition may cause minor injury or negligible damage to the system.

Table 19 Environmental Conditions Severity Definitions

Environmental Probability	Severity			
	I - Irrecoverable	II - Significant	III - Minor	IV – Negligible
A – Probable	AI	AII	AIII	AIV
B – May Occur	BI	BII	BIII	BIV
C - Unlikely	CI	CII	CIII	CIV
D – Highly Unlikely	DI	DII	DIII	DIV

Table 20 Environmental Concerns Assessment Matrix

Environmental Concerns				
Category	Hazard	Effect	Probability/ Severity	Mitigation
Environmental Risks to Rocket and Payload	Terrain	Hazardous terrain such as steep slopes or rough surface could pose a risk of damaging the rocket and payload upon landing.	DIII	The team will launch only at sanctioned launch sites where there is large area of open and flat terrain.
	Low Visibility	Unable to track the location of the launch vehicle and payload during flight.	DII	The team will not launch the rocket in low visibility conditions.
	High Temperatures	Overheated motors or energetics could start a fire and light any flammable objects in the area. This could also be a danger to circuits.	DIII	The electronics will be inspected and tested to prevent shorts and anything else that could cause overheating. Motors will be safely installed and arranged in

				a way to prevent them from stalling or being affected by other things that may overheat them.
	Low Temperatures	Low temperatures could cause batteries as well as circuits to not perform properly. This may also cause shrinkage in the airframe or other components dependent on the structural properties of the material.	AIII	LiPo batteries will be used as they function better compared to others in cold temperatures. Material selected will be less likely to shrink in the cold or the tolerance of such shrinkage will be accounted for in design.
	Trees	Due to winds or an unpredicted flight path, the launch vehicle or payload could end up hitting or landing in a tree.	DIII	The launch vehicle will be launched in an open field and aimed in a direction with wind in mind and far from any trees to ensure the best chance of avoiding trees.
	Birds	If the launch vehicle hits a bird, it could damage the	DIII	The rocket will not be launched when they are any birds in

		launch vehicle and alter its trajectory depending on the size of the bird. It will also harm the bird.		close proximity to the launchpad.
	Low flying aircraft, drones	If the launch vehicle hits an aircraft or drone, significant damage would occur to the launch vehicle and to the aircraft or drone. Passengers could also be put in danger.	DII	The rocket will not be launched in proximity to any drones. Members will monitor for low flying aircraft, and the rocket will not be launched with any in the area. Launches will be approved with the FAA to alert pilots to the danger.
	High Humidity/ Rain	If components get significantly wet in any way, it could cause some material to warp or damage electronics. If it lands in water, it may also disturb animals or plants within the body of water it lands.	DII	The launch vehicle will not be launched near a significantly large unfrozen body of water, nor in severe or prolonged rain. Members will also refrain from working on components near open containers of liquid.
	Strong Winds	Unsafe alterations to launch vehicle's	DII	Alter course and adjust trajectory to

		trajectory including excessive drift after parachute deployment.		prevent launch vehicle's landing from leaving the exclusion zone. If the RSO deems the winds to be too high, the team will wait for the winds to die down.
	Sand	If the launch vehicle lands in sand or has sand blown into it, it could disrupt or get stuck in small components.	DIII	The launch vehicle will not be launched near a significantly sandy area.
	Plants and Animals	Launching too close to animals and plants could result in it damaging plants and possibly any animals in the area as well as the deployed payload.	DIII	The launch vehicle will not be launched in a field with animals or protected plants in significant number close by.
	Obstruction	A plant, rock, or other object could get in the way of the system(s) deploying and get damaged or prevent the system from functioning.	DII	The systems will be designed to deploy slowly in order to minimize potential damage to it and to any surroundings.

Environmental Risks to Personnel	Hot Weather	High temperature conditions may pose health risks to team members and bystanders including sunburn, dehydration, heat exhaustion, and heat stroke.	CIII	The team will monitor the weather forecast before outdoor events. If high temperatures are predicted the team will bring sunscreen and extra water and find shade in order to prevent sunburn and heat sickness.
	Cold Weather	Low temperature conditions may pose health risks to team members and bystanders such as frostbite and hypothermia.	AIII	The team will monitor the weather forecast before outdoor events. If low temperatures are predicted the team will instruct team members to dress warm and bring layers and will provide hand warmers and blankets to protect against the cold.
	Wet Conditions	Wet weather conditions may pose health risks to team members and bystanders such as	CIII	The team will monitor the weather before outdoor events and know when to expect wet conditions.

		hypothermia, and will cause terrain to be slippery and more treacherous.		Members will be instructed to bring adequate raingear and extra dry clothing and will be warned about treacherous wet terrain.
	Thunderstorms /Severe Weather	Severe weather events such as heavy wind and rain events, thunderstorms, tornados, and blizzards may pose a serious threat to the safety of team members and bystanders.	DI	The team will monitor the weather before outdoor events and know if severe weather is expected. The team will not attempt any launch activities when storms or severe weather are expected. In the event of an unexpected severe weather event, the team will immediately cease outdoor activities and move to a safe indoor location.
	Uneven or Hazardous Terrain	Traversing uneven or hazardous terrain poses a risk to team members of tripping or falling and suffering	CIII	The team will only conduct outdoor events and launch activities in large, flat, open spaces where the danger posed by

		resulting injuries.		terrain is minimal.
	Wildlife	Interactions with wild animals may create a dangerous situation for team members and bystanders.	DII	The team will conduct outdoor events in open areas where dangerous wild animals are unlikely to be found. In the event that a wild animal approaches and does not flee, the team will avoid confrontation and move to another location.
	Unsafe Landing Location	In the event that the rocket or payload lands in an unexpected or unsafe location, retrieval of the rocket and payload could pose a risk of injury to team members.	CI	The team will minimize the risk of an unsafe landing location by only launching at sanctioned launch sites where there is a large open area, and wind will be considered in angling the launch rail away from potential hazards. The team will never attempt to retrieve the rocket or

				payload from an excessively dangerous locations such as treacherous terrain, across busy highways, or on powerlines.
Adverse Effects on the Environment	Fire at Launchpad	High temperature exhaust from the motor has a chance to light flammable objects on fire if they are too close.	DII	The vehicle will be launched on a launch rail with a blast deflector. The area will be cleared of flammable materials.
	Expulsion of Debris During Flight	Any parts or debris from the rocket expelled during flight and left behind at the launch site could have detrimental effects on the natural environment.	CIII	The launch vehicle and payload will be designed such that all components will remain intact and retained by the airframe. Any parts or debris lost during flight will be located and removed from the launch site to the best of the team's ability.
	Destruction of Launch Vehicle	In the event that the launch vehicle explodes or	DII	The team will design the rocket and payload with

		suffers severe damage such that it is irrecoverable, the debris left behind could have detrimental effects on the natural environment.		safety in mind and test all systems before launch in order to minimize the risk of catastrophic failure. In the event of such a failure, surviving parts and debris will be removed from the launch site to the best of the team's ability.
	Hazardous Material Spill	Environmental damage could as the result of a leak or spill of a hazardous material, such as chemicals contained within the batteries and rocket motor.	DIII	The team will take care to carefully inspect the launch vehicle and payload, batteries, and motor before all rocket launches and follow relevant safety checklists to minimize the risk of a hazardous material spill.
	Collision with Structure or vehicle	In the event that the rocket or payload collides with any structure, vehicle, or other object, damage may result to	DII	The team will only launch at sanctioned launch sites and fly rockets on trajectories away from any structures or

		the object as a result of the impact.		vehicles. The launch vehicle will be designed to have an acceptable kinetic energy at landing to minimize the impact of any collisions.
	Destruction of Environment During Retrieval	Team members may cause damage to the environment when going to locate and retrieve the rocket and payload after the completion of the mission.	BIV	Team members will take care to minimize environmental impact when going out to retrieve the rocket and payload and will follow designated paths whenever possible.
	Improper Waste Disposal	Damage to the natural environment may occur if team members or bystanders do not properly dispose of all waste generated during outdoor activities and launch events.	BIII	WPI HPRC will reduce environmental impact by properly disposing of all waste in designated containers and will carry out all materials and leave nothing behind when the team departs to the launch site.

Table 21 Environmental Concerns

8.5 Material Safety Data Sheets (MSDS)

The WPI HPRC team maintains current revisions of Material Safety Data Sheets for all potentially hazardous materials used in the construction and fabrication of the rocket and payload. MSDS sheets serve as the first resource for material safety and will always be consulted before the handling or use of any material which may pose a health risk to team members. All relevant MSDS information for the specific hazardous materials planned to be used in construction for this year's rocket and payload can be found in the appendix section. The table below provides a list of these materials, their intended uses, and the location of the relevant MSDS information in the appendix section.

Material	Use	MSDS Sheet
Carbon Fiber	Fin Construction	[6]
Aluminum	Bulkheads, fasteners, coatings, shielding	[7]
Fiberglass	Airframe	[8]
NylonX	3D printed components	[9]
PLA	3D printed components	[10]
Epoxy Resin	Conjoining parts of the rocket, filling holes	[11]
Delrin Plastic	Airbrake system	[12]
LiPo Battery	Payload Component	[13]
Black Powder	Separation of airframe sections	[14]
Ammonium Perchlorate Composite Propellant (APCP)	Used in rocket motor	[15]
Igniter Pyrogen	Motor ignition	[16]

Table 22 Material Safety Data Sheets

9 Project Plan

9.1 Requirements Verification

9.1.1 NASA Requirements

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
General Requirements				
NASA-1.1	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). Teams will submit new work. Excessive use of past work will merit penalties.	Inspection	WPI HPRC will maintain records of member participation. Members will complete and submit milestone documents. Mentors will not contribute to the reports, or design and construction of the vehicle except for providing general guidance.	In Progress
NASA-1.2	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.	Inspection	WPI HPRC will create a maintained project plan in the form of a Gantt chart by including it in documentation. This will be reviewed throughout the project process.	In Progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-1.3	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 on site at Marshall Space Flight Center.	Inspection	WPI HPRC will notify NASA of foreign nationals via the mode specified by NASA.	Verified
NASA-1.4	The team must identify all team members who plan to attend Launch Week activities by the Critical Design Review (CDR). Team members will include:	Inspection	WPI HPRC will not be attending NASA Launch Week activities. NASA will officially be notified of this by Critical Design Review (CDR).	Not Verified
NASA-1.4.1	Students actively engaged in the project throughout the entire year.	Inspection	WPI HPRC will track attendance and maintain a list of active members.	Verified
NASA-1.4.2	One mentor (see requirement 1.13).	Inspection	WPI HPRC will identify this mentor: Jason Nadeau.	Verified
NASA-1.4.3	No more than two adult educators.	Inspection	WPI HPRC will identify their adult educator: John Blandino.	Verified

NASA Requirements				
Requirement No.	Description	Verification Method	Verification Plan	Status
NASA-1.5	The team will engage a minimum of 200 participants in educational, hands-on science, technology, engineering, and mathematics (STEM) activities. These activities can be conducted in-person or virtually. To satisfy this requirement, all events must occur between project acceptance and the FRR due date. The STEM Engagement Activity Report must be submitted via email within two weeks of the completion of each event. A template of the STEM Engagement Activity Report can be found on pages 36-38.	Inspection	WPI HPRC will host and/or participate in outreach events in the Worcester area. The team will take attendance at events and the engagement officer will submit all STEM Engagement Activity Reports on time.	In Progress
NASA-1.6	The team will establish a social media presence to inform the public about team activities.	Inspection	WPI HPRC will demonstrate this by having the PR officer consistently posting content on social media.	In Progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-1.7	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. Late submissions of milestone documents will be accepted up to 72 hours after the submission deadline. Late submissions will incur an overall penalty. No milestone documents will be accepted beyond the 72-hour window. Teams that fail to submit milestone documents will be eliminated from the project.	Inspection	WPI HPRC will demonstrate this by submitting documentation early or on time.	In Progress
NASA-1.8	All deliverables must be in PDF format.	Inspection	WPI HPRC will ensure all deliverables are PDFs and end in a .pdf file extension, as monitored by the documentation officer.	In Progress
NASA-1.9	In every report, teams will provide a table of contents including major sections and their respective sub-sections.	Inspection	WPI HPRC will use Microsoft Word's automatic table of contents feature, as monitored by the documentation officer.	In Progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-1.10	In every report, the team will include the page number at the bottom of the page.	Inspection	WPI HPRC will use Microsoft Word's automatic page numbering feature, as monitored by the documentation officer.	In Progress
NASA-1.11	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.	Inspection	Each team member of WPI HPRC will inspect their own personal audio and visual equipment prior to presentations to ensure they are in working order.	In Progress
NASA-1.12	All teams attending Launch Week will be required to use the launch pads provided by Student Launch's launch services provider. No custom pads will be permitted at the NASA Launch Complex. At launch, 8-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.	Inspection	The team will demonstrate this by designing and constructing the subscale launch vehicle using 1010 rail buttons and the full scale launch vehicle using 1515 rail buttons. Although the team will not be present during Launch Week activities.	In Progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-1.13	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 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.	Inspection	WPI HPRC will choose their mentor, Jason Nadeau, based on the qualifications outlined. The team will include the information of its mentor in documentation.	Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-1.14	Teams will track and report the number of hours spent working on each milestone.	Inspection	WPI HPRC will record attendance at all subteam, division and general body meetings. These hours of meetings will be totaled and in design review documentation.	In Progress
Vehicle Requirements				
NASA-2.1	The vehicle will deliver the payload to an apogee altitude between 3,500 and 5,500 feet above ground level (AGL). Teams flying below 3,000 feet or above 6,000 feet on Launch Day will receive zero altitude points towards their overall project score and will not be eligible for the Altitude Award.	Analysis	WPI HPRC will simulate the vehicle in OpenRocket and with a custom simulator to ensure the apogee falls within bounds.	In Progress
NASA-2.2	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.	Inspection	WPI HPRC will report a target apogee in the PDR report.	Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.3	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. This altimeter may also be used for deployment purposes (see Requirement 3.4)	Inspection	WPI HPRC will verify the final design of the vehicle calls for at least one commercial altimeter. The current design calls for two.	In progress
NASA-2.4	The launch vehicle will be designed to be recoverable and reusable. Reusable is defined as being able to launch again on the same day without repairs or modifications.	Demonstration	WPI HPRC will reuse the vehicle after test flights for competition	Not Verified
NASA-2.5	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.	Inspection	WPI HPRC will verify the final design does not exceed 4 independent sections.	In progress
NASA-2.5.1	Coupler/airframe shoulders which are located at in-flight separation points will be at least 1 body diameter in length.	Inspection	WPI HPRC will ensure the couplers extend at least ½ body diameter into each airframe section.	In progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.5.2	Nosecone shoulders which are located at in-flight separation points will be at least ½ body diameter in length.	Inspection	WPI HPRC will ensure the nosecone coupler extends ¼ body diameter into the body section.	In progress
NASA-2.6	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.	Demonstration	WPI HPRC will demonstrate the vehicle preparation during test launches	Not Verified
NASA-2.7	The launch vehicle and payload 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, although the capability to withstand longer delays is highly encouraged.	Testing	WPI HPRC will test electronics in flight ready configurations for at least 2 hours.	Not Verified
NASA-2.8	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.	Inspection	WPI HPRC will ensure the selected motor can be ignited by a standard firing system	In progress
NASA-2.9	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).	Inspection	WPI HPRC will ensure the vehicle does not require any external ground support equipment	In progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.10	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).	Inspection	WPI HPRC will select motors from only commercially available sources.	In progress
NASA-2.10.1	Final motor choices will be declared by the Critical Design Review (CDR) milestone.	Inspection	WPI HPRC will report final motor choices in the CDR report.	Not Verified
NASA-2.10.2	Any motor change after CDR must be approved by the NASA Range Safety Officer (RSO). Changes for the sole purpose of altitude adjustment will not be approved. 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.	Inspection	WPI HPRC will seek approval for any motor change post-CDR.	Not Verified
NASA-2.11	The launch vehicle will be limited to a single stage.	Inspection	WPI HPRC will verify the launch vehicle does not use more than one stage.	In progress
NASA-2.12	The total impulse provided by a College or University launch vehicle will not exceed 5,120 Newton-seconds (L-class). The total impulse provided by a High School or Middle School launch vehicle will not exceed 2,560 Newton-seconds (K-class).	Inspection	WPI HPRC will verify the selected motor falls in or below the L-class category.	In progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.13	Pressure vessels on the vehicle will be approved by the RSO and will meet the following criteria:	Inspection	WPI HPRC will present the vehicle to the RSO for inspection.	Not Verified
NASA-2.13.1	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.	Analysis	WPI HPRC will simulate pressure vessels, and the pressures will be compared to known burst pressures.	Not Verified
NASA-2.13.2	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.	Inspection	WPI HPRC will inspect pressure systems to ensure the relief value is suitable.	Not Verified
NASA-2.13.3	The full pedigree of the tank will be described, including the application for which the tank was designed and the history of the tank. This will include the number of pressure cycles put on the tank, the dates of pressurization/depressurization, and the name of the person or entity administering each pressure event.	Inspection	WPI HPRC will present tank history in reports.	Not Verified
NASA-2.14	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.	Analysis	WPI HPRC will use OpenRocket simulations to determine rail exit stability.	In progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.15	Any structural protuberance on the rocket will be located aft of the burnout center of gravity. Camera housings will be exempted, provided the team can show that the housing(s) causes minimal aerodynamic effect on the rocket's stability.	Analysis	WPI HPRC will determine the burnout CG using OpenRocket and compare protuberance locations.	In progress
NASA-2.16	The launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit.	Analysis	WPI HPRC will simulate the rail exit velocity in OpenRocket.	In progress
NASA-2.17	All teams will successfully launch and recover a subscale model of their rocket prior to CDR. The subscale flight may be conducted at any time between proposal award and the CDR submission deadline. Subscale flight data will be reported at the CDR milestone. Subscales are not required to be high power rockets.	Inspection	WPI HPRC will launch a subscale vehicle and present the flight results at the CDR milestone	In progress
NASA-2.17.1	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.	Inspection	WPI HPRC will design the vehicle to resemble the full scale vehicle.	In progress
NASA-2.17.2	The subscale model will carry an altimeter capable of recording the model's apogee altitude.	Inspection	WPI HPRC will include an altimeter on the subscale vehicle.	In progress
NASA-2.17.3	The subscale rocket must be a newly constructed rocket, designed and built specifically for this year's project.	Inspection	WPI HPRC will construct an all new rocket for the subscale.	In progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.17.4	Proof of a successful flight shall be supplied in the CDR report. Altimeter data output may be used to meet this requirement.	Inspection	WPI HPRC will include subscale flight data in the CDR report.	Not Verified
NASA-2.18	All teams will complete demonstration flights as outlined below.			
NASA-2.18.1	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.). The following criteria must be met during the full-scale demonstration flight:	Demonstration	WPI HPRC will launch and recover the full-scale vehicle prior to the FRR deadline	Not Verified
NASA-2.18.1.1	The vehicle and recovery system will have functioned as designed.	Demonstration	WPI HPRC will demonstrate the recovery system's functionality in the full-scale test flight	Not Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.18.1.2	The full-scale rocket must be a newly constructed rocket, designed and built specifically for this year's project.	Inspection	WPI HPRC will ensure the full-scale vehicle is all new for this competition year.	Not Verified
NASA-2.18.1.3	The payload does not have to be flown during the full-scale Vehicle Demonstration Flight. The following requirements still apply:			
NASA-2.18.1.3.1	If the payload is not flown, mass simulators will be used to simulate the payload mass.	Inspection	WPI HPRC will ensure either the payload or a mass simulator is included during the full scale test flight.	Not Verified
NASA-2.18.1.3.2	The mass simulators will be located in the same approximate location on the rocket as the missing payload mass.	Inspection	WPI HPRC will verify the mass simulator lies at the same CG of the payload.	Not Verified
NASA-2.18.1.4	If the payload changes the external surfaces of the rocket (such as 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.	Inspection	WPI HPRC will ensure any active systems on payload are functional for the test flight.	Not Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.18.1.5	Teams shall fly the Launch Day motor for the Vehicle Demonstration Flight. The team may request a waiver for the use of an alternative motor in advance if the home launch field cannot support the full impulse of the Launch Day motor or in other extenuating circumstances.	Demonstration	WPI HPRC will launch the full-scale vehicle on the launch day motor, or an approved alternative.	Not Verified
NASA-2.18.1.6	The vehicle must be flown in its fully ballasted configuration during the full-scale test flight. Fully ballasted refers to the maximum 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.	Inspection	WPI HPRC will ensure all ballast is added to any ballast systems included on the vehicle.	Not Verified
NASA-2.18.1.7	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).	Inspection	WPI HPRC will receive RSO approval for changes after the full-scale test flight.	Not Verified
NASA-2.18.1.8	Proof of a successful flight shall be supplied in the FRR report. Altimeter data output is required to meet this requirement.	Inspection	WPI HPRC will supply flight data in the FRR report.	Not Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.18.1.9	Vehicle Demonstration flights must be completed by the FRR submission deadline. No exceptions will be made. 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.	Demonstration	WPI HPRC will ensure the vehicle test flight takes place before the FRR deadline.	Not Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.18.2	Payload Demonstration Flight - All teams will successfully launch and recover their full-scale rocket containing the completed payload prior to the Payload Demonstration Flight deadline. The rocket flown must be the same rocket to be flown on Launch Day. The purpose of the Payload Demonstration Flight is to prove the launch vehicle's ability to safely retain the constructed pay-load during flight and to show that all aspects of the payload perform as designed. A successful flight is defined as a launch in which the rocket experiences stable ascent and the payload is fully retained until it is deployed (if applicable) as designed. The following criteria must be met during the Payload Demonstration Flight:	Demonstration	WPI HPRC will launch and recover the full-scale vehicle with an active payload system.	Not Verified
NASA-2.18.2.1	The payload must be fully retained until the intended point of deployment (if applicable), all retention mechanisms must function as designed, and the retention mechanism must not sustain damage requiring repair.	Demonstration	WPI HPRC will demonstrate that the payload was retained until the intended deployment.	Not Verified
NASA-2.18.2.2	The payload flown must be the final, active version.	Inspection	WPI HPRC will verify the payload flown is the final, active version.	Not Verified

NASA Requirements				
Requirement No.	Description	Verification Method	Verification Plan	Status
NASA-2.18.2.3	If the above criteria are 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.	Inspection	WPI HPRC will verify the payload demonstration flight requirements are met during at least one test flight.	Not Verified
NASA-2.18.2.4	Payload Demonstration Flights must be completed by the FRR Addendum deadline. NO EXTENSIONS WILL BE GRANTED.	Inspection	WPI HPRC will ensure the payload demonstration flight is completed by the FRR Addendum deadline	Not Verified
NASA-2.19	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.	Inspection	WPI HPRC will produce a FRR Addendum if required.	Not Verified
NASA-2.19.1	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 a final competition launch.	Inspection	WPI HPRC will verify the FRR Addendum is submitted accordingly before the competition launch.	Not Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.19.2	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 a final competition launch.	Inspection	WPI HPRC will verify the Payload Demonstration flight was completed successfully before the competition launch.	Not Verified
NASA-2.19.3	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 granted if the RSO or the Review Panel have any safety concerns.	Inspection	WPI HPRC will ensure the Payload demonstration flight was completed successfully and will petition the RSO for permission to fly at launch week if necessary.	Not Verified
NASA-2.20	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.	Inspection	WPI HPRC will ensure contact info is present on the airframe before launch.	Not Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.21	All Lithium Polymer batteries will be sufficiently protected from impact with the ground and will be brightly colored, clearly marked as a fire hazard, and easily distinguishable from other payload hardware.	Inspection	WPI HPRC will ensure LiPo batteries are appropriately marked before launch.	Not Verified
NASA-2.22	Vehicle Prohibitions			
NASA-2.22.1	The launch vehicle will not utilize forward firing motors.	Inspection	WPI HPRC will ensure the vehicle design does not use forward firing motors.	Verified
NASA-2.22.2	The launch vehicle will not utilize motors that expel titanium sponges (Sparky, Skidmark, Metal Storm, etc.)	Inspection	WPI HPRC will ensure selected motors are not of the type described.	Verified
NASA-2.22.3	The launch vehicle will not utilize hybrid motors.	Inspection	WPI HPRC will ensure the vehicle design does not use hybrid motors.	Verified
NASA-2.22.4	The launch vehicle will not utilize a cluster of motors.	Inspection	WPI HPRC will ensure the vehicle design does not use a motor cluster.	Verified
NASA-2.22.5	The launch vehicle will not utilize friction fitting for motors.	Inspection	WPI HPRC will ensure the motor retention design does not rely of friction fitting.	Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-2.22.6	The launch vehicle will not exceed Mach 1 at any point during flight.	Analysis	WPI HPRC will simulate the vehicle's flight in OpenRocket to verify the vehicle does not exceed Mach 1.	In progress
NASA-2.22.7	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 an unballasted weight of 40 lbs. on the pad may contain a maximum of 4 lbs. of ballast).	Inspection	WPI HPRC will ensure the ballast weight does not exceed 10% of the vehicle weight before launch.	In progress
NASA-2.22.8	Transmissions from onboard transmitters, which are active at any point prior to landing, will not exceed 250 mW of power (per transmitter).	Analysis	WPI HPRC will verify the total telemetry output does not exceed 250 mW during flight	Not Verified
NASA-2.22.9	Transmitters will not create excessive interference. Teams will utilize unique frequencies, hand-shake/passcode systems, or other means to mitigate interference caused to or received from other teams.	Analysis	WPI HPRC will ensure transmitters on the vehicle to not interfere with one another.	Not Verified
NASA-2.22.10	Excessive and/or dense metal will not be utilized in the construction of the vehicle. Use of light-weight metal will be permitted but limited to the amount necessary to ensure structural integrity of the airframe under the expected operating stresses.	Analysis	WPI HPRC will simulate the rocket using FEA and other structural simulation methods to ensure metal used only where necessary.	In progress
Recovery System Requirements				

NASA Requirements				
Requirement No.	Description	Verification Method	Verification Plan	Status
NASA-3.1	The full scale 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.	Inspection, Analysis	WPI HPRC will ensure the design of the recovery system maintains a reasonable descent energy as determined by the RSO	In progress
NASA-3.1.1	The main parachute shall be deployed no lower than 500 feet.	Demonstration	WPI HPRC will demonstrate the main parachute deployment altitude in the test flights.	Not Verified
NASA-3.1.2	The apogee event may contain a delay of no more than 2 seconds.	Inspection	WPI HPRC will ensure backup altimeters are set with a delay of no more than 2 seconds.	In progress
NASA-3.1.3	Motor ejection is not a permissible form of primary or secondary deployment.	Inspection	WPI HPRC will ensure redundant systems do not include motor ejection.	In progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-3.2	Each team will perform a successful ground ejection test for all electronically initiated recovery events prior to the initial flights of the subscale and full scale vehicles.	Testing	WPI HPRC will perform a ground ejection test for the drogue and main deployment before launching.	Not Verified
NASA-3.3	Each independent section of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf at landing.	Analysis	WPI HPRC will use OpenRocket and a custom descent simulator to verify landing kinetic energies per section.	In progress
NASA-3.4	The recovery system will contain redundant, commercially available altimeters. The term "altimeters" includes both simple altimeters and more sophisticated flight computers.	Inspection	WPI HPRC will ensure the recovery system includes redundant commercial altimeters.	In progress
NASA-3.5	Each altimeter will have a dedicated power supply, and all recovery electronics will be powered by commercially available batteries.	Inspection	WPI HPRC will ensure the recovery system includes redundant commercial power systems	In progress
NASA-3.6	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.	Inspection	WPI HPRC will ensure the altimeter arming switches are in place and accessible for launch.	In progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-3.7	Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	Inspection	WPI HPRC will use switches shown to be capable of withstanding flight forces without triggering.	In progress
NASA-3.8	The recovery system electrical circuits will be completely independent of any payload electrical circuits.	Inspection	WPI HPRC will ensure separation between recovery and payload electronics	In progress
NASA-3.9	Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.	Inspection	WPI HPRC will ensure shear pins of appropriate size and number are used to secure parachute bays.	In progress
NASA-3.10	The recovery area will be limited to a 2,500 ft. radius from the launch pads.	Analysis	WPI HPRC will simulate drift during recovery, and will not launch in winds which would cause the rocket to drift more than 2500 ft.	In progress
NASA-3.11	Descent time of the launch vehicle will be limited to 90 seconds (apogee to touch down). The jettisoned payload (planetary lander) is not subject to this constraint.	Analysis, Demonstration	WPI HPRC will simulate descent time, and verify calculations during test launches	In progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-3.12	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.	Inspection, Testing	WPI HPRC will verify the inclusion of a tracking device and will test the transmission capabilities prior to launch.	In progress
NASA-3.12.1	Any rocket section or payload component, which lands untethered to the launch vehicle, will contain an active electronic tracking device.	Inspection	WPI HPRC will verify the inclusion of tracking devices on all sections descending separately.	In progress
NASA-3.12.2	The electronic tracking device(s) will be fully functional during the official flight on Launch Day.	Inspection, Demonstration	WPI HPRC will verify the functionality of tracking devices before launch.	Not Verified
NASA-3.13	The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	Analysis, Testing	WPI HPRC will determine the effects of other electronic systems on the recovery system and will test to ensure functionality.	In progress
NASA-3.13.1	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.	Inspection	WPI HPRC will locate the recovery system separate from other electronic systems	Verified

Requirement No.	Description	Verification Method	Verification Plan	Status
NASA Requirements				
NASA-3.13.2	The recovery system electronics will be shielded from all onboard transmitting devices to avoid inadvertent excitation of the recovery system electronics.	Analysis, Testing	WPI HPRC will verify that other electronics cannot interfere with recovery electronics.	In progress
NASA-3.13.3	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.	Analysis, Testing	WPI HPRC will verify that devices that generate a magnetic field cannot interfere with recovery electronics.	In progress
NASA-3.13.4	The recovery system electronics will be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	Analysis, Testing	WPI HPRC will verify that other electronics cannot interfere with recovery electronics.	Not Verified
Payload Experiment Requirements				
NASA- 4.1	High School/Middle School Division – Teams may design their own science or engineering experiment or may choose to complete the College/University Division mission. Data from the science or engineering experiment will be collected, analyzed, and reported by the team following the scientific method.			

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-4.2	College/University Division – Teams will design a planetary landing system to be launched in a high-power rocket. The lander system will be capable of being jettisoned from the rocket during descent, landing in an upright configuration or autonomously uprighting after landing. The system will self-level within a five-degree tolerance from vertical. After autonomously uprighting and self-leveling, it will take a 360-degree panoramic photo of the landing site and transmit the photo to the team. The method(s)/design(s) utilized to complete the payload mission will be at the teams' discretion and will be permitted so long as the designs are deemed safe, obey FAA and legal requirements, and adhere to the intent of the challenge. An additional experiment (limit of 1) is allowed, and may be flown, but will not contribute to scoring. If the team chooses to fly an additional experiment, they will provide the appropriate documentation in all design reports so the experiment may be reviewed for flight safety.	Inspection	WPI HPRC will be participating in the College/University Division Payload Mission.	Verified
NASA-4.3	Primary Landing System Mission Requirements			

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-4.3.1	The landing system will be completely jettisoned from the rocket at an altitude between 500 and 1,000 ft. AGL. The landing system will not be subject to the maximum descent time requirement (Requirement 3.11) but must land within the external borders of the launch field. The landing system will not be tethered to the launch vehicle upon landing.	Analysis +Testing	WPI HPRC will design the recovery system and utilize decent calculations to ensure landing within the field. The payload will also perform deployment tests	In Progress
NASA-4.3.2	The landing system will land in an upright orientation or will be capable of reorienting itself to an upright configuration after landing. Any system designed to reorient the lander must be completely autonomous	Analysis +Testing	WPI HPRC will design a self-righting system to orient the payload into an upright position post landing. WPI HPRC will conduct tests on the system post construction.	In Progress
NASA-4.3.3	The landing system will self-level to within a five-degree tolerance from vertical.	Analysis +Testing	WPI HPRC will utilize the stabilization system to level the payload within 5 degrees of vertical post self-righting.	In Progress
NASA-4.3.3.1	Any system designed to level the lander must be completely autonomous.	Analysis	WPI HPRC will design the control system for the payload to be entirely autonomous.	In Progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-4.3.3.2	The landing system must record the initial angle after landing, relative to vertical, as well as the final angle, after reorientation and self-levelling. This data should be reported in the Post Launch Assessment Report (PLAR).	Analysis	WPI HPRC will design the control system for the payload to record and stream orientation data to the ground station throughout the process.	In Progress
NASA-4.3.4	Upon completion of reorientation and self-levelling, the lander will produce a 360-degree panoramic image of the landing site and transmit it to the team.	Analysis +Testing	WPI HPRC will utilize a 360-degree panoramic camera to take a photo of the environment.	In Progress
NASA-4.3.4.1	The hardware receiving the image must be located within the team's assigned prep area or the designated viewing area.	Inspection	WPI HPRC will ensure that all equipment for receiving images and telemetry will be located in the prep area.	In Progress
NASA-4.3.4.2	Only transmitters that were onboard the vehicle during launch will be permitted to operate outside of the viewing or prep areas.	Design	WPI HPRC's payload will only utilize the transmitters onboard connected to the main microprocessor.	In Progress

NASA Requirements				
Requirement No.	Description	Verification Method	Verification Plan	Status
NASA-4.3.4.3	Onboard payload transmitters are limited to 250 mW of RF power while onboard the launch vehicle but may operate at a higher RF power after landing on the planetary surface. Transmitters operating at higher power must be approved by NASA during the design process.	Design	WPI HPRC will utilize a LORA transceiver for streaming telemetry. The camera will utilize a high-power LTE transmitter to transmit the photos.	In Progress
NASA-4.3.4.4	The image should be included in your PLAR.	Inspection	WPI HPRC will ensure all images captured by the camera system are included in the PLAR report.	In Progress
NASA-4.4	General Payload Requirements			
NASA-4.4.1	Black Powder and/or similar energetics are only permitted for deployment of in-flight recovery systems. Energetics will not be permitted for any surface operations.	Design	WPI HPRC will ensure all black powder charges used in the deployment will only be fired during decent while in the air	In Progress
NASA-4.4.2	Teams must abide by all FAA and NAR rules and regulations.	Design	WPI HPRC will ensure all designs abide by FAA and NAR rules and Regulations	In Progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-4.4.3	Any experiment element that is jettisoned, except for planetary lander experiments, during the recovery phase will receive real-time RSO permission prior to initiating the jettison event.	Analysis	WPI HPRC will not be flying additional payloads	Verified
NASA-4.4.4	Unmanned aircraft system (UAS) payloads, if designed to be deployed during descent, will be tethered to the vehicle with a remotely controlled release mechanism until the RSO has given permission to release the UAS.	Analysis	WPI HPRC will not be creating a UAS	Verified
NASA-4.4.5	Teams flying UASs will abide by all applicable FAA regulations, including the FAA's Special Rule for Model Aircraft (Public Law 112-95 Section 336; see https://www.faa.gov/uas/faqs).	Analysis	WPI HPRC will not be creating a UAS	Verified
NASA-4.4.6	Any UAS weighing more than .55 lbs. will be registered with the FAA and the registration number marked on the vehicle.	Analysis	WPI HPRC will not be creating a UAS	Verified
Safety Requirements				
NASA-5.1	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.	Inspection	WPI HPRC will verify this requirement by including the final launch and safety checklists in the FRR report.	Not Verified

NASA Requirements				
Requirement No.	Description	Verification Method	Verification Plan	Status
NASA-5.2	Each team must identify a student safety officer who will be responsible for all items in section 5.3.	Inspection	WPI HPRC has designated Michael Beskid to be the student safety officer responsible for all items in section 5.3.	Verified
NASA-5.3.1	The role and responsibilities of the safety officer will include, but are not limited to: Monitor team activities with an emphasis on safety during: 5.3.1.1. Design of vehicle and payload 5.3.1.2. Construction of vehicle and payload components 5.3.1.3. Assembly of vehicle and payload 5.3.1.4. Ground testing of vehicle and payload 5.3.1.5. Subscale launch test(s) 5.3.1.6. Full-scale launch test(s) 5.3.1.7. Launch Day 5.3.1.8. Recovery activities 5.3.1.9. STEM Engagement Activities	Inspection	WPI HPRC will maintain a safe environment during all design, construction, assembly, and testing activities at the direction of the team safety officer Michael Beskid. The safety officer will further be responsible for overseeing safety at all launch and recovery activities, in addition to STEM engagement activities and other events.	In Progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-5.3.2	Implement procedures developed by the team for construction, assembly, launch, and recovery activities.	Inspection	WPI HPRC will develop checklists outlining safety procedures for construction, assembly, launch, and recovery activities.	Not Verified
NASA-5.3.3	Manage and maintain current revisions of the team's hazard analyses, failure modes analyses, procedures, and MSDS/chemical inventory data.	Inspection	WPI HPRC will maintain current revisions of the team's hazard analyses, failure modes analyses, safety procedures, and MSDS/chemical inventory data at the direction of the safety officer, and include current revisions in PDR, CDR, and FRR reports.	In Progress
NASA-5.3.4	Assist in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	Inspection	WPI HPRC will complete and submit required safety documentation including hazard analyses, failure mode analyses, and safety procedures at the direction of the safety officer.	In Progress

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-5.4	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.	Inspection	WPI HPRC will clearly communicate its intentions to the local club President and RSO before attending NAR or TRA sanctioned launch events. The team agrees to abide by all rules put into effect by the local rocketry club and will readily follow all guidance provided by the RSO on site. These items will be verified in a pre-launch checklist before all flights.	Not Verified
NASA-5.5	Teams will abide by all rules set forth by the FAA.	Inspection	WPI HPRC will carefully inspect the rocket and payload ahead of all flights, using a checklist to ensure compliance with all FAA regulations.	Not Verified
Final Flight Requirements				
NASA-6.1	NASA Launch Complex			

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-6.1.1	Teams must complete and pass the Launch Readiness Review conducted during Launch Week.	Inspection	WPI HPRC will not be attending NASA Launch Week in person.	Not Verified
NASA-6.1.2	The team mentor must be present and oversee rocket preparation and launch activities.	Inspection	WPI HPRC will not be attending NASA Launch Week in person.	Not Verified
NASA-6.1.3	The scoring altimeter must be presented to the NASA scoring official upon recovery.	Inspection	WPI HPRC will not be attending NASA Launch Week in person.	Not Verified
NASA-6.1.4	Teams may launch only once. Any launch attempt resulting in the rocket exiting the launch pad, regardless of the success of the flight, will be considered a launch. Additional flights beyond the initial launch, will not be scored and will not be considered for awards.	Inspection	WPI HPRC will not be attending NASA Launch Week in person.	Not Verified
NASA-6.2	Commercial Spaceport Launch Site			

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-6.2.1	The launch must occur at a NAR or TRA sanctioned and insured club launch. Exceptions may be approved for launch clubs who are not affiliated with NAR or TRA but provide their own insurance, such as the Friends of Amateur Rocketry. Approval for such exceptions must be granted by NASA prior to the launch.	Inspection	WPI HPRC will be competing remotely and will use a Commercial Spaceport Launch Site for launches, including the final flight. WPI HPRC will schedule the final flight at a NAR or TRA sanctioned launch or seek approval from NASA if a different launch site is required for team purposes.	In Progress
NASA-6.2.2	Teams must submit their rocket and payload to the launch site Range Safety Officer (RSO) prior to flying the rocket. The RSO will inspect the rocket and payload for flightworthiness and determine if the project is approved for flight. The local RSO will have final authority on whether the team's rocket and payload may be flown.	Inspection	WPI HPRC will submit the launch vehicle and payload to the RSO. The team will abide by the RSO's decision on approval for flight.	Not Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-6.2.3	The team mentor must be present and oversee rocket preparation and launch activities.	Inspection	WPI HPRC will choose a launch date where the team mentor, Jason Nadeau, can be in attendance for launch day preparation and activities.	Not Verified
NASA-6.2.4	BOTH the team mentor and the Launch Control Officer shall observe the flight and report any off-nominal events during ascent or recovery on the Launch Certification and Observations Report.	Inspection	WPI HPRC will provide the Launch Control Officer and the team mentor, Jason Nadeau, with the Launch Certification and Observations Report to record any off-nominal events. This completed documentation will be submitted to NASA.	Not Verified
NASA-6.2.5	The scoring altimeter must be presented to BOTH the team's mentor and the Range Safety Officer.	Inspection	WPI HPRC will present the scoring altimeter to both the RSO and team mentor, Jason Nadeau.	Not Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
NASA-6.2.6	The mentor, the Range Safety Officer, and the Launch Control Officer must ALL complete the applicable sections of the Launch Certification and Observations Report. The Launch Certification and Observations Report document will be provided by NASA upon completion of the FRR milestone and must be returned to NASA by the team mentor upon completion of the launch.	Inspection	WPI HPRC will provide the Launch Control Officer, RSO and the team mentor, Jason Nadeau, with the Launch Certification and Observations Report to complete and required sections. The team mentor, Jason Nadeau, will submit this completed documentation to NASA.	Not Verified
NASA-6.2.7	The Range Safety Officer and Launch Control Officer certifying the team's flight shall be impartial observers and must not be affiliated with the team, individual team members, or the team's academic institution.	Inspection	WPI HPRC will choose a launch location with no affiliation to WPI itself, individual team members or the team itself.	Not Verified
NASA-6.2.8	Teams may launch only once. Any launch attempt resulting in the rocket exiting the launch pad, regardless of the success of the flight, will be considered a launch. Additional flights beyond the initial launch will not be scored and will not be considered for awards.	Inspection	WPI HPRC will only launch the full scale launch vehicle with payload once and recognizes this is the launch that will be scored.	Not Verified

9.2 Derived Requirements

Derived Requirements					
Requirement No.	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
Vehicle Requirements					
WPI-1.1	The vehicle shall consist of a 6 in diameter airframe	A smaller airframe would restrict the room for payload, the airbrake system, and the fin can beyond acceptable limits. A larger airframe would bring additional cost in the form of airframe materials and motors, and labor due to larger internal components	Inspection	WPI HPRC will design the vehicle to use airframes within the 6 in range, depending on the material used	Verified
WPI-1.2	The airframe material shall be resistant to warpage from humidity and temperature changes	In previous project years, the airframe changing shape caused significant issues with assembly	Inspection	WPI HPRC will design the vehicle to use airframes with materials shown not to warp	Verified
WPI-1.3	The airframe material shall be resistant to zippering and shearing from bolts and other attachment hardware placed through it	The team uses bolts for attaching components over adhesives such as epoxy to increase the modularity of the launch vehicle. Materials prone to zippering would not safely retain internal components	Inspection	WPI HPRC will use materials that do not shear or tear easily when concentrated loads are applied	Verified

Derived Requirements					
Requirement No.	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
WPI-1.4	The airframe and coupler tubes shall be dimensionally compatible	The airframe and coupler must slide smoothly together, so must have compatible outer and inner diameters	Inspection	WPI HPRC will ensure the airframe and coupler tubes are compatible	Verified
WPI-1.4.1	The airframe and coupler tubes shall be made from the same material	There will be fewer issues with thermal expansion and binding if the materials are the same	Inspection	WPI HPRC will ensure the airframe and coupler tubes are made from the same material	Verified
WPI-1.5	Structural components of the vehicle shall have a safety factor of at least 2 times the maximum expected load	An additional safety factor is essential to ensure safety and prevent damage to the vehicle in the event unexpected flight forces are encountered	Analysis	WPI HPRC will simulate components analytically or numerically, and compared against expected flight loads	In progress
WPI-1.6	The vehicle shall use a 75 mm CTI motor reload	The team already possesses motor hardware for a CTI 75mm motor, and the purchase of an additional motor hardware set would place an undue financial burden on the team.	Inspection	WPI HPRC will limit its acceptable motors to CTI 75mm motors	Verified

Derived Requirements					
Requirement No.	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
WPI-1.7	Fins shall be made removable and replaceable	Fin damage is the most likely damage to the launch vehicle during landing. Permanently attached fins would present a significant challenge to replace if damaged	Inspection	WPI HPRC will ensure fins can be replaced easily on the launch vehicle	In progress
WPI-1.8	The avionics system will both store onboard and transmit all collected data to the ground	Access to flight data is essential for post-flight analysis and determining the processes behind a successful or unsuccessful launch	Demonstration	WPI HPRC will demonstrate data storage and transmission capabilities during test launches	Not verified
Recovery Requirements					
WPI-2.1	The ejection charges shall produce a pressure at least 1.5 times that necessary to break the shear pins. The backup charge shall produce a pressure twice the necessary pressure	The ejection charges must break the shear pins with enough force to continue to separate the vehicle and allow the parachutes and payload to exit the vehicle. The	Analysis	WPI HPRC will calculate the expected pressure generated by each ejection charge, and compare to the calculated force needed to break the shear pins	In progress

Derived Requirements					
Requirement No.	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
WPI-2.1.1	If ground testing realizes the need for additional black powder for a safe ejection, the backup charge shall be increased by a proportional amount	The backup ejection charge must be larger than the primary ejection charge to provide safe redundancy in the event the primary charge is not powerful enough to separate the vehicle	Inspection	WPI HPRC will increase the size of ejection charges proportionally	Not verified
WPI-2.2	Payload deployment shall be made independent from deployment of the main parachute	Due to the possibility of complications from releasing the payload, the main parachute could be prevented from opening, which would cause significant damage to the launch vehicle.	Inspection, Demonstration	WPI HPRC will show the main and payload separation events to be independent in design and during test flights	In progress
WPI-2.3	Recovery hardware attachment points shall consist of a U-Bolt	U-Bolts provide two attachment points, increasing strength, and preventing the possibility of rotational forces disconnecting a device such as an eyebolt	Inspection	WPI HPRC will ensure all shock cord attachment points consist of a U-Bolt	Verified
Payload Requirements					

Derived Requirements					
Requirement No.	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
WPI-3.1	The Payload shall fit comfortably within the 6in airframe	Fitting within the airframe comfortable will allow for ease of installation into the rocket and will prevent damage from vibration	Design + Inspection	WPI HPRC will design the payload with tolerance to fit within the 6in ID airframe and upon completion run fitting tests with airframe pieces	In Progress
WPI-3.2	The Payload shall be designed in a modular way	Reducing the amount of people required for final assembly will allow for assembly to happen in rapid fashion allowing us more time for testing	Design	WPI HPRC will design the subsystems of the payload to be assembled individually then assembled	In Progress
WPI-3.3	The Payload shall be at most 5lbs	Keeping the weight to a minimum will allow for better rocket performance	Design + Inspection	WPI HPRC will keep constant checks on the mass of the payload and will weight all parts after manufacturing to ensure expected weights are achieved	In Progress

9.3 Budget

HPRC's treasurer, Kevin Schultz, is responsible for keeping a detailed budget and handling purchases for WPI HPRC. Due to WPI's, ongoing ban of student travel, the team is not planning to attend the NASA Launch Week activities in person this competition year. In the beginning of the year, as an officer board, the team has transferred half of the given logistics budget, $2466.90, towards our component budget. The remaining $2366.90 of the logistics budget is reserved for launches as school affiliated travel is approved on a case-by-case basis. As of the time of submission, two students are permitted to attend and perform the subscale launch. Their gas expenses will be paid for through the remaining logistics funds. This logistic budget will also fund future WPI approved launches.

It is important to note the overall budget is somewhat stagnant. Due to the funding received from WPI TinkerBox, the only item WPI HPRC has had to pay for from our account has been the last item seen in Table 9.2 - Items Purchased at Time of PDR. Other methods of funding, including TinkerBox, are discussed further in Section 9.4.

Base Anticipated Budget		
Expense	Amount	Notes
Aerostructures	$669.95	Airframe, couplers, and nosecone
Avionics	$400.00	Electronics
Airbrakes	$172.84	Materials and COTS parts
Propulsion	$559.97	Motor casing and retention components
Recovery	$310.00	Drogue and main parachutes
Motors	$878.97	Primary and backup
Payload	$1,000.00	All components for the payload
Subscale Rocket	$600.00	All components for subscale rocket
General Hardware	$33.54	General nuts, bolts, screws, etc.
Tools	$750.00	3D printer, Dremmel kit, soldering iron, iFixit Toolkit, flap sander and sanding pads
Total Expenses	$5,535.27	
Extra Costs	$1,0000.00	Overspending expectation
Total Anticipated Expenses	**$6,535.27**	

Table 9.1 - Base Anticipated Budget

Items Purchased at Time of PDR Submission					
Item - General Description	Item – Specific Description	Vendor Name	Base Unit Price (USD)	Quantity	Total (Including Tax and Shipping)
Raspberry Pi 3 B+	Pi with Power Block and Heatsink	PiShop.us	$46.4	1	$46.4

Raspberry Pi Zero W	Aluminum Heatsink for Raspberry Pi Zero (K2B-1306)	PiShop.us	$16.2	1	$16.2
Lipo Bag	Zeee Lipo Safe Bag Fireproof Explosionproof Bag	Amazon	$12.99	1	$12.99
Small Lipo-Bag	Teenitor Fireproof Explosionproof Lipo Battery Safe Bag	Amazon	$7.99	2	$15.98
360 Degree Camera	PICAM360-CAMPT8MP (CAMPT8MP)	Picam360	$95	1	$95.00
Lipo Battery Charger	SKYRC B6 AC V2 50W LiPo LiFe LiIon NiMH NiCd Battery Charger Discharger (B01MZ1ZZ7Z)	Amazon	$48.49	1	$48.49
Transceiver	Ebyte E32-915T30D LoRa Transceiver SX1276 915MHz 1W SMD Wireless Module	Ebyte	$11.5	4	$46.00
Safety Glasses	Standard safety glasses (SKU: SFTEYSG1000021190)	Discount Safety Gear	$0.89	10	$8.90
Safety Glasses	Safety glasses that go over normal glasses (SKU: UAT9800)	Discount Safety Gear	$1.30	5	$6.50
Face Shields	Safety Face Shield, Transparent Reusable Glasses, 2 Pack Full Face Protective Visor with Eye & Mouth Protection	Walmart	$7.99	2	$15.98
Transceiver	Ebyte E32-915T20D LoRa Transceiver SX1276 915MHz 100mW Wireless Module	PiShop.us	$21.99	1	$21.99
GSM Raspberry Pi Shield	GSM/GPRS/GNSS/Bluetooth HAT for Raspberry Pi	PiShop.us	$33.99	1	$33.99
SIM card	GSM SIM Card from Ting & Adafruit	Adafruit/Ting	9.00	1	$9.00
Magnetometer	MLX90393	Sparkfun	$14.95	1	$14.95
GPS	NEO-M9N, U.FL	Sparkfun	$64.95	1	$64.95
GPS Antenna	GNSS Antenna (10mm)	Sparkfun	$2.95	1	$2.95
Microcontroller	Teensy 3.2	Sparkfun	$19.80	3	$59.4
Nitrile Gloves	Nitrile Exam Gloves - 50ct - Up&Up™	Target	$7.99	2	$15.98
Photos for Sponsors	Photos to give to Sponsors to say thank you.	Walmart	$13.86	1	$13.86
TOTAL SPENT	**$549.51**				

Table 9.2 - Items Purchased at Time of PDR

Overall Current Budget	
Components	
Budget Given by AIAA	+ $2,654.85
Funds Moved from Logistics Budget	+ $2,366.90
Expenses Thus Far	-$549.51
TinkerBox Funding (up to $3,000)	+ $535.65
Sponsorship	+ $1,000
Total in Component Budget as of PDR	**$6,007.89**
Logistics	
Budget Given by AIAA	+ $4,733.80
Funds Taken from Logistics Budget for Components	-$2,366.90
Total in Logistic Budget as of PDR	**$2,366.90**
Total in Account	**$8,374.79**

Table 9.3 Overall Current Budget

9.4 Funding

A significant portion of this year's funding for 2020-2021 WPI HPRC will come from WPI TinkerBox Cohort 4. TinkerBox is a program hosted by WPI's Innovation and Entrepreneurship department that provides seed funding for WPI student-initiated innovation and entrepreneurship ideas. WPI HPRC has been granted $3,000 of funding which can be used until the end of the calendar year. All purchases (up to $3,000) pertaining to components will be reimbursed through TinkerBox. There will be another opportunity to reapply for this grant during the second semester of this academic year.

In addition to TinkerBox, WPI HPRC will also be receiving funds from the WPI AIAA chapter on campus. The AIAA receives their annual budget from the Student Government Association (SGA) on campus which is responsible for governing undergraduate organizations on campus. This year, HPRC will be the only competitive rocketry team in AIAA so all the funding for high powered rocketry competitions will be going to HPRC. The amount that the AIAA has allocated to HPRC is reflected in the budget above. Any additional funds will have a request submitted to the Student Government Association.

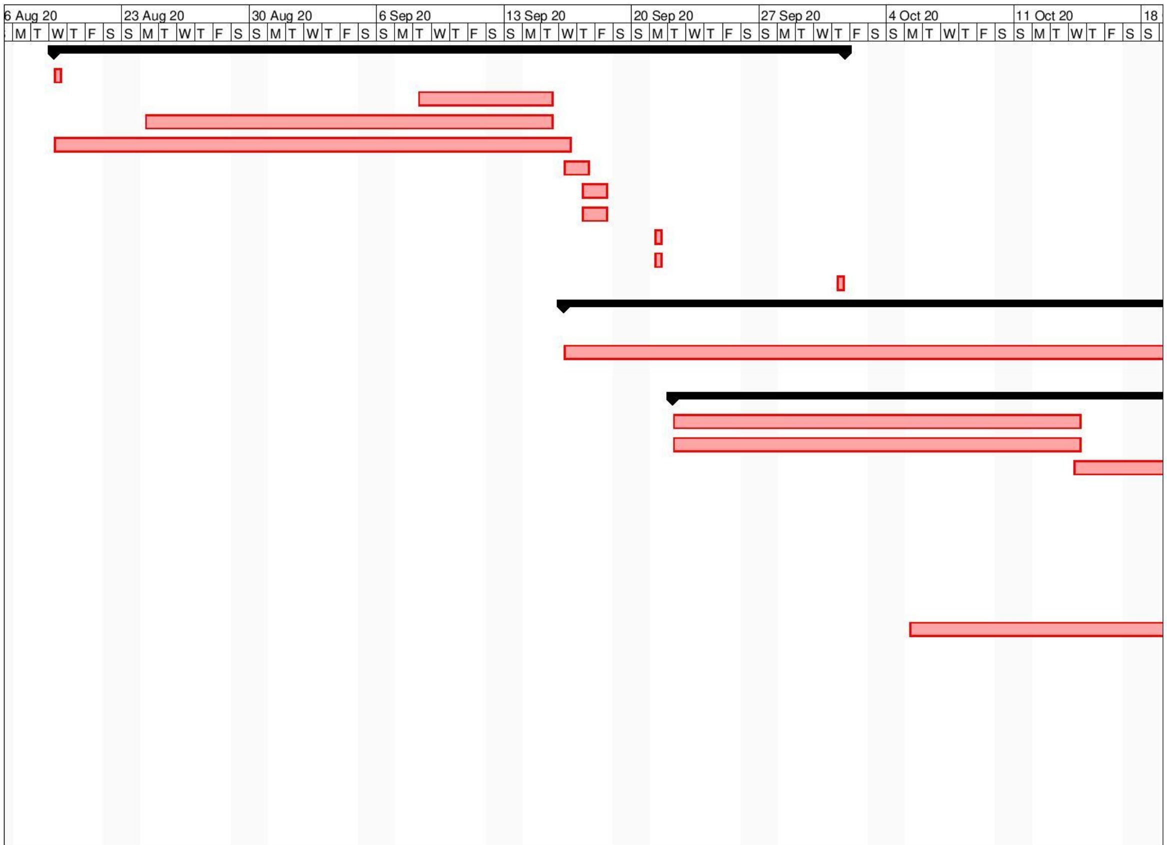
Another way the team raises funds is through corporate sponsorship. The Sponsorship Officer, Julia Sheats, is responsible for gathering funds from corporate sponsors and communicating with the Financial Services Department of WPI to ensure all proper transfer of funds is being done so appropriately. The Sponsorship Officer created a sponsorship package to present to companies primarily located in the local Worcester area, and will continuously reach out to companies in the area throughout the year. The corporate sponsorship package is approved by the Division of University Advancement on WPI's campus before it is presented to our potential sponsors. Each sponsor interested in funding the team will be provided with the selection of several packages. In order of increasing sponsor funding value, these sponsorship levels are Bronze, Silver, Gold and Platinum.

One of the team's primary goals in funding this year is to create strong and lasting relationships with these sponsors so they will be interested in working with us again in the following competition years. The team is creating "thank you" packages for past sponsors that include photos and thank you notes signed by the team. Currently the team has confirmed two returning sponsors from last year that will be continuing their support for our team into the coming year. Thus far, the team has acquired $1,000 from these returning sponsors. If the team has any extra funding from corporate sponsors at the end of the competition season, the money will roll over to be used in the next competition year in 2021-2022.

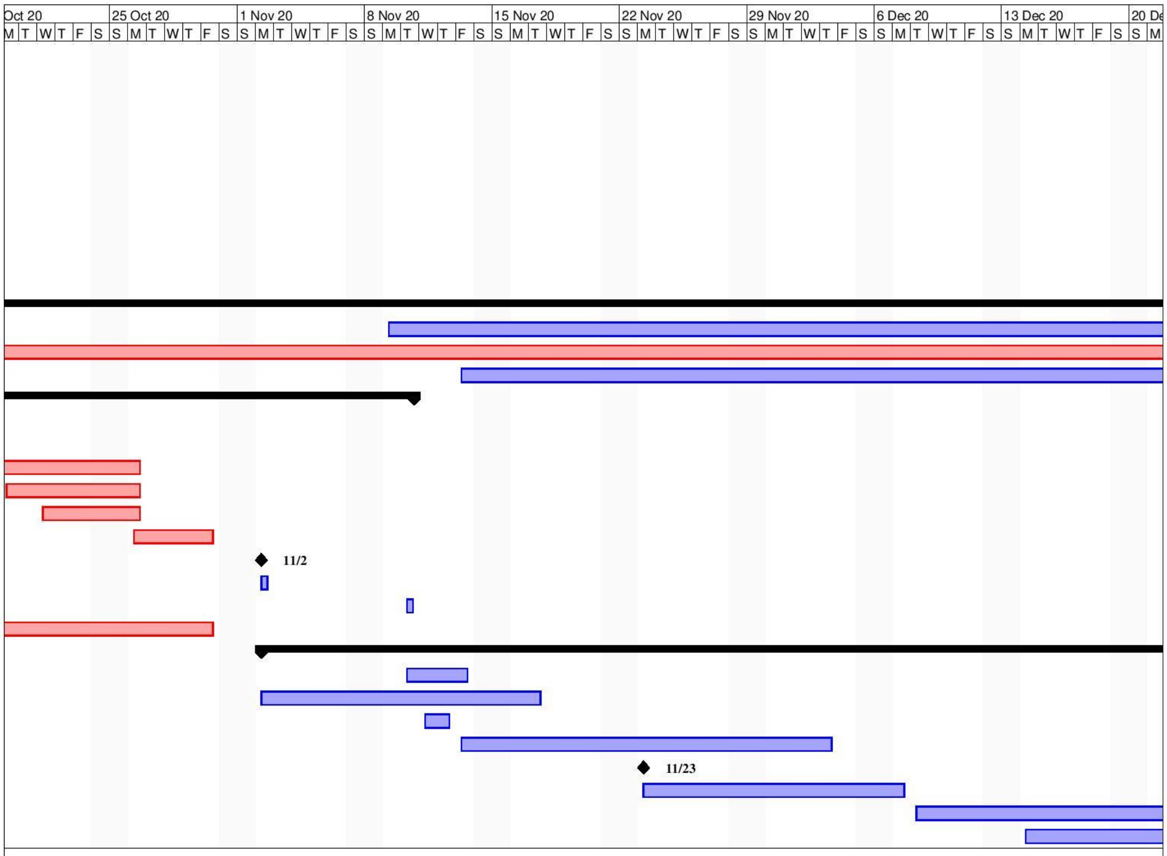
9.5 Gantt Chart

		Name	Duration	Start	Finish	Predecessors	Resource Names	1 S
1		**Proposal**	**32 days?**	**8/19/20 7:00 AM**	**10/1/20 5:00 PM**			
2		Proposal Announcement	1 day?	8/19/20 7:00 AM	8/19/20 5:00 PM			
3		Orientation	6 days?	9/8/20 7:00 AM	9/15/20 5:00 PM			
4		Recruiting	17 days?	8/23/20 7:00 AM	9/15/20 5:00 PM			
5		Brainstorming	21 days?	8/19/20 7:00 AM	9/16/20 5:00 PM			
6		Presenting Ideas	2 days?	9/16/20 7:00 AM	9/17/20 5:00 PM			
7		Finalizing Ideas	2 days?	9/17/20 7:00 AM	9/18/20 5:00 PM			
8		Proposal Writing	2 days?	9/17/20 7:00 AM	9/18/20 5:00 PM			
9		Proposal Revision/Collec...	1 day?	9/20/20 7:00 AM	9/21/20 5:00 PM			
10		Submission	1 day?	9/21/20 7:00 AM	9/21/20 5:00 PM			
11		Awarded Proposal	1 day?	10/1/20 7:00 AM	10/1/20 5:00 PM			
12		**General**	**168 days?**	**9/16/20 7:00 AM**	**5/7/21 5:00 PM**			
13		Outreach Events	86 days?	11/7/20 8:00 AM	3/8/21 5:00 PM			
14		Workshops	168 days?	9/16/20 7:00 AM	5/7/21 5:00 PM			
15		Payload Construction	58 days?	11/13/20 8:00 AM	2/2/21 5:00 PM			
16		**PDR**	**36 days?**	**9/22/20 7:00 AM**	**11/10/20 5:00 PM**			
17		Rocket Design	17 days?	9/22/20 7:00 AM	10/14/20 5:00 PM			
18		Payload Design	17 days?	9/22/20 7:00 AM	10/14/20 5:00 PM			
19		Internal Design Presenta...	9 days?	10/14/20 7:00 AM	10/26/20 5:00 PM			
20		Design Revisions	6 days?	10/17/20 7:00 AM	10/26/20 5:00 PM			
21		Safety analysis	4 days?	10/21/20 7:00 AM	10/26/20 5:00 PM			
22		PDR Writing	5 days?	10/26/20 7:00 AM	10/30/20 5:00 PM			
23		PDR Revision/Collection	0 days?	11/1/20 8:00 AM	11/2/20 5:00 PM			
24		PDR Submission	1 day?	11/2/20 8:00 AM	11/2/20 5:00 PM			
25		PDR Presentation	1 day?	11/10/20 8:00 AM	11/10/20 5:00 PM			
26		Subscale Design	20 days?	10/5/20 7:00 AM	10/30/20 5:00 PM			
27		**CDR**	**62 days?**	**11/2/20 8:00 AM**	**1/26/21 5:00 PM**			
28		Revisions Suggested By ...	4 days?	11/10/20 8:00 AM	11/13/20 5:00 PM			
29		Testing Components	12 days?	11/2/20 8:00 AM	11/17/20 5:00 PM			
30		Internal Design Presenta...	2 days?	11/11/20 8:00 AM	11/12/20 5:00 PM			
31		Subscale Construction (s...	15 days?	11/13/20 8:00 AM	12/3/20 5:00 PM			
32		Subscale Flight	0 days?	11/21/20 8:00 AM	11/23/20 5:00 PM			
33		Flight Analysis	11 days?	11/23/20 8:00 AM	12/7/20 5:00 PM			
34		CDR Writing	24 days?	12/8/20 8:00 AM	1/8/21 5:00 PM			
35		Winter Break	22 days?	12/14/20 8:00 AM	1/12/21 5:00 PM			

WPI HPRC Ghant Chart Updated - page1

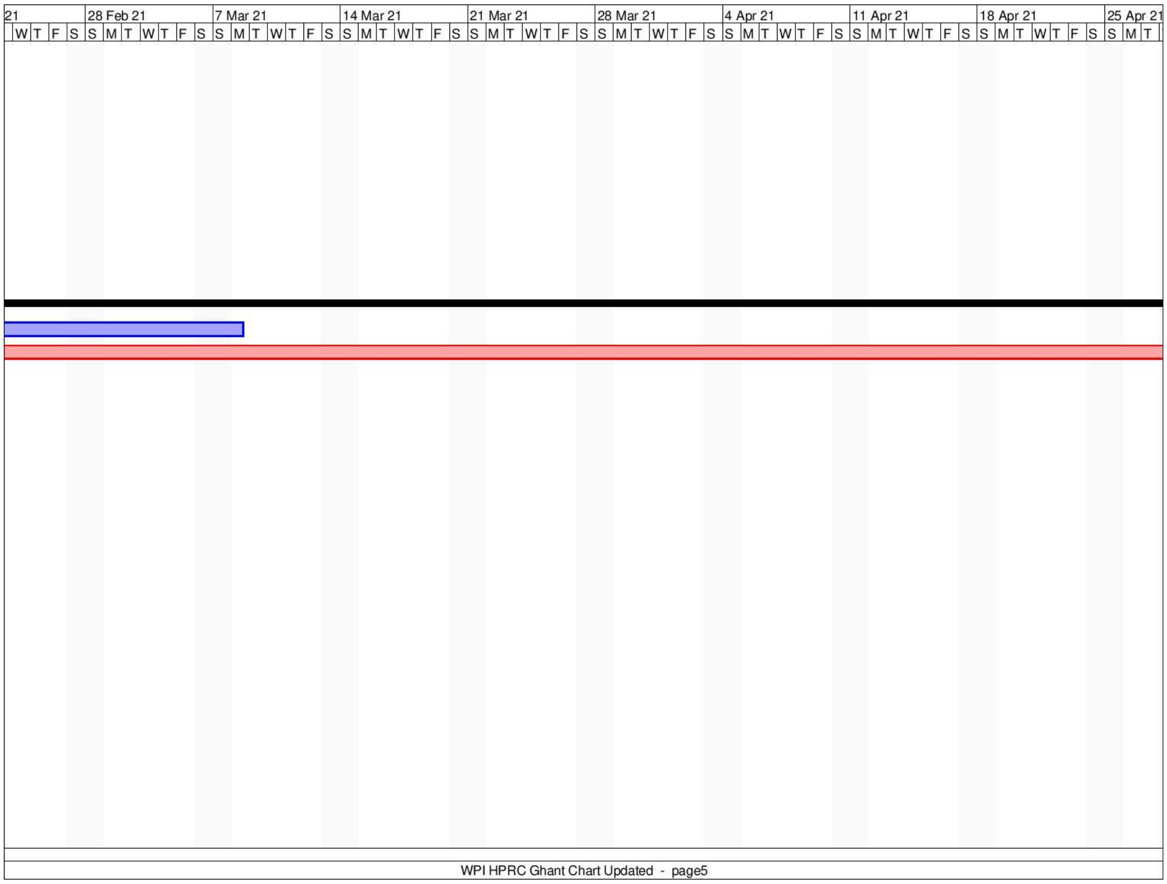


WPI HPRC Ghant Chart Updated - page2



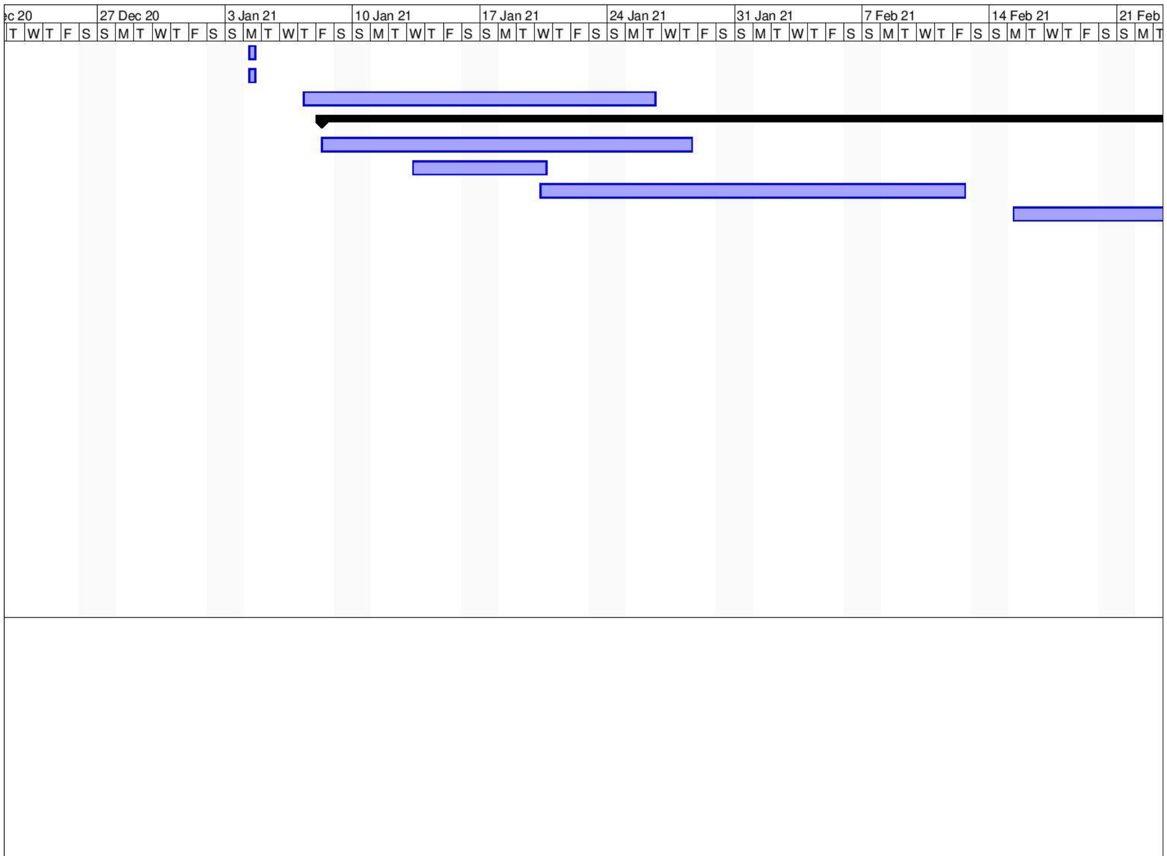
WPI HPRC Ghant Chart Updated - page3

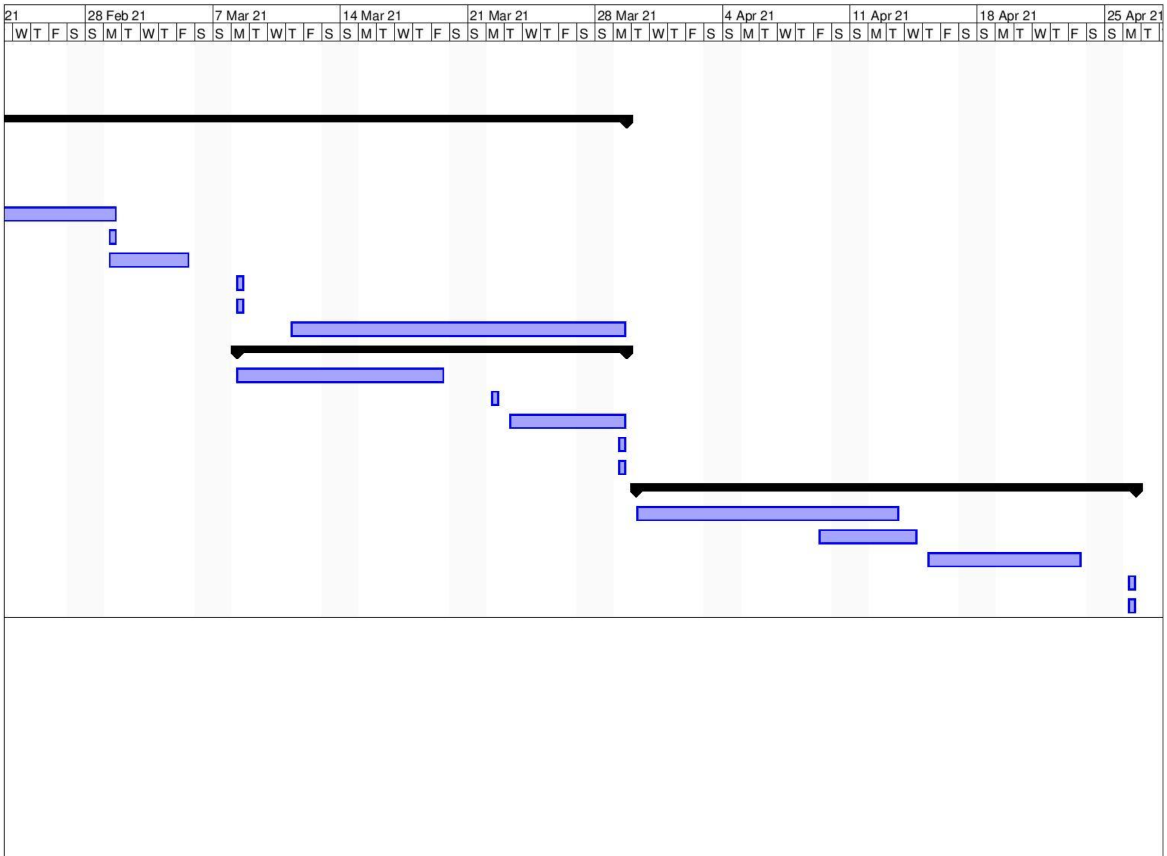
WPI HPRC Ghant Chart Updated - page4


21
28 Feb 21
7 Mar 21
14 Mar 21
21 Mar 21
28 Mar 21
4 Apr 21
11 Apr 21
18 Apr 21
25 Apr 21
W T F S S M T W T F S S M T W T F S S M T W T F S S M T W T F S S M T W T F S S M T W T F S S M T W T F S S M T W T F S S M T
WPI HPRC Ghant Chart Updated - page5


2 May 21
9 May 21
16 May 21
23 May 21
30 May 21
6 Jun 21
13 Jun 21
20 Jun 21
27 Jun 21
W T F S S M T W T F S S M T W T F S S M T W T F S S M T W T F S S M T W T F S S M T W T F S S M T W T F S S M T W T F S S M T W
WPI HPRC Ghant Chart Updated - page6

		Name	Duration	Start	Finish	Predecessors	Resource Names
36		CDR Revision/Collection	1 day?	1/2/21 8:00 AM	1/4/21 5:00 PM		
37		CDR Submission	1 day?	1/4/21 8:00 AM	1/4/21 5:00 PM		
38		CDR Presentation	14 days?	1/7/21 8:00 AM	1/26/21 5:00 PM		
39		**FRR**	**57 days?**	**1/8/21 8:00 AM**	**3/29/21 5:00 PM**		
40		Revisions Suggested By ...	15 days?	1/8/21 8:00 AM	1/28/21 5:00 PM		
41		Testing Components	6 days?	1/13/21 8:00 AM	1/20/21 5:00 PM		
42		Full scale Construction (s...	18 days?	1/20/21 8:00 AM	2/12/21 5:00 PM		
43		Full Scale Flight	11 days?	2/13/21 8:00 AM	3/1/21 5:00 PM		
44		Flight Analysis	1 day?	3/1/21 8:00 AM	3/1/21 5:00 PM		
45		FRR Writing	5 days?	3/1/21 8:00 AM	3/5/21 5:00 PM		
46		FRR Revision/Collection	1 day?	3/7/21 8:00 AM	3/8/21 5:00 PM		
47		FRR Submission	1 day?	3/8/21 8:00 AM	3/8/21 5:00 PM		
48		FRR Presentation	13 days?	3/11/21 8:00 AM	3/29/21 5:00 PM		
49		**FRR Addendum**	**16 days?**	**3/8/21 8:00 AM**	**3/29/21 5:00 PM**		
50		Full Scale Flight	10 days?	3/8/21 8:00 AM	3/19/21 5:00 PM		
51		Flight Analysis	1 day?	3/20/21 7:00 AM	3/22/21 5:00 PM		
52		FRR Addendum Writing	5 days?	3/23/21 7:00 AM	3/29/21 5:00 PM		
53		FRR Addendum Revision...	1 day?	3/28/21 7:00 AM	3/29/21 5:00 PM		
54		FRR Addendum Submission	1 day?	3/29/21 7:00 AM	3/29/21 5:00 PM		
55		**PLAR**	**20 days?**	**3/30/21 7:00 AM**	**4/26/21 5:00 PM**		
56		Full Scalecompetition Flight	11 days?	3/30/21 7:00 AM	4/13/21 5:00 PM		
57		Flight Analysis	4 days?	4/9/21 7:00 AM	4/14/21 5:00 PM		
58		PLAR Addendum Writing	7 days?	4/15/21 7:00 AM	4/23/21 5:00 PM		
59		PLAR Addendum Revisio...	1 day?	4/24/21 7:00 AM	4/26/21 5:00 PM		
60		PLAR Addendum Submis...	1 day?	4/26/21 7:00 AM	4/26/21 5:00 PM		


c 20
27 Dec 20
3 Jan 21
10 Jan 21
17 Jan 21
24 Jan 21
31 Jan 21
7 Feb 21
14 Feb 21
21 Feb



WPI HPRC Ghant Chart Updated - page12

10 Appendix

10.1 Descent Simulator


```
function [t,x,h] = descentsim(conds,stages,params,sim,solver)
%descentsim Simulator for the descent of the vehicle
%    Inputs:
% conds = initial conditions of form [px;py;pz;vx;vy;vz]
% stages = flight stage events of form
%    [state_var_index_1,des_value_1;state_var_index_2,des_value_2;...]
% params = physical parameters during each stage of form
%    [mass_1,chute_cd_1,chute_area_1;mass_2,chute_cd_2,chute_area_2;...]
% sim = key simulation parameters of form [initial_time;step_value]
% solver = selected ODE solver (either rk4 or rkf45)
%    Outputs:
% t = time vector
% x = states matrix

preal = 10000; % preallocated matrix length for t and x

h = sim(2);
h_rec = zeros(1,preal); % preallocate simulation step vector
h_rec(1) = sim(2); % set first index of step vector to inital step

t = zeros(1,preal); % preallocate time vector
t(1) = sim(1); % set first index of time vector to initial time

x = zeros(6,preal); % preallocate state matrix
xCurr = conds; % set xCurr to the initial conditions
x(:,1) = xCurr; % set first index of state matrix to initial conditions

ii = 1;
stage = 1; % set first flight stage as active stage
while true

    ii = ii+1; % increment index by 1
    t(ii) = t(ii-1)+h; % increment time by dt

    env = environment(xCurr); % calculate environmental variables

    [xCurr,h] = solver(@chute_xDot,xCurr,params(stage,:),env,h,t(ii)); % Run rk4 solver ↙
step

    x(:,ii) = xCurr; % pass xCurr to state matrix
    h_rec(ii) = h;

    if xCurr(stages(stage,1)) <= stages(stage,2) % check if necessary to increment stage if stage
        == size(stages,1) % verify not on final stage
            break
        else
```

```
            stage = stage + 1; % increment stage by 1
        end
    end

    if ii >= length(t) % Check if reached the end of preallocated matrix size t =
        [t,zeros(1,preal)];
```



```
        x = [x,zeros(6,preal)];
        h_rec = [h_rec,zeros(1,preal)];
    end

    if xCurr(3) >= conds(3)*5 % check if altitude has far exceeded initial altitude warning('Vehicle
        altitude has far exceeded initial altitude. Verify wind profile ↙
and inputs.')
break
    end

    if max(t) >= 1200 % check if simulated descent time has exceeded 20 min
        warning('Simulation time has exceeded 20 min. Verify inputs') break

    end

end
  t = t(1:ii);
  x = x(:,1:ii);
  h = h_rec(1:ii);
```



```
function env     = environment(x)
%environment    Environmental parameters calculator
%     Inputs:
% x = state variables of form [px;py;pz;vx;vy;vz]
%     Outputs:
% env = environmental parameters of form [g,rho,wind_x,wind_y,wind_z]

g = gravitywgs84(x(3),41.553223);
[~, ~, ~, rho] = atmosisa(x(3));
w_x = 0;
w_y = 0;
```

```
w_z = 0;

% g = 9.81;
% rho = 1.08;

env = [g,rho,w_x,w_y,w_z];

end
```



```
function [x,h] = rk4(func,x,params,env,h,~)
%rk4 4th Order Runge-Kutta solver
%     Detailed explanation goes here

k1 = h * func(x,params,env);
k2 = h * func(x + (k1 / 2),params,env);
k3 = h * func(x + (k2 / 2),params,env);
k4 = h * func(x + k3,params,env);

x = x + ((1/6) * (k1 + 2*k2 + 2*k3 + k4));

end
```



```
function xD = chute_xDot(x,params,env)
%chute_xDot State functions for descent
%     Inputs:
% x = state variables of form [px;py;pz;vx;vy;vz]
% params = vehicle parameters of form [mass,chute_cd,chute_area]
% env = environmental parameters of form [g,rho,wind_x,wind_y,wind_z]
% Outputs:
% xD = state derivatives

dir = [((-x(4)+env(3))/abs((-x(4)+env(3))));
       ((-x(5)+env(4))/abs((-x(5)+env(4))));
       ((-x(6)+env(5))/abs((-x(6)+env(5))))];

dir_check = isnan(dir);
dir(dir_check) = 1;

xD = [x(4);
            x(5);
            x(6);
            dir(1)*(0.5*params(2)*params(3)*env(2)*(x(4)-env(3))^2)/(params(1));
            dir(2)*(0.5*params(2)*params(3)*env(2)*(x(5)-env(4))^2)/(params(1));
            (dir(3)*(0.5*params(2)*params(3)*env(2)*(x(6)-env(5))^2)/(params(1)))-env(1)];
end
```

```
function descentplot(t,x)
%descentplot Summary of this function goes here
%     Inputs:
% t = time vector
% x = states matrix

figure()
title('Positions and Velocities')
subplot(3,2,1)
plot(t,x(1,:));
title('Vehicle x Position vs Time');
xlabel('t (s)');
ylabel('p_{x} (ft)');
set(gca,'XGrid','off','YGrid','on')
subplot(3,2,3)
plot(t,x(2,:));
title('Vehicle y Position vs Time');
xlabel('t (s)');
ylabel('p_{y} (ft)');
set(gca,'XGrid','off','YGrid','on')
subplot(3,2,5)
plot(t,x(3,:));
title('Vehicle z Position vs Time');
xlabel('t (s)');
ylabel('p_{z} (ft)');
set(gca,'XGrid','off','YGrid','on')
subplot(3,2,2)
plot(t,x(4,:));
title('Vehicle x Velocity vs Time');
xlabel('t (s)');
ylabel('v_{x} (ft/s)');
set(gca,'XGrid','off','YGrid','on')
subplot(3,2,4)
plot(t,x(5,:));
title('Vehicle y Velocity vs Time');
xlabel('t (s)');
ylabel('v_{y} (ft/s)');
set(gca,'XGrid','off','YGrid','on')
subplot(3,2,6)
plot(t,x(6,:));
title('Vehicle z Velocity vs Time');
xlabel('t (s)');
ylabel('v_{z} (ft/s)');
set(gca,'XGrid','off','YGrid','on')

figure()
plot3(x(1,:),x(2,:),x(3,:))
title('Vehicle Trajectory')
axis padded
xlabel('p_{x} (ft)')
```

```
ylabel('p_{y} (ft)')
```



```
zlabel('p_{z} (ft)')
grid on

end
```

10.2 Mass Budget

10.2.1 Lower Airframe

Component	Part Number	Component Mass (lb)	Quantity	Mass Margin	Mass (lb)	Notes
LOWER AIRFRAME	U21-1-1-002	3.7400	1	0	**3.740**	
MOTOR TUBE	U21-1-1-003	0.4950	1	0	**0.495**	
THRUST PLATE	U21-1-1-004	0.4000	1	0.1	**0.440**	
RADIAL BRACKET - A	U21-1-1-005	0.0212	6	0.1	**0.140**	
RADIAL BRACKET - RB	U21-1-1-005	0.0200	2	0.1	**0.044**	
RADIAL BRACKET - C	U21-1-1-005	0.0210	4	0.1	**0.092**	
FIN RING	U21-1-1-006	0.5000	2	0.1	**1.100**	
FIN BRACKET	U21-1-1-007	0.0289	16	0.1	**0.509**	
FINS	U21-1-1-008	0.3000	4	0.2	**1.440**	Approximate 2D Density = 0.0054 lb/in^2
TAILCONE	U21-1-1-009	0.4500	1	0.1	**0.495**	
Rail Button		0.0213	2	0	**0.043**	
Rail Button Bolt	91253A540	0.0278	2	0	**0.056**	
#8-32 x 3/8" Screw	91255A192	0.0029	22	0	**0.064**	
#8-32 x 1/2" Screw	91255A194	0.0035	40	0	**0.140**	
#8-32 Hex Nut	91841A009	0.0031	8	0	**0.025**	
#8 Washer	92141A009	0.0035	32	0	**0.112**	
#8-32 M-F Standoff	91780A194	0.0014	4	0	**0.006**	
3/8-16 1.25" Screw	91253A626	0.0420	1	0	**0.042**	
Pro75 4G Hardware		4.0400	1	0	**4.040**	
CTI L1395		5.1700	1	0	**5.170**	
Total Mass					**18.19**	

10.2.2 Middle Airframe

Component	Part Number	Component Mass (lb)	Quantity	Mass Margin	Mass (lb)	Notes
MIDDLE AIRFRAME	U21-1-2-002	3.5	1	0	**3.5**	
Rail Button		0.0213	1	0	**0.02125**	
Rail Button Bolt	91253A540	0.0278	1	0	**0.0278**	
120" Main Chute		1.5625	1	0	**1.5625**	
Swivel		0.05	1	0	**0.05**	
5/16" Quick Link		0.163	3	0	**0.489**	
Shock Cord		0.0022	300	0	**0.66**	Quantity measured in inches
Reefing Ring		0.071	1	0	**0.071**	
Total Mass					**6.38155**	

10.2.3 Upper Airframe

Component	Part Number	Component Mass (lb)	Quantity	Mass Margin	Mass (lb)	Notes
UPPER AIRFRAME	U20-1-3-002	2.75	1	0	**2.75**	
NOSECONE	U20-1-3-003	3.6	1	0	**3.60**	
PISTON COUPLER	U21-1-3-004	0.635	1	0	**0.64**	
PISTON BULKHEAD	U21-1-3-005	0.224	1	0.2	**0.27**	
NOSECONE BULKHEAD	U21-1-3-006	0.223	1	0.1	**0.25**	
RADIAL BRACKET - C	U21-1-1-005	0.0210	4	0.1	**0.09**	
#8-32 x 1/2" Screw	91255A194	0.0035	8	0	**0.03**	
#8-32 x 3/8" Screw	91255A192	0.0029	4	0	**0.01**	
32" Drogue Chute		0.113	1	0	**0.11**	
Swivel		0.05	1	0	**0.05**	
5/16" Quick Link		0.163	4	0	**0.65**	
Shock Cord		0.0022	300	0	**0.66**	Quantity measured in inches

Payload		5	1	0.1	**5.50**	
Total Mass					**14.61**	

10.2.4 Avionics Bay

Component	Part Number	Component Mass (lb)	Quantity	Mass Margin	Mass (lb)	Notes
AVIONICS BAY COUPLER	U21-1-4-002	1.74	1	0	**1.740**	
AVIONICS BAY UPPER BULKHEAD	U21-1-4-003	0.2403	1	0.2	**0.288**	
AVIONICS BAY LOWER BULKHEAD	U21-1-4-004	0.2089	1	0.2	**0.251**	
AVIONICS BAY SPINE	U21-1-4-005	0.1922	1	0.1	**0.211**	
AIRBRAKE BAND	U21-1-4-006	0.116	1	0	**0.116**	
AVIONICS BAY CONNECTION RING	U21-1-4-007	0.272	1	0.1	**0.299**	
AVIONICS SLED	U21-1-4-008	0.5	1	0.2	**0.600**	
AVIONICS BAY SPINE LOCK	U21-1-4-009	0.0786	1	0.2	**0.094**	
AVIONICS BAY SPINE ADAPTER	U21-1-4-010	0.0432	1	0.1	**0.048**	
RADIAL BRACKET - C	U21-1-1-005	0.0210	4	0.1	**0.092**	
1/4-20 x .75" Screw	91251A540	0.0141	1	0	**0.014**	
1/4" Washer	92141A029	0.0040	1	0	**0.004**	
5/16" U-Bolt	8880T8880	0.2	1	0	**0.200**	
#8-32 x 1/2" Screw	91255A194	0.0035	8	0	**0.028**	
#8-32 x 5/8" Screw	91255A196	0.00404	8	0	**0.032**	
MPU-6050		0.0028	1	0.3	**0.004**	
RFM-95W		0.00456	1	0.3	**0.006**	
MPL3115A2		0.00224	1	0.3	**0.003**	
MLX90393		0.004	1	0.3	**0.005**	
NEO-M9N, U.FL		0.00832	1	0.3	**0.011**	
GNSS Antenna (10mm)		0.001	1	0.3	**0.001**	
Teensy 3.2		0.00392	1	0.3	**0.005**	

Battery		0.0794	2	0	**0.159**	
Total Mass					**4.212**	

10.2.5 Recovery Bay

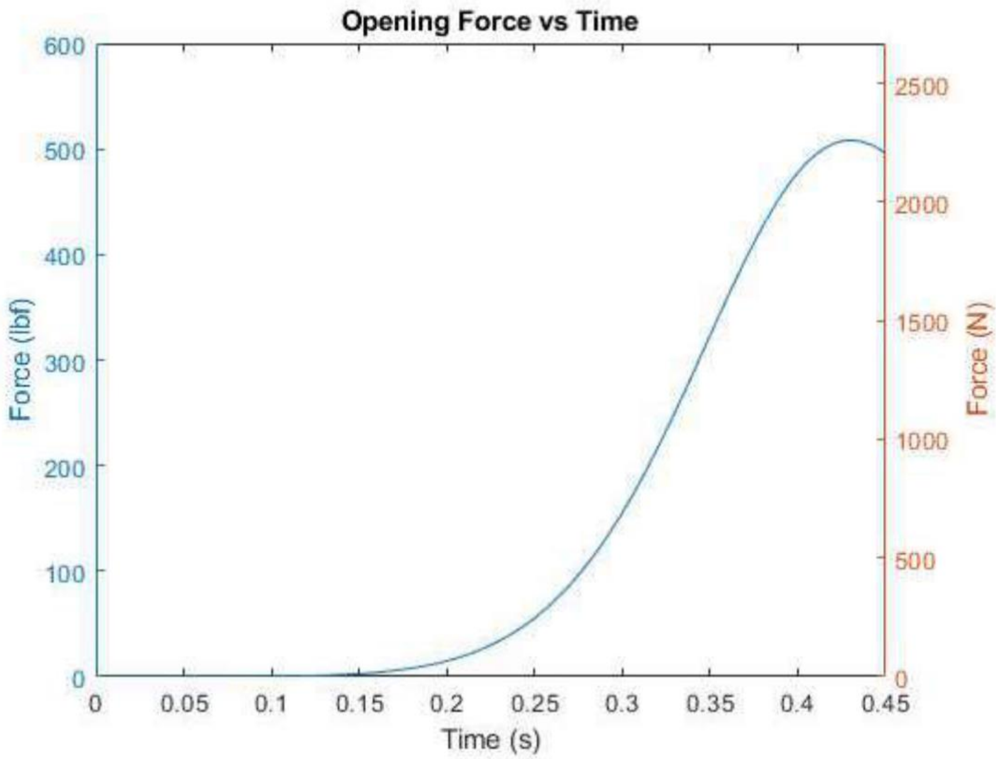
Component	Part Number	Component Mass (lb)	Quantity	Mass Margin	Mass (lb)	Notes
RECOVERY BAY COUPLER	U21-1-5-002	1.2100	1	0	**1.210**	
RECOVERY BAY BULKHEAD	U21-1-5-003	0.2300	2	0.1	**0.506**	
RECOVERY BAY SPINE	U21-1-5-004	0.1553	1	0.1	**0.171**	
RECOVERY BAY ADAPTER	U21-1-5-005	0.1471	2	0.1	**0.324**	
SWITCH BAND	U21-1-5-006	0.1160	1	0	**0.116**	
RECOVERY BAY SLED	U21-1-5-007	0.4650	1	0.1	**0.512**	
#4-40 Shear Pin	93135A109	0.0002	8	0	**0.002**	
#8-32 Thumb Screw	91830A206	0.0149	4	0	**0.060**	
#8-32 x 1/2" Screw	91253A194	0.0035	4	0	**0.014**	
#8 Washer	92141A009	0.0035	4	0	**0.014**	
#8-32 Hex Nut	91841A009	0.0031	4	0	**0.012**	
1/4-20 x .75" Screw	91251A540	0.0141	2	0	**0.028**	
1/4" Washer	92141A029	0.0040	2	0	**0.008**	
5/16" U-Bolt	8880T8880	0.2000	2	0	**0.400**	
Stratologger CF		0.0240	2	0	**0.048**	
Nano-Tech 370 mAh Lipo		0.0860	2	0	**0.172**	
Rotary Switch		0.0140	2	0	**0.028**	
2 Pole Terminal Block		0.0080	4	0	**0.032**	
3g Charge Well		0.0240	4	0	**0.096**	
22 AWG Wire		0.0020	2	0	**0.004**	Quantity measured in feet
JST Connector		0.0005	8	0	**0.004**	
Total Mass					3.760	

10.2.6 Airbrakes

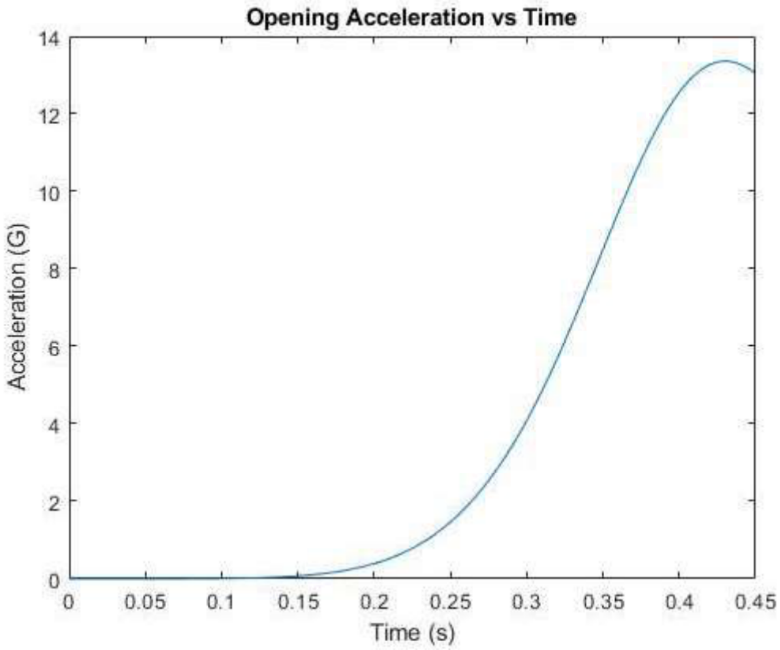
Component	Part Number	Component Mass (lb)	Quantity	Mass Margin	Mass (lb)
SPINE RING	U21-1-6-002	0.0092	3	0.1	**0.03**
GUIDE PLATE	U21-1-6-003	0.1523	1	0.1	**0.17**
MOTOR PLATE	U21-1-6-004	0.1745	1	0.1	**0.19**
ACTUATOR PLATE	U21-1-6-005	0.1483	1	0.1	**0.16**
AIRBRAKE FINS	U21-1-6-006	0.0343	4	0.1	**0.15**
DRIVE GEAR	U21-1-6-007	0.0716	1	0	**0.07**
SERVO GEAR	U21-1-6-008	0.04385	1	0	**0.04**
FIN PIN	U21-1-6-009	0.0054	4	0.1	**0.02**
GUIDE PIN	U21-1-6-010	0.00053	8	0.1	**0.00**
Airbrake Servo	HS-7985MG	0.1323	1	0	**0.13**
#8-32 x 1/2" Screw	91255A194	0.0035	8	0	**0.03**
Servo Screw		0.00155	4	0	**0.01**
Servo Nut		0.00022	4	0	**0.00**
Aluminum Standoff	93330A471	0.00517	4	0	**0.02**
Ball Bearing	60355K505	0.042	1	0	**0.04**
Spine Mount Screw		0.00155	8	0	**0.01**
Spine Mount Nut		0.00022	8	0	**0.00**
Extra Weight		0.4	1		**0.40**
Total Mass					**1.49**

Parachute Opening Force Calculator

```
clear variables; close all; clc

diam = 120/39.37; % Parachute Diameter (in -> m)
diam_packed = 6/39.37; % Packed Diameter (in -> m)
Cd = 2.2; % Drag coefficient
n = 4; % Canopy fill constant - Dont change unless you know

v = 89/3.281; % Descent velocity (ft/s -> m/s)
alt = 600/3.281; % Deployment altitude (ft -> m)

m = 38.03/2.205; % Vehicle mass (lb -> kg)

area = pi*(diam/2)^2;
area_packed = pi*(diam_packed/2)^2;

Chute_Area = @(A, Ap, t, tf) A * ((1 - (Ap / A)) * (t / tf)^3 + (Ap / A))^2;

dt = 0.001;
t_inf = (n*diam)/v;
t = 0:dt:t_inf;

x = zeros(2,length(t));
x(:,1) = [alt;-v];

u = zeros(1,length(t));

A = zeros(1,length(t));
A(1) = area_packed;

for i = 2:length(t)
    A(i) = area * ((1-(area_packed/area))*(t(i)/t_inf)^3+(area_packed/area))^2;
    [~,~,~,rho] = atmosisa(x(1,i-1));
    u(i) = (0.5*rho*x(2,i-1)^2*A(i)*Cd)/m;
    x(:,i) = rk4(@xdot_1D,x(:,i-1),u(i),dt);
end

F = (u.*m)/4.448;

figure()
yyaxis left
plot(t,F)
range = ylim();
title('Opening Force vs Time')
xlabel('Time (s)')
ylabel('Force (lbf)')
yyaxis right
ylabel('Force (N)')
ylim(range.*4.448)
```




```
figure()
plot(t,u./9.81)
title('Opening Acceleration vs Time')
xlabel('Time (s)')
ylabel('Acceleration (G)')
```




```
fprintf('Opening Time = %0.2f sec',t_inf)
```

```
Opening Time = 0.45 sec
```

```
fprintf('Maximum Acceleration = %0.1f G',max(u./9.81))
```

```
Maximum Acceleration = 13.4 G
```

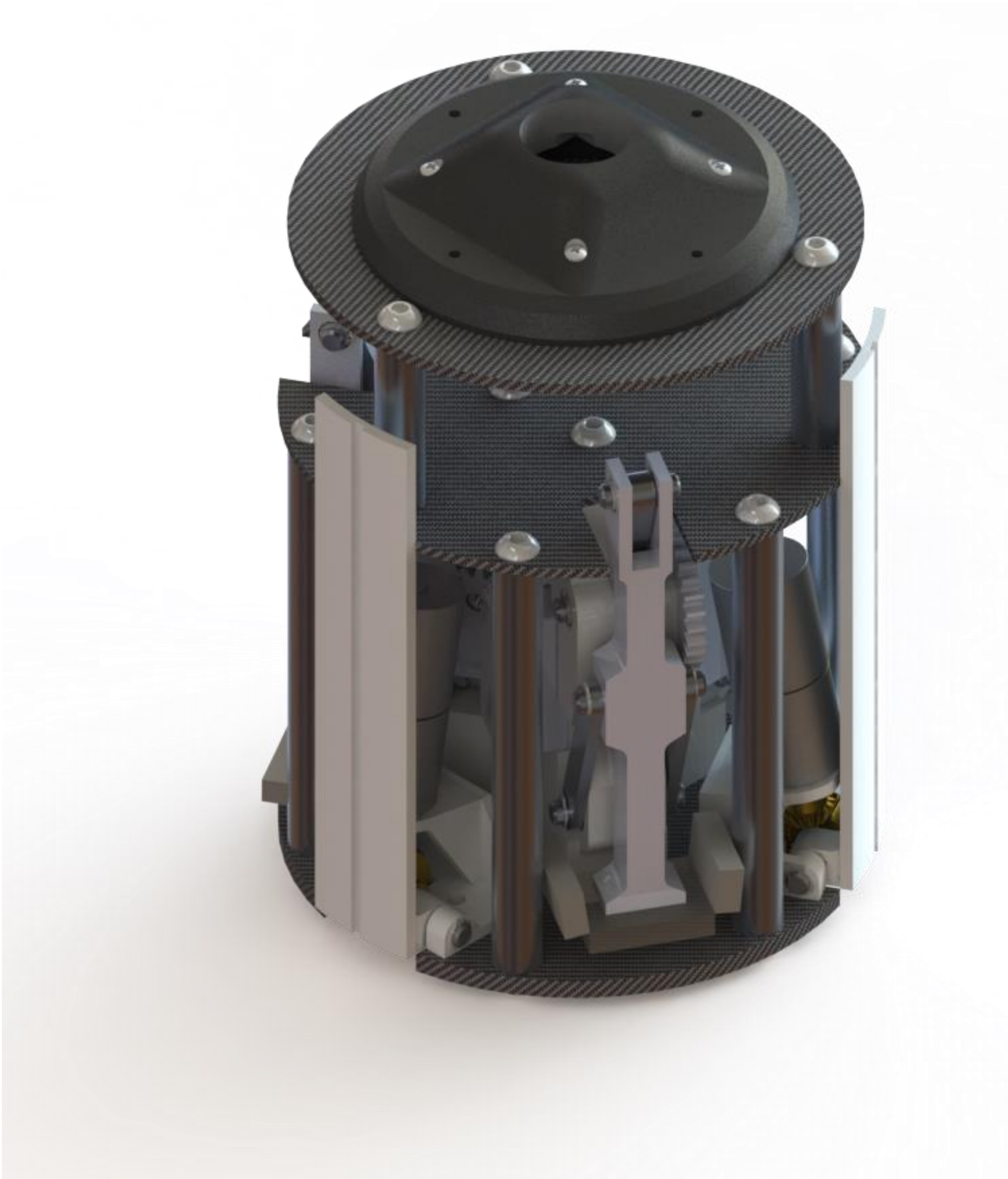
```
fprintf('Maximum Opening Force = %0.2f lbf (%0.2f N)',max(F),max(F).*4.448)
```

```
Maximum Opening Force = 508.19 lbf (2260.44 N)
```

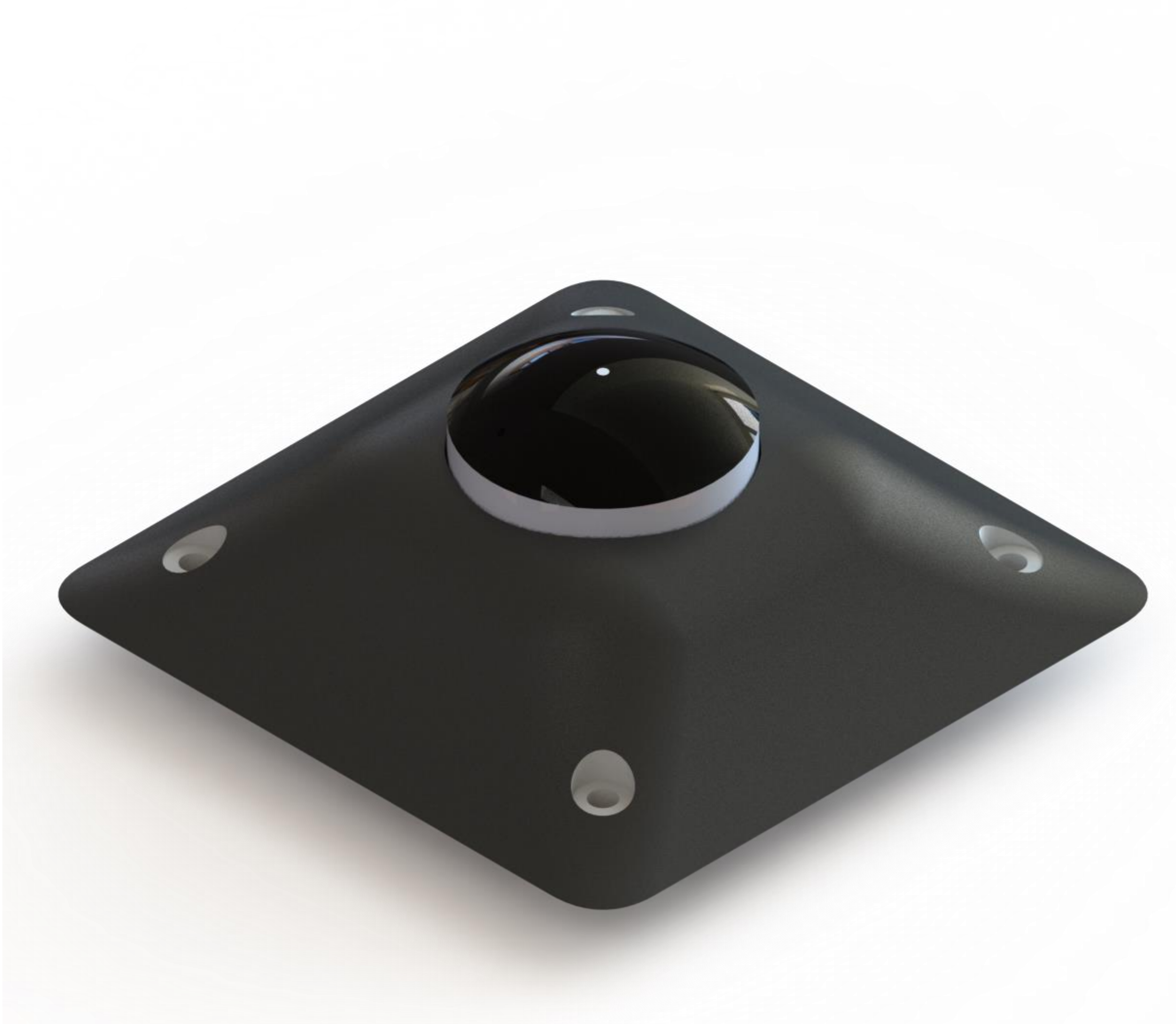
```
function x = rk4(func,x,u,h)

k1 = h * func(x,u);
k2 = h * func(x + (k1 / 2),u);
k3 = h * func(x + (k2 / 2),u);
k4 = h * func(x + k3,u);

x = x + ((1/6) * (k1 + 2*k2 + 2*k3 + k4));

end

function xdot = xdot_1D(x,u)
    xdot = [x(2);
        u-9.81];
end
```

10.4 Large Renders

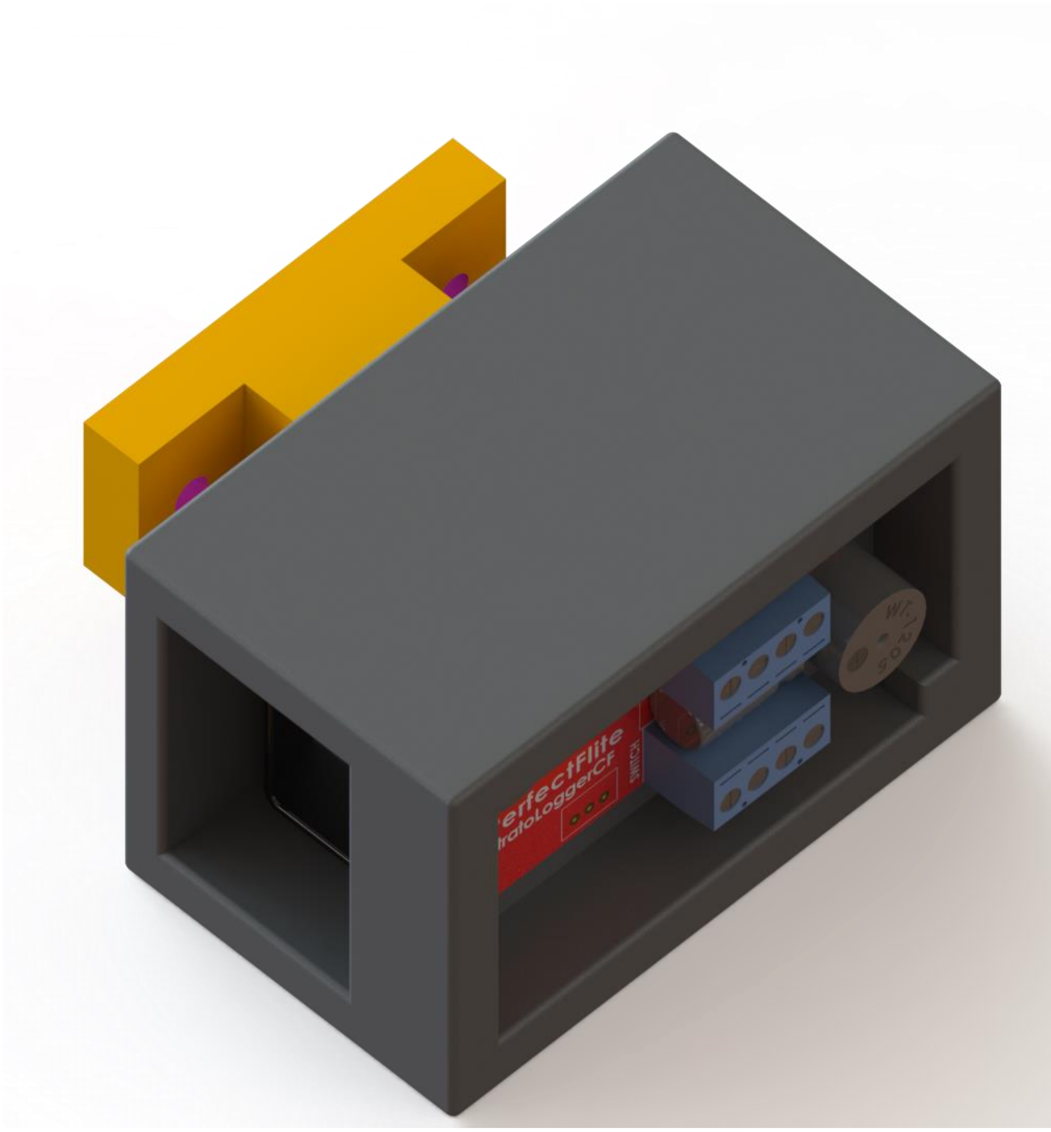
10.4.1 Payload Isometric View



10.4.2 Payload Photography Isometric View

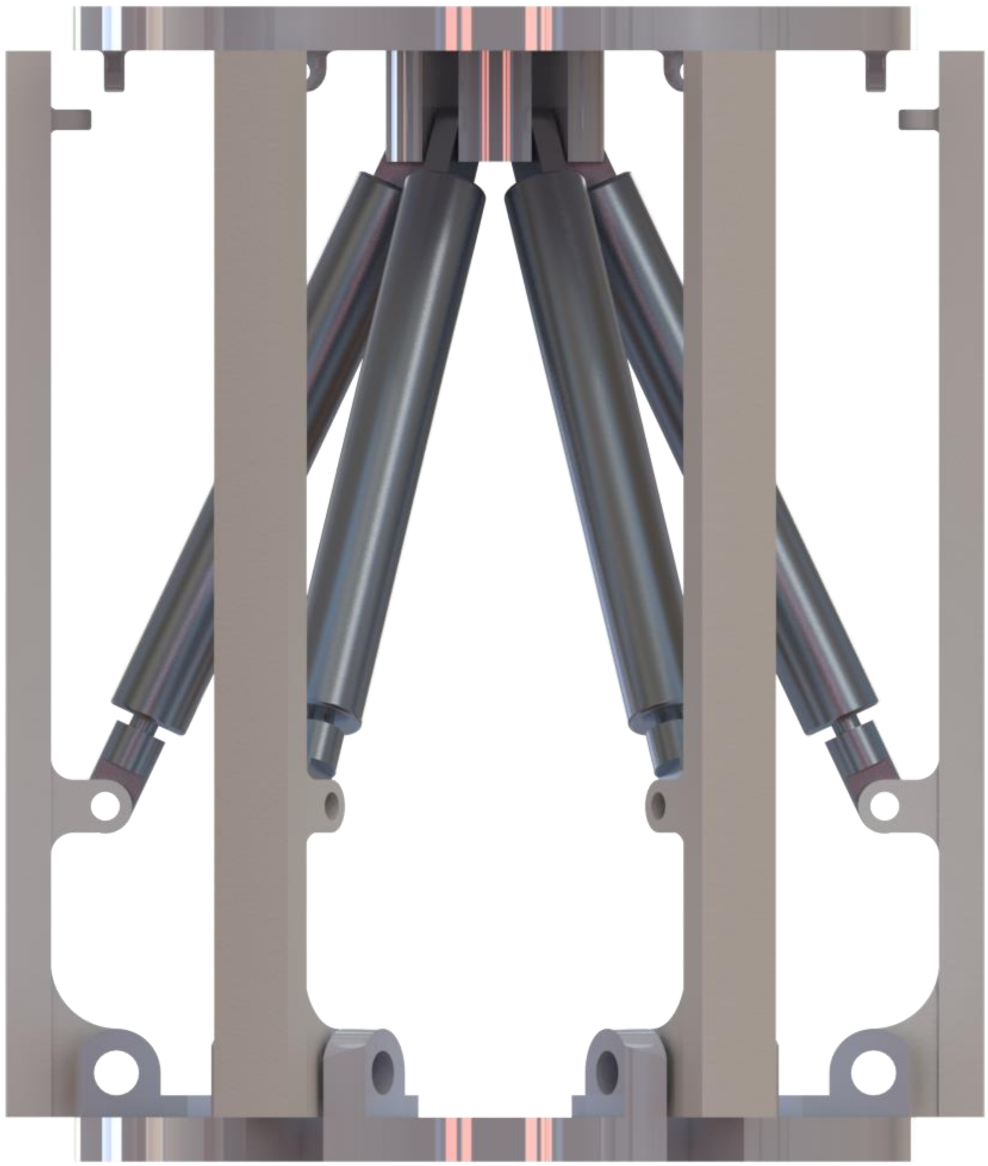


10.4.3 Payload Retention Tender Descender Isometric View

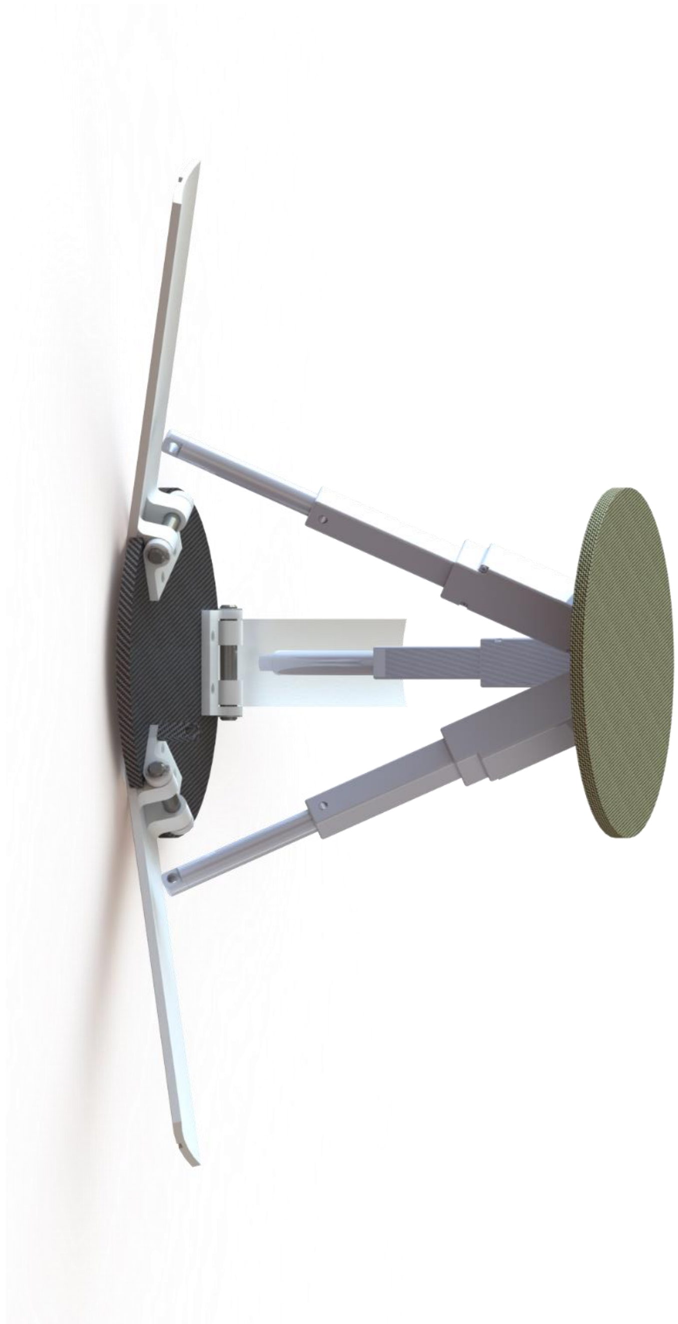


10.4.4 Payload Self Righting System Deployed View

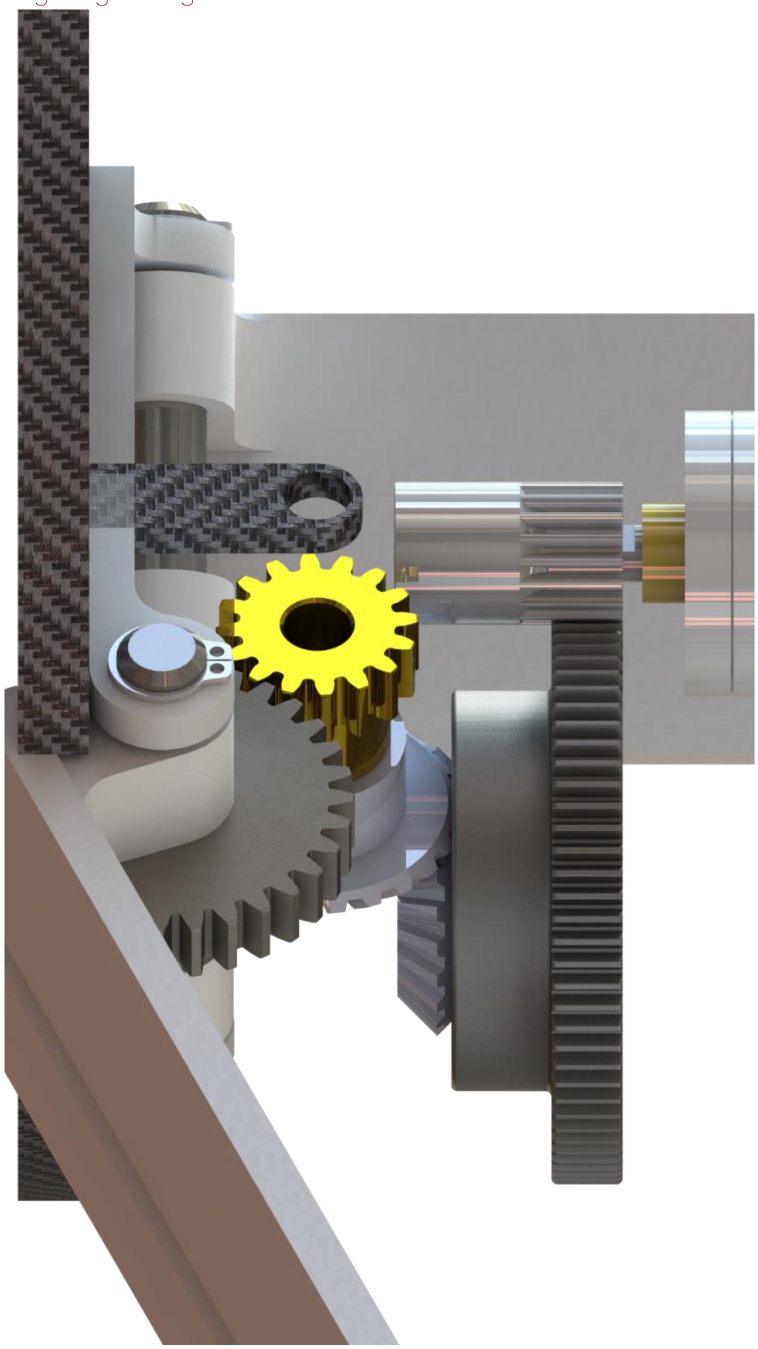
10.4.5 Payload Self Righting Gas Springs Front View



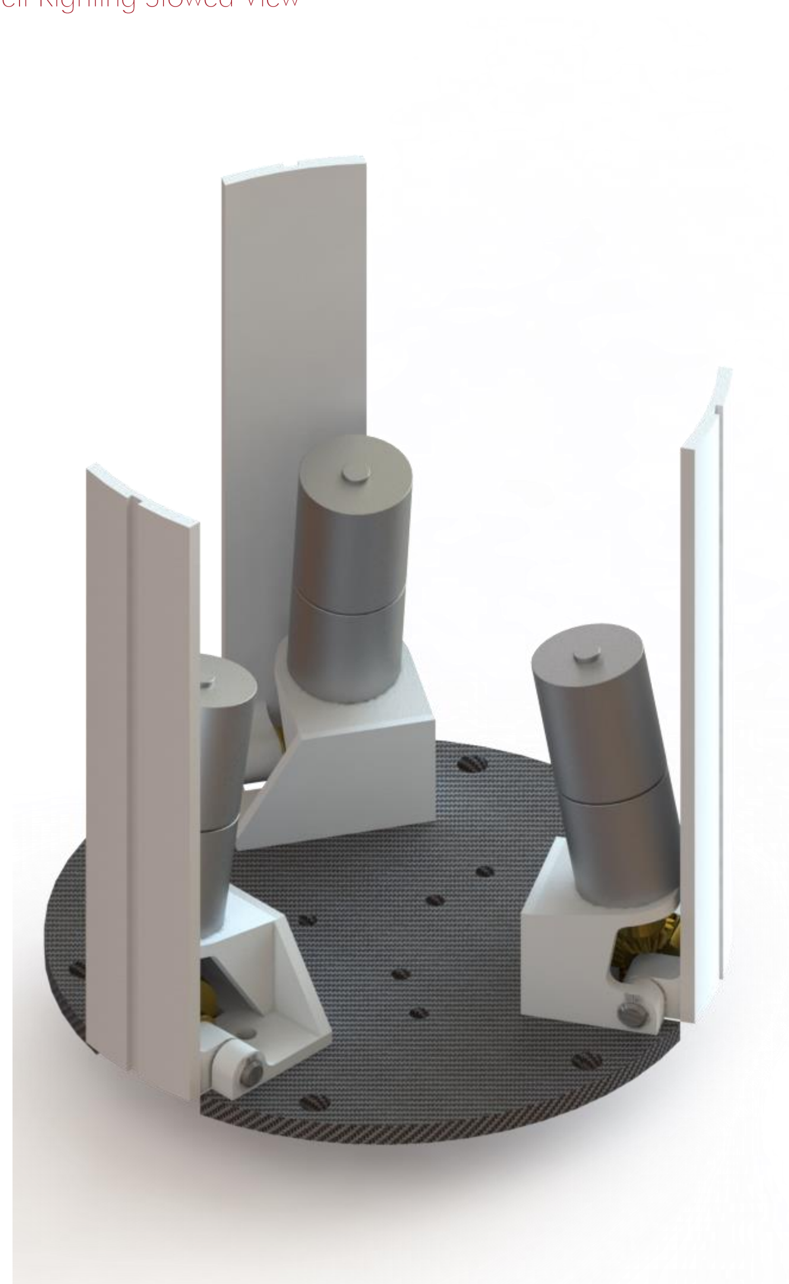
10.4.6 Payload Self Righting Linear Actuators



10.4.7 Payload Self Righting Multigear

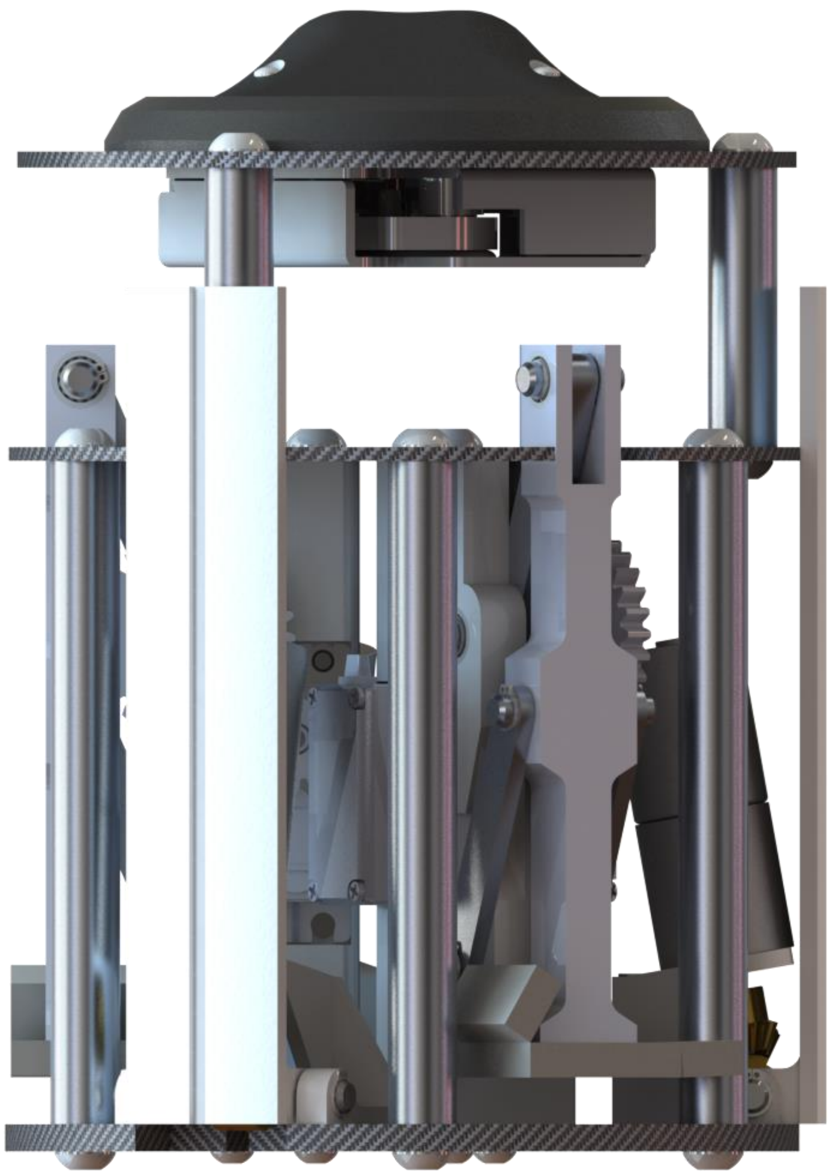


10.4.8 Payload Self Righting Stowed View



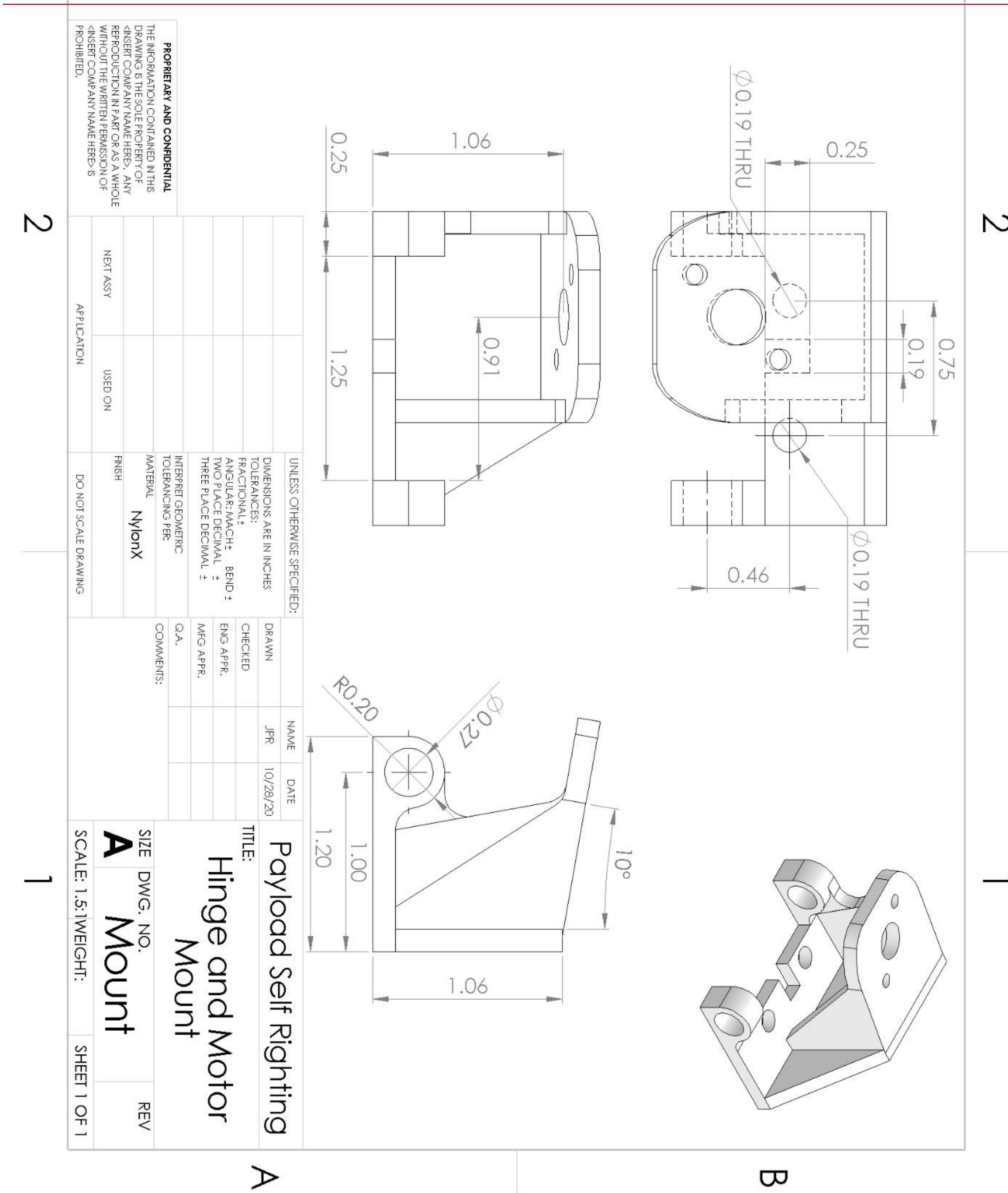
10.4.9 Payload Side Camera Field of View

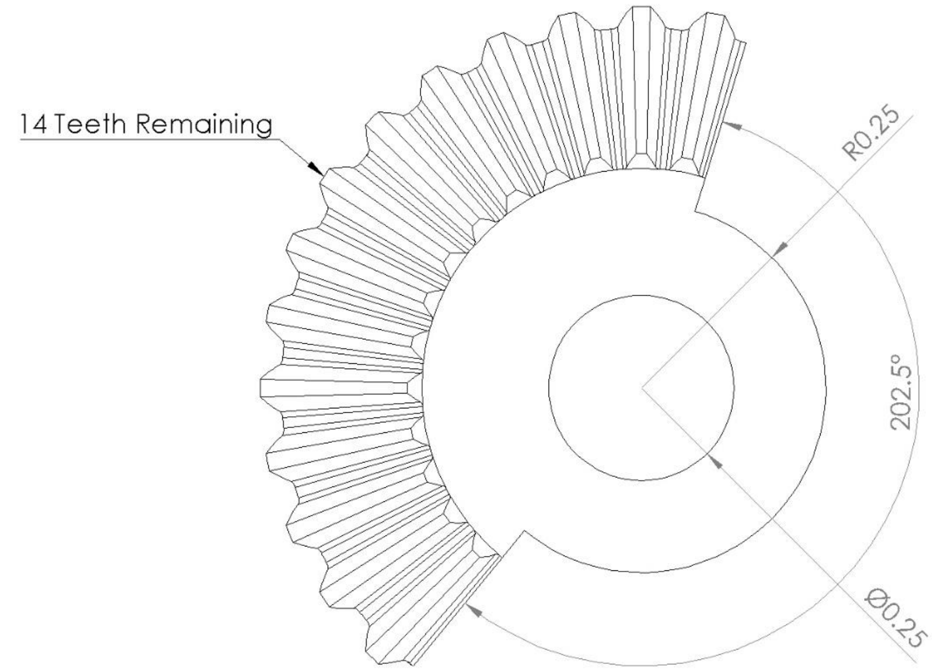
10.4.10 Payload Side View

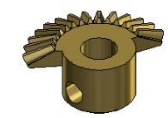


10.4.11 Payload Top View

10.5 Drawings

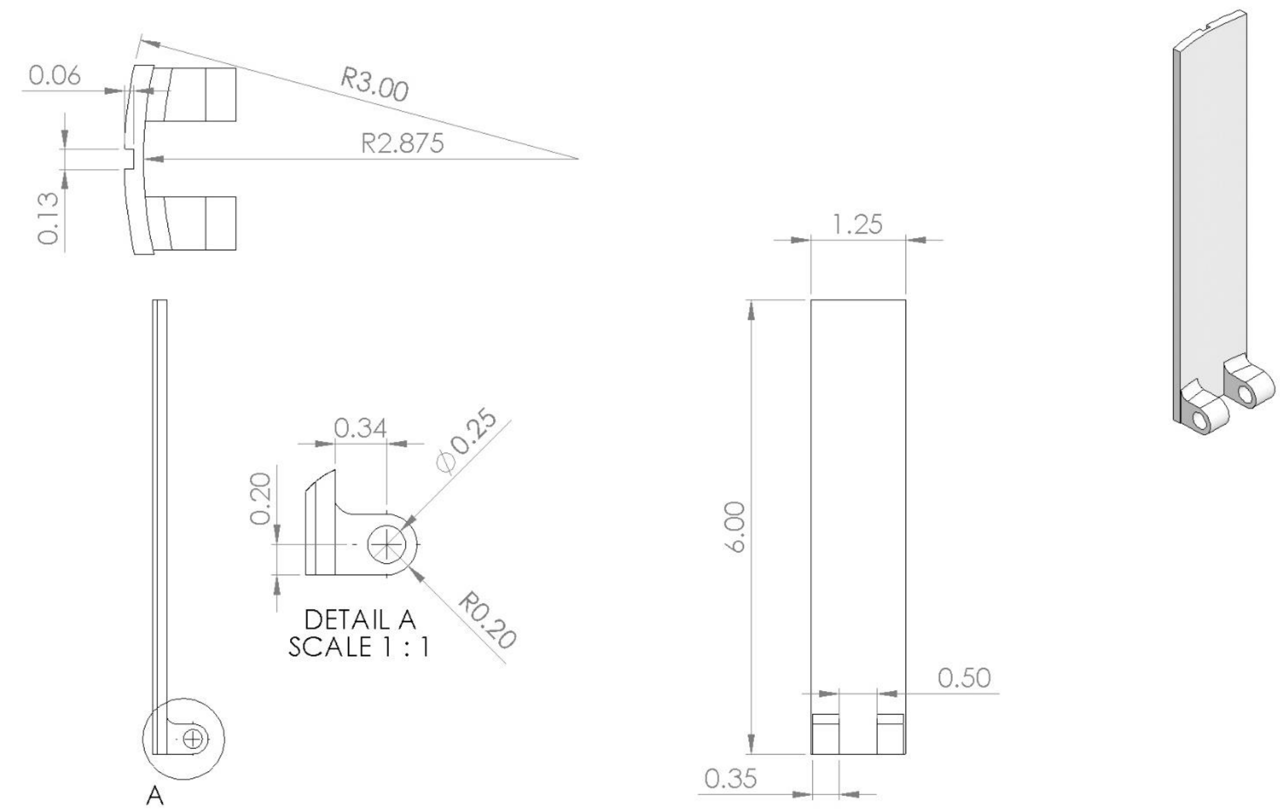






UNLESS OTHERWISE SPECIFIED:		NAME	DATE	Payload Self Righting	
DIMENSIONS ARE IN INCHES TOLERANCES: FRACTIONAL ± ANGULAR: MACH ± BEND ± TWO PLACE DECIMAL ± THREE PLACE DECIMAL ±	DRAWN	JPR	10/28/20	TITLE:	
	CHECKED			Modified 615444 Bevel Gear	
	ENG APPR.				
	MFG APPR.				
INTERPRET GEOMETRIC TOLERANCING PER:	Q.A.				
MATERIAL Brass	COMMENTS:			SIZE A	DWG. NO. Bevel Gear	REV
FINISH					
DO NOT SCALE DRAWING				SCALE: 5:1 WEIGHT:	SHEET 1 OF 1

NEXT ASSY	USED ON
APPLICATION	



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NEXT ASSY	USED ON
APPLICATION	

UNLESS OTHERWISE SPECIFIED:

DIMENSIONS ARE IN INCHES
TOLERANCES:
FRACTIONAL±
ANGULAR: MACH± BEND ±
TWO PLACE DECIMAL ±
THREE PLACE DECIMAL ±

INTERPRET GEOMETRIC TOLERANCING PER:

MATERIAL: NylonX

FINISH

DO NOT SCALE DRAWING

	NAME	DATE
DRAWN	JPR	10/28/20
CHECKED		
ENG APPR.		
MFG APPR.		
Q.A.		
COMMENTS:		

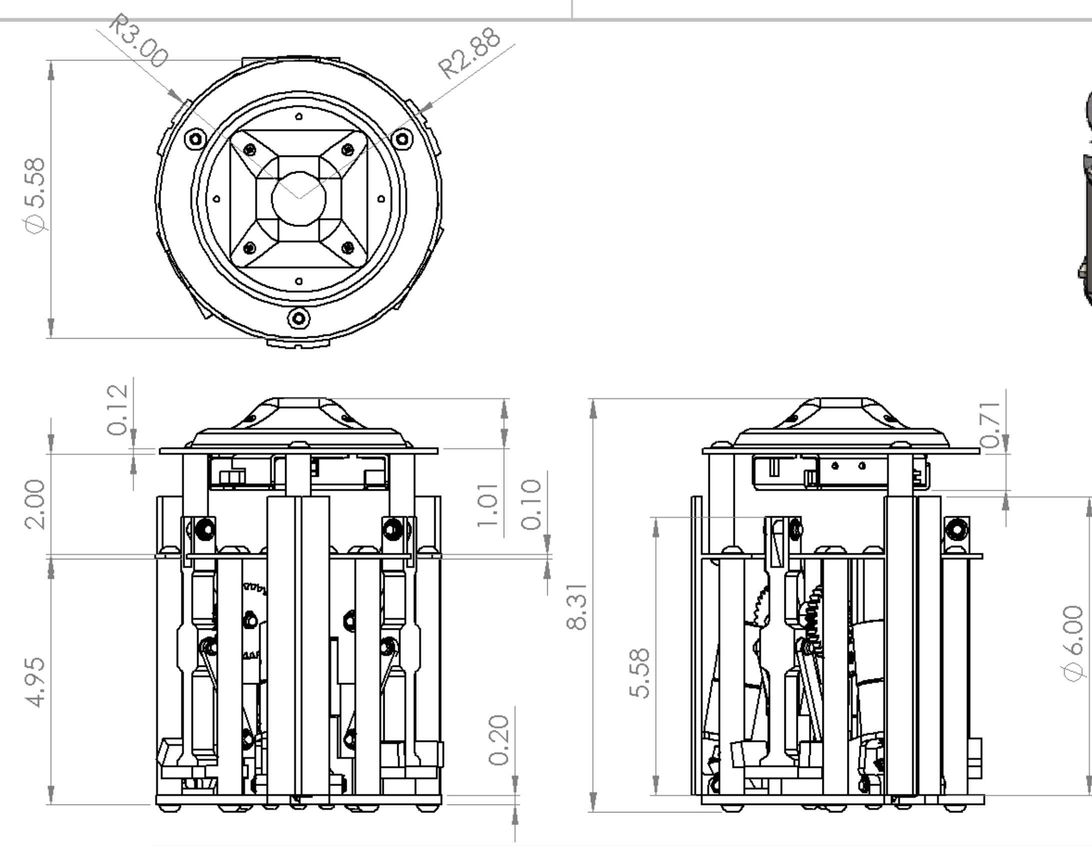
Payload Self Righting

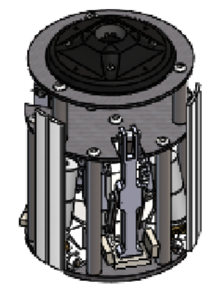
TITLE:

Self Righting Petal

SIZE: **A** DWG. NO. Petal REV

SCALE: 1:2 WEIGHT: SHEET 1 OF 1





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APPLICATION	

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DIMENSIONS ARE IN INCHES
TOLERANCES:
FRACTIONAL ±
ANGULAR: MACH ± BEND ±
TWO PLACE DECIMAL ±
THREE PLACE DECIMAL ±

INTERPRET GEOMETRIC TOLERANCING PER:

MATERIAL

FINISH

DO NOT SCALE DRAWING

	NAME	DATE
DRAWN	JPR	11/1/20
CHECKED		
ENG APPR.		
MFG APPR.		
Q.A.		

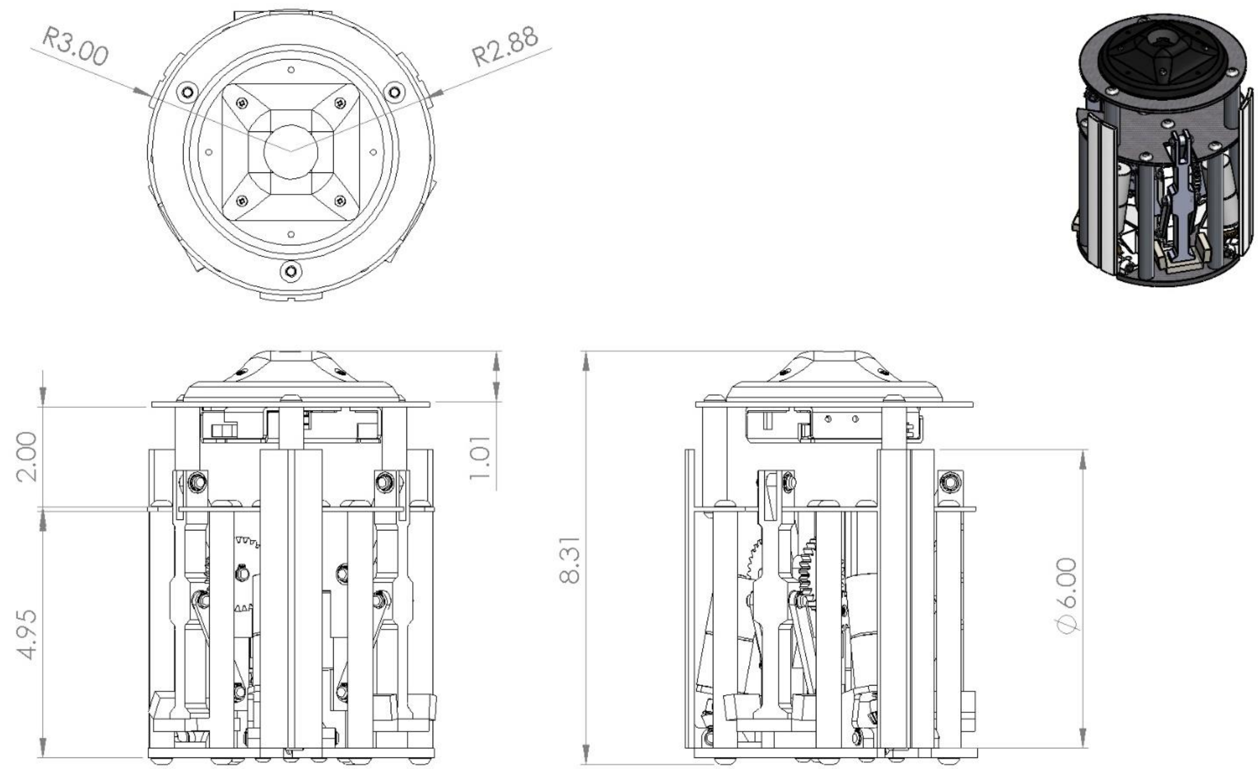
COMMENTS:

TITLE:

Payload Top Assembly

SIZE	DWG. NO.	REV
A	Top Assembly_V2	

SCALE: 1:3 WEIGHT: SHEET 1 OF 1



		UNLESS OTHERWISE SPECIFIED:		NAME	DATE	
		DIMENSIONS ARE IN INCHES TOLERANCES: FRACTIONAL ± ANGULAR: MACH ± BEND ± TWO PLACE DECIMAL ± THREE PLACE DECIMAL ±	DRAWN	JPR	11/1/20	TITLE: Payload Top Assembly
			CHECKED			
			ENG APPR.			
			MFG APPR.			
		INTERPRET GEOMETRIC TOLERANCING PER:	Q.A.			
		MATERIAL	COMMENTS:			SIZE A — DWG. NO. Top Assembly — REV
NEXT ASSY	USED ON	FINISH				
APPLICATION		DO NOT SCALE DRAWING				SCALE: 1:3 — WEIGHT: — SHEET 1 OF 1

11 References

[1] Weather Spark, "Average Weather in April in Saint Albans," [Online]. Available: https://weatherspark.com/m/24994/4/Average-Weather-in-April-in-Saint-Albans-Vermont-United-States#Sections-Wind.

[2] T. V. Milligan, "What is the best fin shape for a model rocket," *Peak Of Flight,* 2nd May 2017.

[3] "Lug Analysis," MechaniCalc, [Online]. Available: https://mechanicalc.com/reference/lug-analysis.

[4] NC State University, "20 20 NASA Student Launch Flight Readiness Review," 2020.

[5] L. Westra, "SIZING A PARACHUTE," University of Idaho, [Online]. Available: https://www.webpages.uidaho.edu/dl2/on_target/tv.htm.

[6] Hexcel, "Material Safety Data Sheet Carbon Fiber," 1 10 2002. [Online]. Available: https://www.tapplastics.com/image/pdf/MSDS%20Carbon%20Fiber%20Sheet.pdf.

[7] Reliable Source Metals, "Safety Data Sheet Aluminum," 28 February 2019. [Online]. Available: https://irp-cdn.multiscreensite.com/46cb8fc8/files/uploaded/rs-metals-sds-msds-aluminum-final.pdf.

[8] Owens Corning, "SAFE USE INSTRUCTION SHEET Fiberglass," 23 10 2019. [Online]. Available: https://dcpd6wotaa0mb.cloudfront.net/mdms/sds/OCCM10021_CIUS_EN.PDF?v=1604299605950.

[9] Matter Hackers, "SAFETY DATA SHEET - NylonX," [Online]. Available: https://www.matterhackers.com/r/A7mInq.

[10] Makerbot, "Safety Data Sheet - PLA 3D Printer Filament," 26 09 2016. [Online]. Available: https://images.makerbot.com/support/production/SDS-000002ENA.pdf.

[11] Nuplex, "Material Safety Data Sheet ES180 Epoxy Resin," 4 8 2016. [Online]. Available: https://www.amcsupplies.com.au/manuals/R180%20Epoxy%20Resin%20-%20SDS.pdf.

[12] Quadrant, "MATERIAL SAFETY DATA SHEET Delrin," 18 7 2009. [Online]. Available: https://www.tapplastics.com/image/pdf/MSDS%20Carbon%20Fiber%20Sheet.pdf.

[13] Intellect Battery , "Safety Data Sheet LiPo," 1 1 2020. [Online]. Available: https://www.spectro.com/-/media/ametekspectro/documents/msds/msds%20lithium%20polymer%20battery%20artno75040816%2017%20en082017.pdf?la=en.

[14] Environmental Protection Agency, "Material Safety Data Sheet MSDS-BP (Potassium Nitrate)," 17 3 2009. [Online]. Available: https://www.epa.gov/sites/production/files/2015-05/documents/9530608.pdf.

[15] Cesseroni Technology Incorporated, "Pro-X Rocket Motor Reload Kits & Fuel Grains Safety Data Sheet," 12 3 2020. [Online]. Available: http://www.pro38.com/MSDS/SDS--ProX--ver4-03-EN_FR.pdf.

[16] Charles River Endosafe, "Material Safety Datasheet - Pyrogen," 15 7 2002. [Online]. Available: http://aiaaocrocketry.org/wp-content/uploads/2010/11/pyrogen-msds.pdf.