HPRC
WPI AIAA

Critical Design Review

NASA University Student Launch Initiative
January 25th, 2020

Worcester Polytechnic Institute
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WPI STUDENT BRANCH

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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 CDR

4.1 Team Summary

4.1.1 Team Information

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4.1.2 Team Mentor

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4.2 Launch Vehicle Summary

WPI's launch vehicle will have an outer diameter of 6.17 in, a length of 111 in, and a wet mass of 48.757lb. The vehicle will utilize a Cesaroni Technologies Incorporated (CTI) L1395 as its primary motor. The vehicle is expected to reach an apogee of 4550 ft, launching off a 12 ft 15x15 launch rail. The recovery system consists of a dual bay, dual deployment layout, ejecting a 36 in drogue parachute at apogee, and a 120 in main parachute at 600 ft. Redundant ejection charges will be ignited by 2 redundant altimeters with a 1 second delay.

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 PDR

5.1 Rocket Criteria

The length of the launch vehicle has changed from 108 in to 111 in, and the wet mass of the vehicle without payload has decreased slightly from 44.03 lb to 43.07 lb. The lower airframe increased in length slightly from 29.95 in to 30 in, and the tailcone reducing in length from 2 in to 1.95 in to retain the same overall subassembly length. The bolts through the tailcone have also been replaced with heat set inserts used to attach the tailcone to the lower fin ring. The upper airframe increased in length from 22 in to 25 in. The avionics bay spine diameter was changed from 0.5 in to 15mm, and 2 grooves for snap rings were added to the spine. The recovery bay sled was modified to reduce weight and simplify manufacturing. Within the recovery bay the backup altimeter has changed from a StratoLoggerCF to a Raven 4, and the backup altimeter battery has changed from a 2S 370mAh lithium polymer (LiPo) battery, to a 1S 300mAh LiPo battery. The separation point for the middle airframe has moved to the avionics bay, and the drogue parachute has increased in size from 32 in to 36 in.

The airbrakes system has changed significantly since PDR. The fins now slide on aluminium rails attached to the bottom plate, and the guide pins have been removed from the fins. PTFE adhesive film has been added to the rails and fins to reduce friction during operation. The driving pin on each fin now includes a bushing to reduce rolling friction on the driving plate. The original Hi-Tec 7985 MG servo has been replaced with a GoBILDA Super Speed 2000 Series Dual Mode servo for increased torque and operation speed. The original gears have been replaced with GoBILDA gears that mount directly to the servo and bolt to the driving plate, producing a 2:1 gear ratio. The driving plate now rotates on a 32mm ball bearing constrained by retaining rings attached to the spine.

5.2 Payload Criteria

Since PDR the payload has changed in minor ways. Both the self-righting and stabilization mechanisms had zeroing limit switches and a potentiometer added to them to allow for ease of control. The thickness of the bottom plate of the payload was reduced from 5mm to 3mms after force analysis. The foot geometry and materials in the stabilization system were changed after qualitative testing.

5.3 Project Plan

Leading towards the end of B-Term, our team was denied access to our on-campus machine shop and the ability to launch our subscale on campus. In addition to these setbacks and what they implicated for the future, since WPI had chosen to extend our winter break, our time and resources to work together on campus have been cut short. Due to the current circumstances at WPI, the officer and executive boards felt the team could no longer pursue the NASA USLI Launch Competition nor the NASA USLI Design Competition.

The team still plans on continuing with the project, just not formally with NASA due to the time and schedule conflicts. The team still plans on machining and constructing the rocket and payload, in addition to trying to launch it by the end of the year if possible. We realize this may be less feasible with not participating in competition but feel this will be more beneficial overall to team members as they will be more likely to have the chance to make their own designs come to life on a more flexible schedule.

The team also plans on creating CDR and FRR documents and presentations, with the FRR due date TBD if we launch. In order to compensate both our sponsors and our team, we plan on presenting these documents and presentations to our sponsors and mentors. In addition, we hope this more flexible schedule will allow us to hold more internal and external workshops.

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.

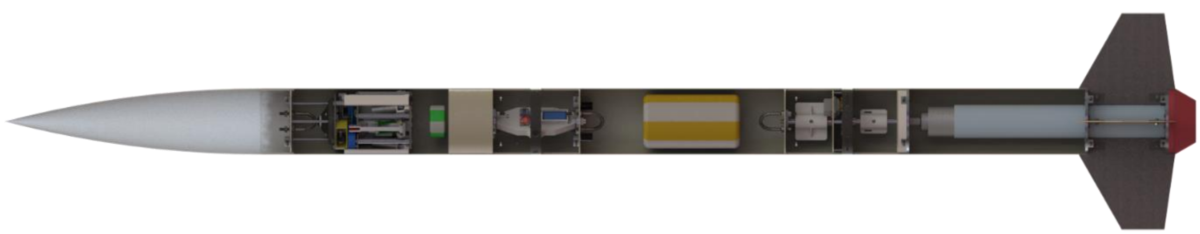


Figure 6-1 Launch Vehicle Section View

6.2 Mission Performance Predictions

6.2.1 Vehicle Mass Budget

A mass budget for the vehicle was created using reported masses for purchased components, and estimations based on material and geometry for custom components. Reported masses for the major sections of the vehicle can be found in Table 1. Listed values are for the wet mass of the vehicle, with payload in the upper airframe. A full breakdown of mass per component can be found in Appendix 10.1 and 10.2.

Sub Assembly	Mass (lb)
Lower Airframe	18.246
Middle Airframe	6.331
Upper Airframe	15.401
Avionics Bay	3.944
Recovery Bay	3.612
Airbrakes	1.224
Grand Total	48.757

Table 1 Sub Assembly Masses

6.2.2 Motor Selection

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 shown in Figure 6-2.

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 2 L1395 Motor Specifications

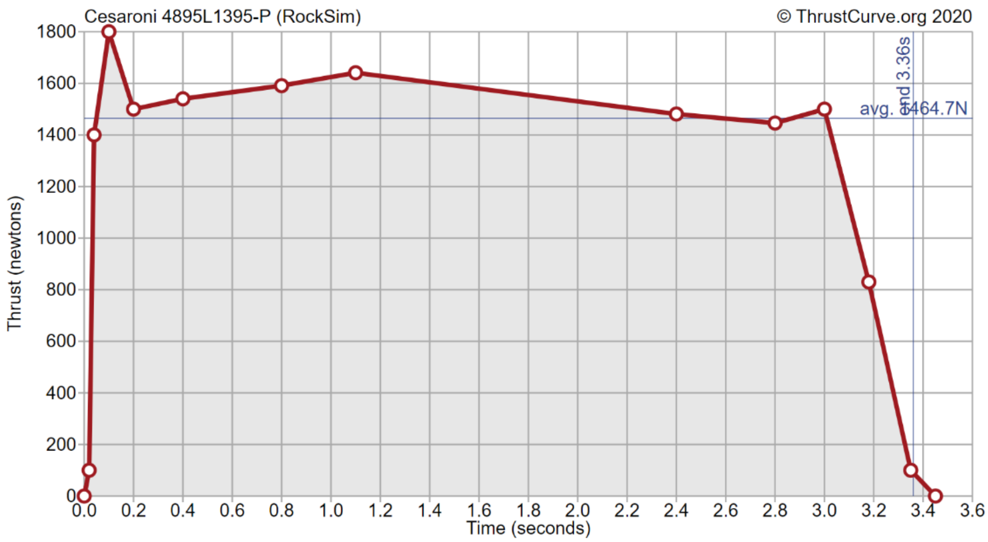


Figure 6-2 L1395 Thrust Over Time Graph

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 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 shown in Figure 6-3.Figure 6-3 L2375 Thrust Over Time Graph

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 3 L2375 Motor Specifications

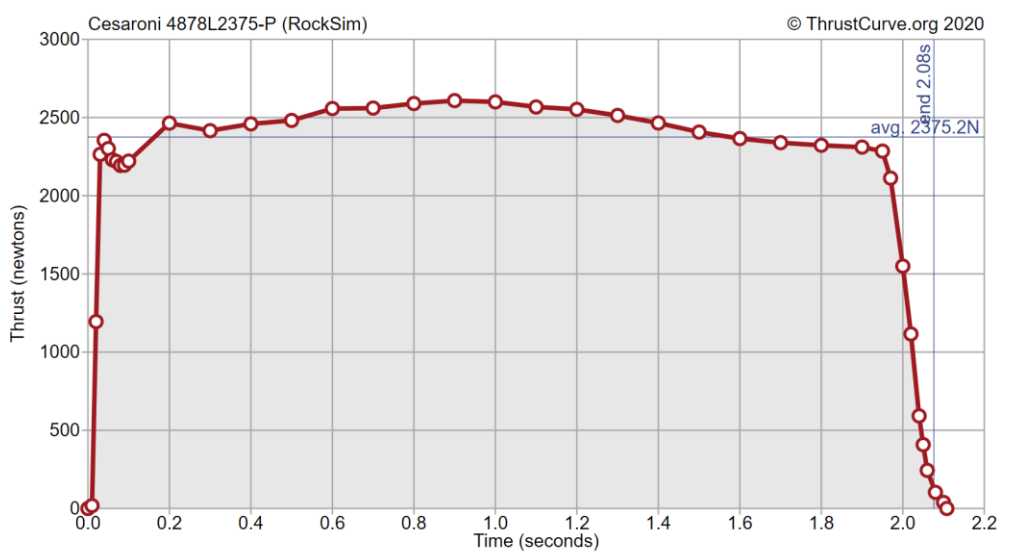


Figure 6-3 L2375 Thrust Over Time Graph

6.2.3 Stability

The stability margin, C_p, and C_g locations for the vehicle are shown in Figure 6-4.

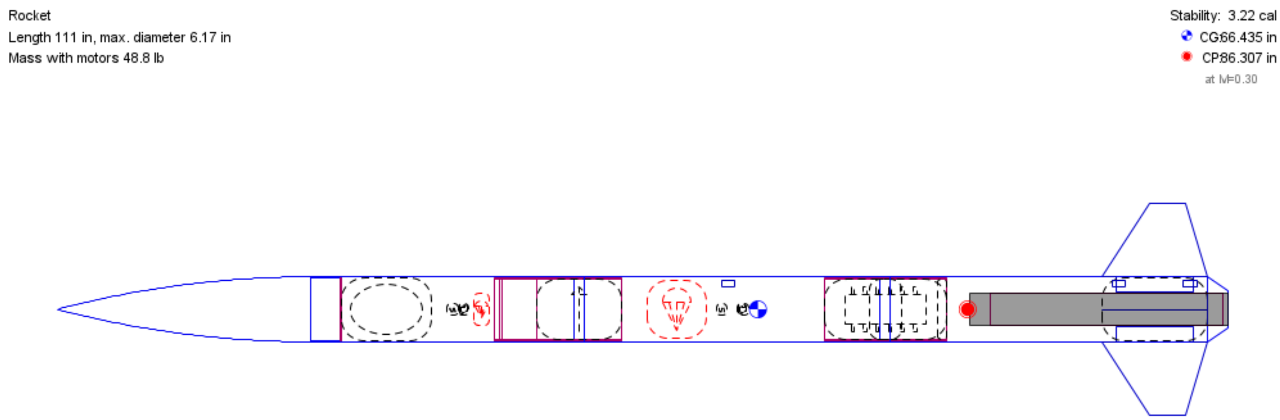


Figure 6-4 Vehicle Stability Parameters

The vehicle's stability margin during flight is shown in Figure 6-5. The vehicle leaves the launch rod with a stability margin of 3.25 cal, and at motor burnout has a stability margin of 3.97 cal. The vehicle remains within the acceptable stability range during all points in flight.

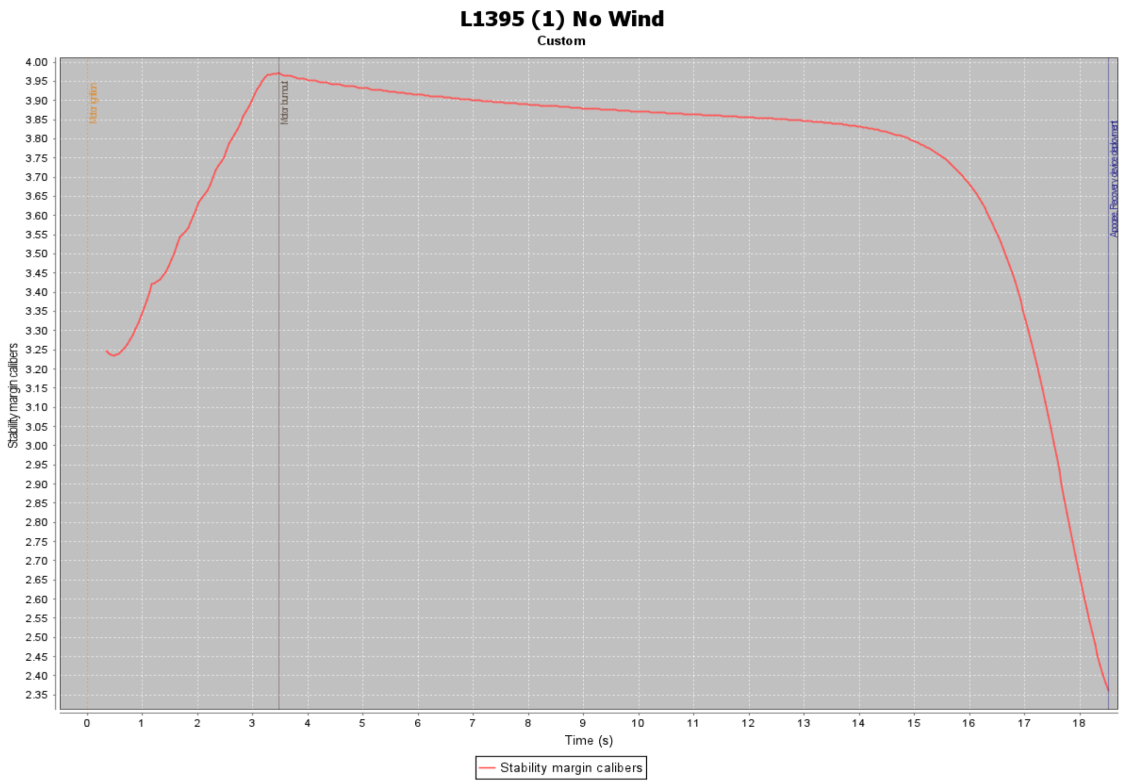


Figure 6-5 Vehicle Stability Plot

6.2.4 Vehicle Ascent

The vehicle's target apogee is set to 4550 ft, which it will achieve using the active airbrake system. Flight simulations were conducted in OpenRocket 15.03, using varying values of wind speed and launch angle to understand the range of possible apogees we can expect to reach without the airbrakes active.

Simulation	Wind Speed (mph)	Launch Rod Angle (deg)	Apogee (ft)
Best Case	3	5	5142
Standard Case	8	7.5	4934
Worst Case	20	10	4380

Table 4 OpenRocket Unguided Simulation Results

To account for the effect of the airbrakes, an extension was developed for OpenRocket to calculate and apply the necessary drag from the airbrakes to reach the target apogee. The development and functionality of this extension, as well as the control system used is discussed in section 6.7.3.2.

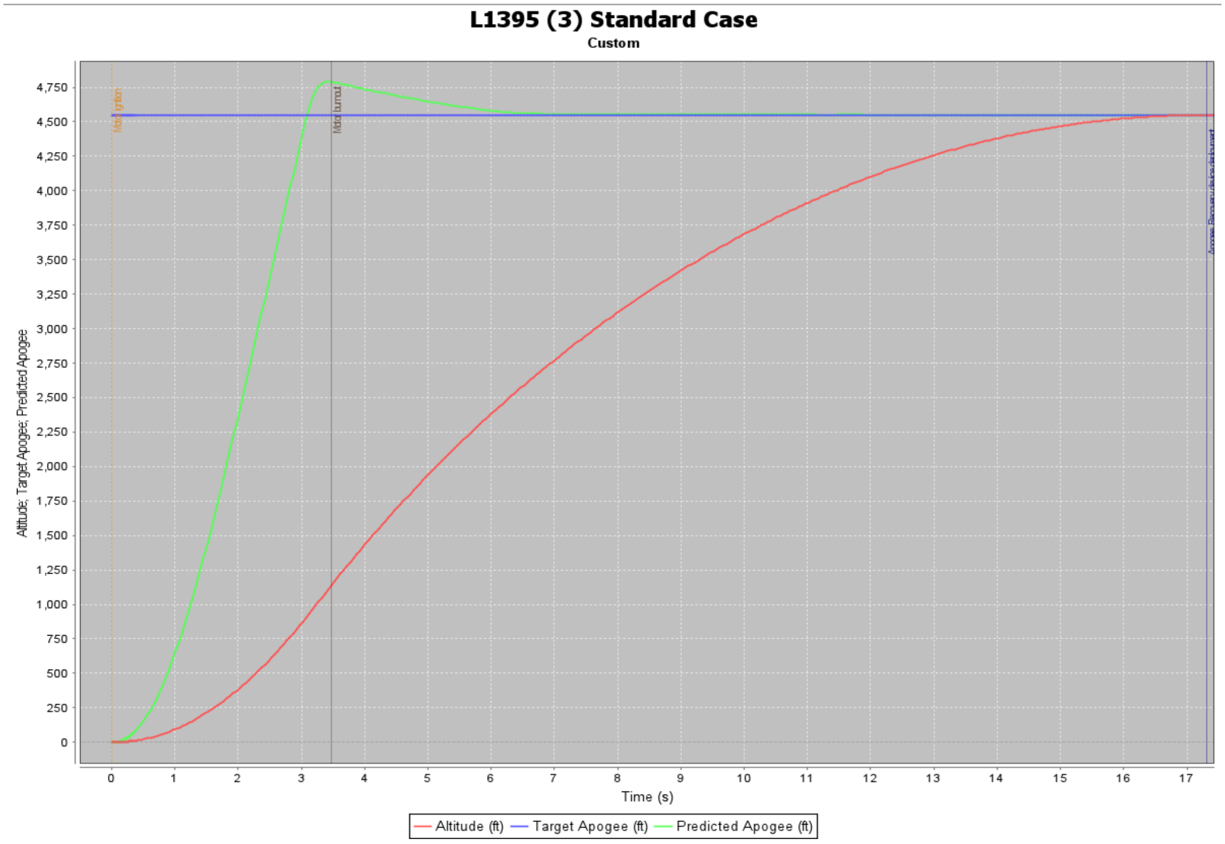


Figure 6-6 Controlled Flight Profile

Figure 6-6 shows the simulated flight profile with an active airbrake system. At burnout, the predicted apogee (in green) exceeds the target apogee (in blue). The airbrakes actuate between burnout and apogee to reduce the predicted apogee until the target apogee is reached. Table 5 lists the predicted apogee for the guided simulations.

Simulation	Wind Speed (mph)	Launch Rod Angle (deg)	Apogee (ft)
Best Case	3	5	4679
Standard Case	8	7.5	4548
Worst Case	20	10	4388

Table 5 OpenRocket Guided Simulation Results

As can be seen, both the worst case and best case simulations are not in range of the target apogee even with active control. However, it is unlikely that the actual conditions on launch day match either of these simulations, and with the most likely conditions the target apogee is well within range. The airbrakes are constrained structurally and mechanically, and thus can only offer so much altitude change. Despite this, the current system offers a reasonable range of apogees, as well as tolerance to changes in launch conditions and vehicle mass.

6.2.5 Vehicle Descent

Due to the competition requirements, this year's payload must detach from the launch vehicle during descent and fall under its own parachute. Although the vehicle's descent is typically simulated using OpenRocket, the simulation does not allow for internal mass components such as payload to be ejected as a stage. Since payload ejection will cause a significant mass difference, the vehicle's descent is simulating using a 3 DOF descent simulator developed and verified during the 2019-2020 competition year.

The initial conditions start the vehicle at our target apogee of 4550 ft, with zero velocity in all directions. The dry mass of the rocket, including the payload, at apogee is 43.59 lb. The drogue parachute deploys at apogee, and the vehicle begins to descend at a rate of 83.7 ft/s. At an altitude of 1000 ft, the payload detaches from the vehicle and the mass drops to 37.90 lb. The descent velocity decreases to 78.43 ft/s, and at 600 ft the main parachute is deployed. After main deployment, the vehicle descends with a constant mass until it lands with a ground hit velocity of 13.58 ft/s.

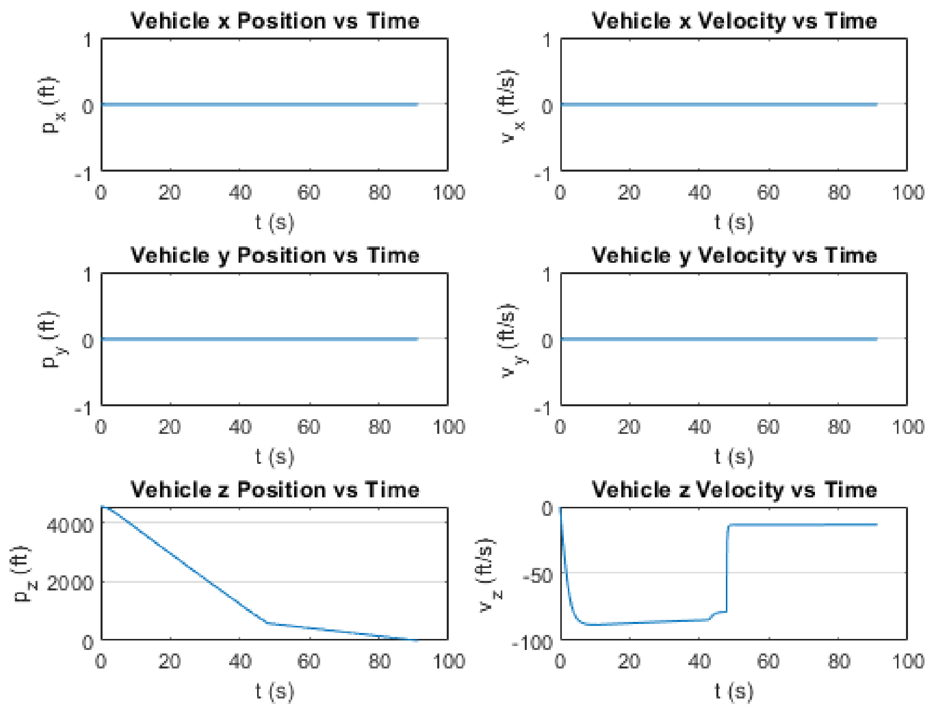


Figure 6-7 Simulated Vehicle Descent Profile

With a landing velocity of 13.58 ft/s for the launch vehicle, and 15.01 ft/s for the payload, each independent section has a kinetic energy and descent time shown in Table 6.

Section	Section Mass (lb)	Kinetic Energy (ft-lbf)	Descent Time (sec)
Lower Section	18.24	52.27	91.2
Middle Section	7.16	20.52	
Upper Section	7.35	21.06	
Payload	5.68	19.89	90.8

Table 6 Descent Parameters

As previously mentioned, OpenRocket cannot provide completely accurate descent parameters due to issues with mass, it was still used to validate our custom descent simulator, as shown in

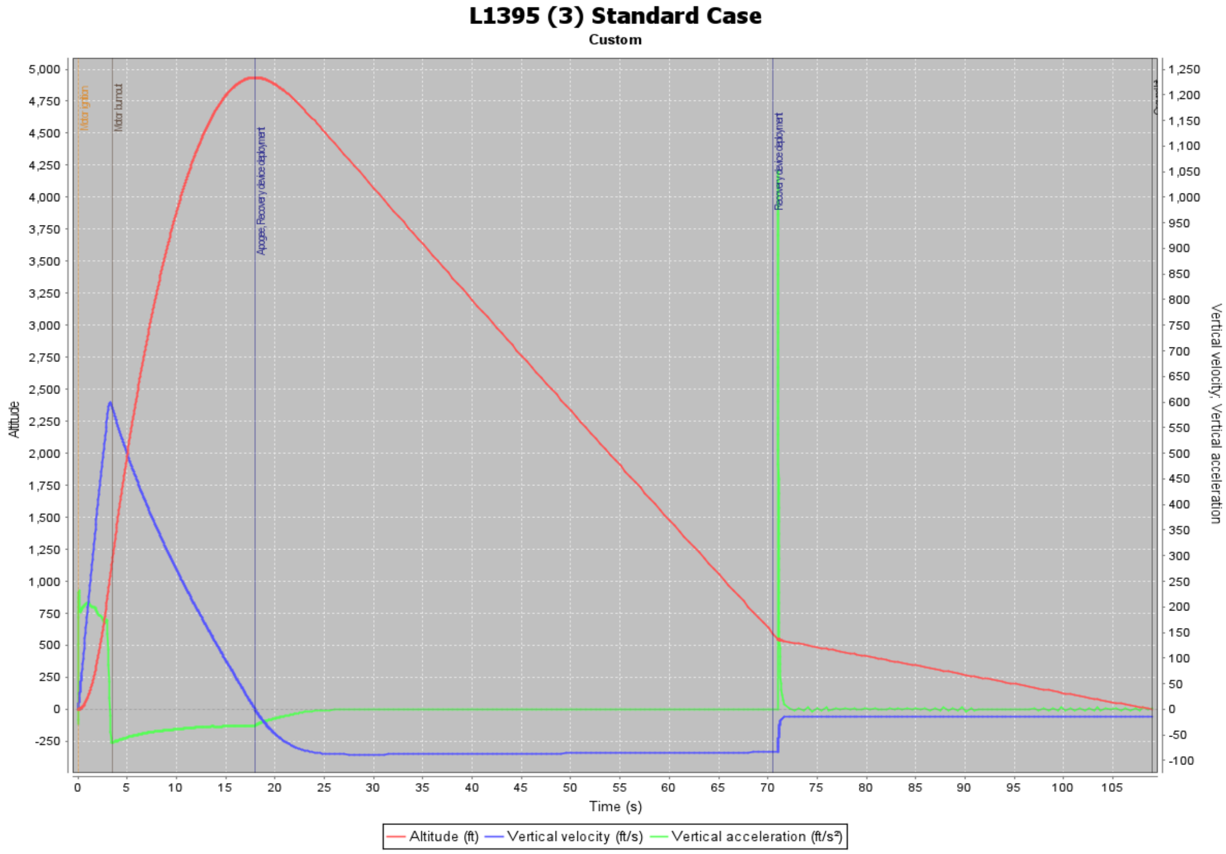


Figure 6-8 and Table 7. The landing velocity of the launch vehicle increases to 14.4 ft/s and the descent time decreases to 86.9 s due to the extra weight of payload. The additional payload mass is not included in the kinetic energy calculation of the upper section.

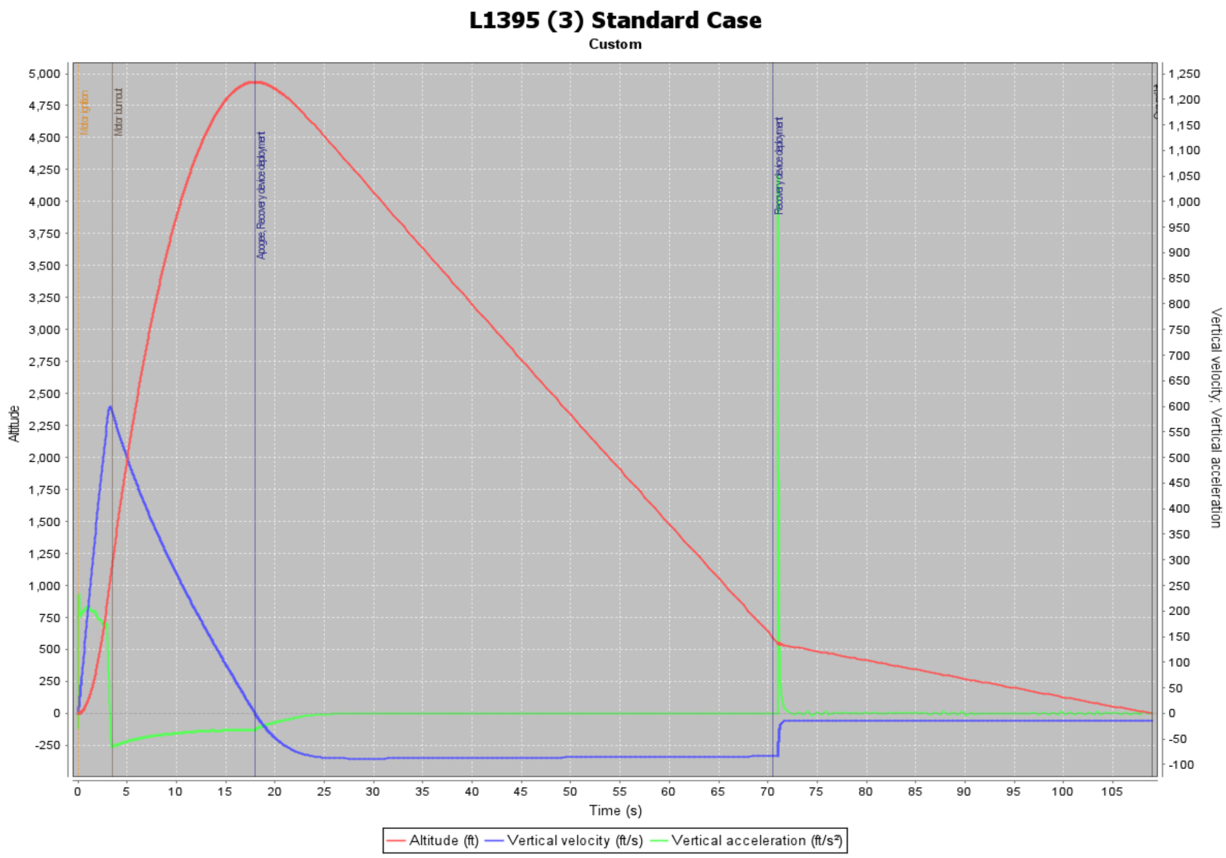


Figure 6-8 OpenRocket Descent Profile

Section	Section Mass (lb)	Kinetic Energy (ft-lbf)	Descent Time (sec)
Lower Section	18.24	58.78	86.9
Middle Section	7.16	23.07	
Upper Section	7.35	23.69	

Table 7 OpenRocket Descent Parameters

Although the descent parameters found using OpenRocket do not match those found with the custom simulator, the differ reasonably and in the expected direction. Further calculations will use the parameters in Table 6, since these are the most accurate values for the vehicle's descent.

With the calculated descent time, we were also able to calculate the drift of both the vehicle and payload in different wind conditions assuming the sections travel at constant velocities with the wind, as shown in Table 8.

Section	0 mph (ft)	5 mph (ft)	10 mph (ft)	15 mph (ft)	20 mph (ft)
Launch Vehicle	0	668.80	1337.60	2006.40	2675.19
Payload	0	665.86	1331.73	1997.60	2663.45

Table 8 Section Drift

6.3 Aerostructures

6.3.1 Nosecone

This rocket will utilize a 24 in, filament-wound fiberglass nosecone. The nosecone will be a 4:1 tangent ogive shape and will be purchased from Madcow Rocketry. As a result of COVID-19, we have experienced limitations on group work and access to manufacturing resources. So, purchasing from an online retailer is our best choice.

Figure 6-9 Vehicle Nosecone

Fiberglass was chosen as a material for its high strength and proven durability from launches in past years. The metal tip on the nose cone brings the center of mass closer to the front of the rocket, improving stability. This vehicle will travel at subsonic speeds, so a 4:1 ogive is the best option of those available online. Under the conditions in which the rocket will be launched, this shape provides the lowest drag from skin friction, and the reduced weight relative to a 5:1 ogive allows the rocket to reach a higher apogee.

The nosecone will incorporate a bulkhead attached through the coupler using radial brackets as discussed in section 6.4.1. The bulkhead will serve as the mounting point of the shock cord through a U-bolt. Additionally, 4 aluminum threaded standoffs will be attached to the bulkhead that will interface with the payload when the vehicle is packed, as shown in Figure 6-10. During flight, these pins will retain the payload laterally and rotationally, preventing damage from occurring to the airframe or the payload itself due to vibrations and accelerations during flight.

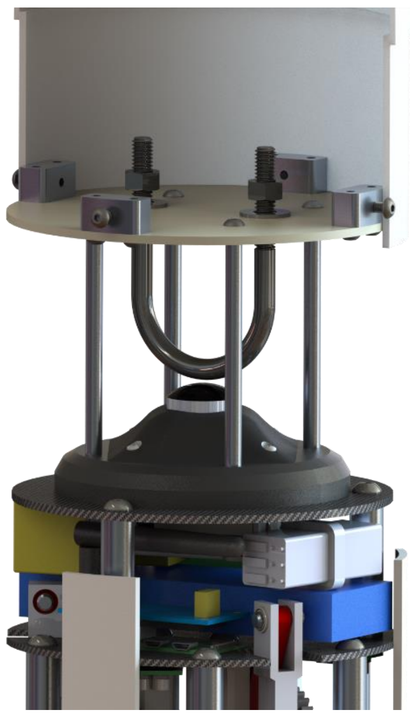


Figure 6-10 Payload Interface

6.3.2 Airframe

The airframe will be made of G12 Filament Wound Fiberglass which comes in readily available 30-inch tubes from Madcow Rocketry. It will be divided into three sections, the upper, middle, and lower. It has an inner diameter of 6 in, to allow for ample room to house the electronics bay, 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. The upper airframe will be 24 in long and houses the payload and the drogue parachute. The middle airframe will be 28 in long and houses the main parachute. The lower airframe is 30 in long and serves as an attachment point for the fin can and the motor retention system. Previously, the length of the lower airframe was 29.95 in, however, this was changed for ease of manufacturing. As previously stated, the airframes come in 30 in lengths from Madcow Rocketry, so rather than cut 0.05 in off the airframe we extended the length to 30 in. This specific length is desired so that the motor retention hardware will interface with the avionics bay.

Figure 6-11 Vehicle Lower Airframe

G12 filament wound fiberglass was chosen for its considerable strength while still maintaining a low weight. Also, it is beneficial to use fiberglass since it is transparent to radio waves and has high heat resistance. The criteria for material strength are based on NAR guidelines, where it states that the airframe of a rocket must be able to withstand forces 40-60 times the total weight of the rocket. On top of that, bolt shearing must be considered as well. In 2020, NC State University tested the same airframe we will be using this year for bolt shearing and published their findings in their Flight Readiness Review. Their results concluded that 4 #6 bolts held approximately 1500lb before failing, which proves that our airframes will not fail due to bolt shearing, as the maximum applied load will be 400 lbf at the thrust plate.

Figure 6-12 NC State Bolt Shearing Tests, Before and After

6.3.3 Fins

The materials for the fins of our launch vehicle this year will consist of a birch plywood core with a carbon fiber overlay on the exterior. Birch plywood was chosen due to its stiffness to weight ratio and hardness compared to the other materials. These properties are important in order to save weight where we can, and to avoid any crushing of the fins due to the relatively small attachment area afforded by the fin brackets. Carbon fiber was chosen for the exterior since it possesses the best stiffness to weight ratio, which will be useful for preventing a failure in the fins due to fluttering. Plain weave 3K T300 carbon fiber/epoxy lamina will be used for the skin, with the primary axes layered parallel to each fin's central axis. The fins will mostly experience bending loads due to flight, so a quasi-isotropic layup would be unnecessary and would require extra weight to reach the same stiffness offered by a $[90,0]_s$ layup. Similarly, since the fins are flat, the pliability of a twill weave would be unnecessary.

Figure 6-13 Vehicle Fin

The shape of the fins will be trapezoidal, with a root chord of 10 in, a tip chord of 3.5 in, a sweep length of 4.5 in, and a height of 7 in. Trapezoidal fins were chosen since they would be easy to manufacture, and because have favorable aerodynamic properties. These properties include a lower amount of induced drag, and a good amount of strength and stiffness at the root of the fin. The fins will be secured to the fin can using two bolts on through the fin brackets.

6.3.3.1 Fin Flutter

One possible failure the fins could experience is fluttering. Fin fluttering occurs when the rocket surpasses a certain velocity, and the rocket's fins will rapidly oscillate and increase their energy until they are destroyed. To ensure the fins do not fail due to fluttering, the maximum velocity of the launch vehicle must be lower than the flutter velocity. The following formula can be used to calculate flutter velocity.

$$V_f = a\sqrt{\frac{G}{\frac{1.337AR^3P(\lambda+1)}{2(AR+2)(\frac{t}{c})^3}}}$$

Equation 1 Fin Flutter Velocity

While most parameters in this equation are straightforward to calculate from the geometry of the fins or from the flight simulations, the shear modulus G is somewhat more complex. The equation given assumes that the fin material be isotropic, or that it has the same material properties in any direction. As a composite laminate, our fins do not meet that requirement. It is possible to use classical laminate theory with the expected material properties of each lamina to calculate the effective material properties for the laminate, though only the in-plane shear modulus G_{xy} can be easily found this way. Due to the nature of a composite sandwich material, the core dominates the in-plane shear modulus, resulting in a predicted G_{xy} of ~30000 psi, very close to that of pure plywood. We know that the stiffness of the laminate will be much greater than that of plywood alone due to the carbon fiber layers, however the values for G_{xz} and G_{yz}, which would better estimate the stiffness of the fin, are more difficult to calculate.

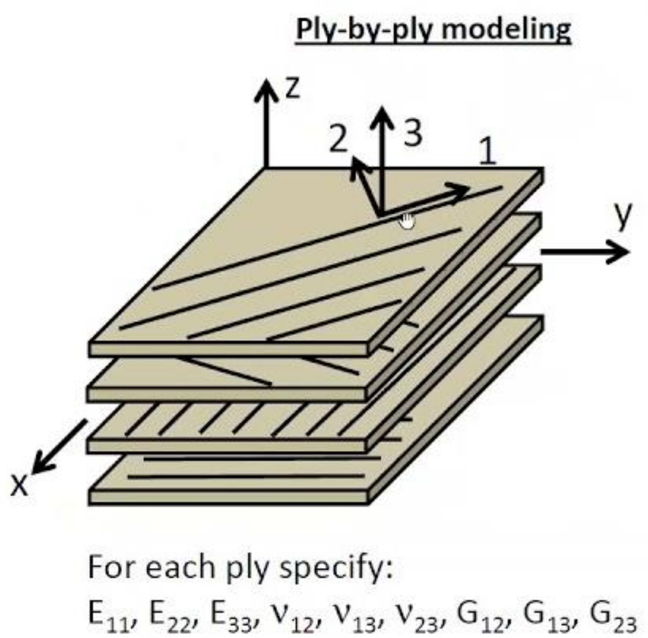


Figure 6-14 Laminate Modeling

Having constructed fins in a similar manner last year, we know the stiffness of the fin, and therefore the shear modulus, is actually dominated by the properties of the carbon fiber skin. We will take a conservative approach in estimating that the true shear modulus of the fin is just 25% of the 50 GPa of a single T300/epoxy lamina. This assumption gives us the shear modulus listed in Table 9.

V_f	Flutter Velocity	1027.6 ft/s
c_r	Root Chord	10 in
c_t	Tip Chord	3.5 in
b	Fin Height	7 in
t	Fin Thickness	0.15 in
G	Shear Modulus	1183000 psi
h	Altitude at Max Velocity	1130 ft
T	Atmospheric Temperature	54.98°F
P	Atmospheric Pressure	14.12 psi
a	Speed of Sound	1112.5 ft/s

S	Fin Surface Area	47.25 in^2
λ	Taper Ratio	0.35
AR	Aspect Ratio	1.037

Table 9 Fin Flutter Calculation Values

The flutter velocity of the fins is 1027.6 ft/s which is greater than the maximum velocity the launch vehicle achieves of 618 ft/s, with a safety factor of 1.66. The conservative estimation of shear modulus also adds to the effective safety factor, making the team confident no flutter will occur.

6.3.4 Tail Cone

The tail cone will be 3D printed using Polyethylene terephthalate (PETG) filament. This filament was chosen because it is strong and durable. These characteristics are important because it must be able to hold the weight of the rocket during assembly and handling and absorb the energy of the rocket upon impact.

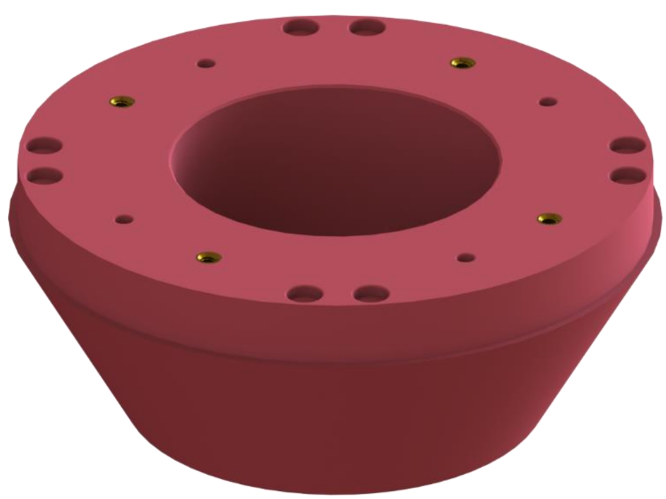


Figure 6-15 Tail Cone

The tail cone will be attached through the fin brackets using threaded inserts. This will make the tail cone smooth on the bottom. There will be a lip on the top, so the tail cone is flush and concentric with the airframe. The eight 11/32 in holes in the top of the tail cone provide clearance for screw heads that go through the fin ring. This way the tail cone can fit tight against the fin ring.

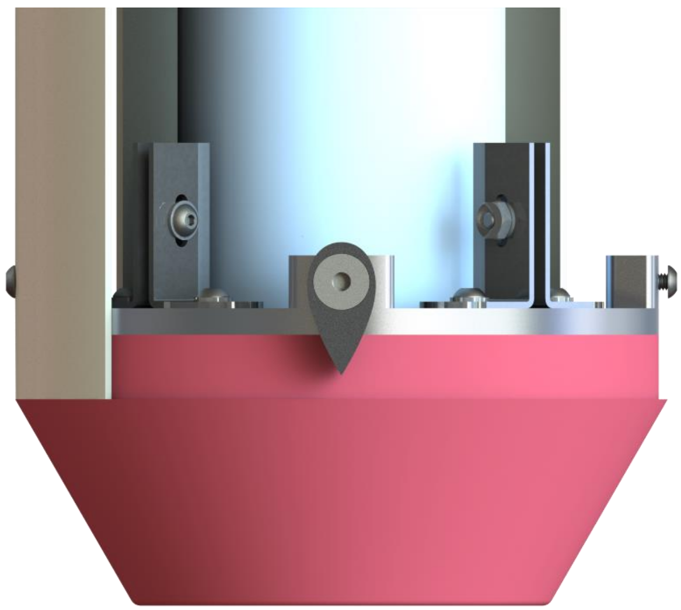


Figure 6-16 Tail Cone Attachment

The most important aspect of the tail cone is for it to withstand the impact of landing. The tail cone is reinforced by the retaining ring, so if it were to deform significantly, the motor casing would absorb the energy from impact. PETG is flexible enough to absorb the energy from impact without shattering, but also rigid enough to not deform during flight.

6.4 Propulsion Integration

6.4.1 Fin Can Design

The fin can is designed with the intent to ensure the fins remain secure during flight. The system consists of four fins, two centering rings, right-angle fin brackets, radial brackets, rail buttons, and a tail cone as shown in Figure 6-17. The fins will be secured by the right-angle fin brackets that fasten to the centering rings. Each centering ring will be attached to the airframe by four radial brackets.

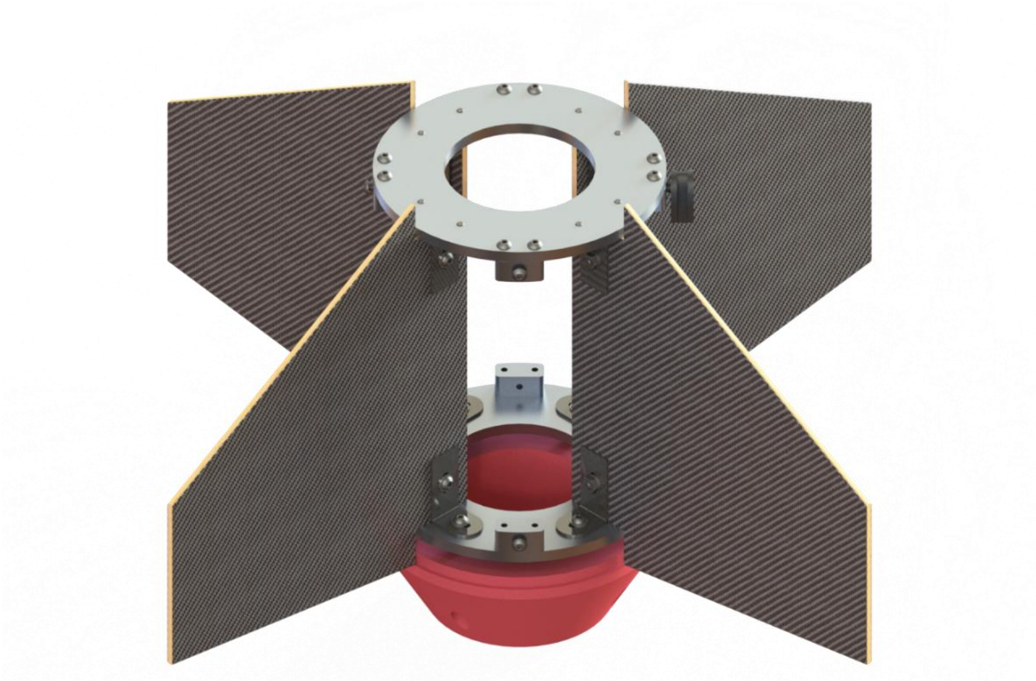


Figure 6-17 Fin Can Assembly

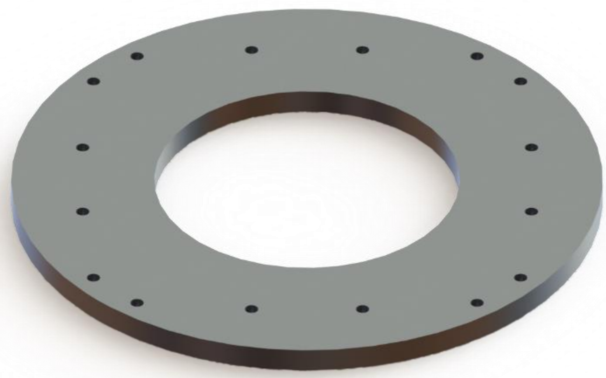


Figure 6-18 Centering Ring

The purpose of the centering ring is act as an intermediary component upon which the fin brackets and radial brackets are secured, while also centering the motor. There are two centering rings, one above and one below the fin tabs extending along the inside of the lower airframe body tube. Both centering rings will have clearance #8 holes for securing the radial brackets and tapped #8 holes for the fin brackets.

The centering rings will be manufactured from a six-inch square, 0.25-inch thick piece of 6061-T6 aluminum plate. The centering ring make use of a water-jet cutter to cut the holes, center bore and outer profile. The holes used to secure the right-angle fin brackets will then be hand tapped.

In the event that a water-jet cutter cannot be accessed, the centering ring will be machined using a CNC milling machine. An aluminum fixture plate, shown in Figure 6-19, would be made to secure the centering ring during the machining process. This fixture will also be used to manufacture the thrust plate. The CNC programs for both components and the fixture will be made using the computer aided manufacturing software, Esprit. The fixture will have sixteen #8-32 threaded holes, eight to secure the centering rings during manufacturing and eight to secure the thrust plate and will be faced flat on a milling machine. The raw aluminum centering ring material will be clamped to the fixture, held in a milling vice in a Haas mini mill, while the component's holes are drilled, and the holes used to attach the right angle brackets are tapped. At a program stop, the part will be screwed to the fixture, the clamps will be removed, and the program will finish by contouring and pocketing the centering ring.

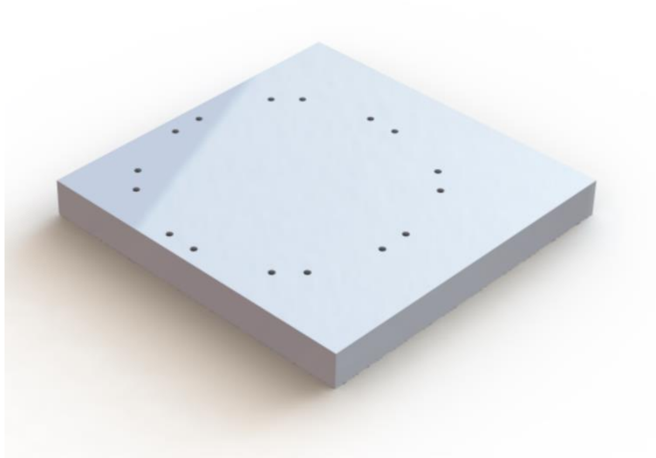


Figure 6-19 Fixture for Centering Rings

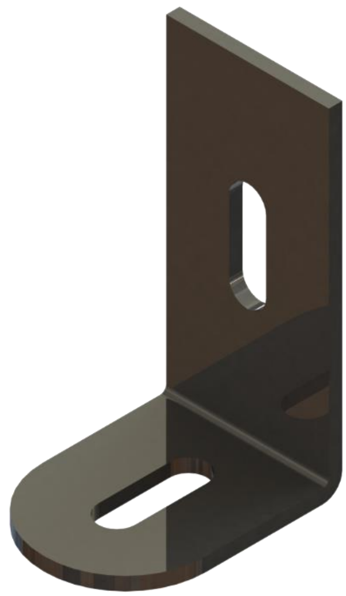


Figure 6-20 Right-angle Fin Bracket

The right-angle brackets will be fastened to both the centering ring and the fins with #8 bolts, securing the fins in place. Some important qualities considered when designing the right-angle brackets included a large surface area for more stability, and slots where the bracket interfaces with the centering ring and the fin. These slots are necessary to increase the level of allowable tolerance, compensating for errors in the tolerances resulting from the cutting and bending the brackets.

These right-angle brackets will be manufactured out of 1/16 inch sheet metal. The primary approach to manufacturing this part would be water jet. The outer perimeter of the right-angle bracket will be cut along with the 2 inner slots. If a water jet is not available for use, our secondary approach would be to outsource this part to a sponsor with access to a water jet. Regardless of the method, we will be using a bending break to create the 90-degree angles in the right-angle bracket between the 2 inner slots.

Step 1: Cut the sheet metal. (Possible Options: water jet, outsource to sponsor)

Step 2: Bend part to 90 degrees using bending break.

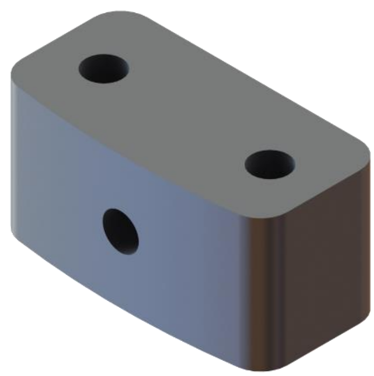


Figure 6-21 Radial Bracket

The primary function of each radial bracket is to act as a fastener securing the two centering rings and the thrust plate to the airframe. Four radial brackets are positioned on each centering ring and on the thrust plate. The radial brackets are secured to the ring or plate by two #8-32 bolts in vertical holes and connected to the airframe by a #8-32 bolt for most brackets through a single horizontal hole. The radial brackets securing the rail buttons are held by ¼-20 bolts. The brackets have a curved profile on the side interfacing with the body tube to ensure that the piece will fit flush with the inside of the airframe.

Our radial brackets will be manufactured using a custom pallet system. These brackets are necessary at numerous locations throughout the rocket, emphasizing the importance of a rapid and efficient manufacturing process.

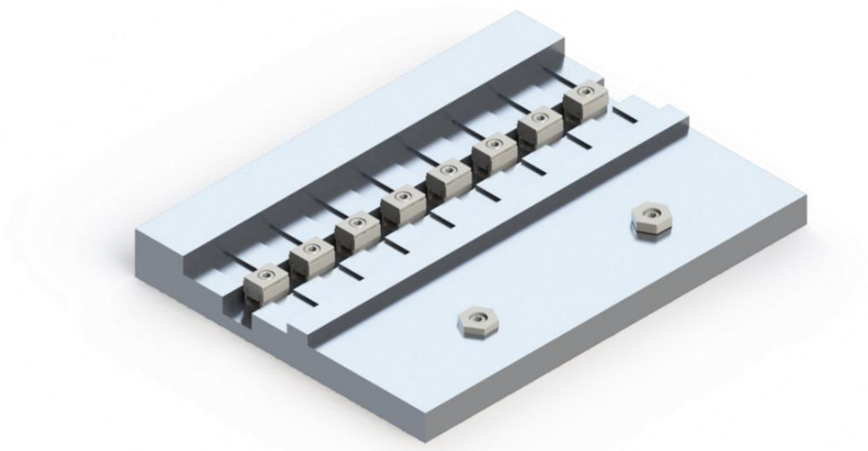


Figure 6-22 Pallet Fixture for Radial Brackets

Step 1: Facing operation, Vertical hole x2 and tap, Contouring operation (1st half)
Step 2: Contouring operation (2nd half)
Step 3: Drill and tap radial hole

The pallet system will be machined out of 6061-T6 Aluminum stock and consist of 3 separate sections or "steps". This pallet will include several clamps to hold the stock in place and pockets for the stock in the second and third operation to self-align. Radial brackets will be made from 6061-T6 Aluminum bar stock which will be cut to length to fit in the pallet. The bar stock from the first operation will be held in place by two low-profile fixture clamps, while the stock in the second and third operation will be secured by dual clamping fixture clamps. In preparation for rerunning the milling machine, the stock from the first operation will be rotate 180 degrees along the axis of the length of the bar and placed into the second operation. Then the stock from the second operation, where the radial brackets should be separate parts, will be rotated such that the curved surface will face upwards and place into the third operation. The use of this pallet system will allow us to run multiple operations in sequence and ultimately create eight finished radial brackets at a time.

Of the 24 radial brackets located throughout the rocket, some are designed with certain specifications to fit different sizing restraints and functionality. In the motor retention system, the curved surface of four radial brackets has a slightly smaller radius in order to fit against the coupler. Another two radial brackets that secure the centering rings to the body tube have a ¼-20 tap in the radial hole to accompany the two rail buttons. These two design modifications will be taken into consideration when creating the CAM for the radial brackets.

6.4.2 Motor Retention Design

The motor retention design consists of the thrust plate, the centering rings, and the motor tube. The goal of the assembly is to keep the motor centered and from moving out of place, while maintaining a lightweight design. The thrust plate also distributes the thrust to the airframe, where it is attached with four radial brackets and screws. Both the thrust plate and centering rings are made of 6061-T6 aluminum.

Figure 6-23 Motor Retention System

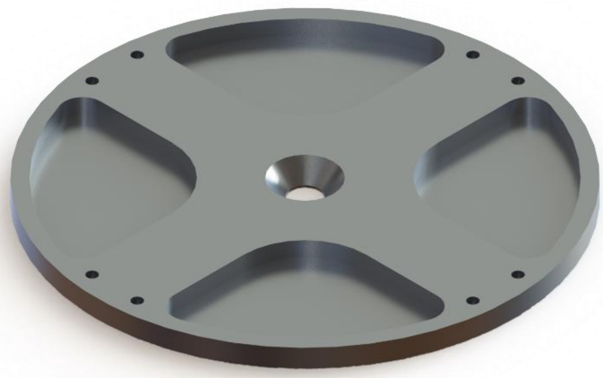


Figure 6-24 Thrust Plate

The purpose of the thrust plate assembly is to evenly transfer the thrust exerted by the motor to the airframe. The thrust plate is attached to the airframe via four radial brackets and secured 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 electronics bay from the motor. The intent of this feature is to help block heat emitted by the motor, which could be potentially dangerous to the rocket's operation due to close proximity to sensitive electronics. Minimizing material and creating a design with a minimum safety factor of 3 was considered in the design process and verification steps.

The thrust plate will be machined from a 6061-T6 aluminum plate, similarly to the fin rings, using the same fixture. The material will be clamped to the fixture while the through holes are drilled and the center hole is countersunk, so it can be secured to the fixture using eight #8-32 screws at a program stop and the clamps can be removed. The shape can then be contoured, and the pockets will be cut.

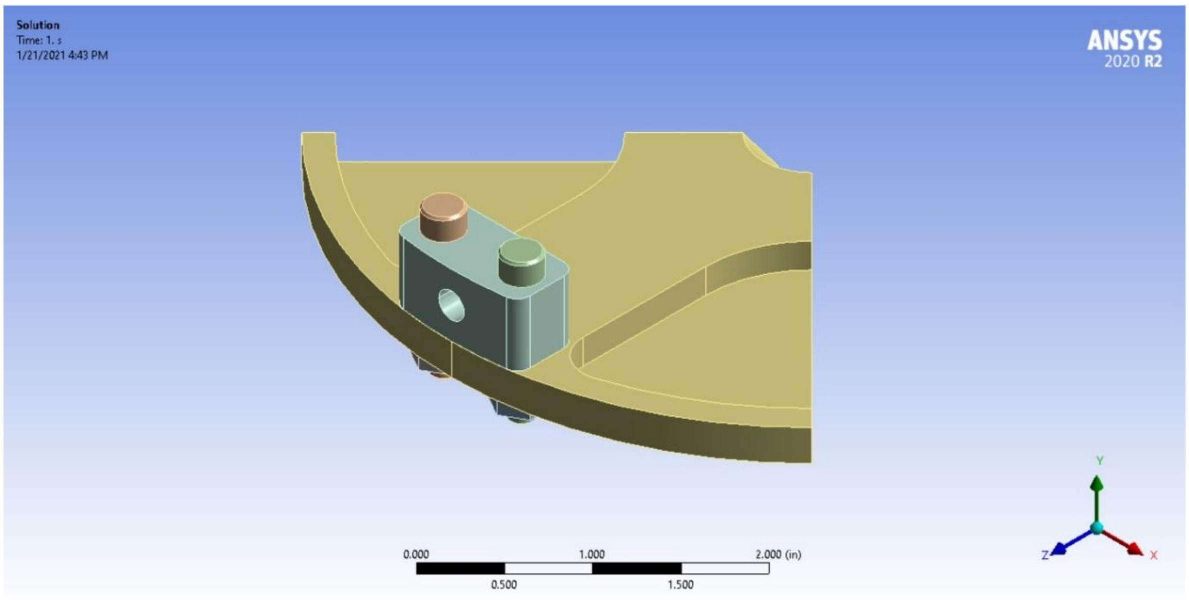


Figure 6-25 Thrust Plate Simulation Model Set Up

To verify whether the thrust plate is capable of withstanding thrust from the primary motor, a static structural analysis of the thrust plate and radial bracket assembly was conducted using ANSYS simulation software. Locations of high displacement, high von mises stresses, and minimum safety factory were solved for to verify the reliability of the thrust plate. Utilizing symmetry tools in ANSYS, we analyzed a quarter section of the thrust plate to reduce computing time and allow for greater mesh refinement.

Our primary motor, the L1395, has a peak thrust of approximately 400 lbf. Thus, a 100 lbf load was applied over the area where the motor casing interfaces with the quarter section of the thrust plate modeled in the analysis, as shown in model set-up in Figure 6-25. The radial bracket's single horizontal hole was set as fixed. A bolt pretension of 700 lbf, determined using Futek Advanced Sensor Technology's bolt torque calculator, was applied to each vertical bolt connecting the thrust plate to the radial bracket [1]. The pretension can be calculated as shown in Equation 2, where F is the preload tension force, A_t is the tensile shear area of the bolt, and S_p is the proof load of the bolt.

$$F = 0.75 \times A_t \times S_p$$

Equation 2 Required Bolt Preload

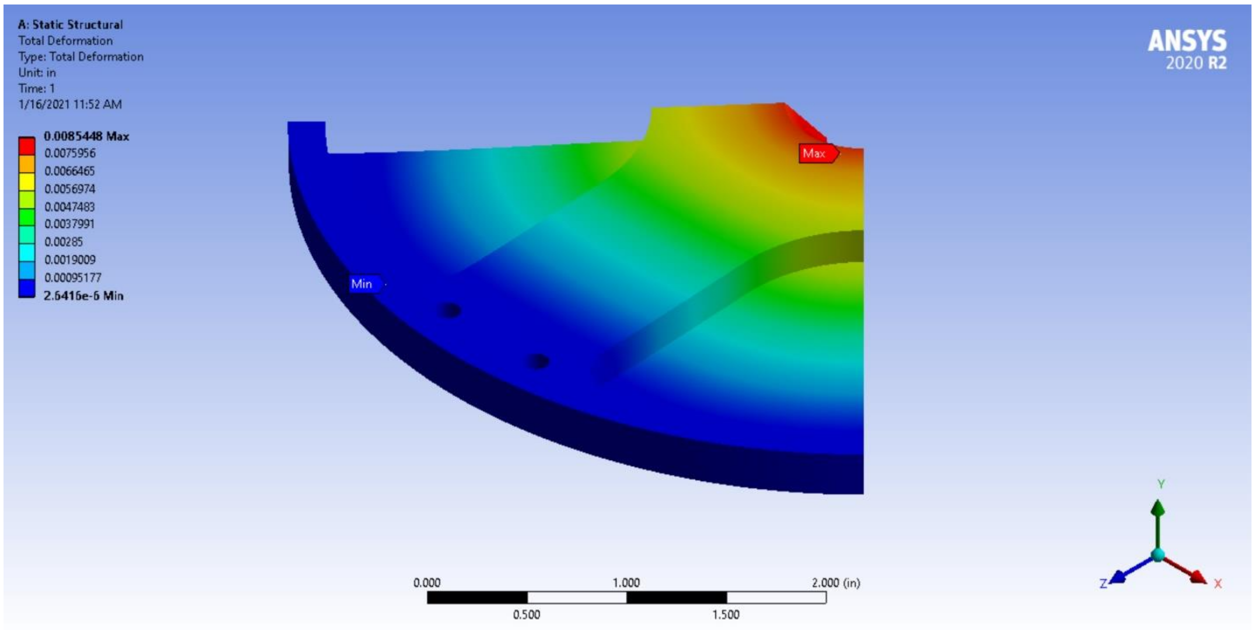


Figure 6-26 Thrust Plate Deformation Plot

The deformation plot of the thrust plate depicts a maximum displacement value of 0.0085 inches, located where the thrust plate bolts down to the motor casing, as shown in Figure 6-26. The minimum displacement is located around the outer edge of the thrust plate where it bolts into the radial bracket and into the airframe. The maximum displacement value is small enough to be negligible, suggesting that the thrust place will not deform significantly or lead to component failure during launch.

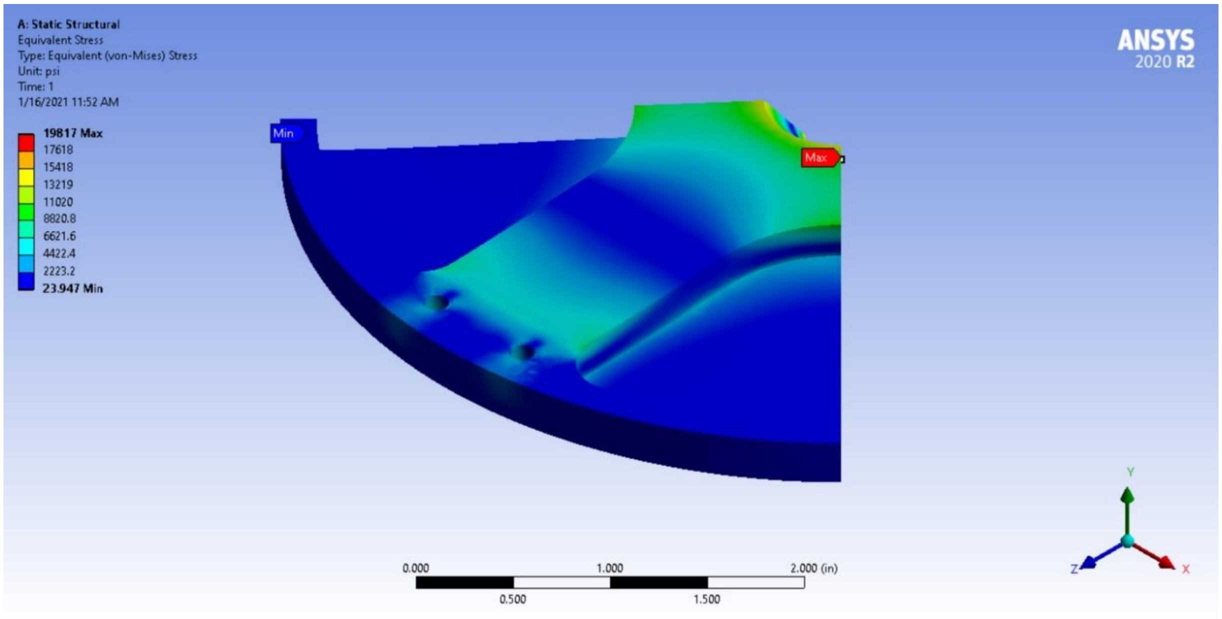
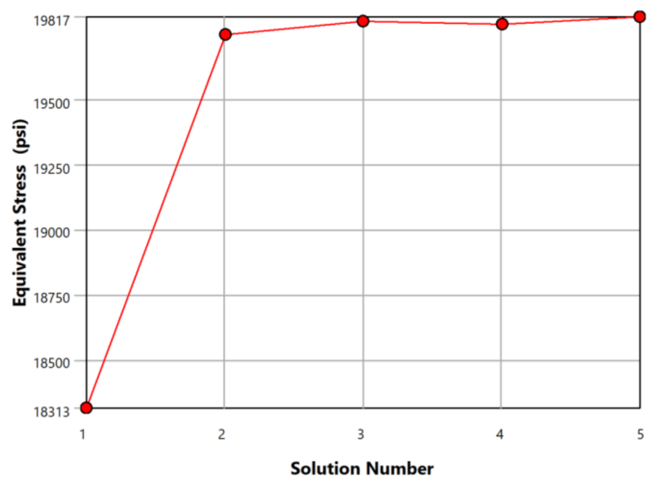


Figure 6-27 Thrust Plate Equivalent Stress Plot



	Equivalent Stress (psi)	Change (%)	Nodes	Elements
1	18313		5166	2188
2	19746	7.5264	23533	13385
3	19799	0.27116	96310	61436
4	19786	-6.6631e-002	226044	151041
5	19817	0.15615	348557	236595

Figure 6-28 Thrust Plate Equivalent Stress Convergence Plot

An adaptive convergence plot was created for the equivalent stress of the thrust plate to expose possible singularities in the simulation by running five iterations of mesh refinement. Referring to Figure 6-28, the equivalent stress rises after the first iteration, but then flattens out and converges to approximately 19,817 psi. The convergence plot verifies that the calculated results are mesh independent, and that no singularities exist in the model.

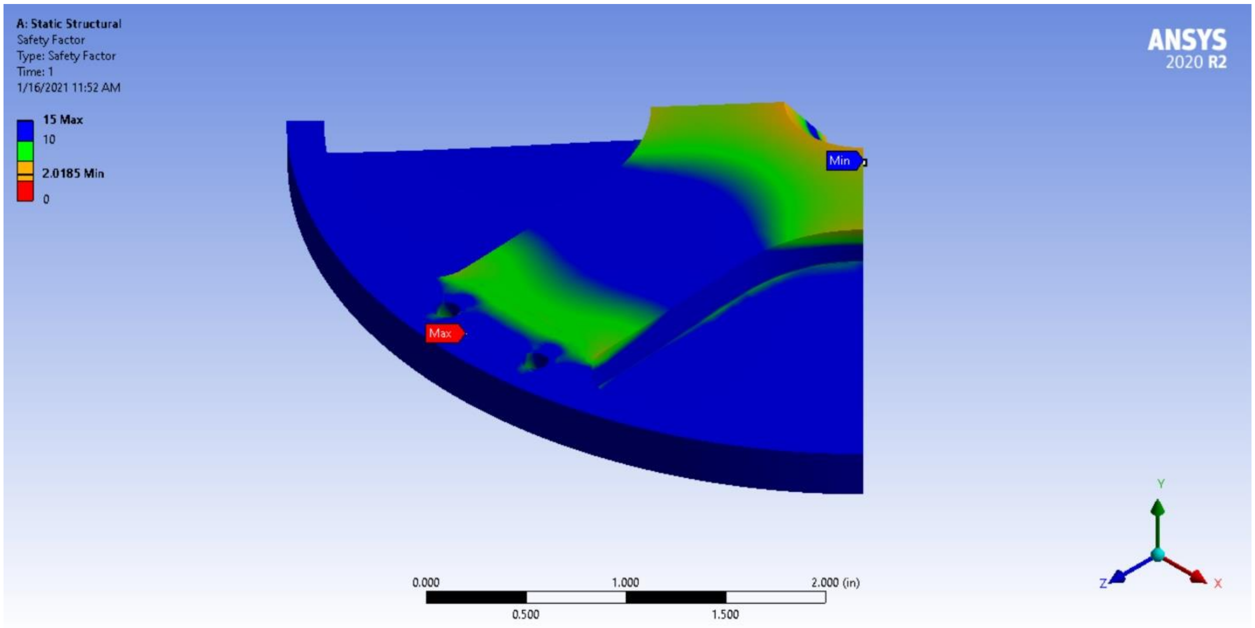


Figure 6-29 Thrust Plate Safety Factor Plot

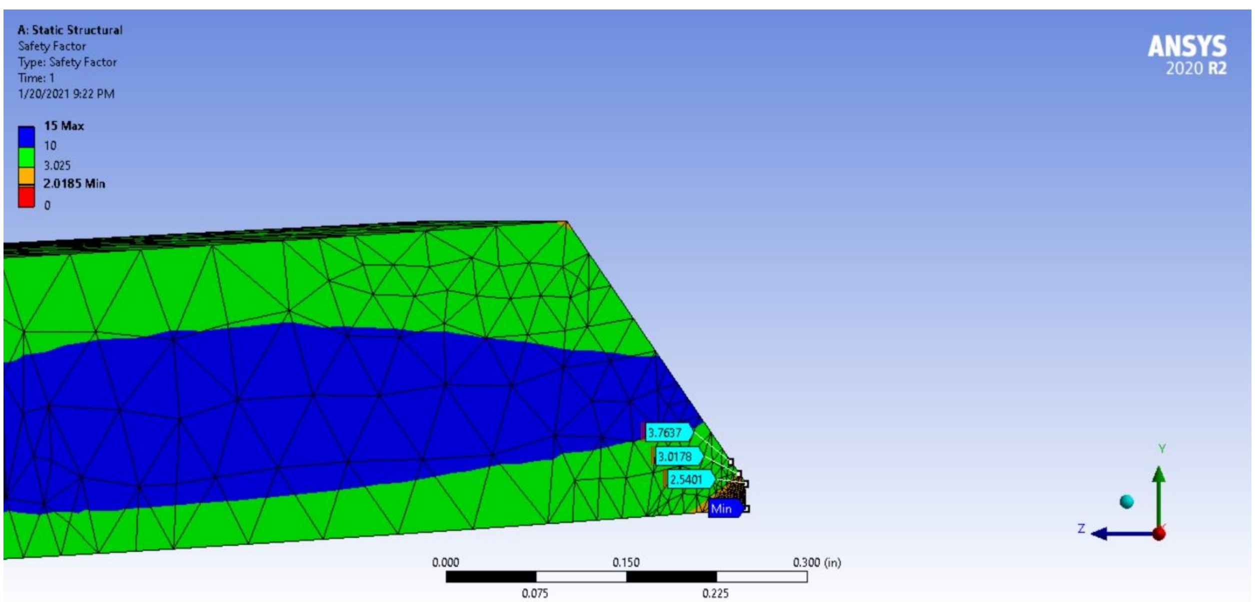


Figure 6-30 Thrust Plate Safety Factor Plot with Probe Detail

The safety factor plot of the thrust plate in Figure 6-29 exhibits a minimum safety factor of 2.02 located where the thrust plate bolts down to the motor casing. However, in the ANSYS model a clamping force of the center bolt that screws the plate down to the motor casing was not modeled. The clamping force of the bolt would keep the thin region from deforming and thus, keep that region above a safety factor of 3.

Due to the convergence of the equivalence stress plot, negligible deformation at central hole and proper location of stress concentrations, we can validate the integrity of the simulation. In addition, we clarified anomalies in the safety factor plot, verifying that the overall thrust plate should have a safety factor above 3, the minimum value that we deemed appropriate for this application. The thrust plate will reliably perform under high loads, while minimizing the amount of unnecessary geometry. Lastly, the results of the static structural analysis, particularly validation from the safety factor plot in Figure 6-30, confirm that the motor retention system will not surpass the yield strength of the thrust plate during the duration of the vehicle's ascent.

To further validate the suitability of the motor retention, the radial brackets and bolts attaching them to the airframe were analyzed using analytical methods. Three failure modes were analyzed: tear out failure, bearing failure, and bolt shear failure. In tear out failure, the material shears along two planes from the bolt hole to the edge of the material. In bearing failure, the bolt hole deforms significantly to the point where the bolt is no longer held securely. Bolt shear failure is when the shear stress in the bolt itself exceed the maximum allowable, and the bolt breaks parallel to the bolting surface.

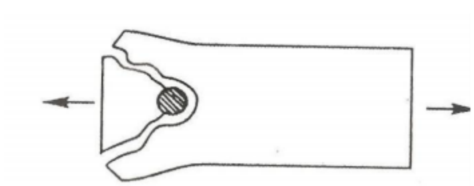


Figure 6-31 Tear out Failure of Plate

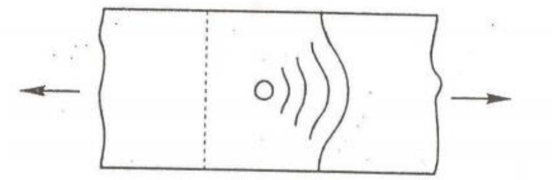


Figure 6-32 Bearing Failure of Plate

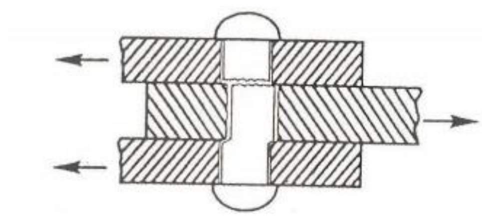


Figure 6-33 Bolt Shear Failure

Tear out failure is predicted by assuming the shear two shear planes to have a length equal to the shortest distance between the bolt hole edge and material edge, a conservative assumption since tear out planes will typically form at angles ~35-45° from the loading direction, increasing the shear area. The shear load that would result in tear out is calculated using Equation 3, with S_{su} as the ultimate shear strength of the material, L_{sp} as the length of one shear plane, and t as the thickness of the material, taken to be the bolt's engagement length.

$$P_{su} = S_{su}A_{su} = S_{su}(2L_{sp}t)$$

Equation 3 Ultimate Tear Out Load

Bearing failure is predicted using the ultimate bearing stress of the material, with the bearing area taken conservatively to be ¼ of the circumference of the bolt, multiplied by the thread engagement. The ultimate tear out load can be found using Equation 4, with S_{bru} as the ultimate bearing strength of the material, and A_{br} as the bearing area.

$$P_{bru} = S_{bru}A_{br}$$

Equation 4 Ultimate Bearing Failure Load

Bolt shear failure can be predicted in a similar manner, finding the minimum cross sectional area of the bolt using the thread minor diameter, and using the shear yield strength of the bolt material in Equation 5.

$$P_s = S_{sb} A_{sb}$$

Equation 5 Bolt Shear Failure Load

Material properties for the radial bracket and the bolt are shown in Table 10.

Property	Value
$\boldsymbol{S_{su}}$ (6061-T6)	26000 psi
$\boldsymbol{A_{su}}$	0.126 in^2
$\boldsymbol{S_{bru}}$ (6061-T6)	56000 psi
$\boldsymbol{A_{br}}$	0.048 in^2
$\boldsymbol{S_{sb}}$ (Alloy Steel)	87000 psi
$\boldsymbol{A_{sb}}$	0.013 in^2

Table 10 Radial Bracket Analysis Material Properties

Inputting these values to the given equations, with an actual shear load of 400 lbf from the motor, or 100 lbf for each bolt, the factors of safety for tear-out failure, bearing failure, and bolt shear failure were 32.8, 27.1, and 11.0, respectively. With high factors of safety in all three analyses, the #8-32 bolts, and the 6061-T6 radial brackets will be able to withstand the 400 lb force without any deformation or damage.

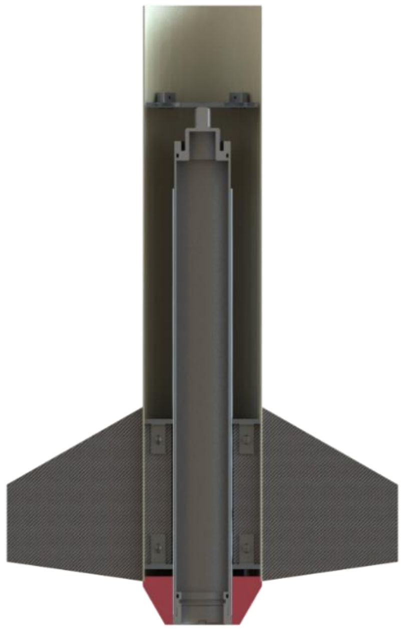


Figure 6-34 Cross Section of Lower Airframe Assembly

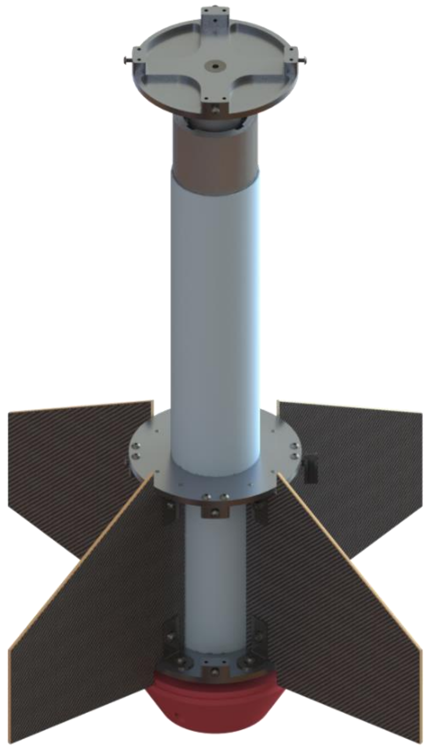


Figure 6-35 Lower Airframe Assembly

6.5 Recovery

6.5.1 Recovery Bay

The recovery bay is located within the 8 in coupler between the upper and middle airframes. It houses the recovery electronics for the launch vehicle, including the primary and backup altimeters, batteries, switches, and ejection charge wells. The recovery system has a dual bay design, with the drogue parachute deployed from the upper airframe and the main parachute deployed from the middle airframe.

Figure 6-36 Recovery Bay

The drogue and main parachutes are secured by shock cord to the 0.3125 in U-bolts on the forward and aft bulkheads. The U-bolts are connected directly to adapter plates, located just beneath the bulkheads, which are connected to the central spine with a single bolt on either end. The parachute opening shock flows directly through the adapter and spine, resulting in minimal loads on the bulkheads. The bulkheads will be water jetted from 1/8 in thick fiberglass plates and secured to the adapters with thumb screws for ease of access at the launch site. The electrical terminals and ejection charge wells for the drogue and main parachutes are mounted to the forward and aft bulkheads, respectively. The aft bulkhead is also secured to the coupler and middle airframe with four radial brackets and bolts. The adapters will be water jetted from ¼ in thick aluminum. The adapters have a 2D profile so that they will be more easily manufacturable and stronger at their connections to the spine.

Figure 6-37 Recovery Bay Upper Bulkhead

The central spine has a 1/2 in hex profile in order to hold the electronics sled in place and lock its rotation. Each end of the spine is tapped, and the spine bolts to the adapters using ¼-20 socket head screws.

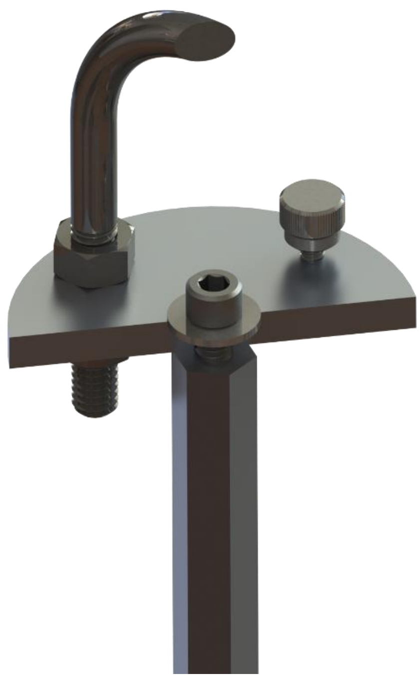


Figure 6-38 Recovery Bay Spine Attachment

The contents of the recovery bay will be easily accessible at the launch site since the electronics sled can be pulled out once the forward bulkhead and adapter are removed. The forward side of the recovery bay coupler will be secured to the upper airframe with up to four 4-40 shear pins. The recovery bay coupler also has four 3/16 in holes so that the barometers can detect the pressure changes during flight and the parachutes will be deployed at the correct altitudes. The holes will be spaced at 90° to cancel out the effect of angle of attack on the measured pressure. The vent hole sizes were found using Equation 6, with D_t as the coupler diameter, L_t as the coupler length, and n as the number of holes. The equation assumes a rule of thumb of one ¼ in hole for each 100 in^3 of airframe volume.

$$D = 0.02216 * D_t * \sqrt{\frac{L_t}{n}}$$

Equation 6 Vent Hole Diameter

The 3D printed electronics sled, which has mounts for the altimeters, batteries, and switches, has a hex-shaped hole down the center so that it can easily slide onto the hex spine. The sled is split into two parts in order to reduce the need for supports during 3D printing. The two sections fit between the adapters and are also held together with glue which we deemed adequate since there are no loads going through the sled. The primary and backup altimeters are held in place with bolts and standoffs, so that there is enough room underneath for the pressure sensors. The walls of the battery mount include slots for zip ties in order to secure the batteries onto the sled and also provide proper ventilation. The sled also has an arm extending out to the switch band, onto which two 3D printed angle brackets will be mounted to provide external access to the switches.

Figure 6-39 Recovery Bay Sled

6.5.2 Verification

Since main deployment will result in a significant opening shock due to the high descent rate and large parachute area, we calculated the opening force to verify that the parts in the recovery bay would be able to withstand both deployment events. Since the vehicle will slow down as the parachute is inflating, the deployment will result in a finite mass loading. Such a case requires a numerical simulation to predict the opening shock of the parachute, and knowledge of the inflation time and the area of the parachute as a function of that time.

In order to predict the parachute inflation time, as well as the area of the parachute as it is inflating, we used equations from T.W. Knacke's Parachute Recovery Systems: Design Manual [2] that relate these unknowns to parachute diameter (D), area (A), packed area (A_p) , vehicle velocity (v), and canopy fill constant (n), which is dependent on the type of parachute.

$$t_{inf} = \frac{n * D}{v}$$

Equation 7 Inflation Time

$$A^{parachute}(t) = A * (1 - \frac{A_p}{A} * \frac{t^3}{t_{inf}} + \frac{A_p}{A})^2$$

Equation 8 Inflation Area

Though the canopy fill constant for each type of parachute is usually determined experimentally, we were unable to find any values listed for annular parachutes similar to that of the main parachute. In addition, the main parachute utilized a reefing ring to increase the inflation time and reduce the opening shock load, which usually doubles the canopy fill constant of a given parachute. We used a conservative estimate of 4 after taking the reefing ring into account, resulting in an opening time of 0.51 seconds. At main deployment, the maximum acceleration experienced by the launch vehicle was calculated to be 10.9G, resulting in a maximum opening force of 413.94 lbf at the main parachute swivel.

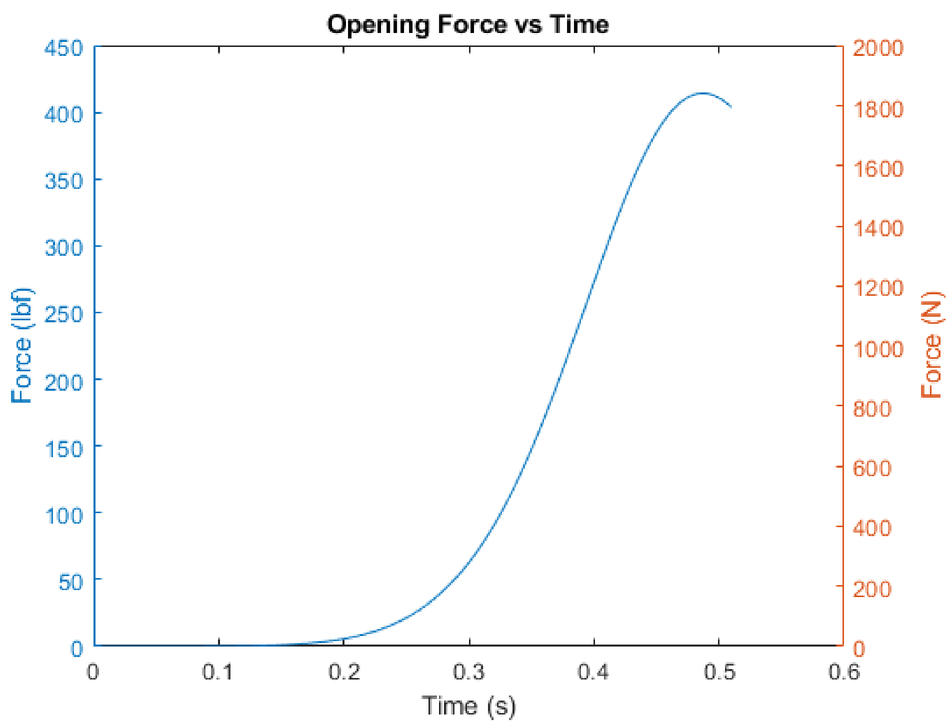


Figure 6-40 Main Parachute Opening Force

To determine the structural loads on the vehicle bulkheads, one can multiply the maximum acceleration by the mass supported by the bulkhead. These values are used for the structural analysis of the recovery bay hardware to verify that they can withstand the parachute opening shock loads. During main deployment, the maximum force at the U-bolt attaching the main parachute to the recovery bay will be 214.32 lbf.

Bulkhead Location	Opening Shock Load (lbf)
Avionics Bay	198.86
Recovery Bay Rear	214.32
Recovery Bay Front	105.94
Nosecone	46.20

Table 11 Bulkhead Shock Loads

The U-bolts used to connect the recovery hardware are rated to withstand a load of up to 600 lbf, providing a safety factor of 2.8 at the recovery bay attachment. From the U-bolt, the forces will travel through the adapter and the central spine, as well as the bolts connecting these.

The tensile stress in the spine can be calculated using Equation 6, with F equal to 214.32 lbf, and A equal to .22 in^2.

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

Equation 9 Stress Calculation

The tensile stress in the spine is 974.2 psi, and with the yield stress of 40000 psi the spine has a safety factor for tensile failure of 41. Using Equation 10 to determine the shear area for the internal thread of the spine, the shear stress at the bolted connection can be determined, using the threads per inch (n), the maximum pitch diameter of the internal thread $\left(E_{n_{max}}\right)$, the minimum pitch diameter of the external thread $\left(D_{s_{min}}\right)$, and the thread engagement (L_e).

$$A_n = \pi n L_e D_{s_{min}} \left(\frac{1}{2n} + 0.57735\left(D_{s_{min}} - E_{n_{max}}\right) \right)$$

Equation 10 Internal Thread Shear Area

With a class 2A and 2B ¼-20 thread, the factor of safety for bolt pull out is 34.5. Though the spine has a high safety factor for both these cases, to ensure the spine does not deform in the area around the bolt hole the thickness of the spine will remain at ½ in.

To verify the integrity of the adapter, a linear static FEA analysis was conducted using SOLIDWORKS Simulation. The 214.32 lbf load was applied to the U-bolt mounting holes, with a fixed face in the center of the bulkhead aligning with the central washer.

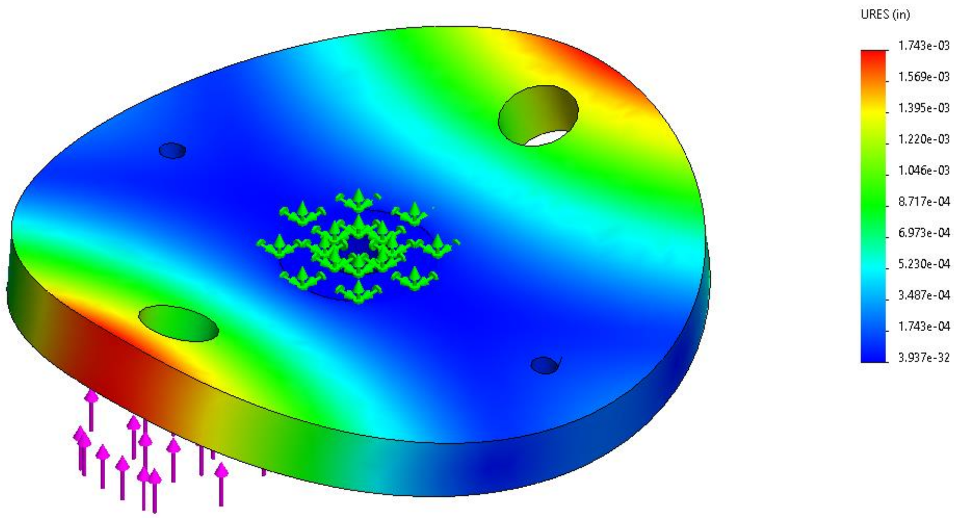


Figure 6-41 Adapter Displacement

The FEA solver did not converge after 5 iterations, though this is due to a singularity at the edge of the fixed face. The maximum displacement of the adapter is .0017 in, and the maximum von Mises stress 3 elements away from the singularity was found to be 10400 psi. With the adapter material as 6061-T6 aluminium, this results in a safety factor for the part of 3.85, verifying the ability of the part to withstand the shock loads.

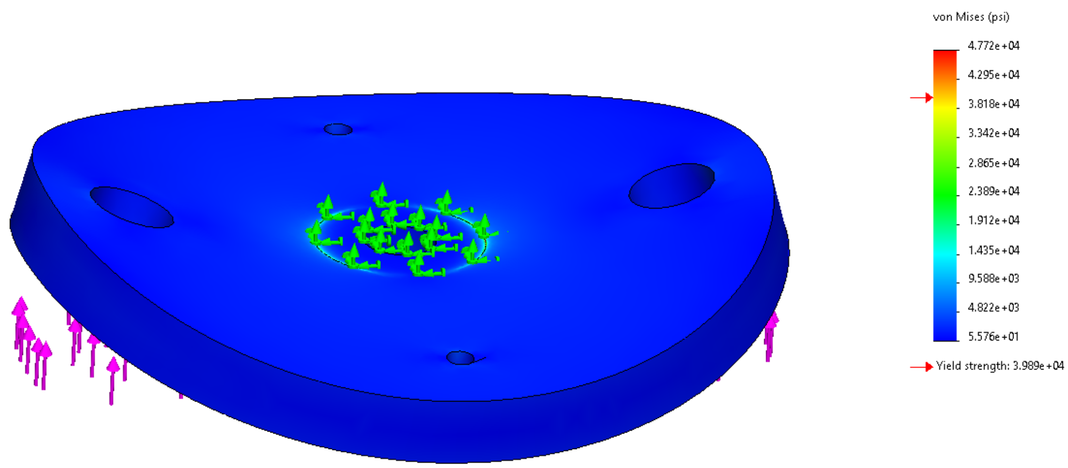


Figure 6-42 Adapter von Mises Stress

6.5.3 Recovery Electronics

The recovery bay electronics sled houses the primary and backup altimeters, batteries, and switches. The primary altimeter is a StratoLoggerCF (SLCF) and the backup altimeter is the Featherweight Raven 4 (Raven 4). Having two altimeters from different manufacturers helps reduce the chance of failure in both altimeters. The electronics diagram for the primary and backup altimeters are shown in Figure 6-43.

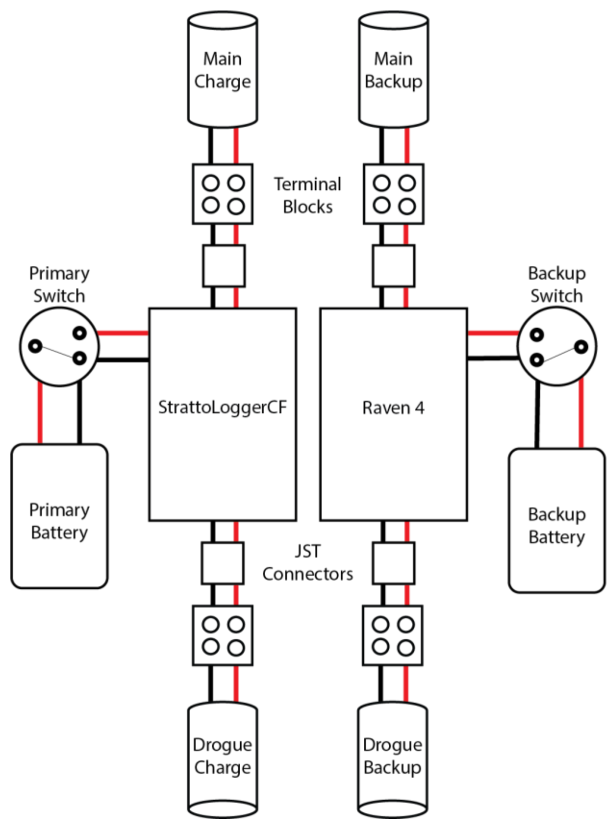


Figure 6-43 Recovery Bay Wiring Diagram

The SLCF itself is both simple and cost-effective, containing two outputs which satisfy our purposes of deploying the drogue and main parachutes in a dual event recovery. It is also accurate, able to deploy the main parachute at altitudes adjustable by one-foot increments between 100 and 9,999 ft, in addition to a barometric precision sensor and 24-bit ADC reported to yield 0.1% accuracy. Thus, we can accurately set the main deployment of our rocket at our altitude target of 600 feet. It is also compatible with a dual altimeter setup with its selectable apogee delay feature, ensuring that there is no overpressure due to simultaneous charge firing in our redundant system. Finally, its current output capable of 5 A for a full one second makes it compatible with almost any e-match for our recovery events.

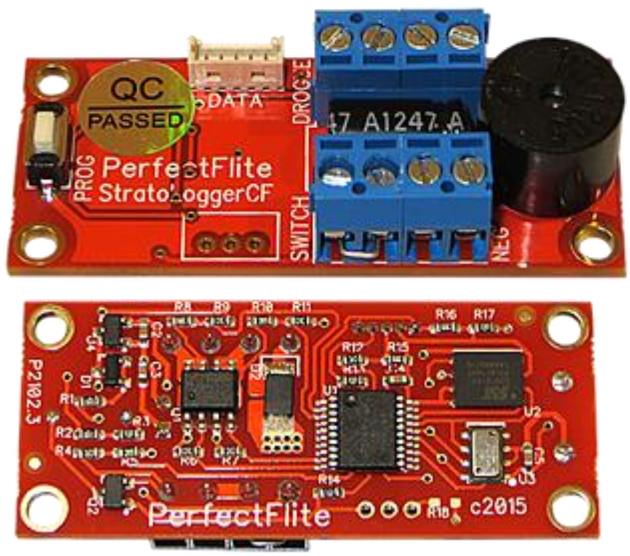


Figure 6-44 StratoLoggerCF

Altitude Resolution	1 ft up to 38,000 ft MSL; < 2 ft to 52,000 ft MSL; < 5 ft to 72,000 ft MSL
Analog to Digital (ADC)	24-bit Sigma Delta
ADC Calibration Accuracy	± 0.05%
ADC Measurement Precision	± (0.01% reading + 1 ft)
Pyro Outputs	2
Max Output Amperage	5 A
Dimensions	2.0 x 0.84 x 0.5 inches
Mass	10.8 grams
Power	4 V - 16 V, Nominal 9 V Battery
Other Recorded Measurements	Temperature, Battery Voltage

Table 12 StratoLoggerCF Specifications

While the SLCF is effective for our purposes, the Featherweight Raven 4 has a different system that further implements redundancy. A prominent feature contained by the Raven 4, which the SLCF lacks, is an accelerometer capable of ±105 G measurements. This paired with a barometer collecting 20 Hz barometric data at ±0.3% accuracy and a 20 Hz high-precision temperature sensor, allows for accurate altitude determination. The Raven is capable of detecting apogee using only the accelerometer. Thus, proper deployment of the drogue and main parachutes, can occur if the recovery bay fails to equalize pressure with the environment, or if the primary altimeter otherwise fails. The use of a different altimeter as our backup also prevents common software or hardware errors from affecting both redundant altimeters, further increasing safety. The Raven 4 has four pyro outputs, each capable of 25 A of current and are compatible with lithium polymer (LiPo) batteries, which our recovery system has utilized over the past few years and continues to do so.

Figure 6-45 Raven 4

Axial Accelerometer Range & Frequency	400 Hz ± 105 Gs
Lateral Accelerometer Range & Frequency	200 Hz ± 105 Gs
Barometer Range	100000 ft
Download Interface	USB Micro
Pyro Outputs	4
Max Output Amperage	22 A
Dimensions	0.79 x 1.77 x 0.5 inches
Mass	6.6 grams
Power	3.8 V DC - 16 V DC
Other Recorded Measurements	Temperature, Event Logic, 4 Battery Voltages

Table 13 Raven 4 Specifications

We have relied on LiPo batteries as they are compact and much better at handling launch forces than alternative battery options. They are also more resistant to the effects of cold temperatures, an important quality considering we conduct our test launches in the Northeast winter months. The SLCF uses a 370 mAh 2S battery, and the Raven 4 uses a 300 mAh 1S battery.

Figure 6-46 StratoLogger CF Battery

The arming 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 110/220 V selector power switch that will be used to switch the mode of the altimeters to safe or armed–the former being especially important during handling stages of the rocket before being placed on the launch pad and pointed in a safe direction. The switches can easily be turned by a flathead screwdriver, allowing for easy arming, and disarming, without the need for specialized tools. With the switches armed, our redundant recovery system will be capable of igniting the two black powder charges placed on both the forward and aft bulkheads of our recovery bay for drogue and main parachute deployment, respectively. Nomex blankets will be used to protect the main parachute from the charges, and the drogue parachute will be shielded by the ejection piston.

Figure 6-47 Arming Switch

In order to make the electronic sled more accessible, locking JST connectors will connect the charge wires from the altimeters to the bulkheads. This will make it easier to completely disconnect the electronics from the charge wells and remove the bulkheads so that the electronics sled can be pulled out of the recovery bay. On the bulkhead the wires will terminate in a terminal block for easy connection to the E-matches.

Figure 6-48 JST Connector

6.5.4 Parachute Selection

The recovery system is a dual deployment system. The first parachute, the drogue, is used to slow down the launch vehicle immediately after reaching the flight's apogee while also minimizing the descent time. It will be located above the recovery bay and ejection piston in the upper airframe. The main parachute will be deployed at an altitude of 600 ft. The main parachute will decrease the rocket's velocity enough so that it has a safe landing, which in this case, means the kinetic energy upon landing does not exceed 75 ft-lbf per independent section of the vehicle. The main parachute will be located below the recovery bay in the middle airframe.

Figure 6-49 Inflated Main Parachute

To determine the size of each parachute, we used the model rocket simulator OpenRocket to perform flight simulations. In these simulations, we estimated the weight of the rocket to be 43.59 lb. The simulations determined the drogue parachute will have a diameter of 36 in and a drag coefficient of 0.75. The main parachute will be 120 in in diameter and will have a drag coefficient of 2.20. The high drag coefficient of the main parachute is to ensure the rocket lands safely while still being small enough to fit within the rocket body. Using these parachute specifications, the simulations determined the rocket would safely land with a maximum kinetic energy of 52.3 ft/s. The maximum drift distance in 20 mph winds was determined to be 2675.19 ft.

Figure 6-50 Spherachutes Drogue Parachute

The drogue parachute will be purchased from Spherachutes and the main parachute will be purchased from Rocketman Enterprises. As discussed in the verification section, the main parachute will also include a reefing ring from Rocketman Enterprises. The reefing ring will reduce shock load during parachute deployment by increasing the amount of time it takes the parachute to open. The ring is designed to work on any Rocketman parachutes, so it is unlikely to cause any 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.

6.5.5 Parachute Deployment

The recovery system is a dual event dual bay system, with the drogue parachute housed in the upper airframe beneath the payload, the main parachute housed in the middle airframe, and the recovery bay housed in the coupler directly between the middle and upper airframes as shown in Figure 6-51. A piston ejection system, used to protect the payload from the ejection gases, consists of a 4 in section of coupler with an epoxied bulkhead to create a chamber directly above the drogue ejection charges. The shock cord running through has a knot on the upper side of the bulkhead, and the piston has a slot for shock cord to run through and be connected to the forward U-bolt on the recovery bay with a quick link. Both parachutes will be deployed using black powder charges, with redundant ejection wells on either side of the recovery bay.

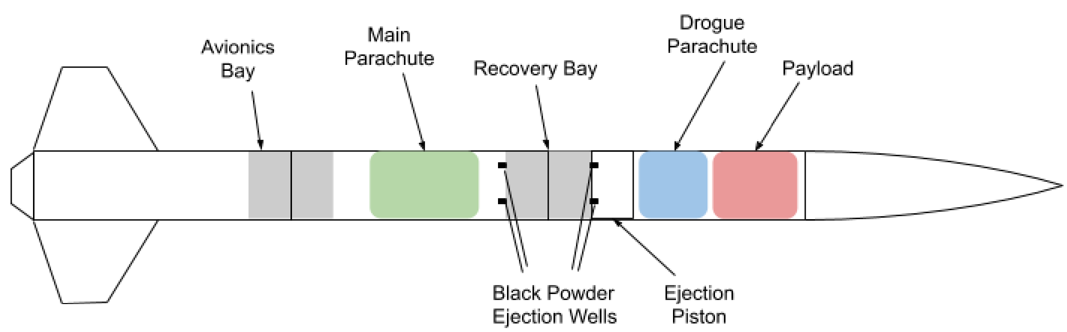


Figure 6-51 Launch Vehicle Configuration

Figure 6-52 Piston Section View

At apogee, the altimeters will signal for the black powder charges to ignite, which will fill the chamber created by the piston and break the shear pins securing the recovery bay coupler to the upper airframe. The piston will then be free to slide out of the upper airframe, pulling the drogue parachute and payload along with it. After drogue deployment, the recovery bay coupler will remain secured to the middle airframe. Once the launch vehicle descends to 1000 feet, the payload will detach and fall under its own parachute.

The main deployment event will occur at an altitude of 600 feet. The altimeters will signal the black powder charges to ignite, breaking the shear pins between the avionics bay and middle airframe and allowing for the release of the main parachute. The launch vehicle will have now split into three independent and tethered sections, as shown in Figure 6-53.

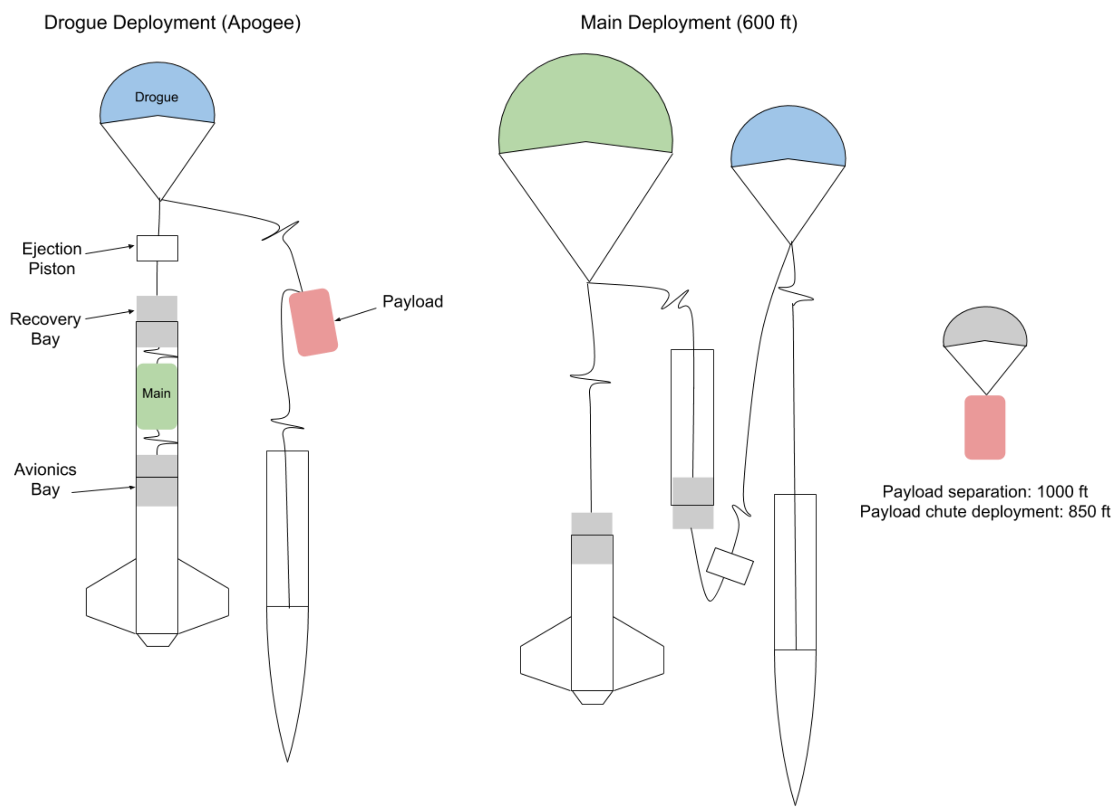


Figure 6-53 Drogue and Main Deployment

For calculating the amount of black powder needed for both drogue and main deployment, we used Equation 11 and assumed a pressure of 12 psi was necessary to break the shear pins. Drogue deployment, during which the 3.375-inch chamber of the ejection piston will be pressurized, will require about 0.59 grams of black powder. Main deployment, which will pressurize the 19-inch middle airframe, will use about 3.32 grams of black powder. These values will be confirmed by conducting ejection testing before launch.

$$Black\ Powder\ (grams) = \frac{PV}{\left(266\ \frac{in\ lbf}{lbm}\right)(3307\ °R)}\left(\frac{454\ g}{lbf}\right)$$

Equation 11 Ejection Charge Calculation

6.6 Mechanical Systems

6.6.1 Airbrakes

6.6.1.1 Summary

The airbrake system is designed to sit within the avionics bay and control the apogee of the vehicle by deploying and retracting 4 fins. The fins slide along aluminium guide rails and are driven by a spiral cam mechanism. Actuation is controlled by a hobby servo driven by the avionics board, as described in section 6.7.3.

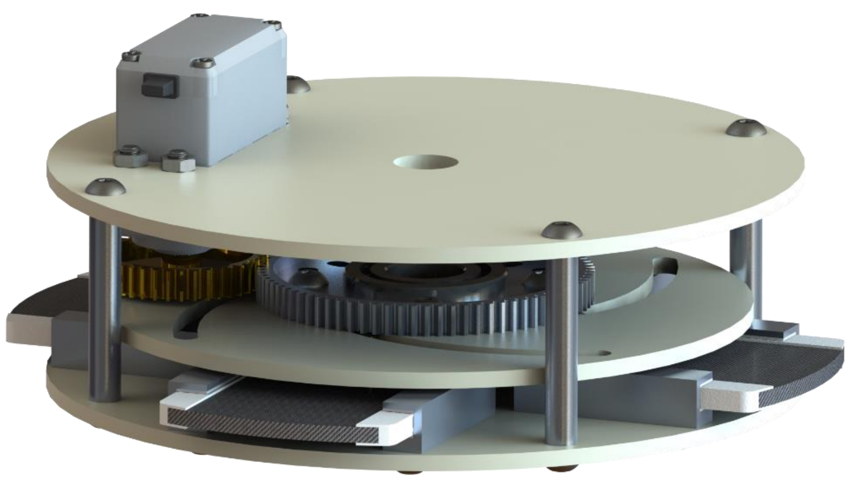


Figure 6-54 Airbrake Assembly

6.6.2 Structural Design

6.6.2.1 Structural Plates

The airbrakes consist of three main plates that each mount to the avionics bay spine. Each plate is made of 1/8 in G10 fiberglass plate, which will be cut to size using a waterjet.

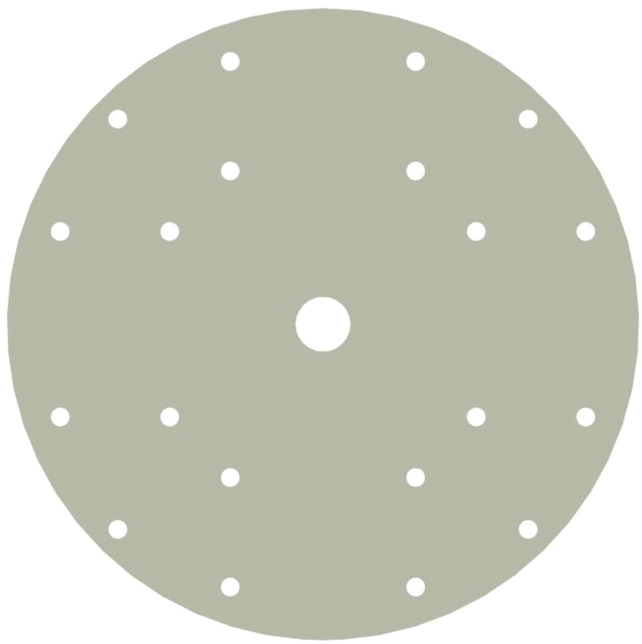


Figure 6-55 Guide Plate

The guide plate sits at the bottom of the airbrakes assembly and serves as the mounting point for the fin rails, as well as for the standoffs that attach to the motor plate. Holes in the guide plate accommodate screws to attach to these components.

Figure 6-56 Actuator Plate

The actuator plate sits just above the fin rails and includes the spiral slots which will be used to drive the fins during flight. The slots are sized to accommodate the fin pins and bushings, and the plate mounts to the central gear using 4 #8-32 bolts. The central hole in the actuator plate has a diameter of 32mm, which matches that of the central gear. A 32mm ball bearing will be pressed into this hole; the bearing has an inner diameter of 15mm, so it will interface with the avionics bay spine.

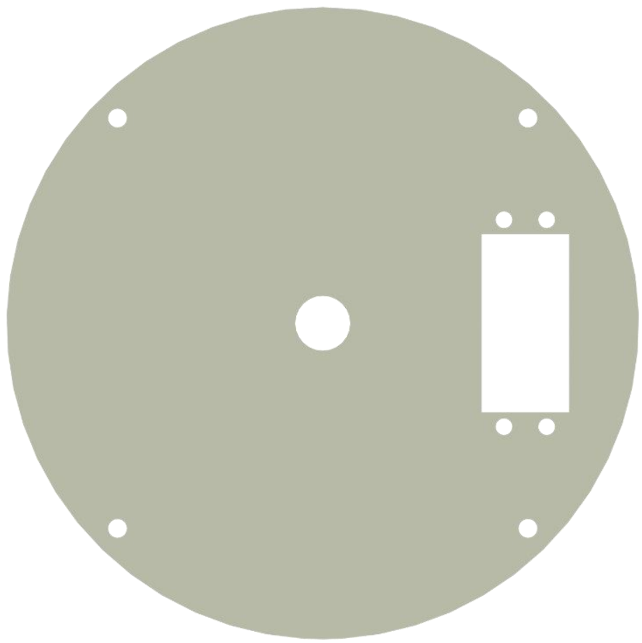
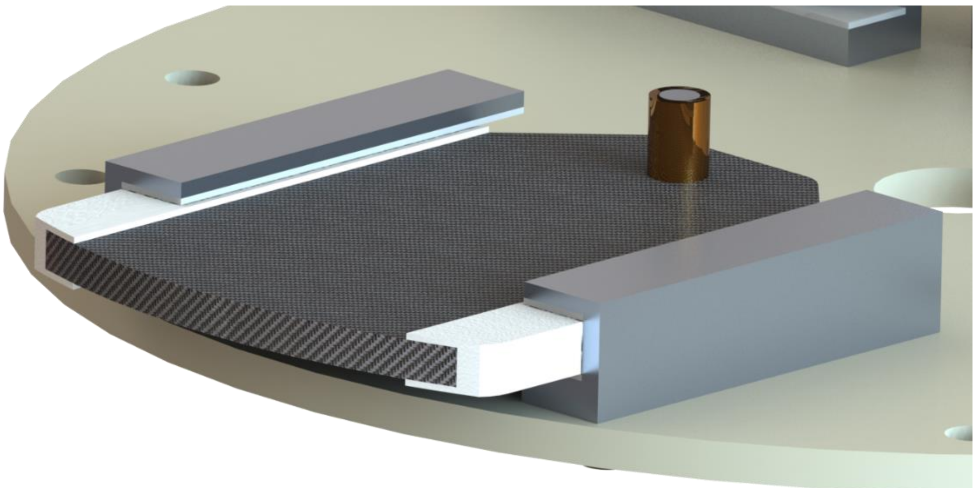


Figure 6-57 Motor Plate

The motor plate is located at the top of the airbrake assembly and attaches to the servo. The servo mounted on a 3D printed spacer to set the distance between the servo gear and the actuator plate. The motor plate itself sits on the aluminum standoffs attached to the guide plate.

6.6.2.2 Fin Rails

The fin rails are U shaped channels of 6061-T6 aluminium with 2 tapped #8-32 bolt holes on their bottom face. The fin rails allow the fins to slide easily while constraining the fins vertically and laterally. The top and bottom surfaces of the fin rails will have adhesive backed PTFE film applied to reduce the friction with the fins.



6.6.2.3 Spine Connection

The airbrakes assembly is connected to the spine via two shaft collars on the top and bottom of the motor plate and guide plate, respectively. The shaft collars fit around the spine and tighten down using a set screw. The actuator plate and bearing are constrained on the spine by two snap rings that fit into grooves milled into the spine.

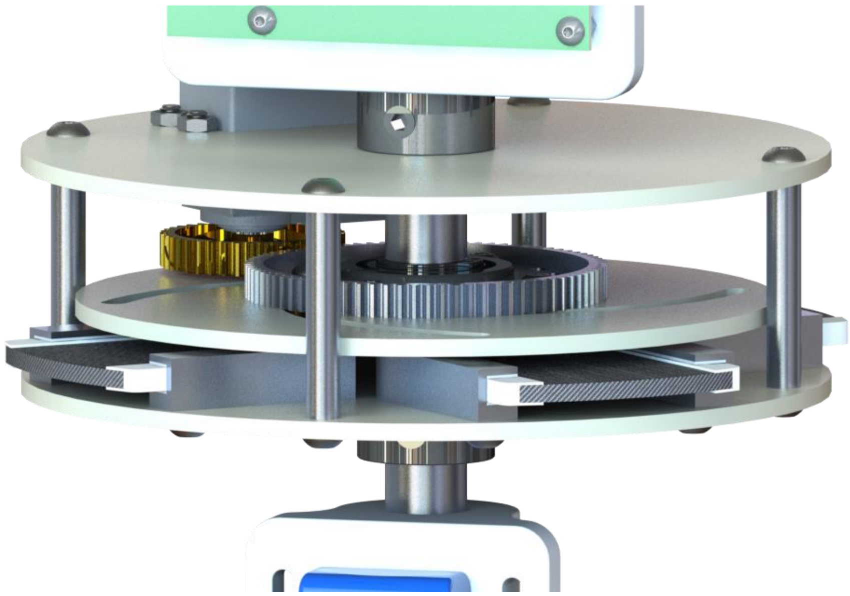


Figure 6-58 Airbrake Mounting

6.6.3 Actuation System

The actuation system uses a goBILDA Super Speed servo motor to drive a 2:1 gear reduction connected to the actuator plate. The actuator plate has 4 equiangular spiral slots cut into its profile, which pins and bearings on each fin fit into. As the servo turns, the bearings roll in and out along these slots, driving the fins with them.

6.6.3.1 Airbrake Fins

The airbrake fins are made of 1/8 in carbon fiber plate, which will be waterjet. Their profile matches that of the outer airframe when stowed, and each fin has a single 1/8 in hole to accept a steel dowel pin. This steel dowel pin will extend into the actuator plate and serve to push the fin in and out as the plate rotates.

Figure 6-59 Airbrake Fin

To reduce the friction between the pin and the plate, a bronze bushing has been added to the fin pin where it interfaces with the actuator plate. With the bushing, as the plate turns there is no longer a large component of sliding friction between the fin and the rough edge of the fiberglass, but instead a much smaller rolling resistance, allowing the system to actuate more freely.

The portions of the fins that slide against the fin rails will also be covered in a layer of 0.02 in thick adhesive backed PTFE film. PTFE has a very low coefficient of friction, even without lubrication, so is an ideal material to reduce the resistance on the airbrake system. Since the airbrake system will not see extended use, there is no concern of the film wearing out and requiring replacement.

6.6.3.2 Gear System

The goal of the gearing sub-system is to transfer the inputted servo power to the actuator plate. This is achieved via two spur gears; a 40-tooth brass gear that mounts directly to the servo's 25 tooth output spline, and an 80-tooth aluminium gear that bolts onto the actuator plate. The gears are both purchased from goBILDA, and thus are designed to mesh together. The paring with the servo provides evidence that the gears will be adequate to handle the torques the servo is capable of producing.

Figure 6-60 goBILDA 80-tooth Hub Gear

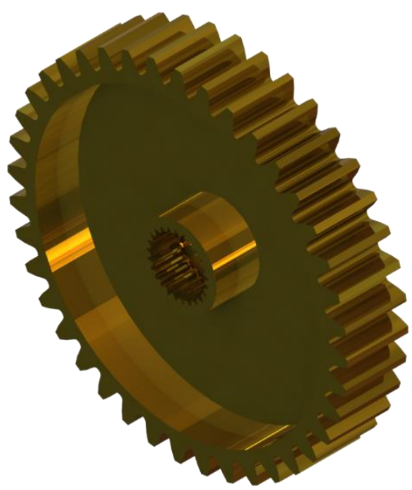


Figure 6-61 goBILDA 40-tooth Servo Gear

The gear ratio produced by these gears is 2:1. This ratio was chosen based on analysis of the required torque to drive the mechanism and the choice of the servo to do so, a process explained further below.

6.6.3.3 Airbrake Kinematics

In order to determine the requirements to drive airbrake mechanism a mathematical model describing the force applied to each fin and the friction force experienced by each fin was developed. Due to the unique nature of the airbrake mechanism, the development of these formulae are described below.

$$F = \frac{T * GR * \cos(\alpha)}{d * \sin(\alpha) * n}$$

Equation 12 Applied Force per Fin

Equation 12 is used to determine the applied force to each fin of the airbrake system due to the torque of the motor. From the motor, the torque (T) is multiplied by the gear ratio (GR) to find the applied torque to the driving plate. The force from this torque is transferred to the fin pin at an angle determined by the polar angle of the slot (α), as shown in Figure 6-62. By using an equiangular spiral slot, this angle remains constant along the length of the spiral, simplifying calculations.

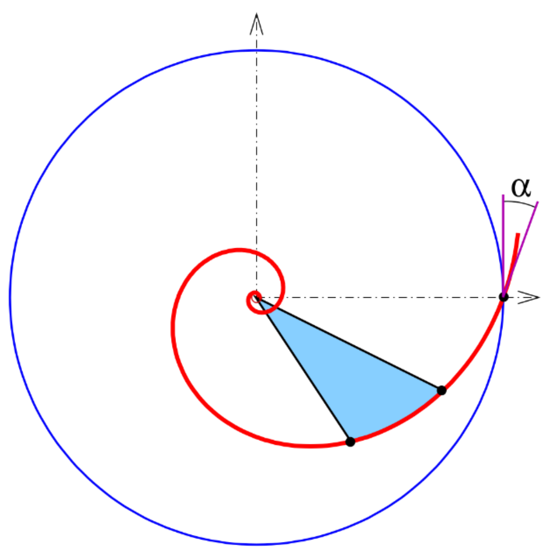


Figure 6-62 Polar Angle of Equiangular Spiral

The effective moment arm becomes the extension of the fin from the center point (d) multiplied by $\sin(\alpha)$. The radial component of the force applied to the fin is then the force multiplied by $\cos(\alpha)$. Dividing by the number of fins (n) gives the applied radial force to each fin for a given configuration and torque.

Knowing the force applied to each fin, we can then calculate the friction this force must overcome to actuate the airbrakes, as shown in Equation 13.

$$F_f = \frac{F_d * \mu_n}{n} + \frac{T * GR * \mu_l}{d * n}$$

Equation 13 Frictional Force per Fin

The friction experienced by each fin is due to two independent factors. One component is due to the drag produced by the airbrake airbrakes (F_d), calculated as the multiple of the total drag force and the coefficient of friction on the top and bottom of the fin rails (μ_n), divided by the number of fins. The second component is due to the lateral component of the actuation force, causing friction between the fin and the rail wall. This component is calculated in a similar fashion to the radial force, instead finding the lateral force component, and multiplying by the friction coefficient between the fin and rail walls (μ_l). By balancing the friction and applied force the torque required to drive the mechanism can be computed.

6.6.3.4 Servo Selection

Using the equations outlined above, we can calculate the frictional force on the airbrakes at maximum drag, and the torque required to drive the airbrakes in this condition. We can also define a minimum actuation time we would like the servo to produce. This actuation time states how long the airbrakes should take to go from closed to full extension, assuming maximum drag force, and is used to determine the RPM requirement for the servo motor. 0.25 seconds was chosen as the minimum actuation time based on consultation with the avionics team. The required servo speed with a 2:1 gear reduction is 120 RPM, and using a maximum drag of 28.6 lbf, and coefficients of friction for the top and bottom of the fin rails as 0.04 and as 0.2 for the walls, the required torque is 14.18 oz-in.

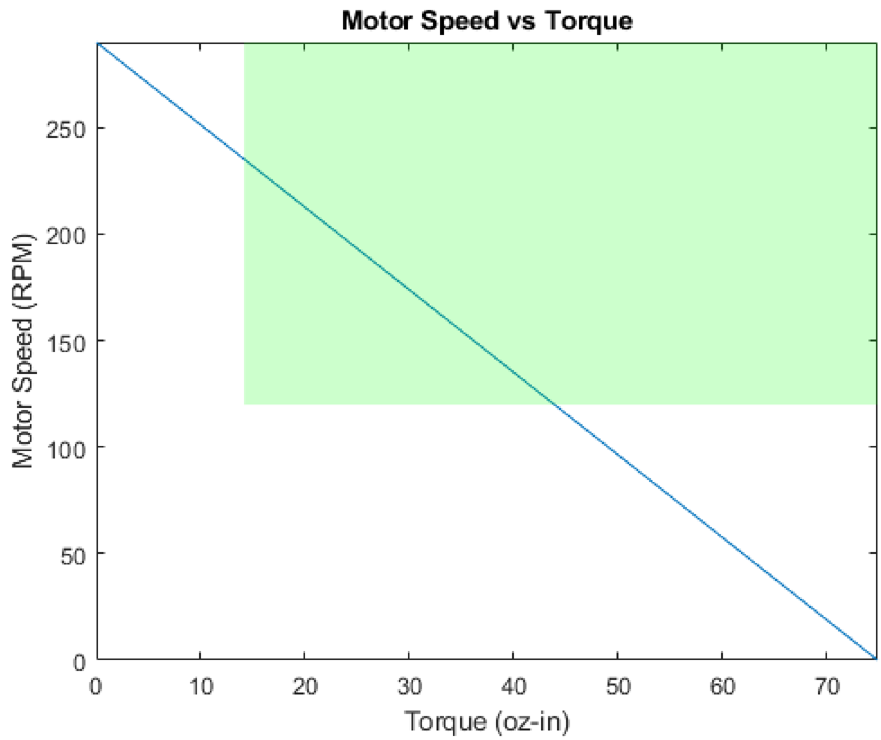


Figure 6-63 Motor Speed vs Torque

With these values computed, we can define a range over which a hypothetical motor would have enough torque and spin fast enough to actuate the servo to our specifications, the green area in Figure 6-63. Assuming a linear relationship between the no-load and stall torque and RPM for the servo, we can then plot the speed as a function of torque onto this graph. If the line passes through the green region, the servo will meet the requirements. The plotted curve is for goBILDA's Super Speed servo motor.

Figure 8 goBILDA Super Speed Servo

Lastly the current drawn by the servo was also an important constraint to consider. The servo operates at 7.4 volts for highest performance operation. Again, assuming a linear relationship between the no-load and stall current, the operating amperage for the motor is at maximum 1.11 A. The avionics system is driven using a 3S LiPo producing 11.1 V, which will be stepped down to drive the servo. The battery is capable of a constant output current of 27 A, so is more than capable of powering the servo.

Under the worst-case assumptions stated above, our minimum deployment time to full airbrake extension is .153 seconds, far exceeding our original goal of .25 seconds.

6.6.3.5 Gear Selection

The 2:1 gear ratio utilized in the system was determined based on optimizing the servo's performance curve. When analyzing the servo's performance curve, we calculated that for every reduction of 0.1 ft-lbs. of torque, the servo's speed would increase by 74.2 RPM. Further, increasing the gear ratio to 3:1 would decrease the servo's required torque by 33%, while only increasing the max rotational speed of the servo under that given torque by 16.4%. However, for the 3:1 gear ratio to make for a faster deployment time, it would have to increase the rotational speed by over 33%. The servo is near to its torque limit using a 1:1

gear ratio, and as such the 2:1 gear ratio is the best option of the gear combinations available to the team.

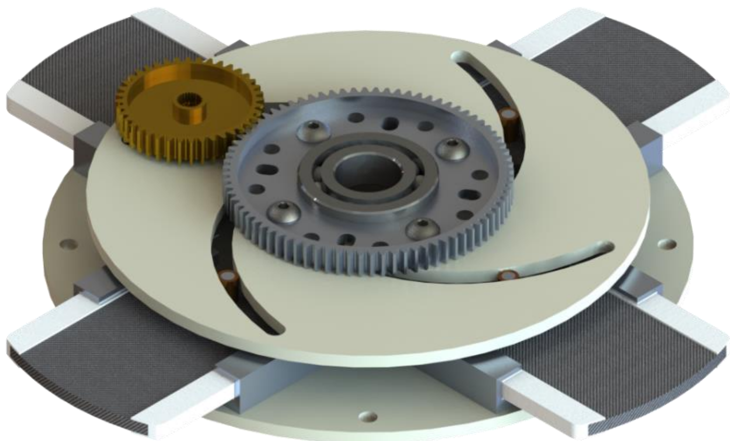


Figure 6-64 Airbrake Gear System

6.6.4 Aerodynamic Analysis

Understanding the forces developed by the airbrakes during flight is essential to both simulating and predicting the flight of the vehicle and to verifying the structural and mechanical integrity of the airbrake system. The variables considered to determine the drag produced by the system are the altitude, velocity, and extension of the fins. The effect of parameters such as angle of attack are not analyzed due to the added computation time needed, and the fact that after motor burnout the vehicle is unlikely to experience large angles of attack until apogee is reached, at which point the airbrakes will not be effective.

Computational Fluid Dynamics (CFD) simulations were chosen as the primary method to analyze the airbrake drag, due to their high accuracy and ability to account for complex effects such as the boundary layer of the vehicle. The tradeoff with using this system is that CFD simulations are computationally intensive, with even optimized simulations taking up to 10 minutes to complete on a reasonably powerful desktop computer. To fully quantify the airbrake drag, simulations must be run at multiple altitudes, velocities, and extensions, so it is important to minimize the number of runs while still retaining important data.

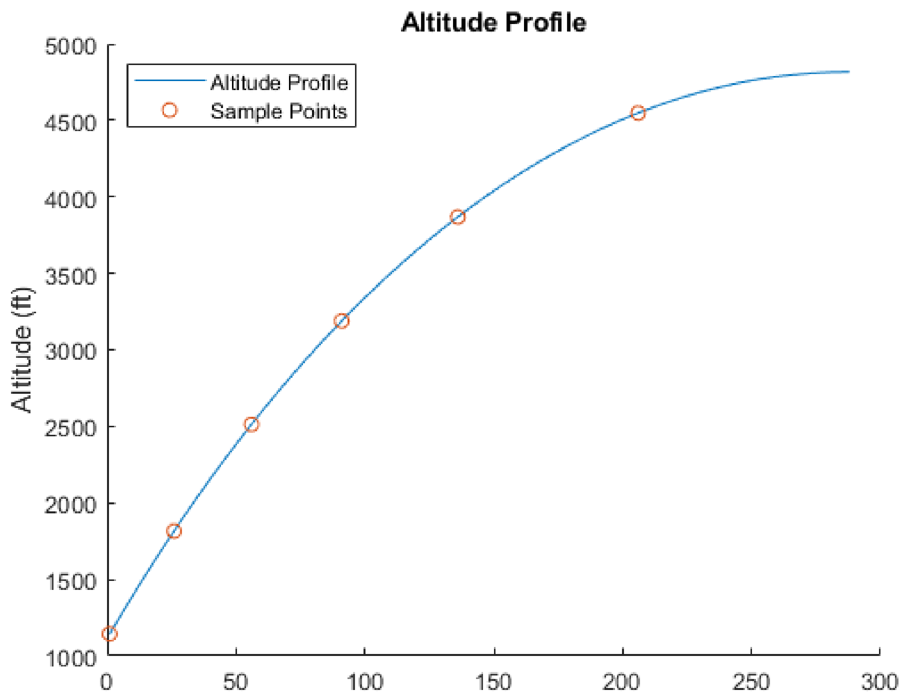


Figure 6-65 Altitude Sample Points

Using data from a simulated standard flight profile, 6 sample points distributed linearly from burnout altitude to our target apogee were selected to serve as the simulated altitudes, as shown in Figure 6-65.

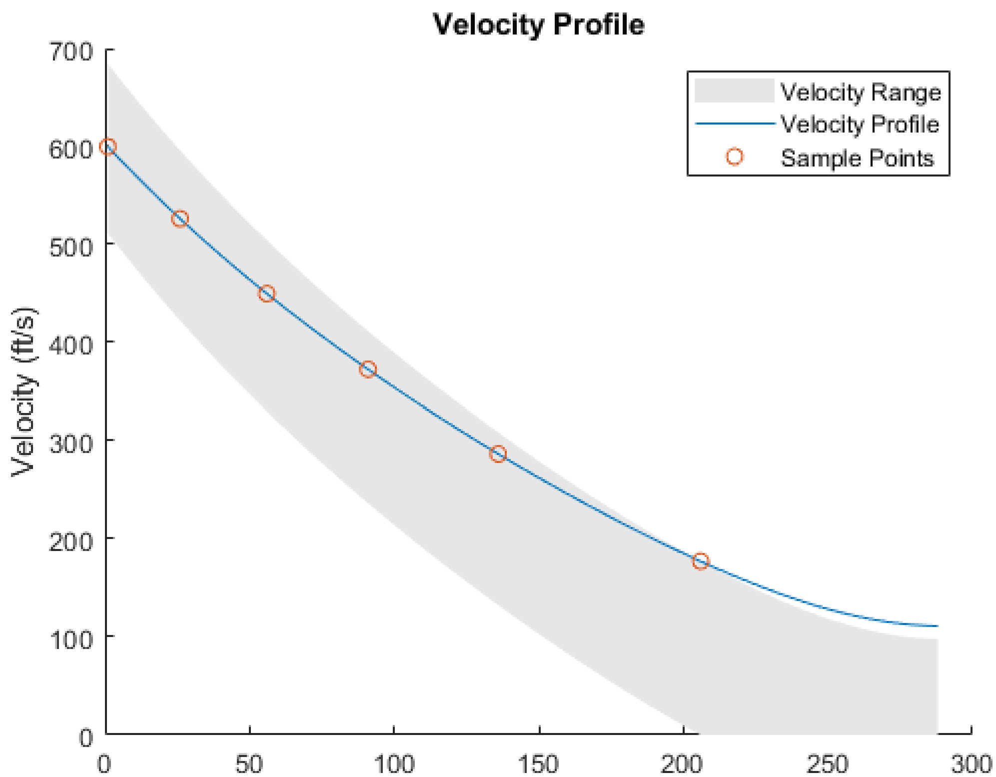


Figure 6-66 Velocity Profile and Sample Range

From the same flight data, the predicted velocity over the flight was determined. A range of possible velocities was created to select the velocity samples from. At burnout, the velocity of the vehicle could vary below or above the predicted velocity, but as the flight continues the airbrake system should actuate to slow down the vehicle. At apogee, the velocity could be as low as zero, depending on the flight angle. Thus, a velocity corridor was created, with a symmetric range at burnout, and ranging from zero to the predicted velocity at our target apogee. Within this range 6 linearly spaced points within this range were assigned to each altitude sample point.

At each combination of altitude and velocity, 6 simulations were run with different levels of extension of the airbrakes. With 6 altitude samples, 6 velocity samples per altitude, and 6 extension samples per combination, a total of 216 simulations were completed, distributed among members to reduce total runtime. SOLIDWORKS Flow Simulation was used to analyze each case, due to the team having access and experience with the SOLIDWORKS environment.

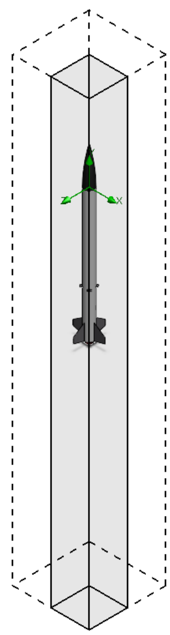


Figure 6-67 Simulation Setup with Computational Domain

Due to the large number of simulations to run, it was important that the simulation setup was well optimized. A single run with a large computational domain and small mesh size was run to establish a ground truth value for the results. The simulation was then optimized by reducing the domain size and through the use of detailed local meshes in the areas required, while the global mesh remained relatively coarse. To further reduce computation time, the domain was split to only include ¼ of the vehicle, and symmetric boundary conditions ensured accuracy was retained. Since a ground truth value could not be established for each simulation, the runs also incorporated adaptive mesh refinement to retain detail in necessary areas. The initial ground truth simulation took 32:03 minutes to solve, while the optimized simulations generally took around 8:30 minutes, a decrease of 116% The results differed by only 2.3% between the simulations.

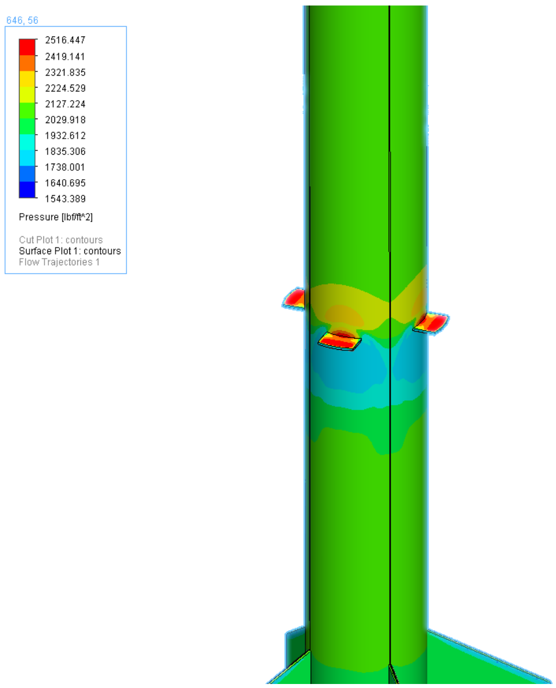


Figure 6-68 Pressure Distribution on Vehicle Body

Figure 6-68 shows the pressure distribution in the area around the airbrakes from one of the simulations. From this it is clear that the airbrakes will not interfere with the vehicle's fins, and that the static ports should be offset 45° from the airbrake fins to reduce the change in air pressure due to their deployment.

The drag data produced by each simulation was analyzed using a custom MATLAB program. Data from each extension level was normalized to zero at zero extension, since the entire vehicle was modeled in the simulation, but we are only interested in the drag produced by the airbrakes. For each level of airbrake extension, a 3D surface was fit to the collected data using altitude and velocity as the function inputs, as shown in Figure 6-69. A 2nd order multivariate polynomial was chosen as the fit equation, due to a r^2 value extremely close to 1 for each surface, and a surface shape that matched expected results.

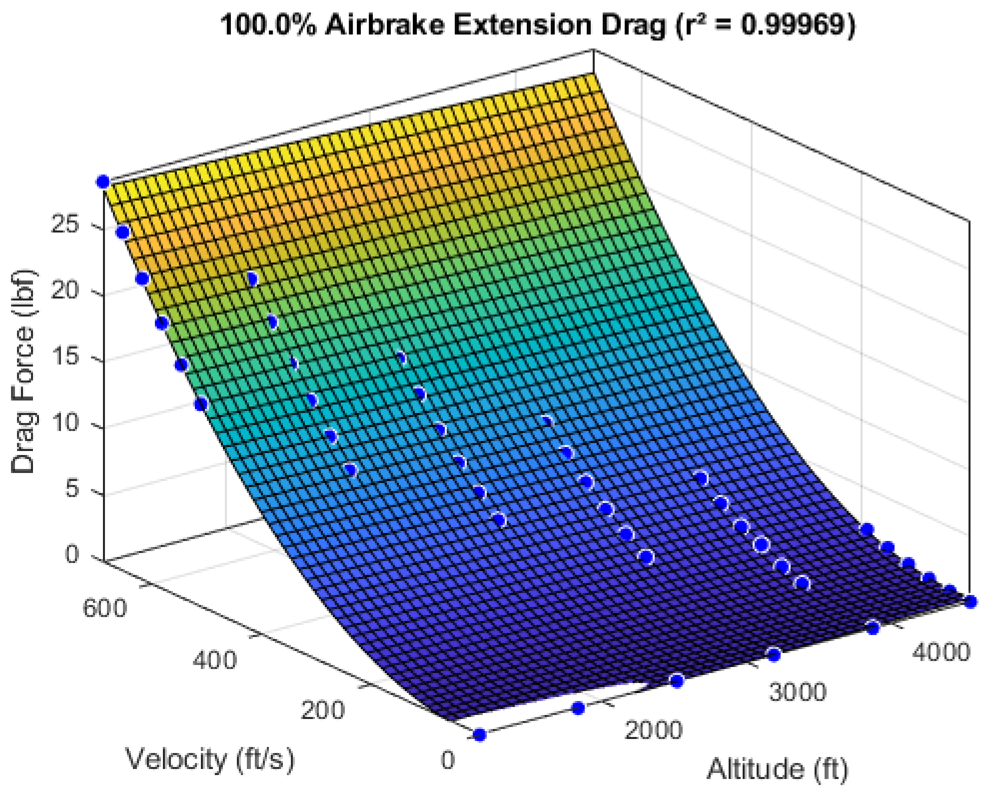


Figure 6-69 Drag Surface for 100% Deployment

With the data from each extension level at each combination of altitude and velocity, the altitude could be used to find the air density using the standard barometric formula and combined with the velocity to calculate dynamic pressure as shown in.

$$q = \frac{1}{2}\rho v^2$$

Equation 14 Dynamic Pressure

By fitting a 3rd order multivariate polynomial with inputs of dynamic pressure and extension percent to the new data, a surface was generated relating these inputs to the drag force on the airbrakes, as shown in Figure 6-70. From a structural standpoint, the full extension data gives us a maximum drag force of 28.6 lbf at full extension and maximum dynamic pressure.

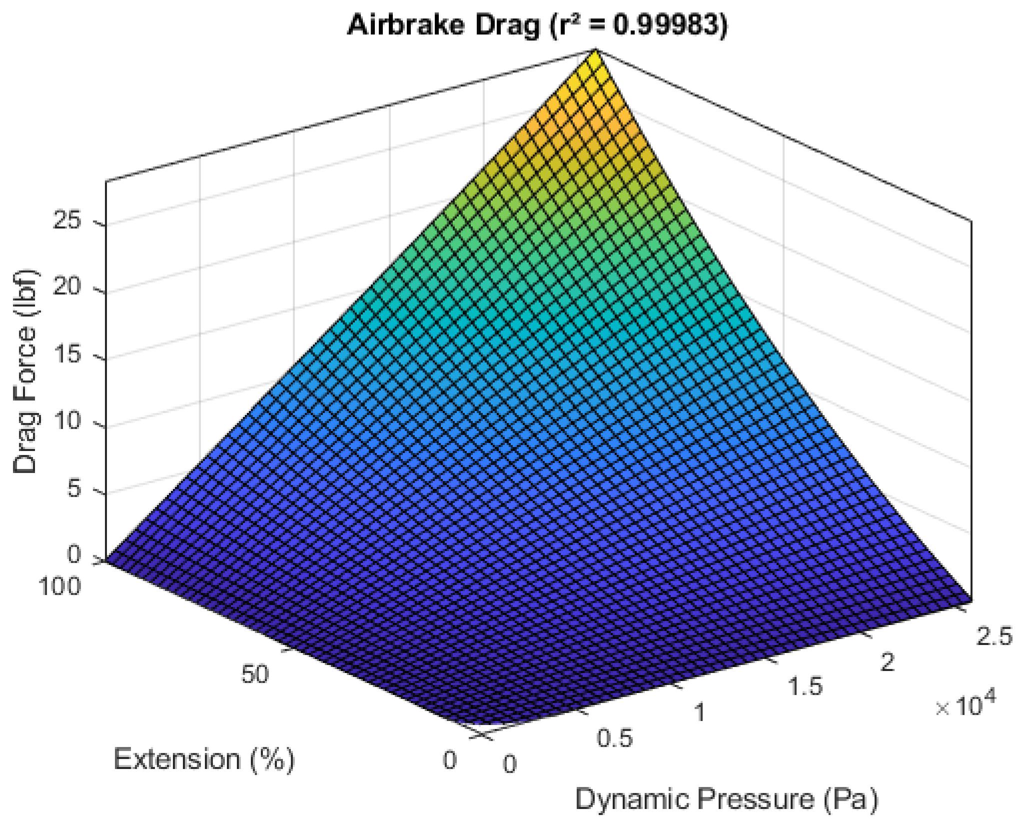


Figure 6-70 Airbrake Drag Surface

This surface's corresponding equation can be used in simulations and on the vehicle's flight computer to predict the drag provided by the airbrakes at any point during flight.

6.6.5 Avionics Bay

The airbrakes and avionics system are housed within the avionics bay. The avionics bay consists of a 12 in long coupler tube, with a fiberglass bulkhead at the top of the coupler and the vehicle's thrust plate at the bottom. On top of the thrust plate a 3D printed annular standoff and another fiberglass bulkhead make up the fixed components of the twist lock mechanism. The coupler and twist lock are bolted into the airframe and are not removed at the launch site.

Figure 6-71 Avionics Bay

Instead, to access the airbrakes and the electronics bay inside, a central spine with a T-profile can be twisted and removed from the lower airframe. This design is strong and simple to use, significantly accelerating launch site operations.

6.6.5.1 Spine

The main structural component of the avionics bay is the cylindrical spine that runs through the center. The spine allows for easy mounting of the airbrake system and hardware used in the rocket's flight calculations. The cylindrical shape will allow for easy manufacturing compared to previous designs such as a hexagonal shaped spine. The spine is 10 inches tall with a 15 mm diameter. The spine is made of 6061-T6 aluminum to minimize weight. On each end of the spine there is a tapped ¼-20 hole for attaching to the adapter plate and T-lock, as well as 2 retaining ring grooves near the center of the spine spaced to hold the airbrakes actuating plate.

Figure 6-72 Avionics Bay Spine

At the top of the spine at 5/16 in U-bolt is connected to an adapter plate made of ¼ in thick 6061-T6 aluminium. This plate will be waterjet cut and will serve to transfer the shock loads of the main parachute deployment directly to the spine, removing the need for a thick forward bulkhead.

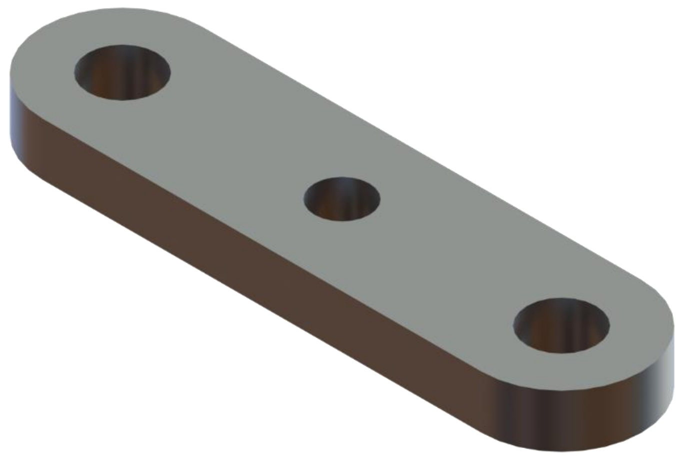


Figure 6-73 Spine Adapter

6.6.5.2 Twist Lock

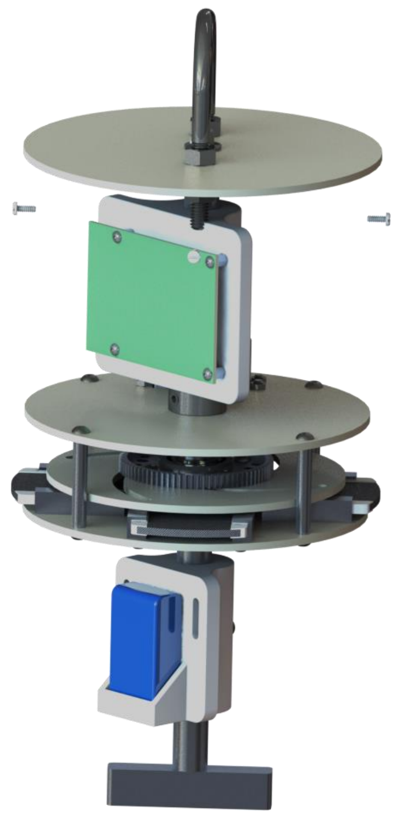


Figure 6-74 Avionics Bay Removable Section

At the bottom of the spine a 6061-T6 aluminum cross-bar is attached at the end of the spine with a ¼-20 bolt to incorporate a twist lock mechanism for the avionics bay. The twist lock method of securing the avionics and airbrake was implemented for simple construction on launch day as well as easy and quick access to electronics inside the rocket without compromising stability of the bay during flight.

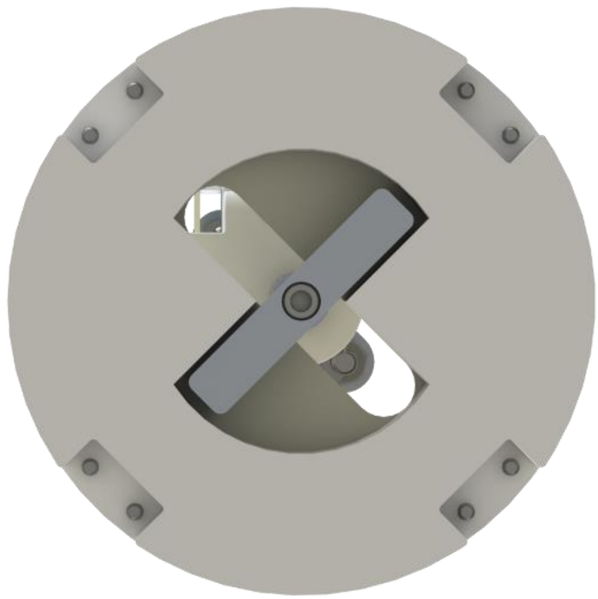


Figure 6-75 Twist Lock Mechanism

The cross-bar reacts against a 1/8 in thick fiberglass plate during parachute deployment, which is bolted securely to the thrust plate. A large contact area ensures the fiberglass will not crack due to point stresses.

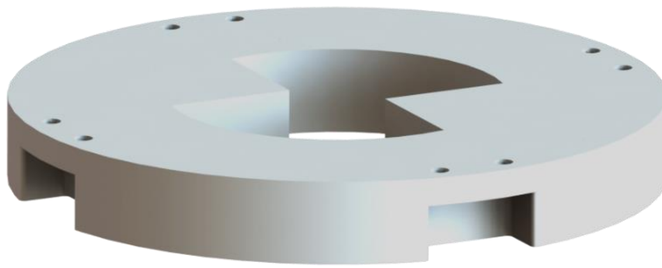


Figure 6-76 Connection Ring

To act as a standoff from the thrust plate and to guide the twist lock mechanism, a 3D printed connection ring will sit between the fiberglass plate and the thrust plate, with gaps to fit the thrust plate's radial brackets and holes for the 8 #8-32 screws attaching the top plate to pass through.

6.6.5.3 Airbrakes Retention

The airbrake system will be held into place using snap rings and shaft collars. The snap rings will be made of plain carbon steel and have an inside diameter of 0.544 inches fully closed. The snap rings will be used to retain the actuator plate by holding the bearing in place on the spine. Two shaft collars will be used to hold the guide plate and the guide plate and motor retention plate in place holding the entire system in place on the spine. This will result in a tight retention of the system so that no movement will occur, and the airbrakes can be deployed properly through the slots in the airframe.

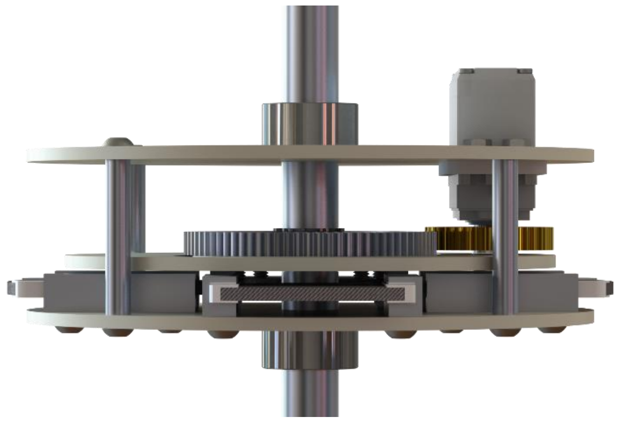


Figure 12: Airbrake Retention in Avionics Bay

6.6.5.4 Avionics Retention

The avionics computer will be attached to the spine using a custom 3D printed sled above the airbrake system. The sled will be printed out of PLA plastic and will fit around the spine with a set screw to hold it in place. 4 threaded inserts will mount standoffs to keep the board way from the plastic and allow clearance for any electrical components.

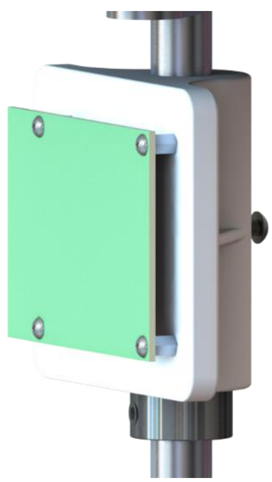


Figure 6-77 Avionics Bay Board Mount

The battery for the avionics system will mount below the airbrakes to provide isolation between the board and its power source, with the power cables running along the aluminum standoffs of the airbrake mechanism. The sled will be similar in design to the avionics computer sled but will include a slot for the battery to fit into and will hold the battery in place using a Velcro strap wrapped through the structure of the sled.

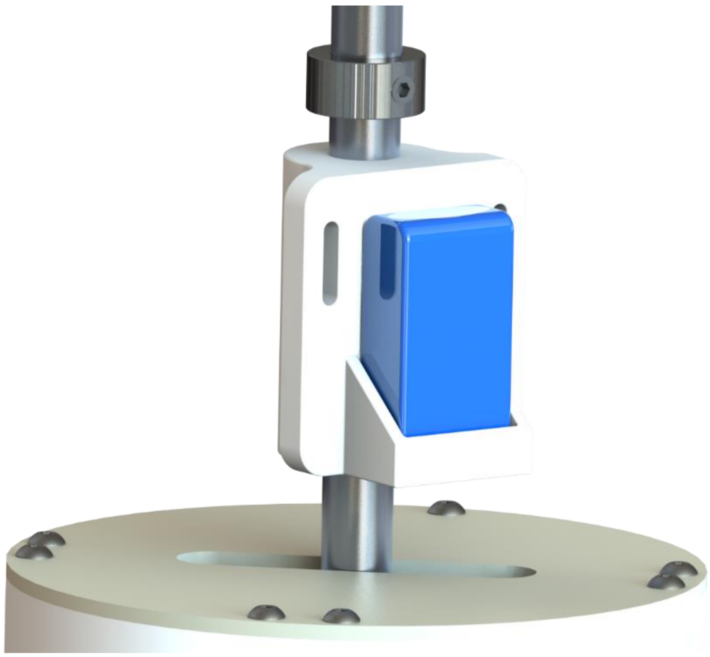


Figure 6-78 Avionics Bay Battery Mount

6.6.5.5 Verification

The avionics bay U-bolt attaches to the shock cord holding the main parachute, which will experience a significant opening shock during deployment. As determined in section 6.5.2, the maximum acceleration during opening will be 10.9 G, resulting in an opening shock load of 198.86 lbf at the avionics bay U-bolt and through the spine, and 142.53 lbf at the cross bar.

The tensile stress in the spine can be calculated as shown in Equation 15, and is determined to be 727.37 psi under maximum load, with a resulting safety factor for tensile failure of 55.

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

Equation 15 Stress Calculation

Equation 10 can be used to calculate the shear area of the internal threads of the spine, and the safety factor for shear failure can be calculated similarly to what is presented above. With values for a class 2A and 2B ¼-20 thread, the factor of safety for bolt pull out is 33.4.

$$A_n = \pi n L_e D_{s_{min}} \left(\frac{1}{2n} + 0.57735 \left(D_{s_{min}} - E_{n_{max}} \right) \right)$$

Equation 16 Internal Thread Shear Area

To verify the integrity of the spine adapter, a linear static FEA simulation was run using the loads described above.

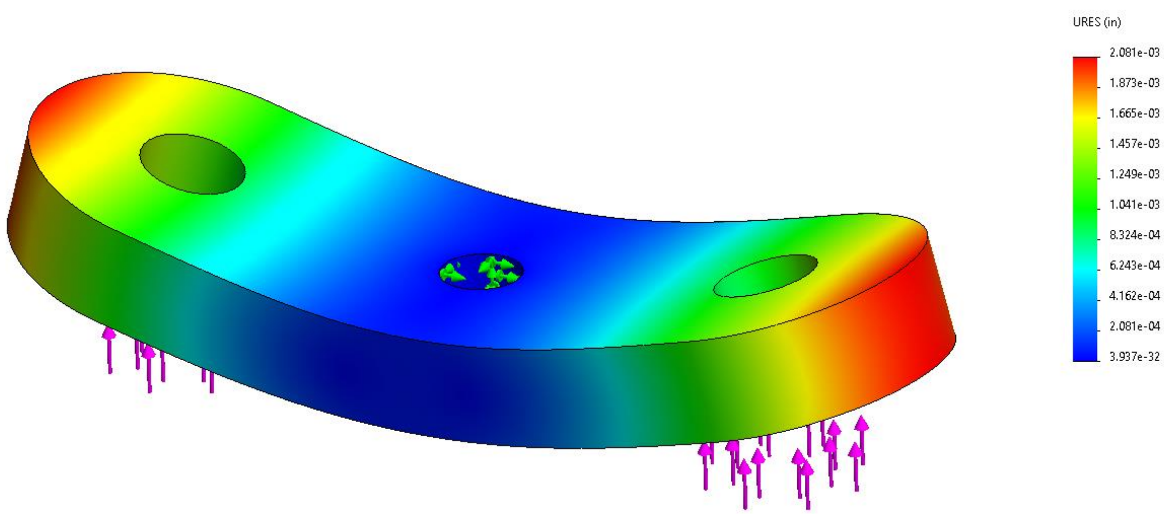


Figure 6-79 Spine Adapter Displacement

After 3 iterations of an adaptive meshing simulation converging with 98% accuracy, the maximum displacement was found to be .002 in at the edges of the part, and the maximum von Mises of 18700 psi, resulting in a safety factor of 2.14.

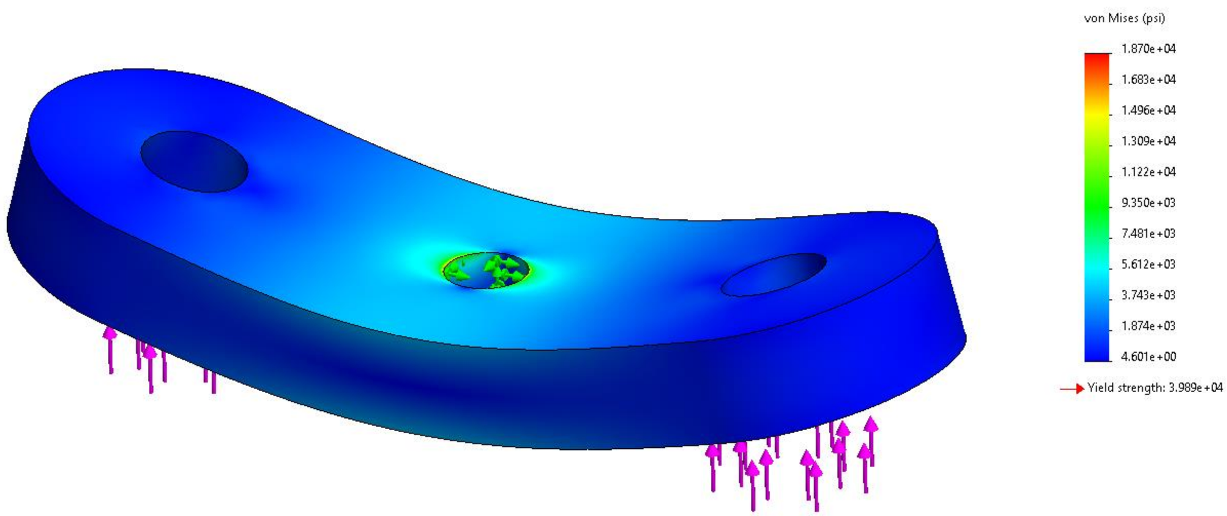


Figure 6-80 Spine Adapter von Mises Stress

A similar study was conducted for the cross bar, with results showing a maximum displacement of 0.0001 in and a maximum von Mises stress 3 elements from a singularity caused by a fixed constraint to be 2239 psi, for a safety factor of 17.8.

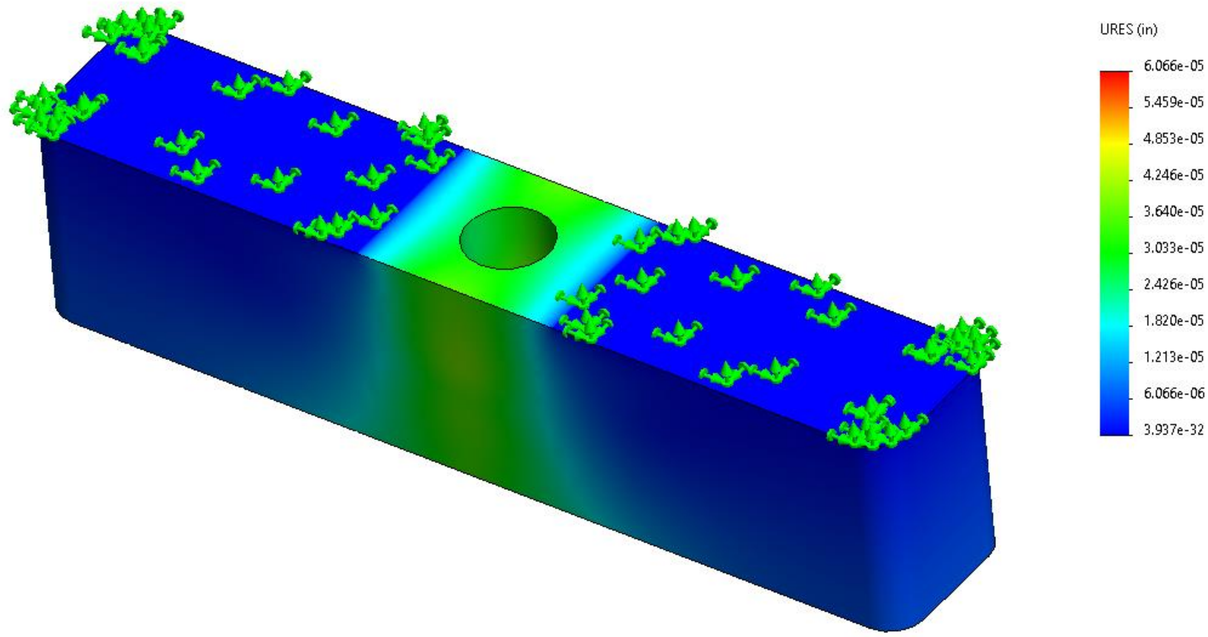


Figure 6-81 Cross Bar Displacement

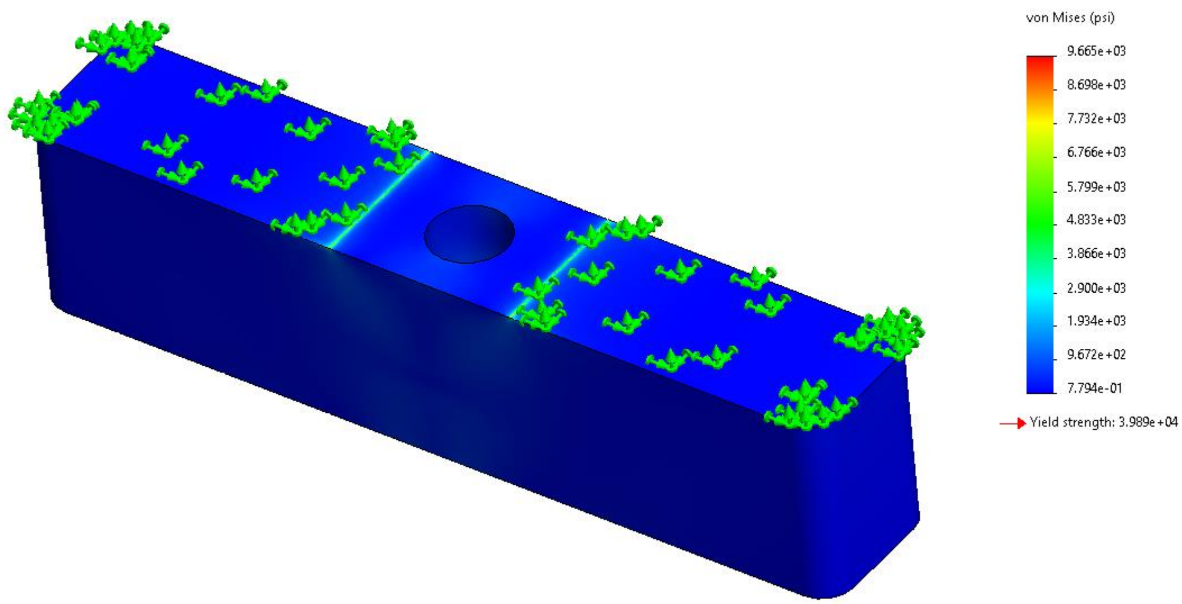


Figure 6-82 Cross Bar von Mises Stress

6.7 Avionics

6.7.1 Electronics

6.7.1.1 Components

The avionics system will include several electronic components and will be responsible for recording and transmitting data during flight, in addition to controlling the vehicle's airbrake system. The system will be controlled with a Teensy 3.2 microcontroller and feature an accelerometer, magnetometer, barometer, and GPS to measure the position, velocity, acceleration, altitude, and orientation of the rocket. A radio transceiver and serial flash memory chip will also be included to transmit live telemetry and log data during flight. The final avionics system for the rocket will utilize a single board solution with all of the electronic components integrating into a custom printed circuit board (PCB).

The team will be using a Teensy 3.2 microcontroller to control the final avionics system for the rocket. This microcontroller features a 32-bit system with a Cortex-M4 core and has 256 kB of memory in addition to 64kB of RAM. It is a good choice for this system because it is adequately equipped to perform the required data processing, easy to program with the Arduino environment, and compact enough to be easily integrated into the final PCB design. The Teensy will be used to interface with the various sensors and peripherals on the final avionics board and process the data received, in addition to controlling the airbrake system. The Teensyduino add-on program is being utilized to allow for programming the Teensy in the Arduino environment.

Figure 6-83 Teensy 3.2 microcontroller breakout board from SparkFun

The MPU-6050 is a triple axis MEMS accelerometer and gyroscope. This particular accelerometer was chosen because it has the capability to measure up to 16 g of acceleration, as many similar sensors can only measure up to 4 or 8 g. This will ensure that the accelerometer will be able to accurately measure the accelerations experienced by the rocket, as the maximum expected acceleration is 10.8 G during main parachute deployment. The MPU-6050 will be used primarily to record acceleration data, which may then be integrated to measure the velocity and position of the rocket. The gyroscope capability will also be used to measure the vehicle's orientation. The accelerometer will use the I2C protocol to communicate with the microcontroller.

Figure 6-84 MPU-6050 Accelerometer breakout board from Adafruit

To measure the heading of the rocket more accurately, an MLX90393 magnetometer will be used in conjunction with the MPU-6050. The MLX90393 is a triple axis magnetometer capable of accurately measuring the rocket's orientation with respect to Earth's magnetic field. The magnetometer has a resolution of 0.161 μT and will provide data to complement the gyroscope for a very accurate determination of the vehicle's orientation. The magnetometer will communicate with the Teensy over I2C.

Figure 6-85 MLX90393 Magnetometer breakout board from SparkFun

The team will be using an MPL3115A2 as the primary sensor for measuring altitude. The MPL3115A2 is a barometric pressure sensor which has a 1.5 Pa resolution, corresponding to an altitude change of approximately 0.3 meters. One distinct advantage of this particular barometer is that it offers a built-in altitude calculator using data from the onboard pressure and temperature sensors, making it much easier to use for altitude measurements than other alternatives. The MPL3115A2 offers very good accuracy for the cost and is suitable for the needs of our team. The barometer will use I2C to communicate with the microcontroller.

Figure 6-86 MPL3115A2 Barometer breakout board from Adafruit

In addition to the several inertial sensors above, a NEO-M9N GPS will be included in the avionics system to determine the position of the rocket both during and after flight. The NEO-M9N will serve as the primary instrument for measuring the absolute position and velocity of the launch vehicle. Furthermore, the rocket will continue to transmit GPS data after landing to assist with the recovery of the rocket in the event that line of sight is lost. This GPS has a max update rate of 25MHz and is accurate to within 1.5m for horizontal position and 0.05m/s for velocity. The GPS will use I2C to communicate with the microcontroller. The team is using the SparkFun breakout board with U.FL connector for testing purposes, but the final system will feature the NEO-M9N integrated into the PCB along with an SMA connector for an external antenna.

Figure 6-87 NEO-M9N, u.FL GPS breakout board from SparkFun

The avionics system will provide live telemetry and tracking data to the ground station throughout the duration of the flight and continue after the rocket has landed. To accomplish this, the team will be using ES32-915T30S radio transceivers to transmit GPS and sensor data from the rocket to the ground station. Long range radio (LoRa) was chosen for communication between the rocket and ground station because it allows for signals to be sent and received over much greater ranges than alternatives such as Bluetooth or Wi-Fi. LoRa transceivers are available in different frequency ranges, however models in the 433MHz range are not viable options for our team because United States regulations do not permit transmission on this frequency for an extended duration like the flight of our rocket. The LoRa module that will be used in the final design transmits at a frequency of 915MHz which should support a sufficiently high transmission rate for our needs as well as adequate range to stay in contact with the ground station. The LoRa transceiver will use SPI protocol to communicate with the microcontroller.

Figure 6-88 Ebyte E32 Wireless Transceiver Module

In addition to transmitting live telemetry data, the avionics system will also log data on the avionics board itself. The team will be using a Winbond 128M-Bit Serial flash memory chip to log GPS and sensor data during flight. This will allow for all of the flight data to be collected for review after the completion of the mission. Moreover, it will provide added security and redundancy such that the flight data can be recovered with the rocket in the event that data is not transmitted during flight. Data is often recoverable even in the case of catastrophic failure as long as the chip can be recovered. This model has sufficient storage capability for our needs and is able to store all important flight data. This component will communicate with the Teensy using SPI.

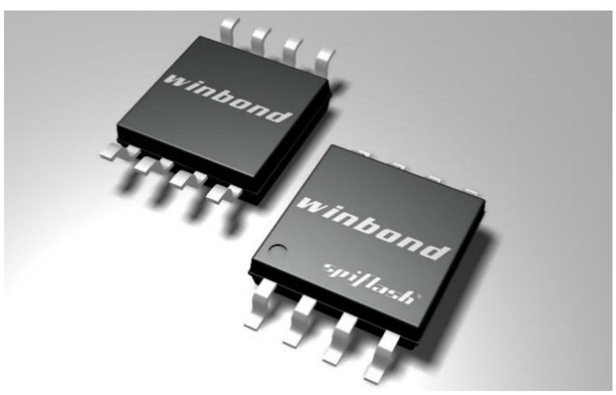


Figure 6-89 Winbond NOR Flash Serial chip

Powering the full avionics system will require a battery capable of providing adequate power to the microprocessor, GPS, radio transceiver, and all sensors being used. To accomplish this, the team will be using a Turnigy 3S 450mAh Lithium Polymer (LiPo) battery. This battery is capable of outputting 27 A continuously, far above the expected 2-3 A maximum required to both power the board and drive the servo concurrently. The 450 mAh capacity is sufficient to allow for a 3-hour pad stay time while keeping the avionics board active. This stay time can be extended by shutting off unnecessary sensors and reducing telemetry transmission output power, reducing the total power draw of the board. The battery will be connected to the final avionics board with an integrated XT-30 connector.

Figure 6-90 3S 450mAh LiPo Battery

In addition to the major components listed above, the final avionics board will also feature a few smaller components and additional connectors. This is intended to allow for the team to make changes and add additional components and functionality to the board without having to completely overhaul the custom PCB design. The board will feature a status LED and a buzzer which may be used to indicate the status of the system. A servo connector will be added to support the servo driven airbrake system. Additionally, there will be a connector for attaching the battery and a USB connector included for programming the Teensy microcontroller. Extra connectors will also be added to allow for the future addition of devices using I2C, SPI, or other GPIO pins on the Teensy.

6.7.1.2 Printed Circuit Board

Our final assembly will consist of our sensors, memory, microcontroller, and other components together on one PCB. We are hoping to have one PCB design that can be used for both the avionics and payload control systems. Using a single PCB instead of many premade breakout boards will make the computer more compact and tidier, preventing the need to have many wires connecting every component. This should also reduce potential failures from wired connections. The board will have a voltage regulator and be connected to a battery mounted below the airbrake system for isolation. Our board is being designed with KiCad using schematics of the Teensy board and our sensor breakout boards as a reference. The PCB will be manufactured by JLCPCB, who will fabricate the board and attach some of the surface mount components.

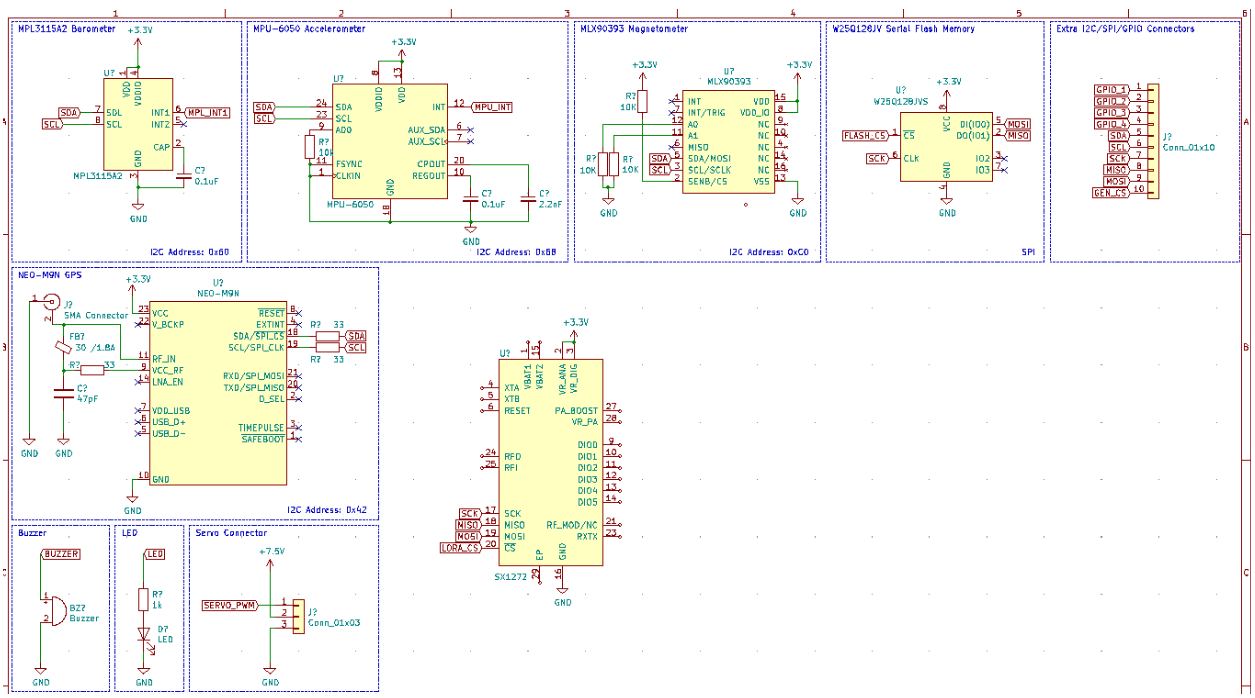


Figure 6-91 Our current KiCad schematic for some sensors included on our PCB

6.7.2 Firmware

The firmware for the system is Avionics Computer, a C++ program written in the Arduino environment. While it is designed for Teensy 3.2 chips, the Arduino environment makes it relatively simple to port to other environments.

At the base level of Avionics Computer is Peripherals. These interface directly with the hardware. They access sensors and actuators via serial communications like SPI and I2C as well as using PWM and GPIO pins. No additional functions beyond what is required to read or operate the Peripherals are included. Examples include the GPS, flash memory, and motor.

Peripherals are utilized by Subystems. Subsystems are responsible for specific functions of the computer and manage their own state. For example, the Airbrakes subsystem is responsible for ensuring the vehicle reaches a target altitude precisely. It runs a control algorithm with input from the state estimator and actuates servos to apply the control algorithm. The Airbrakes subsystem uses state estimator data to determine when it is safe to activate and when its job is done.

Loops are used for frequently executed code. Every Subsystem has its own loop, all though there are also independent loops such as the state estimator. AvionicsComputer is managed by an overarching SystemManager object which holds Subsystems and Loops. It uses the looper to execute loops at a set frequency. Finally, the Constants file is used to manage the overall configuration of the system. It defines peripheral pin addresses, code and physical constants, and active subsystems.

6.7.3 Control System

As discussed in Section 6.6.1, the rocket uses an airbrake system to lower the current predicted apogee to the target apogee. In order to predict and achieve performance in different weather and wind scenarios, a control system which selects and implements an ideal angle of extension and drag force has to be developed. This system must be verified and tuned before flight using flight simulation software and must be integrated into the avionics computer to be run during flight.

6.7.3.1 Control Design

The airbrakes control system will implement a Proportional Integral Derivative (PID) controller. A PID controller analyzes a single process variable, in this case our predicted apogee during flight, and compares it against the desired value or setpoint, which will be our target apogee. This error is passed into the PID controller on each loop. The PID controller itself is made up of 3 gains: the proportional gain, integral gain, and derivative gain. In a continuous system, the proportional term multiplies the current error by the proportional gain value to contribute to the output. The integral term takes the integral of the error over the time the controller has been active and multiplies this value by the integral gain. The derivative term takes the current rate of change of the error value and multiplies it by the derivative gain. These 3 results are all combined, and the controller provides an output value which should change the process variable. In our case, we implemented a discrete system in software using bilinear transformation to get discrete equations, and our output value is the desired drag the airbrakes should produce. By adjusting the values of the gains in a process called tuning, the vehicle can be made to reliably fly to our target apogee.

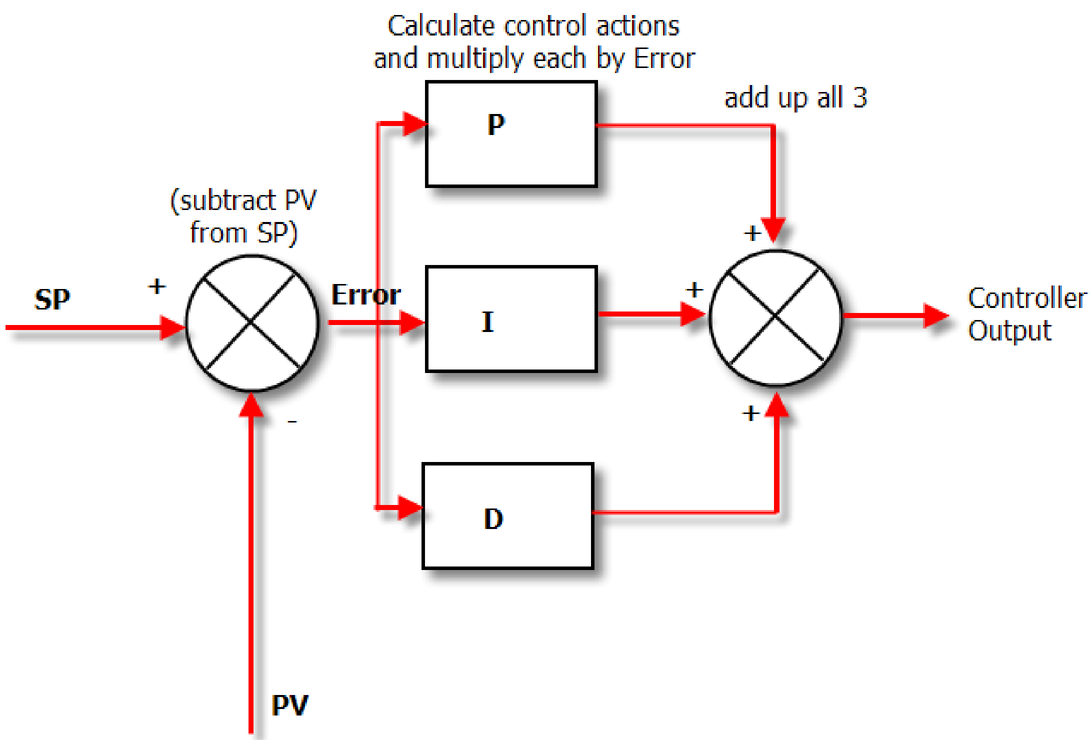


Figure 6-92 PID Controller Block Diagram

The predicted apogee, our process variable, can be computed using Equation 17. The vehicle's predicted apogee (A_P) is found as a function of the vehicle's current altitude (A_C), terminal velocity (v_t), and current vertical velocity (v_z), with the terminal velocity computed using Equation 18. Based the error between the predicted apogee and the target apogee, and on the gains of the system, the PID controller will output a required drag force to bring the vehicle to the predicted apogee. To ensure safe operation, the airbrakes are only active from after motor burnout until apogee is reached, where large angles of attack which could interfere with the airbrake operation are unlikely.

$$A_P = A_C + \left[\frac{{v_t}^2}{2g} \times \log\left(\frac{v_z^2 + {v_t}^2}{{v_t}^2}\right)\right]$$

Equation 17 Predicted Apogee

$$v_t = \sqrt{\frac{2 * m * g}{Cd * A * \rho}}$$

Equation 18 Terminal Velocity

6.7.3.2 Simulation

As established, PID controllers require tuning to ensure they will have acceptable transient response characteristics. To tune the system, one could test different gains in a real-world environment, which would be effective for something like a servo motor which is cheap and easy to operate but is prohibitively expensive and time consuming when each test requires launching our full vehicle. The alternative is to simulate the dynamics of the system and iteratively tune the gains in based on the simulator. OpenRocket is already used to simulate the flight of the vehicle without active control, so it is an ideal candidate for a simulator to use.

To implement the airbrake system in OpenRocket, the team developed a custom simulation extension written in Java known as ORBrake. OpenRocket extensions can be made to run at each simulation timestep and change nearly any property in the simulation. ORBrake works by altering the thrust applied vehicle, applying a negative thrust in response to the desired airbrake drag.

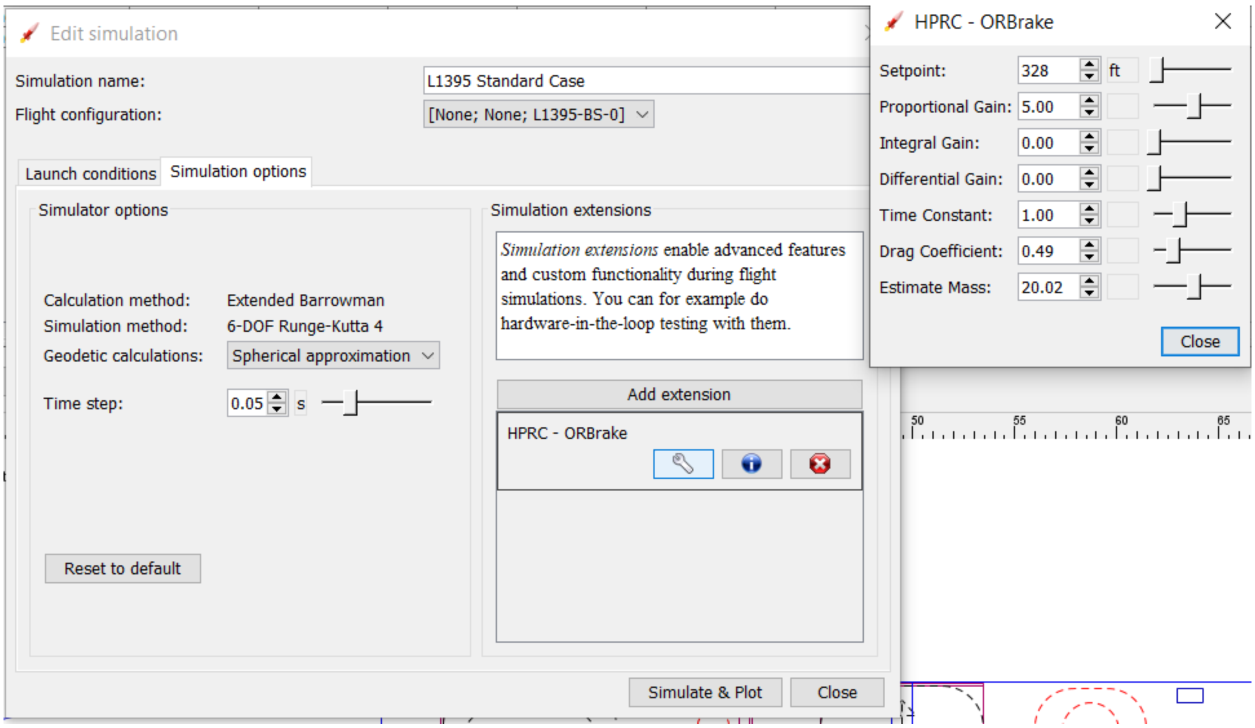


Figure 6-93: ORBrake UI

Internally, ORBrake operates in the same way as will the full-scale flight computer. At each timestep, the extension polls the simulator for the necessary values to calculate the predicted apogee of the vehicle and compares this value to the actual altitude of the vehicle using the PID controller. The controller outputs a desired drag which is checked against the maximum possible drag for the airbrakes. If the requested drag is higher, the drag is set to the full extension drag. The loop then repeats for the next timesteps until apogee.

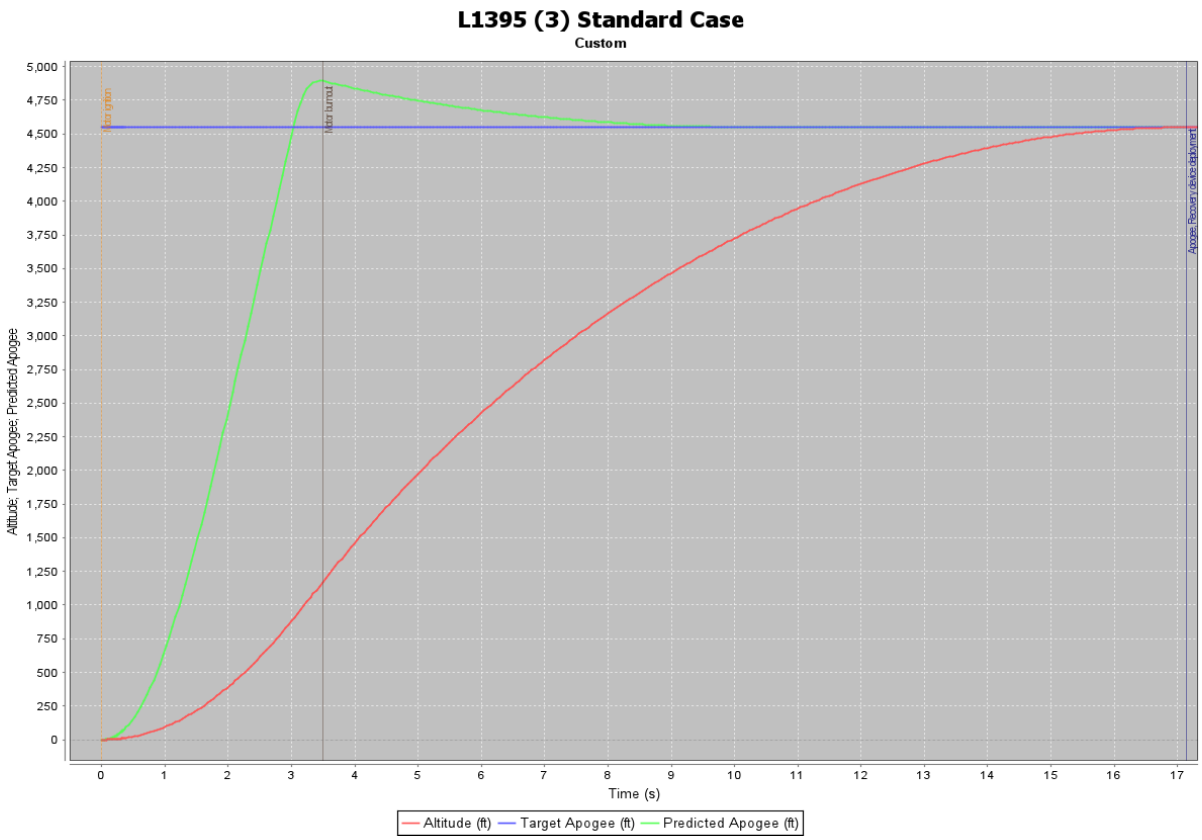


Figure 6-94 Guided Altitude Profile

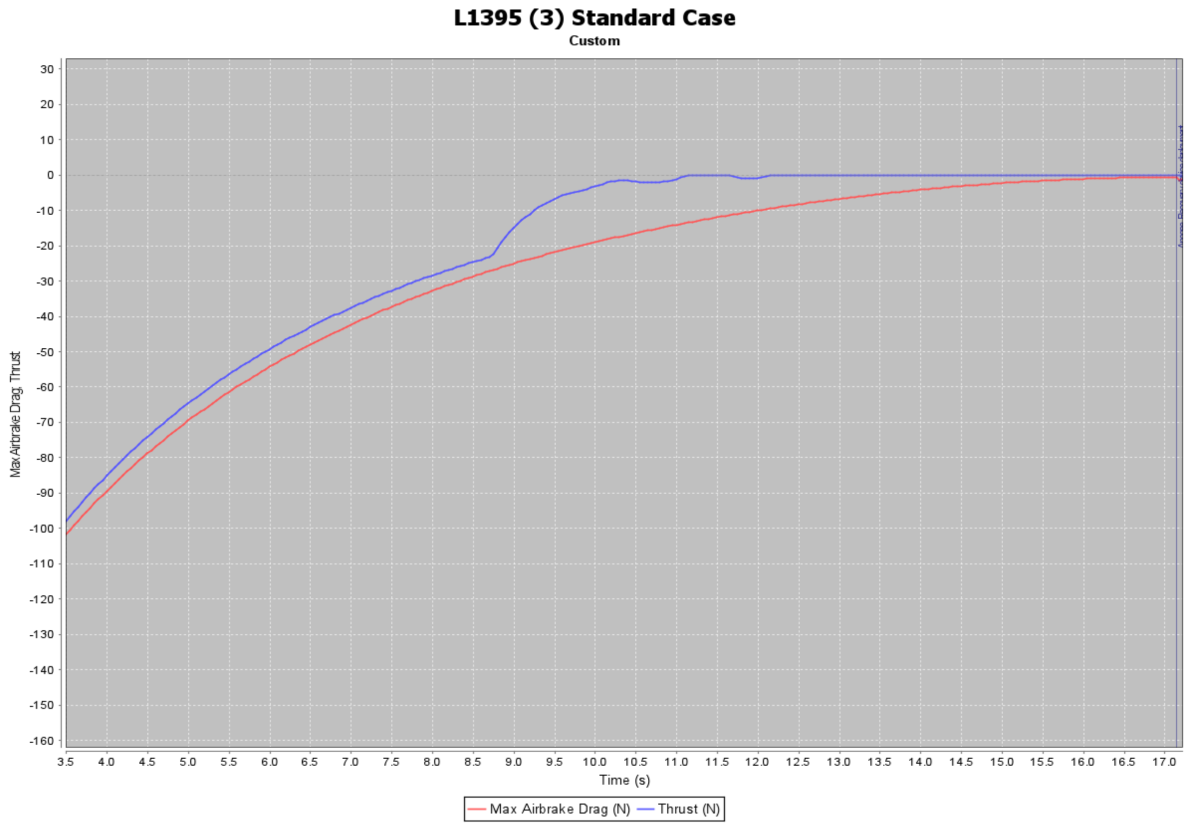


Figure 6-95 Guided Airbrake Drag & Maximum Drag

As shown in Figure 6-94 and Figure 6-95, the predicted apogee decreases after burnout to the target apogee, in response to the changing drag force from the airbrakes.

6.7.3.3 Firmware Implementation

One of the primary reasons for selecting a PID controller was the ease of implementation in low level languages like C++. The primary difference is that the firmware controller operates with a physical system rather than a simulated one. Unlike the plugin controller, the desired drag can not be produced instantaneously due to the actuation time of the servo, which will have to be implemented into the simulation to accurately tune the system. This will slow down the operation and may introduce inaccuracies.

To convert from desired drag to the PWM signal sent to the servo, the current dynamic pressure acting on the vehicle is calculated based on the current altitude and velocity. Using the equation of a surface fit to CFD simulation results, this dynamic pressure and required drag are used to solve for percent extension of the airbrakes. Finally, this is converted to rotation degrees by multiplying by a constant relating extension percentage to degrees rotated before being encoded in the duty cycle of a PWM signal that is sent to the air brake system.

Because of the simplicity of the control system, it is relatively straight forward to implement on the final system. Within the software, the controller will be written as a subsystem that interfaces with the sensors required for state estimation as well as the servo for the airbrake. Because the Teensy 3.2 chip is a 32-bit chip, rather than a slower 8-bit chip, it is able to complete the floating-point computations quickly. It is estimated that the time to complete one iteration of the final loop will be on the order of a few milliseconds which is quick enough to not impact execution of other code.

7 Payload Design

7.1 Payload Summary

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 outer diameter by 8.61in long and weighs 5.68lbs.

Figure 7-1 Payload Upon Landing

Figure 7-2 Payload Self Righting

Figure 7-3 Payload Stabilizing

7.2 Retention and Deployment

7.2.1 Design Overview

The four goals of the payload's retention system are to safely retain the payload during flight, release the payload from the launch vehicle at 1000ft AGL, deploy a parachute for a safe descent of the payload, and release the parachute upon a successful landing. Each of these four goals can be organized into four sections, one for each goal.

7.2.1.1 Alignment Pins

As mentioned in the section 6.3.1, there are four ¼ inch standoffs located in the upper nosecone bulkhead in Figure 7-4. The payload vehicle itself has four corresponding alignment holes located in the camera offset ring located near the top of the vehicle as seen in Figure 7-5. The offset ring is 3D printed out of NylonX and is used to protect electronics while elevating the camera above the vehicle's main body. The four alignment standoffs will slide about a ½ inch into the camera offset ring until they bottom out on the topmost carbon plate. This carbon plate will take the load of any rapid accelerations and decelerations during flight. The purpose of having four alignment pins enables us to lock the payload rotationally during flight. At apogee, the payload is pulled out of the rocket's body tube upon main separation. The wiring diagram for the shock chord can be seen in Figure 7-6.

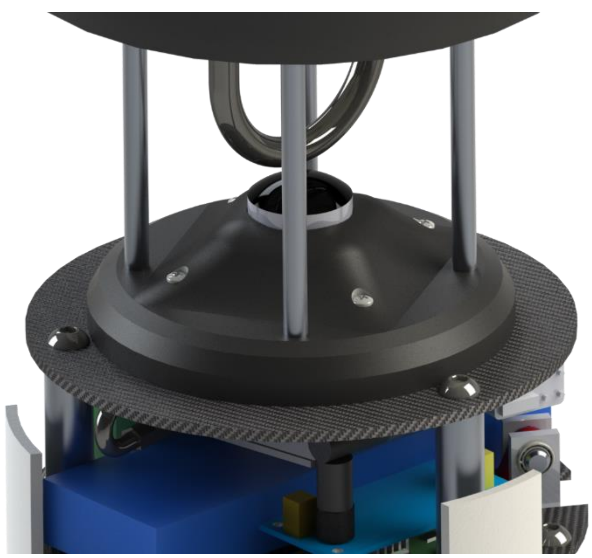


Figure 7-4 Standoffs Inserted Into Payload

Figure 7-5 Corresponding Alignment Holes

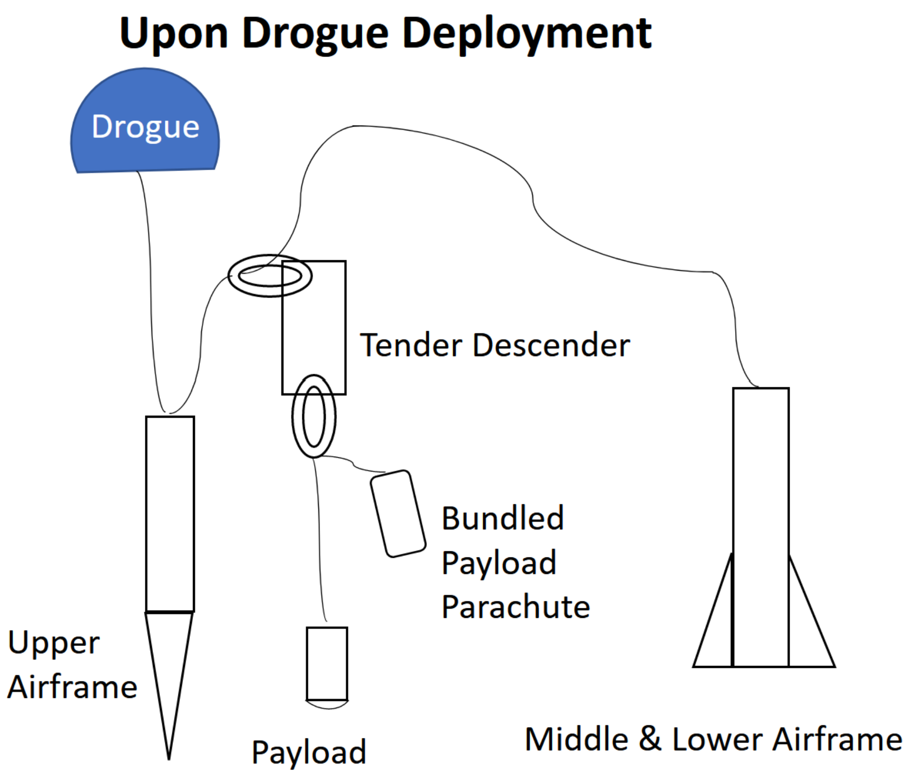


Figure 7-6 Wiring Diagram After Apogee.

7.2.1.2 Payload Releases from Launch Vehicle

The purpose of the Tender Descender assembly is to disconnect the payload from the rocket. This will be achieved via a TinderRocketry L3 tender descender.

A Tender Descender allows two sections of shock chord to be separated via quick links and the detonation of a black powder charge. The Tender Descender's black powder charge will be controlled by a PerfectFlight StratoLogger CF and altimeter. Turnigy Nano-Tech 300mAh 2S 45~90C LiPo Pack supplies power to the subsystem. For safety, there will be a EggtimerRocketry mini-WiFi switch to arm the altimeter.

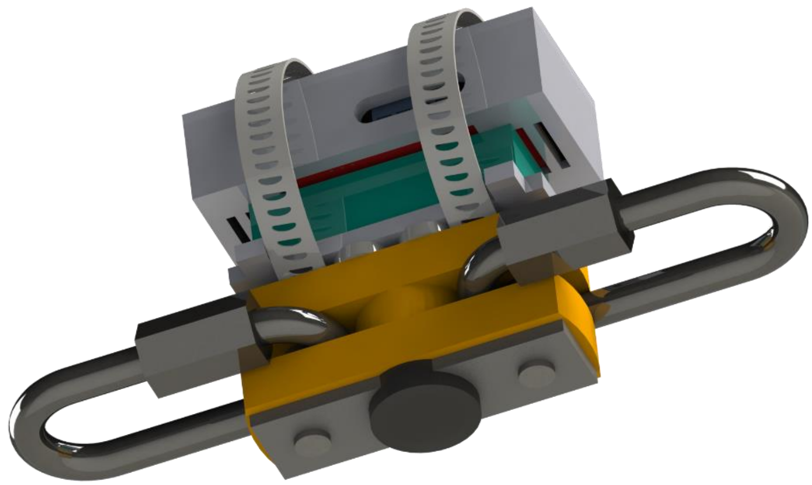


Figure 7-7 First design of assembly

Our initial design was to have all the electronics connected to the Tender Descender; this however had to much difficult machining. In our current design, the electronics are clamped on to the shock cord a few inches from the Tender Descender. This relieves the need to mechanically alter the Tender Descender. To best achieve the design goals, two flat rectangular plates are used to clamp the structure to the shock cord. The clamping allows the electronics to be closer to the Tender Descender which keeps the exposed wires to a minimum. Using slots in one of the clamps a battery will be secured using zip ties. On the other clamp six stand offs will be securely bolted in using heat-set threaded inserts. These stand offs will hold the StratoLoggerCF and the mini-Wi-Fi switch. The electronics are housed in a 3D printed NylonX cage that has a removable top for easy maintenance.

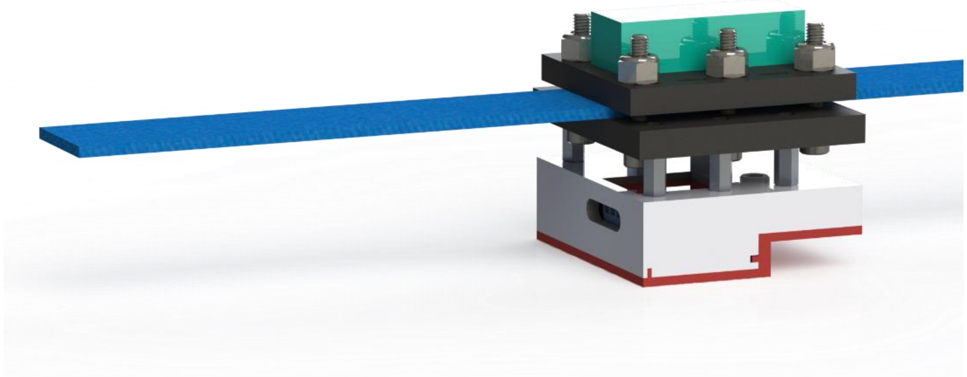


Figure 7-8 Current design of assembly

To detonate the black powder charge of the Tender Descender, some criteria have to be met. Before launch, the StratoLoggerCF has to be turned on. This is done through the Wi-Fi switch that is initially open and is closed shortly before launch. Once this happens the StratoLoggerCF's altimeter must sense that the payload is at the correct altitude (1000ft AGL) during the decent. After these steps happen the Tender Descender's black powder will detonate via an e-match.

Figure 7-9 The tender Descender before detonation (left) and the tender Descender after discharge (right).

Initially the Tender Descender is fully connected. However, after the black powder is detonated the gasses will push up a central peg and pins holding one of the quick links in upwards. These two configurations can be seen in Figure 7-9. This allows the Tender Descender to release the shock cord and separate the payload from the rocket as in Figure 7-10.

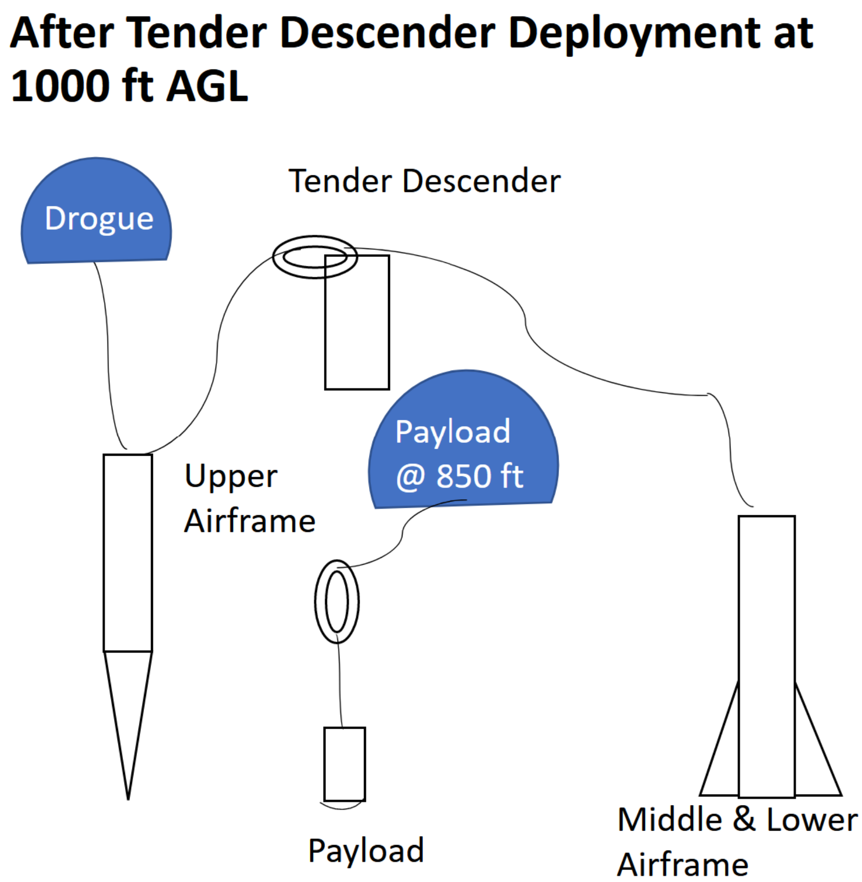


Figure 7-10 Wiring Diagram After Payload Release

7.2.1.3 Jolly Logic Chute Releases

Once the payload vehicle is released as described in section 7.3.2, the payload will enter a free fall for 150 ft AGL until it's two Jolly Logic Chute Releases trigger at 850 ft AGL. The Jolly Logic Chute Release units use a rubber band retainment system to keep the parachute folded until the desired altitude is measured as in Figure 7-11. We are using two chute releases for redundancy in the event that one fails to release the parachute. The parachute released will be a 5ft Rocketman Star PolyConical Parachute. This parachute will give us an estimated decent rate of 15fps. The size of parachute was chosen based on the mass estimates of our payload vehicle. The parachute style was chosen due to the high stability and low oscillation that PolyConical parachutes provide over other options.

Figure 7-11 Jolly Logic Chute Release & Parachute Bundle

7.2.1.4 Parachute Detachment

Once the electronics onboard the payload register that the vehicle has landed, the parachute and shock chord will be released via an electronic rotary latch mechanism. The purpose of releasing the parachute from the vehicle is to ensure that the shock chord and parachute-shroud lines can drift away from the vehicle before it begins the self-righting procedure. Detaching the parachute also reduces the risk that the parachute lands on top of the payload vehicle and obstructs the view of the 360-degree camera. The shock chord that is connected the payload to its parachute and subsequent systems is tied off at a metal quick link. This quick link is secured to the payload through an electronically controlled rotary latch as in Figure 7-13. The SouthCo R4-EM-R21-162 rotary latch was chosen for its electronic actuation and high-strength metal construction. Built for use in aviation and automotive products, the latch has proven simple to control electronically. From the technical drawings SouthCo provided, the rotary latch is rated to a maximum lateral tensile load of 6770 N (1522lbf). We plan to test the load bearing characteristics of the latch with a physical drop test. We have already tested the latch's ability to hold and release while under a load equivalent to 5 lbs.

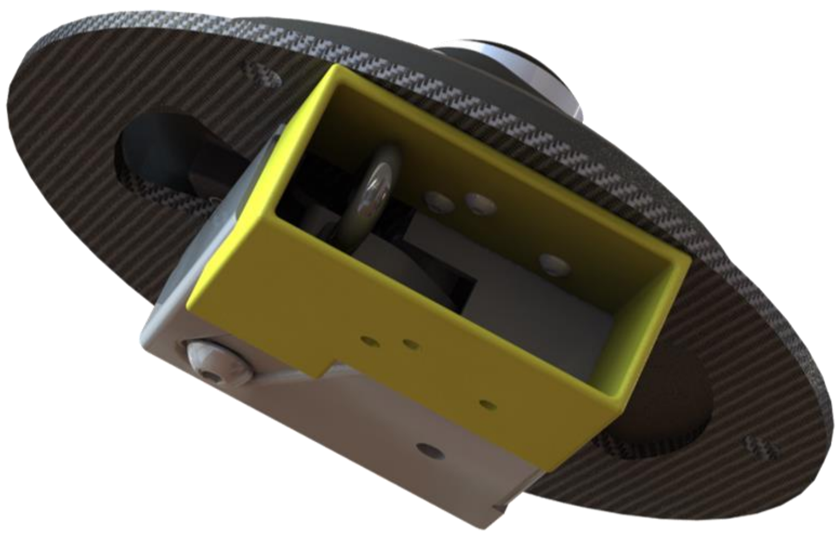


Figure 7-12 Payload Upper Section with Rotary Latch.

7.3 Self-Righting

7.3.1 Design Overview

The payload self-righting system was designed to be able to bring the payload to an upright position from any landing orientation using three petal arms positioned equidistantly about the perimeter of the payload's baseplate, with each petal attached to individual drive systems. Upon landing on the ground, the drive system activates and begins to lower each petal until all three petals have fully deployed, by acting as lever arms against the ground these petals bring the payload to an upright position. Two sensors are involved in this process, the first is a limit switch which will detect whether the petals are in the stowed position before they begin deployment and the second is a potentiometer to measure the rotation of the petals as they deploy and relay this information to the payload control system for finer leveling adjustments.

Most parts will be 3D printed entirely out of Nylon-X. Nylon-X was chosen due to its availability in 3D printing and its high tensile strength when compared to other nylon materials making it optimal for use in the manufacturing of parts which will experience a constant stress such as the petal arms and the gears of the drive system. The petal arms, input and output gears of the drive system, baseplate mounting for the entire apparatus, and the input and output gears of the potentiometer are all 3D printed from Nylon-X. The petal arm was printed as one body with the output gear of the drive system and the input gear of the potentiometer included. The other two gears are also printed out of the same material such that all gears maintain a similar hardness and are unlikely to strip during operation. The motor, drive system, sensors, and petal arm are attached to the printed mount connected to the payload baseplate with the sensors and driving motors connected to the payload control system.

7.3.2 Drive System

Each petal of the self-righting system is driven by a 25mm outer diameter 19RPM Actobotics Econ Gearmotor. This motor has a 499:1 gear ratio providing approximately 167 in-lbs of torque at stall current. Due to packing requirements, the motor is mounted almost vertically and a 2:1 bevel gear drive is used to transmit this torque to the petal. This system provides enough torque for each petal to right the payload independently with the motors operating at a safe fraction of their stall current. With the motor gearing as well as the bevel gear system, each petal is driven with approximately 84 in-lbs of torque with the motor operating at 25% stall current. Given a payload mass estimate of 4.9 lbs and a petal arm length of 6 in, slightly under 30 in-lbs of torque is needed to right the payload. This gives a factor of safety of 2.8 in the worst-case scenario of only one petal being in contact with the ground.

The bevel gear connected to the motor drive shaft is 3D printed as an independent part, and the second bevel gear is incorporated into the structure of the petal, which is also 3D printed. As mentioned in the design overview, these gears are printed from Nylon-X. Because the gears are manufactured from identical material and will experience a few cycles, there is very little risk of the gears stripping or experience other fatigue damage.

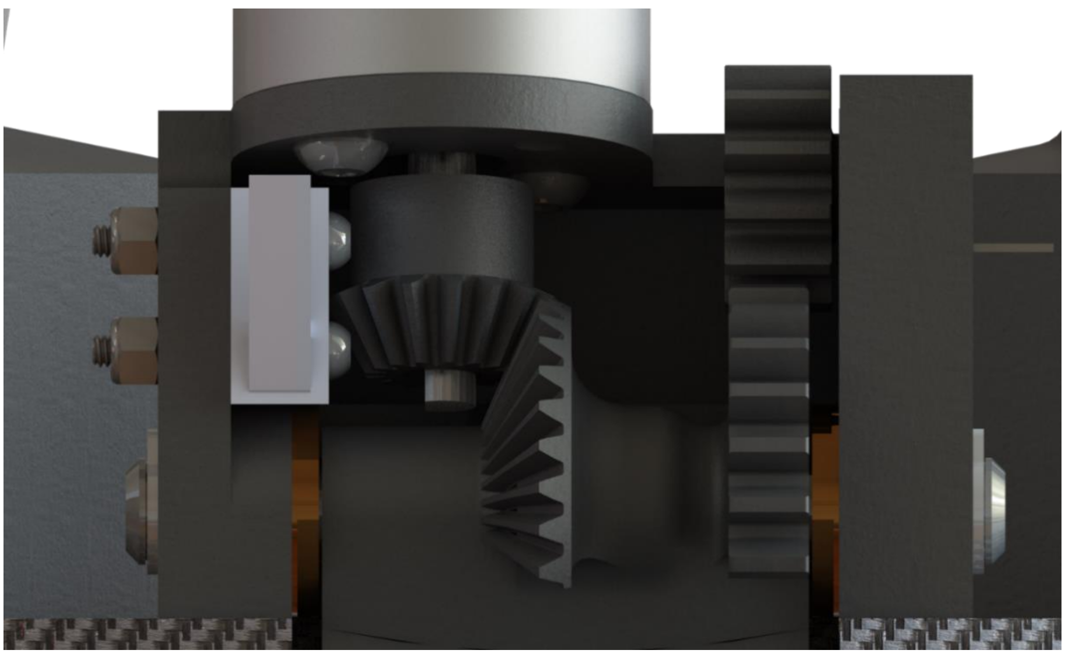


Figure 7-13 Front view of the motor mount housing the drive gears along with both sensors.

To mount the motor in its vertical position and contain the drive system, a motor mount has been designed to be 3D printed. In addition to holding the motor in place, the mount houses the rest of the self righting system. A hinge for the petal is incorporated into the motor mount, which allows for a hinge pin to be inserted through the petal. This hinge pin is secured to the motor mount via set screws inserted behind the pin as well as retaining rings on the outer edges of the mount. A flange sleeve bearing facilitates smooth rotation of the petal. A hollow space beneath the motors accommodates the drive gears and potentiometer gears. The inner side walls of this cavity provide a mounting surface for the sensors required to control the self righting system.

Figure 7-14 Side view of the motor mount with all components attached.

7.3.3 Sensors

There are two sensors integrated into the self-righting mechanism itself: a potentiometer and a limit switch. Both are integrated into the overall combined motor mount / hinge bracket. Both sensors measure the position of the righting petals. The potentiometer is a Bourns 10kΩ Trim Potentiometer and the limit switch is a Mouser UP01DTANLA04 Micro Snap Action Switch. The potentiometer is housed in a toleranced slot and secured by a cover affixed with dovetail joints. The limit switch is bolted to the mount with integrated holes in the switch.

The potentiometer measures the angle of extension of the petal indirectly through a 2:1 gear increase. The drive gear is integrated into the righting petal and the driven gear is 3D printed and affixed to the stem of the potentiometer. The gearing creates more resolution as the potentiometer can rotate through its entire sweep even though the petal only rotates approximately 100 deg. The change in resistance from the potentiometer changes the voltage read by the payload computer and indicate a particular extension so all petals can be synchronized as well as detect if a petal may be on the ground if the landing site is uneven. The gearing is offset to the bracket's own left, and the potentiometer is seated in the left wall of the bracket between two C cutouts and kept in place with a flat cover that slots into a dovetail joint on each side of the potentiometer.

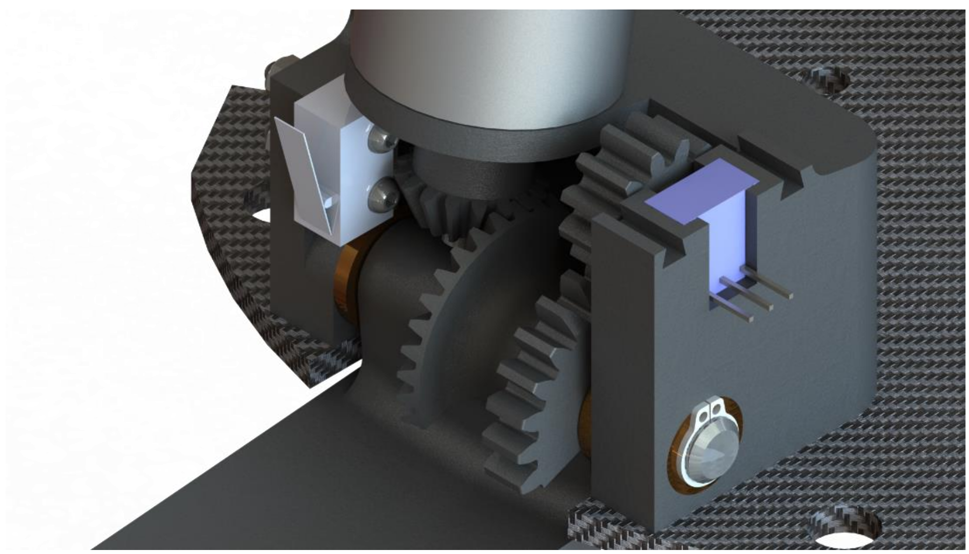


Figure 7-15 Isometric view of potentiometer mount (lid excluded for clarity)

The limit switch tells the payload computer if the petal is closed completely into the stowed position. The lever on the switch is directly activated by the body of the petal when it is fully upright. The limit switch is attached by M2 bolts through its integrated mounting holes to corresponding holes in the hinge bracket's right side.

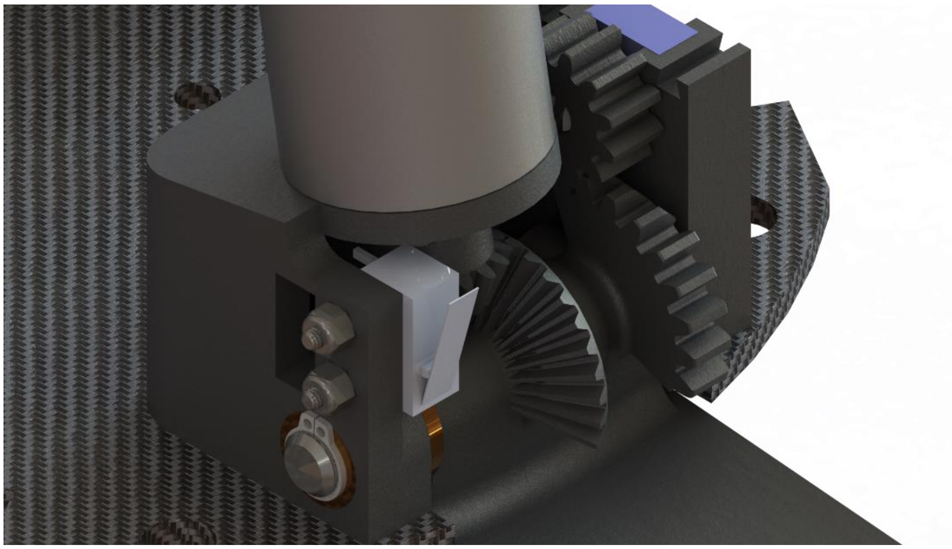


Figure 7-16 Isometric view of limit switch mount

7.3.4 Prototype

To validate the fundamental operation of the designed self righting, a complete prototype was built and tested. The main difference between the final system and the built prototype is that all printed pieces were printed out of PLA rather than the final material of Nylon-X. PLA was chosen due to a combination of its higher availability, lower cost, and ease of printing. While it would have been ideal to prototype with the chosen material, the prototype was not intended to bear as much load as the final self righting mechanism so the inadequate material properties of PLA were acceptable in this case. The other major change in the prototype is the shaft system used. As there was limited access available to the machines needed to cut the shafts as designed, the prototype used a shoulder bolt in place of a machined shaft. Although the shoulder bolt did not have the desired mounting features, the same reasoning of less expected loads was applied.

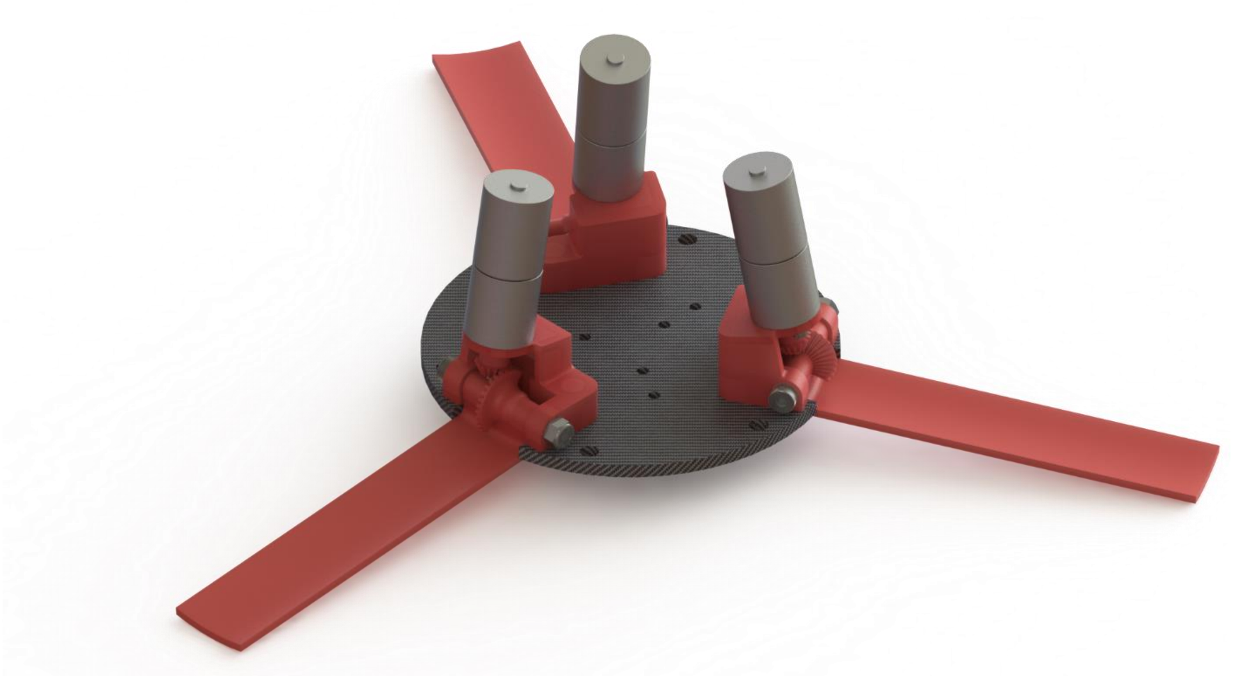


Figure 7-17 Isometric view of the prototype self righting system as designed.

Once assembled, the prototype was able to rotate the petals throughout their entire range of motion while extending, however they were not able to be retracted. The cause of this was determined to be printing artifacts on the top of the petal's bevel gear due to the printer over extruding on that final layer. The issue was remedied by extending the gear teeth all the way to the shaft of the petal to remove the top layer of the gear and once all 3 petals were re-printed, the prototype was able to fully extend and retract the petals with ease. Most importantly, the prototype showed the validity of the designed mechanism and its ability to right the mock payload by extending the petals.

Figure 7-18 Self righting prototype extending petals.

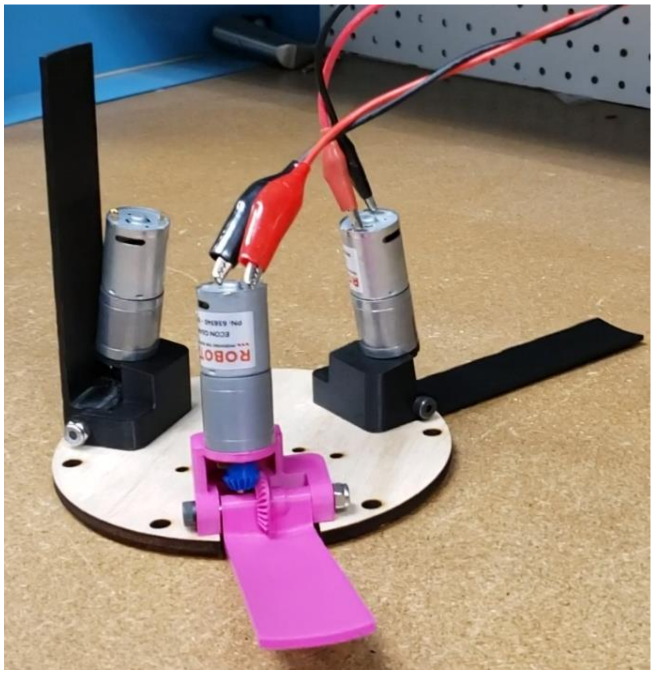


Figure 7-19 Self righting prototype with petals fully extended and righted.

7.4 Electrical System

7.4.1 System overview

The payload lander is fitted with a custom electrical system composed of commercially available sensors and actuators needed to complete its designated mission. These components control their respective subsystems designed to allow the payload to stabilize after landing, self-level within specifications, and transmit a captured panoramic photo back to the ground station. The actuators include the self-righting motors, stabilization servos, and rotary latch for parachute release, details for which can be found in their corresponding subsystem sections. The sensors and processors are almost identical to those of the launch vehicle avionics system with the addition of a Raspberry Pi Zero interfacing with a panoramic camera and GSM (Global System for Mobile communications) board which the team intends to use for transmitting the captured photo to the ground station. After all functionality is verified in testing the sensors with be combined into a single PCB to significantly minimize form factor and increase robustness given less overall wiring. This PCB is also used in the launch vehicle avionics will include additional ports for external peripherals such as the motor signaling wires and the Raspberry Pi connection. The details of this custom electrical system can be found in Section 6.7.1.

An important change to the electrical design of avionics systems developed by the team requires all devices to execute the same software utilizing the same processor and sensor peripherals. This enabled testing and development to be mostly combined between both the payload electrical and launch vehicle avionics sub teams, allowing for a more distributed work effort and fully modular processing board. The result is the same avionics computer PCB running a common code base, the lander program being an extension of the launch vehicle program to handle post-flight operations such as the state machine of its mission criteria.

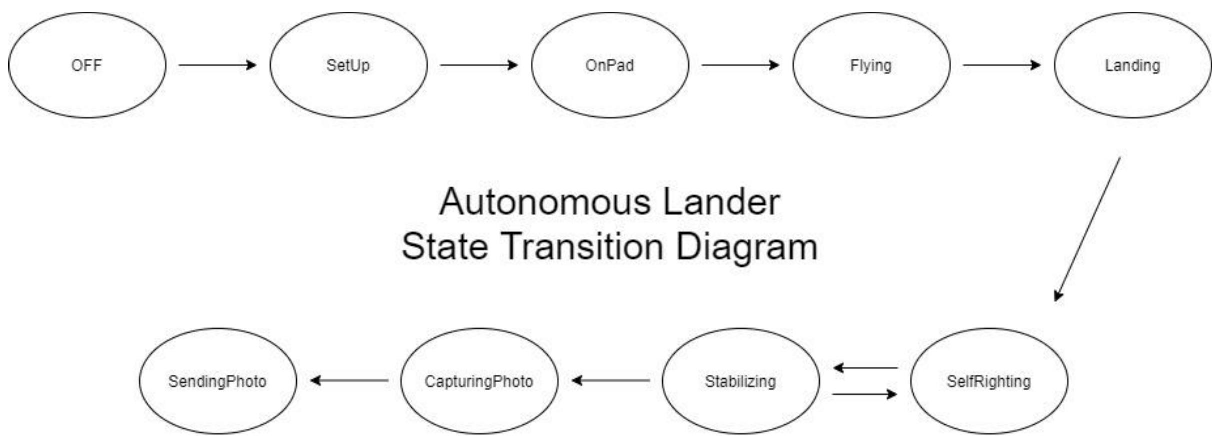


Figure 7-20 High level lander state transition diagram

7.4.2 Power management

Additional electronics specific to the payload lander are the power management systems. External motor and servomotor regulator boards were tested and selected for their compact size and successful use in previous robotics projects developed by the team. The servomotors will each be powered using a BEC (Battery Eliminator Circuit) providing 7.4V for 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. The payload lander will be powered using a Turnigy Nano-tech 12V 2.5Ah LiPo battery selected primarily for its physical size and capacity. With this battery and the low expected idle power required by the lander it will be over 24 hours until fully depleted.

Figure 7-21 3S 2.5Ah LiPo battery used by the lander

7.5 Photography System

The photography system will use the PICAM360 which was selected as this camera possesses a full 360-degree horizontal field of view and a 235-degree vertical field of view. This design allows for a single camera to be used rather than multiple imagers combining to form a single picture as the latter method would undesirably require greater size and computational complexity be allocated to this system. As was intended by the manufacturers, a Raspberry Pi Zero W pictured below will be used to interface with the camera.

Figure 7-22 Raspberry Pi Zero W

This variant of the Raspberry Pi computers was selected for its small form factor and ease of use with a breakout board to transmit the captured photo over MMS (Multimedia Messaging Service). The camera will be mounted atop the payload lander in a 3D printed PLA casing to protect it from impact or vibration while landing. This same casing will additionally house a relatively large GPS antenna underneath the camera to ensure the best connectivity by raising it above ground as the lander levels itself upright.

Figure 7-23 Camera Housing

7.6 Stabilization

The payload Stabilization system's objectives are to bring the payload body to within five degrees of vertical, and to provide a stable and elevated platform for the payload camera to take a panoramic picture. The stabilization system achieves this through three modules evenly distributed about the payload's centerline, one of which is shown in Figure 7-24below. Once the self-righting system rests the payload on its base plate, the stabilization system lowers its three legs through the use of three four bar linkages. During leg deployment, a folded foam composite foot located at the bottom of the leg unfurls to provide extra traction for the leg. A potentiometer relays the position of the leg to the central processor, and under its control, the three modules work together to rotate the payload to the proper orientation as determined by onboard accelerometers. Once the payload is in the proper orientation, the stabilization system holds it in place until the panorama is taken and the mission ends.

Figure 7-24 An isometric view of the payload top assembly. Stabilization modules in stowed configuration are highlighted in red.

7.6.1 Lift Mechanism

The payload leg is actuated by a parallel four bar linkage, as shown in Figure 7-25 below. For the rest of this section, each link will be referred to by their proper names. The crank is the link in red at the top of the assembly, the coupler is in blue on the left, the followers are in red near the bottom of the assembly, and the base link is created by the large base piece in blue. Like-colored pieces have equivalent link lengths; the link length of the base is measured between the centers of the two shaft holes. The red links have lengths of 2 inches, and the blue links have lengths of 2.375 inches. The coupler extends another 2.375 inches downwards and terminates in the foot assembly. As of the time of writing, the height of the foot assembly has not been determined due to a lack of adequate testing equipment. However, the maximum thickness of the entire foot assembly has been determined to be 0.4375 inches.

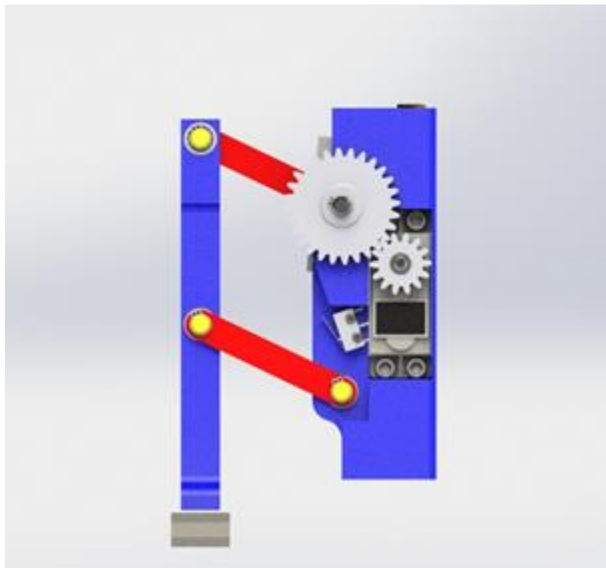


Figure 7-25 A right-hand view of a single stabilization module. Crank and follower links are highlighted in red. Coupler and base links are highlighted in blue.

Due to a lack of access to the Washburn machine shops, there is some ambiguity as to the material the red links will be made of. If access is restored, they will be manufactured from 6061-T6 aluminum alloy. A solid ½ inch thick crank link gives a satisfactory factor of safety, so lightening may be done to save some weight. If access to Washburn is not restored, the red links will be made of NylonX. Since the print settings will affect the mechanical properties of the final piece, some testing is required before the printed parts are allowed on the final payload.

Since PDR, the design of the lift mechanism has changed little apart from minor weight-saving changes. The base and coupler have been slimmed down in multiple places, which reduces the filament needed to print, and therefore the cost to print these parts.

7.6.2 Drive System

As determined through a SolidWorks Motion Study in Figure 7-26 below, the maximum amount of torque required to actuate the leg against a 5 lbf load acting vertically is approximately 10 in-lbf.

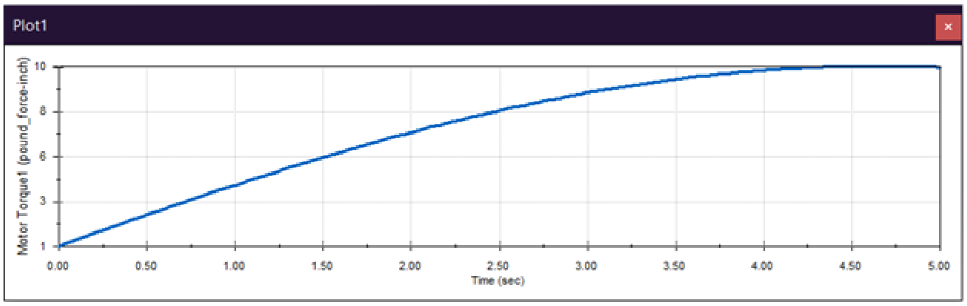


Figure 7-26 A plot of the motor torque at the crank shaft as it rotates at 3 RPM. Plot produces from a SolidWorks Motion Study early in the design process.

To create this motion, the linkage was connected via a 2:1 geartrain to a goBILDA 2000 Series Dual-Mode Servo as shown in Figure 7-27 below. When run at 44 RPM and 7.4 volts, the servo outputs 5.16 in-lbf at 0.86 amps, or 28.6% of stall current. With the geartrain increasing the torque at the crankshaft by a factor of 2 with minimal friction losses, the torque at the crankshaft is enough to counteract the force of the payload. If the maximum allowable current through the servo is 50% of stall current, (1.5 amps) then the factor of safety of this motor is 1.75. According to the mass budget at the time of writing, the payload's current mass is 5.682 lb. While this is greater than the earlier estimate, it is within the safe working limit for the servo.

Figure 7-27 A right-hand view of the geartrain of the lift mechanism. Note the servo driving the 14-tooth servo on the right-hand side of the render

An analysis was also performed on the gears to make sure they could support this load without fracturing. The geartrain consists of a 28-tooth and a 14-tooth gear; the former is mounted to the crank shaft, and the latter is mounted to the servo spline. Depending on whether access to the Washburn machine shops is reinstated, these gears may both be made of either 6061-T6 aluminum alloy or 3D-printed NylonX filament. The stress on each of the gears was found by creating a rectangular section roughly normal to the tooth surface at the gears' contact point. This section is shown in Figure 7-28 below as a red face on the 14-tooth gear. A pressure angle of around 22 degrees was found. Since the 28-tooth gear's pitch circle is 2.8 inches in diameter, the force tangent to the pitch circle from the 10 in-lbf torque is 7.14 lbf. By decomposing the force and dividing the normal and shear forces by the area of the section (0.152 in^2), the stresses on the section are σ = 181 psi, τ = 433 psi. Since the yield strength of 6061-T6 aluminum is 40000 psi, the aluminum gears will not deform an appreciable amount during normal operation. For NylonX gears, the print settings will affect the mechanical properties of the gears, so testing may be required to determine the proper settings for the parts.

Figure 7-28 A skewed view of the stress plane of the driving gear. The section in red is offset 22.68 degrees from the plane tangent to both gears' pitch circles.

Since PDR, the drive system has remained relatively unchanged, except for the addition of a key system to the 28-tooth gear. This change was made so the system worked both inside and outside of CAD.

7.6.3 Foot System

Attached to the bottom of the coupler is the foot system shown in Figure 7-29 and Figure 7-30 below, which provides a compliant tractive surface for the stabilization system. The foot consists of a 1/16-inch polycarbonate layer that bends easily. Adhered to the bottom of the polycarbonate hinge are three three-layer foam sandwich composites. A removable compliant traction layer that lies just below the foam composite completes the assembly. A low-profile bolt attaches the foot to the coupler; the connection is permanent, so long as the foam composite stays in one piece. The purpose of the foot system is to provide a secure support for the stabilization system, which in turn reduces the amount of time required for the payload to be stabilized. The system achieves this by conforming to most surfaces it rests on. By increasing the surface area in contact with the ground, we hope to increase the traction provided, especially on surfaces such as snow or loose dirt that that may give way when high pressures are provided. The system also folds out to increase the surface area in contact with the ground by a factor of three. This is achieved by a thin polycarbonate plate which is initially folded into a U shape before deployment. The plate and its attached foam composite are kept in place by the two structural beams that flank each stabilization module. Once the coupler is deployed far enough, the foot moves away from the beams and deploys from the elastic forces inherent in the polycarbonate. This is an improvement over the original design that included a spring-loaded hinge system due to the reduced mechanical complexity and cost.

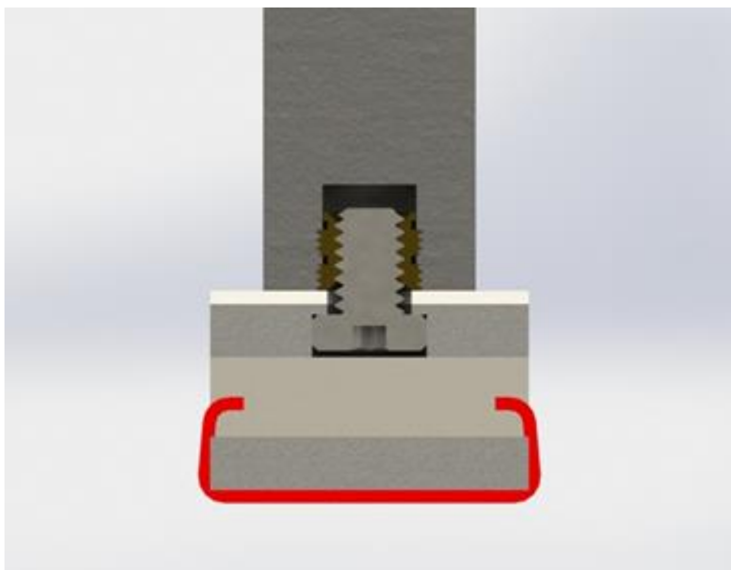


Figure 7-29 A section view of the foot system in the deployed position. Note in white the polycarbonate plate, the darker polyethylene layers, the beige polyurethane layer, and the red traction layer.

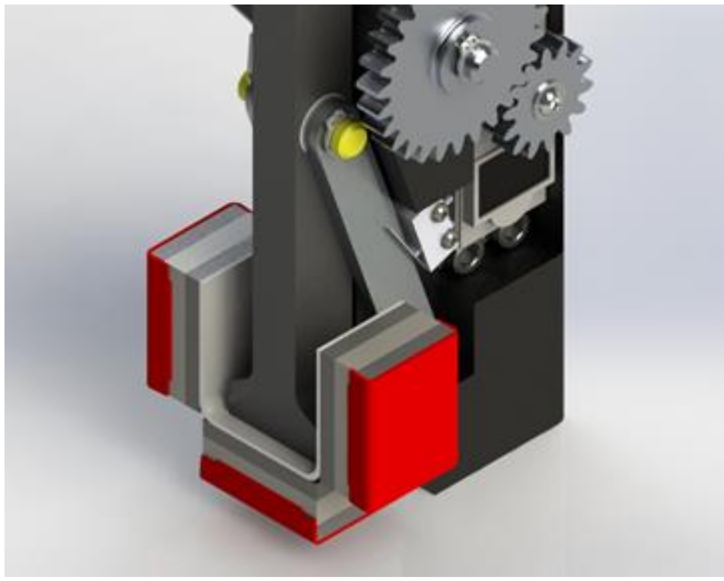


Figure 7-30 An isometric view of the foot system in the stowed position. The polycarbonate plate is bent into a U-shape by two structural beams, which can be spotted in the full payload render in Figure 0.1.

Since PDR, the foot has been the focus of significant testing efforts. The materials of cross-linked polyethylene for the outer layer and polyurethane for the inner layer were determined after testing 16 combinations of foam obtained from a sample pack. Each composite was tested qualitatively for a balance of flexibility and compressive resistance. Then, they were compressed against an uneven surface to test how well they conformed to it. At the end of these tests, two adequate combinations were found. The primary combination was the polyethylene and polyurethane combination above. The secondary kept the polyethylene as the outer layer and had a GUM, EVA, and neoprene foam center. While the latter is more compact and performs almost the same as the former, it is more complex to assemble, which makes it a less attractive option. As mentioned earlier, the spring-loaded hinge system decided upon in PDR has been substituted for a thin polycarbonate plate that provides a similar action to a hinged plate, but at a much lower cost and complexity. The traction layer will be made of compliant CheetahFlex filament and will be mounted to the foot through the use of a slot-and-tab system. A slit will be cut into the polyurethane layer, and the tab of the traction layer will be inserted. This will be adequate for the foot's purposes, as it experiences mostly compressive forces or lateral forces in the radial direction. This system also allows the foot to be changed easily, which aids in testing and assembly, and also allows for multiple feet to be designed for different terrain. Given the orientation and mounting of the tabs, these forces will not unseat it from its position unless they are larger than expected. Testing should be done to find the maximum forces the layer can withstand.

7.6.4 Sensors

For the stabilization modules to be controlled by the central processor, their position must be known at all times. This is handled by a Bourns 38SB-1RB-104 100 kilo-ohm rotary potentiometer and a Mouser UP01DTANLA04 Micro Snap Action Switch. The latter is the same model as the limit switch on the self-righting system, which reduces costs from the switches themselves and the associated mounting hardware. The potentiometer is mounted to each module by a polycarbonate plate as shown in Figure 7-31 below. The mounting plate will come from the same stock as the foot system's polycarbonate plate.

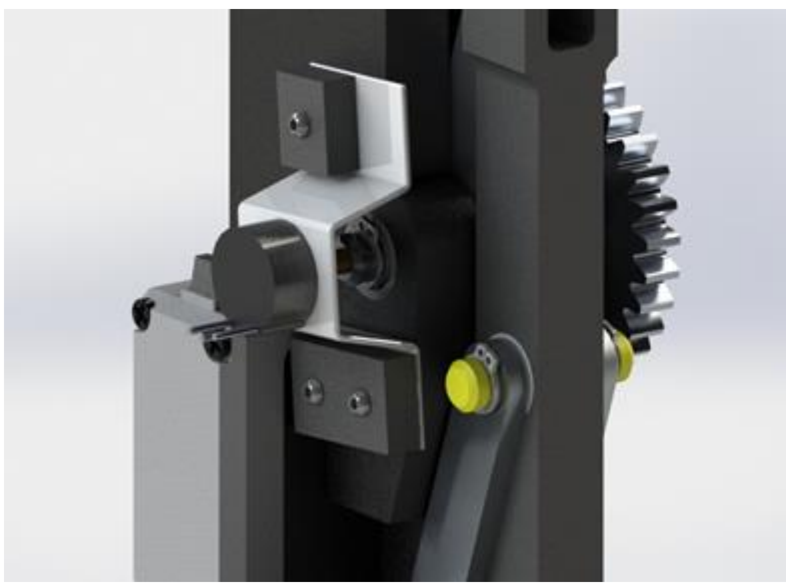


Figure 7-31 A skewed view of the potentiometer mounting system. The potentiometer is mounted to a polycarbonate plate using the nut this particular model ships with. The potentiometer can then interface with the crank shaft.

This particular model allows for 220 degrees of rotation, which is significantly more rotation than the crank shaft's maximum rotation of around 130 degrees. The crank shaft has also been modified to allow a connection with the potentiometer's flatted shaft. This allows the potentiometer to record the angular position of the crank as is required. The limit switch, which can be seen in Figure 0.7 just above the right section of the foot, is used to zero the value for the crank. In combination, these sensors accurately provide the position of the crank, which can be used to calculate the extension of the coupler and to control the stabilization systems.

8 Safety

WPI HPRC is dedicated to creating and maintaining a safe environment at all times. This includes the safety of both team members and others. 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.

8.1 Final Assembly and Launch Procedures

Clearly establishing final assembly and launch procedures is vital to the team's success in completing the mission. The checklists below detail all the steps that must be completed prior to each launch and after each launch. Some important steps require a specific officer to complete or verify; such steps have the appropriate officer’s position listed next to them. WPI HPRC understands the importance of following these checklists to ensure the safety of all team members and bystanders at launch events. First, they ensure that all parts of the launch vehicle and payload are accounted for. Secondly, it helps with preparing our launch vehicle and payload, since it shows all the steps that need to be done. That way we eliminate the risk of forgetting to do something. If a step were to be missed, the safety of the entire mission could be compromised. This misstep could also lead us to fail during the mission, so it is imperative that each of the steps on the checklists be followed carefully. All steps that require PPE are labeled with the proper PPE required to complete them.

Recovery Bay Checklist	
Task	**Verified by (position listed if necessary)**
Setup	
Ensure all wires are properly connected, and that the charges are wired to the correct terminals.	
Ensure batteries are fully charged using a battery tester.	
Ensure structural integrity of recovery bay. Check for loose screws or washers.	
Power on altimeters without charges connected to verify functionality.	Launch Vehicle Lead
Power down altimeters and disconnect batteries for loading and integration.	
Integration	
PPE needed to complete the following steps: safety gloves and safety goggles.	
Test e-match continuity.	
Load disconnected e-matches into charge wells.	
Measure out black powder charges according to values calculated and verified by ground ejection test.	Mentor

Load charges into designated charge wells. Stuff with dog barf and cover completely with a single layer of masking tape.	Mentor
Connect batteries to altimeters. Ensure the altimeters are not active or armed.	
Connect terminal blocks to altimeters.	
Verify charges wired correctly, that black powder is secure in charge wells, that the altimeters are not powered, and that batteries are connected.	Safety Officer
Assemble Recovery Bay into coupler, and attach bulkheads using thumb screws.	
Connect quick links of shock cord to recovery bay U-bolts on either side.	
Pack parachutes and shock cord into upper and middle airframes.	
Before integrating recovery bay with airframes, check that terminal blocks have no live current.	Safety Officer
Attach e-matches to terminal blocks.	
Integrate recovery bay and airframe and install shear pins and radial bolts.	
Ensure vent hole alignment.	
Perform shake test of shear pins.	
After the vehicle is placed on the pad	

Arm StratoLogger.	Launch Vehicle Lead
Confirm StratoLogger beep codes match expected results for continuity and deployment altitude.	Launch Vehicle Lead
Arm Raven 4.	Launch Vehicle Lead
Confirm Raven beep codes match expected results for continuity and deployment altitude.	Launch Vehicle Lead

Table 14 Recovery Bay Checklist

Recovery Hardware Checklist	
Task	**Verified by (position listed if necessary)**
Attach drogue and main swivels to shock cord.	
Verify proper packing of parachutes.	Launch Vehicle Lead
Wrap parachutes in Nomex blankets.	
Attach Payload Tender Descender to shock cord.	
Check shock cord for fraying or damage.	
Bundle shock cord and wrap sections with 1 layer of masking tape.	

Ensure proper attachment of all quick links.	Safety Officer
Attach drogue forward quick link to nosecone, and mount payload to alignment pins.	
Attach upper airframe to nosecone using radial bolts.	
Pack main parachute into middle airframe.	

Table 15 Recovery Hardware Checklist

Avionics Bay Checklist	
Task	**Verified by (position listed if necessary)**
Setup	
Ensure all wires are properly connected.	
Ensure battery is fully charged using a battery tester.	
Ensure structural integrity of Avionics bay. Check for loose screws or washers.	
Power on avionics board and perform test of all sensors, telemetry, and servo control.	
Power off board.	
Integration	

Power on avionics board.	
Insert avionics bay into lower airframe coupler, and twist 90 degrees until locked.	
Ensure alignment of airbrake fins with slots.	
Attach shock cord to avionics bay U-Bolt.	
Pack parachute and shock cord into middle airframe.	
Integrate middle airframe and recovery bay, and insert shear pins into coupler.	
Perform shake test of shear pins.	
Ensure alignment of vent holes.	
Perform test of all avionics board sensors, telemetry, and servo control.	
Set avionics board to low power mode.	
After the vehicle is placed on the pad	
Set avionics board to launch detect.	Launch Vehicle Lead

Table 16 Avionics Bay Checklist

Payload Checklist	
Task	**Verified by (position listed if necessary)**
Verify the structural and mechanical components are in working order including the self-righting petals and stabilization legs.	
Complete Payload Electronics Checklist.	
Power on payload with rotary switch and configure electronics for flight.	
Attach payload parachute and shock cord to payload with rotary latch.	
Ensure that self-righting petals and stabilization legs are folded into closed position.	
Pack payload into the rocket upper airframe.	
Arm jolly logic chute releases, verify chutes set to open at 850ft AGL, and pack into rocket with the parachute.	
Pack tender descender with black powder.	Mentor
Attach battery leads to egg timer and pack tender descender into rocket.	
After the vehicle is placed on the pad	
Arm payload Stratologger altimeter via egg timer WiFi switch.	Payload Lead

Table 17 Payload Checklist

Payload Electronics Checklist	
Task	**Verified by (position listed if necessary)**
Ensure that the payload electronics are wired correctly and that all components are secure.	Payload Lead
Tug on wires, shake unit and pull on the bulkhead to ensure nothing is loose.	
Ensure the battery is fully charged.	
Plug in the battery, power on system, and verify status with power lights on components.	
Test transmitter connection to ground station unit.	

Table 18 Payload Electronics Checklist

Motor Checklist PPE required: Safety glasses	
Task	**Verified by (position listed if necessary)**
Motor Assembly	
Ensure all motor casing components are in working condition and that the reload has not been removed from its package or tampered with in any way.	Mentor
Remove the reload from its packaging and assemble into casing following the manufacturer instructions.	Mentor
Motor Integration	
Verify motor casing and reload are intact and undamaged	Launch Vehicle Lead
Insert motor casing into lower airframe assembly through tailcone	
With the avionics bay removed, screw down the central thrust plate bolt until tight, preventing the motor casing from spinning using the casing wrench	
Verify thrust plate bolt is tight, and that the motor casing does not move when pushed or pulled upon	Launch Vehicle Lead

Table 19 Motor Checklist

Launch Checklist		
Task	**Required Personnel**	**Initials**
Check the weather and wind speed at the launch site to ensure that the vehicle is safe to launch	Logistics Officer	Logistics Officer Log the conditions:
If the vehicle has been flown before, ensure that the Post-Flight Inspection Checklist has been completed. **⚠Operation Hazard: If the vehicle has failed the inspection, it may only be flown after the failure mode has been determined and a mitigation plan has been written, implemented, and verified.**	Safety Officer	Safety Officer
Complete Avionics Bay Checklist except for the "After the vehicle is placed on the pad" section.	Launch Vehicle Lead	Launch Vehicle Lead
Complete Payload Checklist except for the "After the	Payload Lead	Payload Lead

vehicle is placed on the pad" section.		
Complete Recovery Hardware Checklist.	Launch Vehicle Lead	<u>Launch Vehicle Lead</u>
Complete Recovery Bay Checklist except for the "After the vehicle is placed on the pad" section.	Launch Vehicle Lead	<u>Launch Vehicle Lead</u>
Complete Motor Checklist.	Mentor	<u>Mentor</u>
Conduct final visual inspection to ensure launch vehicle is completely assembled. The airframe screws must be fully tightened, shear pins must be inserted properly, and the separation points should be secure.	Launch Vehicle Lead Payload Lead Team Captain Safety Officer	<u>Team Captain</u>
Verify with RSO that vehicle is safe to launch.	RSO	<u>Team Captain</u>

⚠Operation Hazard: The RSO has the final say on the safety of the vehicle.		
Mount launch vehicle on the 1010 launch rail designated by the RSO. If there are high winds, the launch angle may be moved up to 20° from the vertical to compensate. **⚠Setup Hazard: Mounting the vehicle on the launch rail should only occur after the range has been cleared.** **⚠Operation Hazard: The launch angle should never be more than 20° from the vertical. Doing so violates the NAR High Power Rocket Safety Code and risks the vehicle colliding with personnel or objects on the field.**		Team Captain
Complete the "After the vehicle is placed on the pad" section in the Avionics Bay Checklist.	Launch Vehicle Lead	Team Captain
Complete the "After the vehicle is placed on the pad" section in the Payload Checklist.	Payload Lead	Team Captain

Complete the "After the vehicle is placed on the pad" section in the Recovery Bay Checklist.	Launch Vehicle Lead	Team Captain
Secure new ignitor in motor. **⚠Setup Hazard: To avoid premature ignition, do not connect the ignitor to the launch wire in this step.**		Team Captain
Check that the launch wire is not live before connecting the ignitor to the launch wire. Check for igniter continuity. **⚠Setup Hazard: Ensure the ignitor wire is not live before connecting the ignitor to avoid premature ignition.**		Team Captain

Table 20 Launch Checklist

Troubleshooting Checklist	
Task	**Verified by (position listed if necessary)**
Inform the RSO of the issue and follow all instructions given by the RSO.	Team Captain
Remove the launcher's safety interlock.	Team Captain
Wait 60 seconds after the launch attempt before approaching the launch vehicle (as regulated by the NAR High Power Rocketry Safety Code).	Team Captain
Walk to the launchpad and disarm all electronics.	Team Captain
Remove the launch vehicle from the launch rail.	Team Captain
Reinstall the igniter.	Team Captain
Mount launch vehicle on the launch rail.	Team Captain
Re-arm electronics and the e-match igniter.	Team Captain
Retry launching the launch vehicle.	Team Captain

Table 21 Troubleshooting Checklist

Post-Flight Inspection Checklist		
Task	**Required Personnel**	**Initials**
Ensure all components are accounted for. This includes the lower airframe, middle airframe, upper airframe, Recovery Bay, Avionics Bay, nose cone, drogue parachute, 2 main parachutes, Nomex blankets, and the payload (if one was flown).		Safety Officer
Visually inspect the airframe and fins for damage such as dents, zippering, holes, cracks, and anything that would prevent the vehicle from being flown again. This includes checking internal components such as U-bolts and bulkheads.		Safety Officer
Check that all components are attached appropriately. The nose cone should still be secured to its parachute by shock cord. The upper airframe should be secured to the nose cone. The middle airframe should be secured to the recovery bay. The lower airframe should be secured to the avionics bay.		Safety Officer

All four fins should be secured in their slots and should not be able to wiggle.		
Check that the motor and the motor casing are still secured inside of the motor tube and that all ejection charges have been detonated. Properly dispose of the spent motor and ignitors.		Safety Officer
Check that there are no holes or burns in any of the parachutes and that none of the parachute's chords have broken.		Safety Officer
Open the Recovery and Avionics bays and ensure that all components are still secured within it. Visually inspect all electrical components for damage.		Safety Officer
Download flight data from both altimeters.		Safety Officer
If it was flown on the launch vehicle, visually inspect the UAV for damage such as dents, holes, cracks, and anything that would prevent the vehicle from being flown again.		Payload Lead

Verify that all payload electrical components are functional.		Payload Lead

Table 22 Post-Flight Inspection Checklist

8.2 Hazard Analyses

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.

8.2.1 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. If not mitigated, these risks may result in delays, reduction in design quality, or at worse the inability to complete the project and failure of the team's mission. 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 23 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 24 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 25 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	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

		not be ready for launch.			group projects. Sanitizing and PPE gear will be provide by WPI and HPRC and are expected to be used frequently and thoroughly.
Destruction of Full Scale	DI	The team will have to reorganize the schedule to compensate to build a new full-scale rocket. Will cost the team heavily in time and money to build a new rocket, or result in the cancellation and failure of	The budget would have to be increased to compensate for the construction of a new launch vehicle. The team may not be able to afford to construct a new launch vehicle.	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 components together. Analyze and test all electronics within the launch vehicle.

		the team's mission.			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	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	CIII	Little schedule impact unless a shortage of	Budget may have to be supplemented and more money	May impact supplies able to order due to looking for cheaper	WPI HPRC's Treasurer, will keep a detailed budget and

shipping, parts, etc.)		funds results in an incomplete order of needed parts or the faster shipping cannot be afforded for a necessary part.	would have to be raised to offset any additional costs.	options to offset the more expensive ones.	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	Likely unable to recover in time to make another launch due to WPI travel restrictions.	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.

		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 26 Project Risk Overview

8.2.2 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 27 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 28 Personnel Hazard Severity Definitions

Personnel Hazard	Severity			
Probability	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 29 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	Safety officer, leads and/or the lab safety monitor is present during the use of potentially dangerous tools to ensure proper usage and PPE.

					medical attention.	
	Hand Tool Injury	Improper training or human error during the use of tools	Injuries include, but are not limited to cuts, scrapes, even amputation or crushing.	CIII	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.
	Caught in a machine	Loose items of	Partial or complete	DII	Members will not be	Safety officer,

		clothing/jewelry/hair/gloves getting pulled into a machine	destruction of an item pulled in; injuries as severe as amputation.		allowed to use machines while wearing loose items of clothing/jewelry/gloves or having long hair that are not contained.	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 cause a fire to start.	Burns, inhalation of toxic fumes, and in extreme cases, 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 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	BIV	During work with epoxy, members will wear	MSDS sheet for epoxy will be consulted

			reputative skin contact can cause chemical burns.		proper PPE including safety goggles, gloves, and clothes that protect the skin from encountering the material.	and members 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	MSDS sheet for each material will be consulted to make sure members are wearing proper PPE.

					well as a mask to protect their lungs.	
	Exposure to black powder	Loading charges for stage separations or any other contact with black powder.	Serious eye irritation, an allergic skin reaction; can cause damage to organs through prolonged and repetitive exposure.	CIII	Only people who are trained in working with black 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.	Safety officer will ensure that unauthorized members 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	Burns, inhalation of toxic	DII	Members will only work in	Safety officer, leads

		or any other event in which a chemical catches fire.	fumes, death.		facilities with proper fire safety systems installed.	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	WPI HPRC members will provide analysis of

					checked for leaking before use.	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	HPRC Recovery subteam lead, along with others will oversee this process.

					separation chares are weighed 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,	Safety officer will ensure that all safety procedures are followed

					black powder in separation charges will be weighted on an electronic scale.	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	DI	The rocket will not be launched during strong winds, the rocket	Weather conditions will be assessed, the rocket will be launched

			personnel or spectators.		design will be tested through simulations to make sure that it is stable during flight.	only if the RSO considers the weather safe. Multiple simulations will be run to ensure that the rocket is stable.

Table 30 Personnel Hazard Analysis

8.2.3 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 31 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 32 FMEA Probability Definitions

FMEA	Severity			
Probability	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 33 FMEA Assessment Matrix

8.2.3.1 *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	CI	Calculate appropriate ejection charge sizing, and ensure the correct quantities of black powder are used	Testing of the recovery system

		cannot safely deploy.			
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 sustain minor damage and the payload cannot safely deploy.	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
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	DII	Proper installation of the recovery system and select correct sizes of hardware to handle	Testing of recovery system including subscale and full scale testing

		complete the payload mission criteria. It could also injure personnel on the ground due to debris upon impact or impact near a person.		ejection forces.	
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	This could prevent the parachutes	DII	Proper protection and packing	Testing of recovery system

	properly sealed, or the parachutes are not packed correctly.	from slowing the rocket's descent rate, resulting in the possible loss of the rocket and payload.		of the parachutes.	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	DI	The motor will be installed by a certified mentor. The motor retention system will	Subscale and full scale testing

		personnel in the vicinity or destroy the launch vehicle. It could also create free falling debris that could cause harm.		also be inspected prior to launching the rocket.	
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	DI	Launch tower will be setup and maintained by a responsible	Full scale testing

		personnel at a large distance		person at the launch club, and inspected by the safety officer prior to launch	
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, disconnected wires, destruction by black powder charge, or burnt by charge detonation	Incorrect altitude readings and altitude deployment; can result in potential loss of rocket and payload	DI	There will be a backup altimeter with a second power source in case the main altimeter fails. There will also be a set of backup black powder	Altimeter testing included in subscale and full scale testing

				charges connected to the backup altimeter. Both altimeters will also be tested before launch.	
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 charge detonation	Altimeter or recovery system failure	DII	Test the electronic bay and altimeter before launch	Subscale and full scale testing
Descent too fast	Parachute is too small	Potential damage or	DII	Properly size parachute;	Subscale and Full scale

		loss of rocket and payload		test recovery system before launch	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 disqualification of team	DIII	The ground support equipment will be maintained by responsible persons from the launch site club. The motor will be stored according to specified guidelines.	Full scale testing
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	The parachutes would detach from the rocket, leading to	DI	The shock cord will be properly sized to handle ejection	Testing of recovery system including subscale and

	by black powder charge, or burnt by charge detonation	the loss of the rocket and payload.		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.	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	Material testing of the fins, and full scale testing

				descent velocity	
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 recommendations 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. Mechanical actuation will be attempted	Testing of full scale vehicle

				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,	Vehicle overshoots expected	CIII	Test avionics systems before	Full scale testing

	faulty power system	apogee, flight data is not recorded. GPS positions are not transmitted, causing possible loss of vehicle		launch, verify functionality	
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 34 Launch Vehicle FMEA

8.2.3.2 *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	The rocket catches on	DI	WPI HPRC will design	The lander will be run at

	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	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		the lander to be well ventilated to prevent overheating.	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	WPI HPRC will create a payload inspection checklist

				battery's will be charged	

Table 35 Payload FMEA

8.2.4 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 36 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 37 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 38 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	DIII	The launch vehicle will be launched in an open field and aimed in a direction with wind in mind

		landing in a tree.		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 launch vehicle and alter its trajectory depending on the size of the bird. It will also harm the bird.	DIII	The rocket will not be launched when they are any birds in 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 trajectory including excessive drift after parachute deployment.	DII	Alter course and adjust trajectory to 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 hypothermia, and will cause terrain to be slippery and more treacherous.	CIII	The team will monitor the weather before outdoor events and know when to expect wet conditions. 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 resulting injuries.	CIII	The team will only conduct outdoor events and launch activities in large, flat, open spaces where the danger posed by 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 suffers severe damage such that it is irrecoverable, the debris left behind could have detrimental effects on the natural environment.	DII	The team will design the rocket and payload with 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	DIII	The team will take care to carefully inspect the launch vehicle and payload, batteries, and motor before all rocket launches

		batteries and rocket motor.		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 the object as a result of the impact.	DII	The team will only launch at sanctioned launch sites and fly rockets on trajectories away from any structures or 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	BIV	Team members will take care to minimize environmental impact when going out to retrieve the

		rocket and payload after the completion of the mission.		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 39 Environmental Concerns

8.3 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 40 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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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	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**
	mentor attend Launch Week in April.			
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 payload 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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requireme nt No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 progres s
NASA-3.12.2	The electronic tracking device(s) will be fully functional during the official flight on Launch Day.	Inspection, Demonstrati on	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 progres s
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

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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.	[redacted]	[redacted]	[redacted]

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	Inspection	WPI HPRC will be participating in the College/University Division Payload Mission.	Verified

NASA Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
	will provide the appropriate documentation in all design reports so the experiment may be reviewed for flight safety.			
NASA-4.3	Primary Landing System Mission Requirements			
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	Description	Verification Method	Verification Plan	Status
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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-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 Requirements				
Requirement No.	**Description**	**Verification Method**	**Verification Plan**	**Status**
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 Requirements				
Requirement No.	Description	Verification Method	Verification Plan	Status
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					

Derived Requirements					
Requirement No.	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
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

Derived Requirements					
Requirement No.	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
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
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

Derived Requirements					
Requirement No.	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
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					

Derived Requirements					
Requirement No.	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
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
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

Derived Requirements					
Requirement No.	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
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

Derived Requirements					
Requirement No.	**Description**	**Justification**	**Verification Method**	**Verification Plan**	**Status**
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. This logistic budget will 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 CDR. 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	$1,000.00	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

Items Purchased Between PDR and CDR					
Item - General Description	**Item – Specific Description**	**Vendor Name**	**Base Unit Price (USD)**	**Quantity**	**Total (Including Tax and Shipping)**
Subscale Nosecone	PNC-3.00" (75MM) X 12.5"	Apogee Components	$20.28	1	$20.28
Subscale Body Tube	3.00IN LOC BODY TUBE	Apogee Components	$11.17	1	$11.17
Subscale Coupler	3.00" (75MM) LOC COUPLER	Apogee Components	$4.42	1	$4.42
Subscale Epoxy	G5000 ROCKETPOXY - 8-OZ PACKAGE	Apogee Components	$12.50	1	$12.50
Subscale Rail Guide	CONFORMAL RAIL GUIDES FOR 3.1" TUBE	Apogee Components	$6.79	1	$6.79
Subscale Motor Retainer	MADCOW RETAINER 29MM	Apogee Components	$12.79	1	$12.79
Subscale Motors	AEROTECH 29MM LOADABLE MOTOR - G79W-10	Apogee Components	$33.16	3	$99.48
L3 Tender Descender	Fruity Chutes L3 Tender Descender	Fruity Chutes	$129.00	1	$144.75
Organization Box	25 Compartment Box	Zoro.com	$13.08	3	$46.69
Subscale Body Tube	29mmx13" Body tube	Apogee Components	$9.98	1	$48.29
USB cable	Right angle USB mini OTG for RPi Zero	Amazon	$5.99	1	$5.99

Micro SD card	SanDisk 32GB 2-Pack Ultra microSDHC UHS-I Memory Card (2x32GB)	Amazon	$15.88	1	$15.88
DC motor self-righting system with encoder	BEMONOC DC Gear Motor 12V Low Speed 10RPM Encoder Metal Gearmotor with Channel Encoder for DIY Engine Toy	Amazon	$14.32	4	$57.28
Silicone wire 20AWG	BNTECHGO 20 Gauge Silicone Wire Kit 10 Color Each 10 ft Flexible 20 AWG Stranded Tinned Copper Wire	Amazon	$14.98	1	$14.98
Silicone wire 18AWG	BNTECHGO 18 Gauge Silicone Wire Kit Red Black Yellow Brown and Gray Each 25ft 18 AWG Stranded Wire	Amazon	$15.98	1	$15.98
Brass standoffs M2 M3 M4	Hilitchi 360pcs M2 M3 M4 Male Female Brass Spacer Standoff Screw Nut Assortment Kit	Amazon	$18.98	1	$18.98
M2 Male to Female Nylon Hex Standoff Plastic Thread	M2 Male to Female Nylon Hex Standoff Plastic Thread Motherboard Spacer Prototyping Accessories for PCB, Quadcopter Drone, Computer & Circuit Board Assortment Kit;Black	Amazon	$8.99	1	$8.99
M2.5 Female Nylon Hex Standoff Plastic Thread	M2.5 Nylon Hex Standoff Plastic Thread Motherboard Spacer Prototyping Accessories for PCB, Quadcopter Drone, Computer & Circuit Board Assortment Kit (M2.5 Female; Black)	Amazon	$8.99	1	$8.99
Male/female pin headers	DEPEPE 30 Pcs 40 Pin 2.54mm Male and Female Pin Headers for Arduino Prototype Shield	Amazon	$5.59	1	$5.59

Switch	FingerTech 40A Mini Power Switch	RobotShop	$6.78	4	$27.12
Current sensor	ACS724 Current Sensor Carrier 0 to 30A	Pololu	$9.95	2	$19.90
Dual DC motor controller	Dual MC33926 Motor Driver Carrier	Pololu	$29.95	3	$89.85
Servo motor stabilization system	2000 Series Dual Mode Servo (25-2)	ServoCity	$31.99	4	$127.96
Servo programmer board	3102 Series Dual Mode Servo Programmer (1-1)	ServoCity	$9.99	2	$19.98
DC motor self-righting system	19 RPM Econ Gear Motor	ServoCity	$14.99	4	$59.96
32 Pitch, 16 Tooth (4mm Bore) Bevel Gear	32 Pitch, 16 Tooth (4mm Bore) Bevel Gear	ServoCity	$5.99	5	$29.95
32 Pitch, 32 Tooth (.250" Bore) Bevel Gear	32 Pitch, 32 Tooth (.250" Bore) Bevel Gear	ServoCity	$7.99	5	$35.95
Masks	Dasheng 5-Pack Disposable Sanding and Fiberglass Safety Mask	Lowes	$19.99	2	$39.98
SIM Card	GSM SIM Card from Ting & Adafruit	Adafruit	$9.00	2	$18.00
USB mini OTG (RPi Zero) to USBA		Adafruit	$4.95	2	$9.90
DIY USB Cable Parts - Straight Type A Jack		Adafruit	$4.95	2	$9.90
DIY USB Cable Parts - Straight Type A Plug		Adafruit	$4.95	2	$9.90
DIY Cable Parts - 20 cm Ribbon Cable		Adafruit	$1.95	2	$3.90

Microcontroller	PJRC Teensy 3.2 Development Board	Adafruit	$19.95	1	$19.95
USB A to micro B cable		Adafruit	$2.95	5	$14.75
25x 1/4-20 threaded inserts	97171A230	McMaster	$8.91	1	$8.91
50x 1/4-20 1/2" 18-8 SS Button Head Screw	92949A537	McMaster	$5.65	1	$5.65
100x 1/4" Retaining Ring	97633A130	McMaster	$8.13	1	$8.13
1x 1/4" Shaft Bronze Bushing	6338K412	McMaster	$0.82	10	$8.20
100x M3x0.5 6mm 18-8 SS Socket Head Screw	90348A004	McMaster	$7.66	1	$7.66
12" 1/4" 12L14 Steel Shaft	1327K66	McMaster	$8.12	1	$8.12
100x 6-32 Threaded Insert	93365A130	McMaster	$13.35	1	$13.35
100x 6-32 3/16" 18-8 SS Set Screw	92311A143	McMaster	$4.15	1	$4.15
Flanged Stainless Steel Ball Bearing	57155K303	McMaster	$6.67	8	$53.36
1/4-20 Flanged Socket Head Screw	92235A516	McMaster	$3.61	1	$3.61
50x M4x0.7 Threaded Inserts	94180A353	McMaster	$9.37	1	$9.37

50x 8-32 Threaded inserts	94459A310	McMaster	$9.84	1	$9.84
Oval Shaped Threaded Connecting Link	8947T25	McMaster	$3.11	1	$3.11
Aluminum bar stock	Multipurpose 6061 Aluminum 5/8" Thick x 1-1/4" Wide x 2' Length	McMaster	$13.55	1	$13.55
Sheet metal stock	Low-Carbon Steel Bar with Rounded Edges 1/16" Thick, 1" Wide, 3' Length	McMaster	$9.92	1	$9.92
NylonK Kevlar Fiber Filament - 1.75mm (0.5kg)		MatterHackers	$68.00	1	$68.00
Black NylonG Glass Fiber Filament - 1.75mm (0.5kg)		MatterHackers	$51.20	1	$64.00
Rotary Latch	R4-EM-R21-162	DbRoberts	$90.00	1	$101.50
Cheetah Flexable 3D printer Filament	Cheetah™ Flexible 3D Printing Filament	FennerDrives	$45.00	1	$55.30
Motor Casing	CTI 4G 75mm Gen 2 Motor Casing	Sunward	$189.95	1	$189.95
Tapered Heat-Set Inserts for Plastic	97163A135	McMaster	$7.70	1	$7.70
Airframe Tubing	Body Tubes for full scale vehicle, 30", Natural Color	Madcow Rocketry	$114.00	3	$342.00
Nosecone	Full scale nosecone, 4:1 Ogive, Natural Color	Madcow Rocketry	$149.95	1	$149.95
Couplers	Full scale coupler tubing, 12", natural color	Madcow Rocketry	$60.00	3	$180.00

Subscale Altimeter	Pnut Altimeter	PerfectFlite	$54.95	1	$63.87
Main Parachute	120in. Rocketman High Performance CD 2.2 Parachute	Rocketman	$315.00	1	$315.00
Reefing Ring	Stainless Steel Parachute Reefing Ring	Rocketman	$4.00	2	$8.00
18-8 Stainless Steel Button Head Hex Drive Screw 4-40 Thread Size, 7/16" Long	92949A111	McMaster	$3.65	1	$3.65
18-8 Stainless Steel Button Head Hex Drive Screw 4-40 Thread Size, 1/4" Long	92949A106	McMaster	$3.02	1	$3.02
Alloy Steel Shoulder Screw 1/4" Shoulder Diameter, 1-3/4" Shoulder Length, 10-24 Thread	91259A102	McMaster	$1.47	6	$8.82
18-8 Stainless Steel Nylon-Insert Locknut 10-24 Thread Size	91831A011	McMaster	$5.70	1	$5.70
12-Piece Foam Material Sample Pack	2172T21	McMaster	$47.47	1	$47.47
Flame-Retardant Garolite G-10/FR4 Sheet	G10/FR4 Plate	McMaster	$17.87	3	$53.61

12" Wide x 12" Long, 1/8" Thick, Black					
18-8 Stainless Steel Hex Drive Flat Head Screw 82 Degree Countersink Angle, 4-40 Thread Size, 3/8" Long	92210A108	McMaster	$3.54	1	$3.53
316 Stainless Steel Washer for Soft Material Number 4 Screw Size, Regular ASME Designation	92844A122	McMaster	$2.62	1	$2.62
Tapered Heat-Set Inserts for Plastic	97163A138	McMaster	$4.83	2	$9.66
Passivated 18-8 Stainless Steel Pan Head Phillips Screw 4-40 Thread, 11/16" Long	91772A117	McMaster	$4.57	1	$4.57
18-8 Stainless Steel Button Head Hex Drive Screw 1/4"-20 Thread Size, 1-3/8" Long	92949A836	McMaster	$7.25	1	$7.25
TOTAL SPENT	**$3619.28**				

Table 9.2 - Items Purchased Between PDR and CDR

Overall Current Budget	
Components	
Budget Given by AIAA	+ $2,654.85
Funds Moved from Logistics Budget	+ $2,366.90
Expenses Thus Far	-$3619.28
TinkerBox Funding (up to $3,000)	+ $2993.6
Sponsorship	+ $2,500
Total in Component Budget as of CDR	**$7,896.07**
Logistics	
Budget Given by AIAA	+ $4,733.80
Funds Taken from Logistics Budget for Components	-$2,366.90
Total in Logistic Budget as of CDR	**$2,366.90**
Total in Account	**$10,262.97**

Table 9.3 Overall Current Budget

9.4 Funding

A significant portion of this year's funding for 2020-2021 WPI HPRC has 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. By the time of CDR, the team has used this full amount afforded to us through the Tinkerbox program.

In addition to TinkerBox, WPI HPRC receives 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 has created "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. In addition, the team has recently added another sponsor to our family. Thus far, the team has acquired $3,500 from these three sponsors. If the team has any extra funding from corporate sponsors at the end of the competition season, the team is looking into having the money roll over to be used in the next competition year in 2021-2022.

9.5 Timeline

Due to the aforementioned circumstances WPI and their COVID-19 policies have provided us with, our timeline has shifted due to the team's decision to exit the NASA rocketry competitions. The CDR timeline was moved from the previous due date of January 4th to January 26th to accommodate more time for workshops, design, testing, and documentation. The team will still complete the Flight Readiness Review (FRR) but there is no official due date for such, since the team does not know when or if WPI will lift their travel restrictions to attend a launch. The team typically launches their full scale at Lake Winnipesaukee High Powered Rocketry (LWHPR) #834 or Champlain Region Model Rocket Club (CRMRC) #643 and hopes to launch at one of these sites in March or April. Until February 11th, WPI has banned clubs from meeting in person and Washburn Machine Shop is not open for the team's use. Due to this, in person construction has been halted and expected to continue safely as soon as campus bans are lifted. In the meantime, the team will continue to meet online as well as test components and systems individually while taking this time to develop skills such as learning Ansys, developing code, and preparing for full scale construction.

10 Appendix

10.1 Rocket Mass Budget

Component	Part Number	Nominal Mass (lb)	Quantity	Mass Margin	Mass (lb)
Lower Airframe	U21-1-1-002	3.716	1	0	3.716
Motor Tube	U21-1-1-003	0.495	1	0	0.495
Thrust Plate	U21-1-1-004	0.3886	1	0	0.389
Radial Bracket - A	U21-1-1-005	0.0212	6	0	0.127
Radial Bracket - C	U21-1-1-005	0.0212	4	0	0.085
Radial Bracket - RB	U21-1-1-005	0.0204	2	0	0.041
Fin Ring	U21-1-1-006	0.4998	2	0	1.000
Fin Bracket	U21-1-1-007	0.0289	16	0	0.462
Fin	U21-1-1-008	0.3044	4	0.1	1.339
Tailcone	U21-1-1-009	0.6746	1	0.03	0.695
Pro75 4G Hardware		4.3300	1	0	4.330
CTI L1395		5.1700	1	0	5.170
1515 Rail Button		0.0213	2	0	0.043
1515 Rail Button Bolt		0.0278	2	0	0.056
#8-32x0.375 Button Head Screw	91255A192	0.0029	18	0	0.052
#8 Washer	92141A009	0.0010	32	0	0.032
#8-32x0.5 Button Head Screw	91255A194	0.0035	28	0	0.097
#8-32x0.5 Button Head Screw	91255A194	0.0035	12	0	0.042
#8-32 Hex Nut	91841A009	0.0031	8	0	0.024
3/8-18x1.25 Flat Head Screw	91253A626	0.0422	1	0	0.042
#8-32x0.321 Heat Set Insert	94459A330	0.0022	4	0	0.009
Middle Airframe	U21-1-2-002	3.4991	1	0	3.499
120" Rocketman + Swivel		1.5625	1	0	1.563
Reefing Ring		0.0710	1	0	0.071
Quick Link		0.1630	3	0	0.489
300" Shock Cord		0.6600	1	0	0.660
1515 Rail Button		0.0213	1	0	0.021
1515 Rail Button Bolt		0.0278	1	0	0.028

Upper Airframe	U21-1-3-002	3.1241	1	0	**3.124**
Nosecone	U21-1-3-003	3.6000	1	0	**3.600**
Piston Coupler	U21-1-3-004	0.6345	1	0	**0.634**
Piston Bulkhead	U21-1-3-005	0.2249	1	0.1	**0.247**
Nosecone Bulkhead	U21-1-3-006	0.2222	1	0	**0.222**
Radial Bracket - C	U21-1-1-005	0.0212	4	0	**0.085**
36" Spherachutes		0.1125	1	0	**0.113**
Swivel		0.0500	1	0	**0.050**
Quick Link		0.1630	4	0	**0.652**
300" Shock Cord		0.6600	1	0	**0.660**
#8-32x0.5 Button Head Screw	91255A194	0.0035	8	0	**0.028**
#8-32x0.375 Button Head Screw	91255A192	0.0029	8	0	**0.023**
#8-32x0.25x4 Threaded Standoff	93330A563	0.0177	4	0	**0.071**
5/16 U Bolt	8880T88	0.1977	1	0	**0.198**
5/16 Washer	92141A030	0.0057	2	0	**0.011**
Payload	U21-2-0-001	5.6820	1	0	**5.682**
Avionics Bay Coupler	U21-1-4-002	1.8113	1	0	**1.81134499**
Avionics Bay Upper Bulkhead	U21-1-4-003	0.2418	1	0	**0.24184263**
Avionics Bay Lower Bulkhead	U21-1-4-004	0.2089	1	0	**0.20891323**
Avionics Bay Spine	U21-1-4-005	0.2695	1	0	**0.2695003**
Airbrake Band	U21-1-4-006	0.1162	1	0	**0.11619721**
Avionics Bay Connection Ring	U21-1-4-007	0.2271	1	0.03	**0.23388828**
Avionics Sled	U21-1-4-008	0.1301	1	0.03	**0.13397519**
Avionics Bay Spine Lock	U21-1-4-009	0.0786	1	0	**0.07855207**
Avionics Bay Spine Adapter	U21-1-4-010	0.0432	1	0	**0.04323396**
Avionics Battery Sled	U21-1-4-011	0.0926	1	0.03	**0.095372026**
#4-40 Shear Pin	93135A109	0.0002	4	0	**0.0007362**
3S 450mAh Lipo Battery		0.0882	1	0	**0.0881849**
Avionics Board		0.0882	1	0.1	**0.09700339**
6" 22AWG Wire		0.0050	1	0	**0.005**
JST Connector		0.0022	1	0	**0.00220462**
GPS SMA Antenna		0.0397	1	0	**0.0396832**
LoRa Antenna		0.0551	1	0	**0.0551156**

400mm Battery Strap		0.0110	1	0	**0.0110231**
1/4-20x0.75 Socket Head Screw	91251A540	0.0141	2	0	**0.02828252**
1/4 Washer	92141A029	0.0040	1	0	**0.00403252**
5/16 U Bolt	8880T88	0.1977	1	0	**0.19768268**
15mm External Retaining Ring	98541A410	0.0018	2	0	**0.00351318**
15mm Shaft Coupler	6056N23	0.0649	2	0	**0.12974758**
#8-32x0.5 Button Head Screw	91255A194	0.0035	10	0	**0.0347403**
#8-32x0.321 Heat Set Insert	94459A330	0.0022	2	0	**0.00448668**
#4-40x0.1875x0.25 Hex Standoff	91075A101	0.0008	4	0	**0.0030688**
#4-40x0.1875 Button Head Screw	92949A105	0.0008	4	0	**0.0031166**
#4-40x0.226 Heat Set Insert	94459A270	0.0009	4	0	**0.00378292**
Recovery Bay Coupler	U21-1-5-002	1.2629	1	0	**1.26287399**
Recovery Bay Forward Bulkhead	U21-1-5-003	0.2281	1	0	**0.22811086**
Recovery Bay Spine	U21-1-5-004	0.1552	1	0	**0.15517142**
Recovery Bay Adapter	U21-1-5-005	0.1393	2	0	**0.27856506**
Switch Band	U21-1-5-006	0.1213	1	0	**0.12129828**
Recovery Bay Altimeter Sled	U21-1-5-009	0.1393	1	0.03	**0.14344399**
Recovery Bay Battery Sled	U21-1-5-010	0.1629	1	0.03	**0.167763846**
Recovery Bay Aft Bulkhead	U21-1-5-011	0.2271	1	0	**0.22710852**
Left Switch Bracket	U21-1-5-012	0.0133	1	0.03	**0.013715398**
Right Switch Bracket	U21-1-5-013	0.0133	1	0.03	**0.013715398**
Radial Bracket - C	U21-1-1-005	0.0212	4	0	**0.08479916**
#4-40 Shear Pin	93135A109	0.0002	4	0	**0.0007362**
2S 370mAh Lipo Battery		0.0595	1	0	**0.05952479**
1S 300mAh Lipo Battery		0.0198	1	0	**0.0198416**
StratoLoggerCF		0.0238	1	0	**0.02375**
Raven 4		0.0146	1	0	**0.0145505**
Apogee 110-220v 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**
24" 22 AWG Wire		0.0200	1	0	**0.02**
JST Connector		0.0005	8	0	**0.004**

#8-32x0.321 Heat Set Insert	94459A330	0.0022	4	0	**0.00897336**
#8-32x0.375 Button Head Screw	91255A192	0.0029	12	0	**0.03485004**
#4-40x0.226 Heat Set Insert	94459A270	0.0009	6	0	**0.00567438**
#4-40x0.1875x0.25 Hex Standoff	91075A101	0.0008	6	0	**0.0046032**
#4-40x0.1875 Button Head Screw	92949A105	0.0008	6	0	**0.0046749**
5/16 U Bolt	8880T88	0.1977	2	0	**0.39536536**
#8-32x0.375 Knurled Head Screw	91830A206	0.0149	4	0	**0.05966652**
#8-32x0.5 Flat Head Screw	91253A194	0.0035	4	0	**0.013967**
#8 Washer	92141A009	0.0010	4	0	**0.00403292**
#8-32 Hex Nut	91841A009	0.0031	4	0	**0.0122396**
1/4 Washer	92141A029	0.0040	2	0	**0.00806504**
1/4-20x0.75 Socket Head Screw	91251A540	0.0141	2	0	**0.02828252**
5/16 Washer	92141A030	0.0057	4	0	**0.0229876**
#8-32x0.5 Button Head Screw	91255A194	0.0035	4	0	**0.01389612**
Guide Plate	U21-1-6-002	0.2203	1	0	**0.22033312**
Actuator Plate	U21-1-6-003	0.1407	1	0	**0.14068161**
Motor Plate	U21-1-6-004	0.2117	1	0	**0.21165426**
Fin	U21-1-6-005	0.0249	4	0	**0.099445403**
Fin Rail	U21-1-6-006	0.0098	8	0	**0.0781624**
Fin Pin	U21-1-6-007	0.0004	4	0	**0.00179556**
Servo Spacer	U21-1-6-008	0.0043	1	0.03	**0.004450692**
2000 Series Dual Mode Servo	2000-0025-0004	0.1279	1	0	**0.127868**
80 Tooth Hub Gear	2302-0032-0080	0.0551	1	0	**0.0551156**
40 Tooth Servo Gear	2305-0025-0040	0.0397	1	0	**0.0396832**
0.02" PTFE Film	8569K45	0.0179	1	1	**0.035887667**
#8-32x.25x1.5 Threaded Standoff	93330A482	0.0058	4	0	**0.02309688**
32mm Ball Bearing	5972K358	0.0650	1	0	**0.065**
1/8x1/4 Sleeve Bearing	6391K753	0.0012	4	0	**0.00467364**
#8-32x0.5 Button Head Screw	91255A194	0.0035	12	0	**0.04168836**
#8-32 Narrow Hex Nut	90730A009	0.0010	4	0	**0.00397256**
#8-32x0.375 Button Head Screw	91255A192	0.0029	20	0	**0.0580834**

#8-32 Hex Nut	91841A009	0.0031	4	0	**0.0122396**

10.2 Payload Mass Budget

Sub-Assembly	Component	Part Number	Nominal Mass (lb)	Quantity	Mass Margin	Mass (lb)
EnP	MID CARBON PLATE	U21-2-5-003	0.1221	1	0	**0.1221**
EnP	Avionics computer board		0.0330693	1	0	**0.0331**
EnP	Raspberry Pi Zero		0.099208	1	0	**0.0992**
EnP	Pololu Dual MC33926 (motor controller)		0.015625	2	0	**0.0313**
EnP	Servo BEC (battery eliminator circuit)		0.0308647	3	0	**0.0926**
EnP	LiPo battery		0.32408	1	0	**0.3241**
EnP	GPS antenna		0.1241203	1	0	**0.1241**
EnP	Other antennas estimate		0.0330693	1	0	**0.0331**
EnP	Wires, connectors, estimate of everything else		0.0440925	1	0	**0.0441**
						0.0000
Stabilization	LIFT MECHANISM BASE	U21-2-2-003	0.1168	3	0	**0.3504**
Stabilization	LIFT MECHANISM CRANK	U21-2-2-004	0.0184	3	0	**0.0552**
Stabilization	LIFT MECHANISM FOLLOWER	U21-2-2-005	0.0093	6	0	**0.0558**
Stabilization	LIFT MECHANISM COUPLER	U21-2-2-006	0.0375	3	0	**0.1125**
Stabilization	CRANK-BASE SHAFT	U21-2-2-007	0.0069	3	0	**0.0207**
Stabilization	FOLLOWER SHAFT	U21-2-2-008	0.0043	6	0	**0.0258**
Stabilization	CRANK-COUPLER SHAFT	U21-2-2-009	0.0038	3	0	**0.0114**
Stabilization	14T GEAR	U21-2-2-010	0.0062	3	0	**0.0186**
Stabilization	28T GEAR	U21-2-2-011	0.0284	3	0	**0.0852**
Stabilization	POLYCARBONATE HINGEPLATE	U21-2-2-012	0.0030	3	0	**0.0090**
Stabilization	FOOT OUTER LAYER	U21-2-2-013	0.0032	18	0	**0.0576**
Stabilization	FOOT INNER LAYER	U21-2-2-014	0.0006	9	0	**0.0054**
Stabilization	FOOT TRACTION LAYER <Hex Tread>	U21-2-2-015	0.0018	9	0	**0.0162**
Stabilization	FOOT TRACTION LAYER <Spike Tread>	U21-2-2-015	0.0018	9	0	**0.0162**
Stabilization	POTENTIOMETER MOUNTING BRACKET	U21-2-2-016	0.0023	3	0	**0.0069**
Stabilization	BRACKET BOLT SPACER <Two-Pin>	U21-2-2-017	0.0018	3	0	**0.0054**

Stabilization	BRACKET BOLT SPACER <One-Pin>	U21-2-2-017	0.0008	3	0	**0.0024**
Stabilization	Stainless Steel Ball Bearing, 57155K303		0.0016	24	0	**0.0384**
Stabilization	Black Oxide Alloy Steel SHCS, 91251A192		0.0038	12	0	**0.0456**
Stabilization	18-8 SS Flanged BHS, 97654A673		0.0010	3	0	**0.0030**
Stabilization	Button Head HDS, 92095A454		0.0005	15	0	**0.0075**
Stabilization	Steel Ext. Retaining Ring, 97633A130		0.0002	24	0	**0.0048**
Stabilization	Heat-Set Insert for Plastic, 94459A110		0.0003	12	0	**0.0036**
Stabilization	Heat-Set Insert for Plastic, 94459A310		0.0013	21	0	**0.0273**
Stabilization	Heat-Set Insert for Plastic, 97171A230		0.0044	3	0	**0.0132**
Stabilization	goBILDA Servo, 2000-0025-0002		0.1325	3	0	**0.3975**
Stabilization	Limit Switch, UP01DTANLA04		0.0007	3	0	**0.0022**
Stabilization	Bourns Potentiometer, 39SB-1RB-104		0.0088	3	0	**0.0264**
Stabilization	Type 316 SS Flat Washer, 90107A029		0.0030	6	0	**0.0180**
Stabilization	Bourns Pot. Nut, H-38-15		0.0009	3	0	**0.0027**
Stabilization	18-8 SS Low Profile SCS, 93615A317		0.0024	3	0	**0.0072**
						0.0000
Self-Righting	PAYLOAD BOTTOM PLATE	U21-2-1-002	0.1279	1	0	**0.1279**
Self-Righting	SELF RIGHTING HINGE MOUNT	U21-2-1-003	0.0287	3	0	**0.0860**
Self-Righting	SELF RIGHTING HINGE PIN	U21-2-1-004	0.0281	3	0	**0.0843**
Self-Righting	SELF RIGHTING PETAL	U21-2-1-005	0.0397	3	0	**0.1190**
Self-Righting	Heat-Set Insert for Plastic, 97171A230		0.0044	6	0	**0.0265**
Self-Righting	18-8 SS BHCS, 92949A537		0.0088	6	0	**0.0529**
Self-Righting	Bronze Bushing, 6338K412		0.0066	6	0	**0.0397**
Self-Righting	Steel Retaining Ring, 97633A130		0.0002	6	0	**0.0012**
Self-Righting	Heat-Set Insert for Plastic, 93365A130		0.0010	6	0	**0.0060**
Self-Righting	18-8 SS Cup Point Set Screw, 92311A143		0.0022	6	0	**0.0132**

Self-Righting	Gearmotor, 638340		0.2161	3	0	**0.6482**
Self-Righting	M3x0.5 6mm BHCS, 92095A176		0.0010	6	0	**0.0063**
Self-Righting	INPUT BEVEL GEAR	U21-2-1-007	0.0022	3	0	**0.0066**
Self-Righting	ECE Potentiometer		0.0022	3	0	**0.0066**
Self-Righting	POTENTIOMETER SPUR GEAR	U21-2-1-008	0.0022	3	0	**0.0066**
Self-Righting	POTENTIOMETER CAP	U21-2-1-010	0.0022	3	0	**0.0066**
Self-Righting	Limit Switch, UP01DTANLA04		0.0220	3	0	**0.0661**
Self-Righting	BHCS, 92095A503		0.0007	6	0	**0.0044**
Self-Righting	18-8 SS Nylon Locknut		0.0006	6	0	**0.0033**
						0.0000
Retention	Rotary Latch, R4-EM-R21-162		0.5604	1	0	**0.5604**
Retention	TOP PLATE	U21-2-3-006	0.1183	1	0	**0.1183**
Retention	Camera Offset Ring	U21-2-3-004	0.0728	1	0	**0.0728**
Retention	Release Routing	U21-2-3-005	0.0265	1	0	**0.0265**
Retention	Camera Protector	U21-2-3-003	0.0265	1	0	**0.0265**
Retention	PICAM 360		0.0617	1	0	**0.0617**
						0.0000
Tender Descender	StrattoLogger CF		0.0238	1	0	**0.0238**
Tender Descender	L3 Tender Descender		0.3125	1	0	**0.3125**
Tender Descender	Eggtimer Mini Wifi Switch		0.0110	1	0	**0.0110**
Tender Descender	Clamping Plate	U21-2-4-007	0.0220	1	0	**0.0220**
Tender Descender	Turnigy Nano-Tech 300mAh 2S 45~90C LiPo Pack		0.0419	1	0	**0.0419**
Tender Descender	StrattoLogger CF Cage	U21-2-4-004	0.0243	1	0	**0.0243**

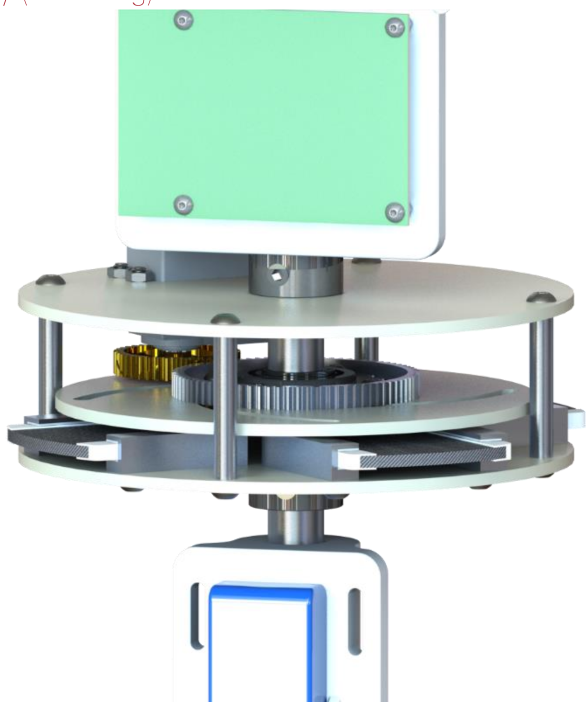
Tender Descender	Bottom Battery Plate	U21-2-4-003	0.0176	1	0	**0.0176**
Tender Descender	StrattoLoggerCF Top Cover	U21-2-4-002	0.0176	1	0	**0.0176**
Tender Descender	18-8 Stainless Steel Button Head Hex Drive Screw	92949A106	0.0009	6	0	**0.0052**
Tender Descender	Black-Oxide Alloy Steel Socket Head Screw	91251A829	0.0067	6	0	**0.0405**
Tender Descender	18-8 Stainless Steel Nylon-Insert Locknut	91831A009	0.0055	6	0	**0.0329**
Tender Descender	Brass Heat-Set Inserts for Plastic	94459A270	0.1044	6	0	**0.6263**
Tender Descender	Male-Female Threaded Hex Standoff	93505A102	0.0010	6	0	**0.0060**

10.3 Assembly Renders

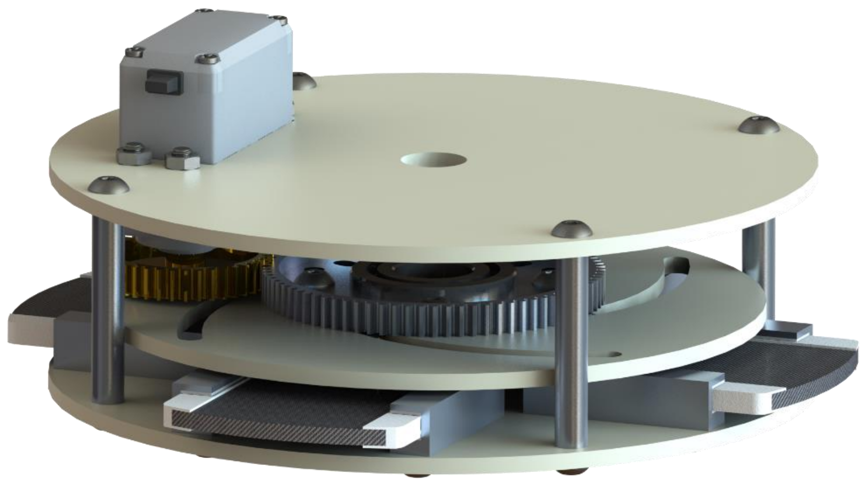
10.3.1 Actuator Plate

10.3.2 Airbrake Fin

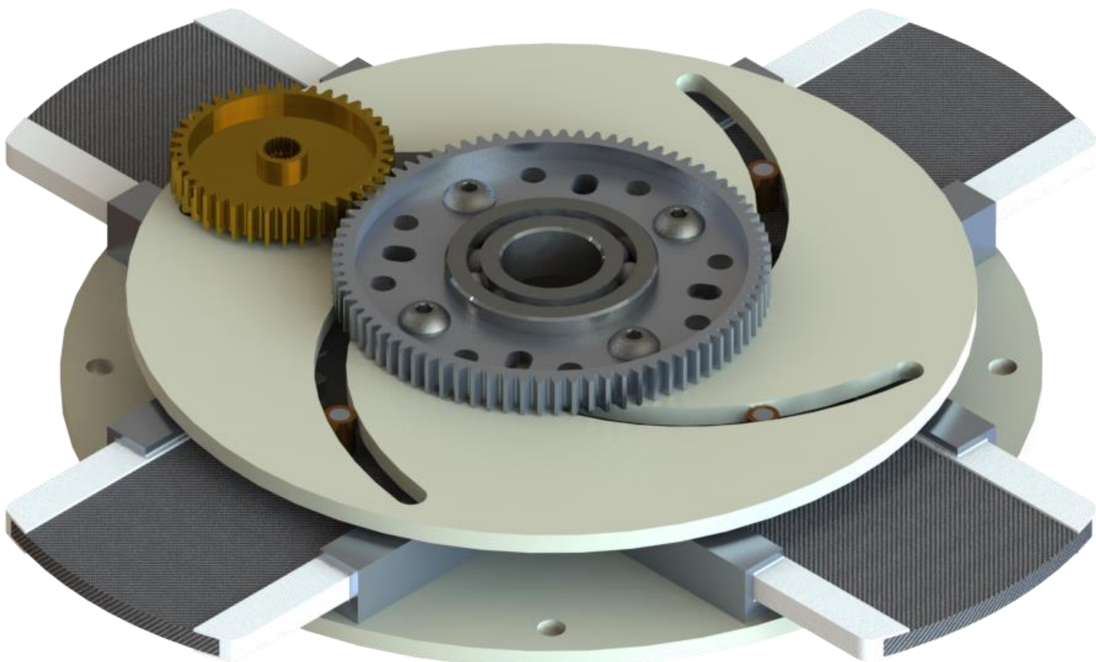
10.3.3 Airbrake Assembly (Mounting)



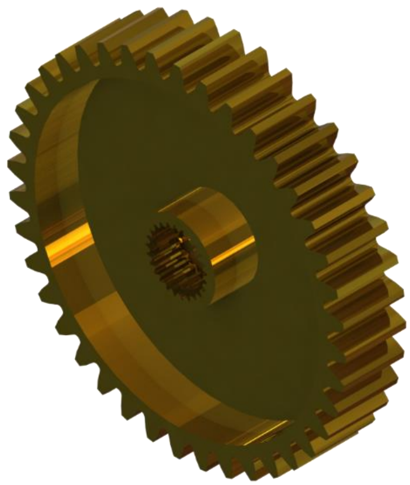
10.3.4 Airbrake Assembly



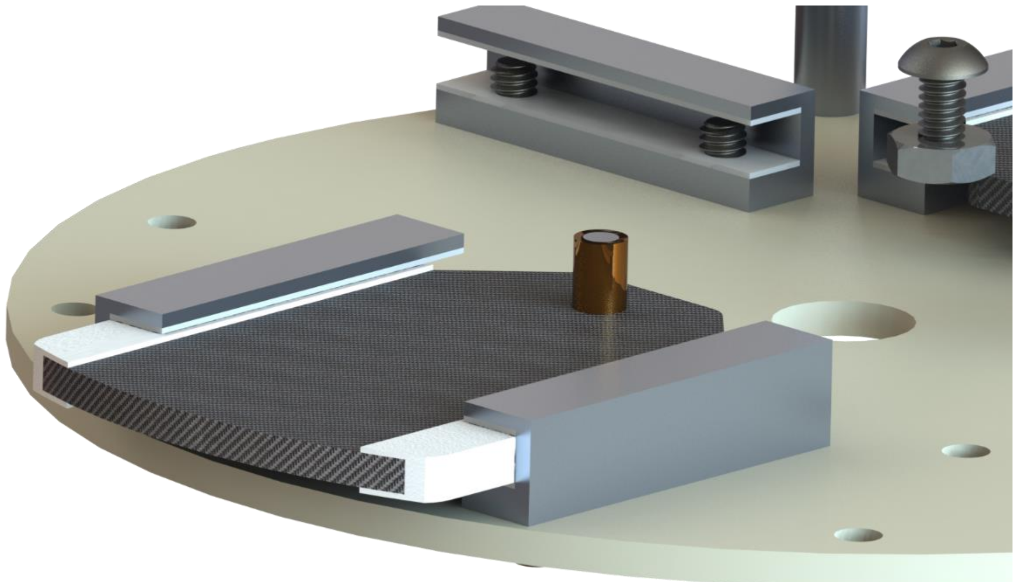
10.3.5 Airbrake Assembly (Gears Exposed)



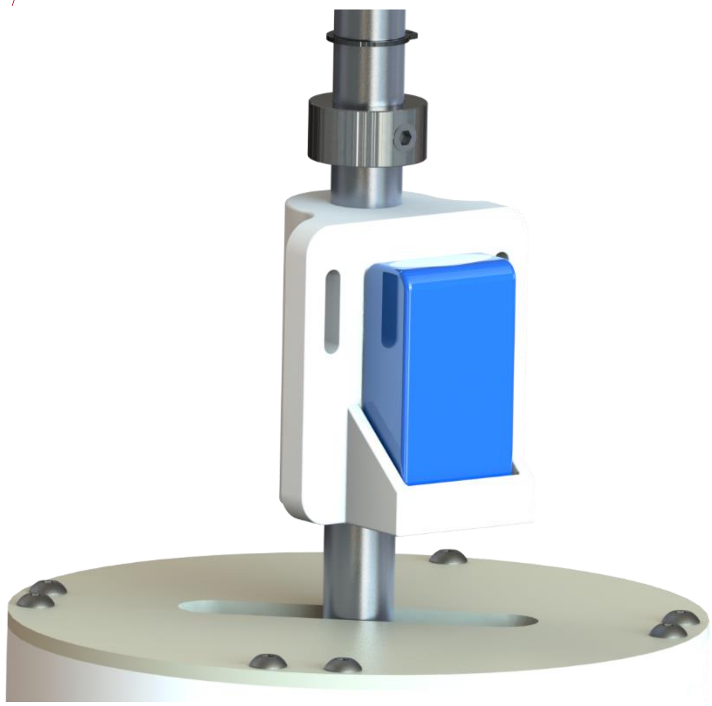
10.3.6 Servo Gear



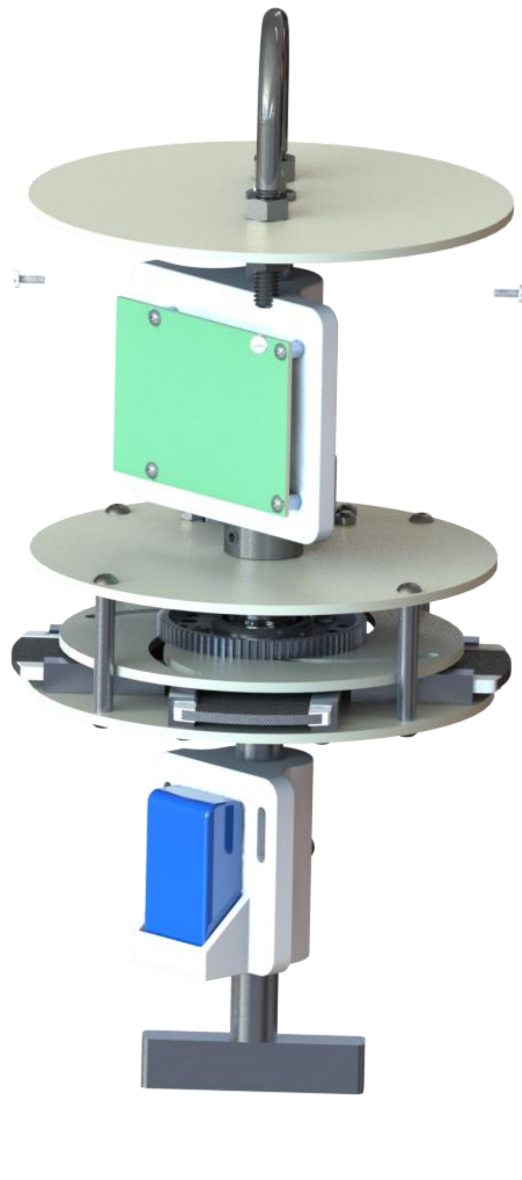
10.3.7 Fin and Fin Rail



10.3.8 Avionics Battery Mount

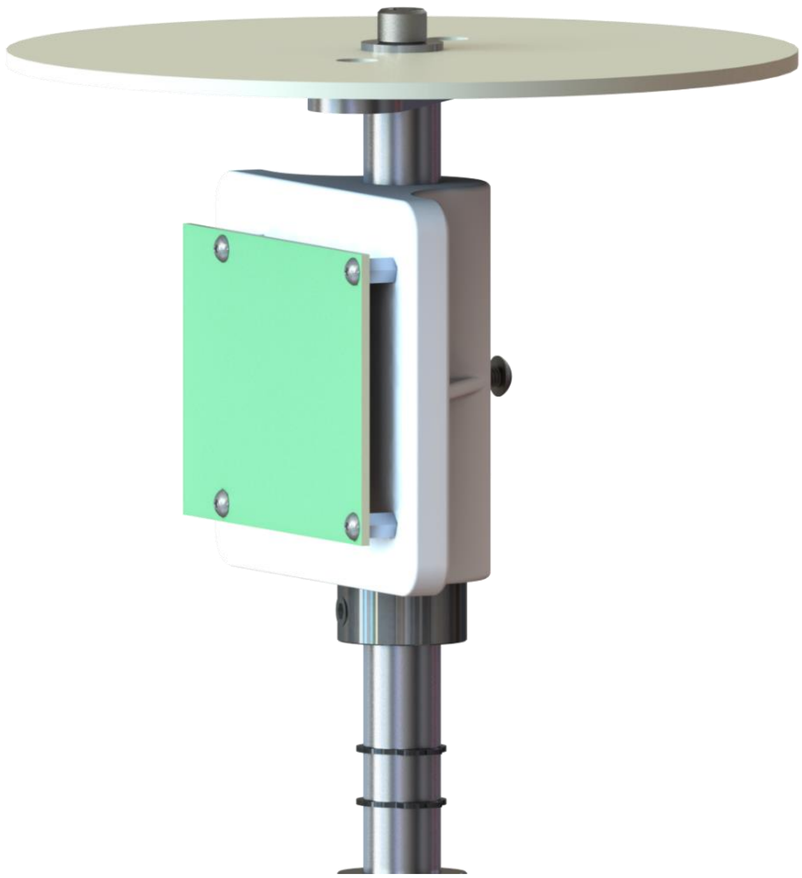


10.3.9 Avionics Twist Lock Section

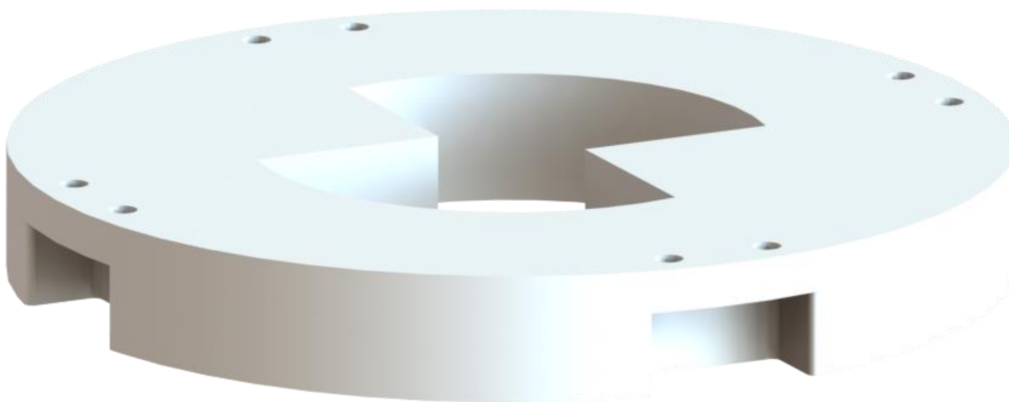


10.3.10 Avionics Bay

10.3.11 Avionics Board Mount



10.3.12 Connection Ring



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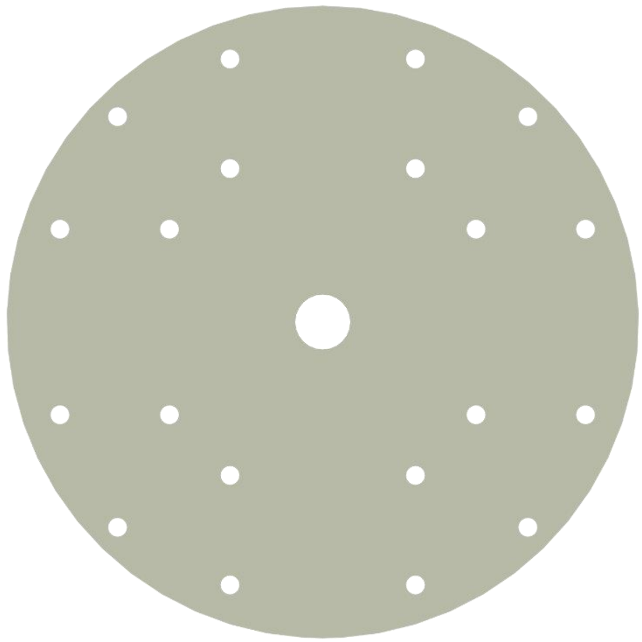
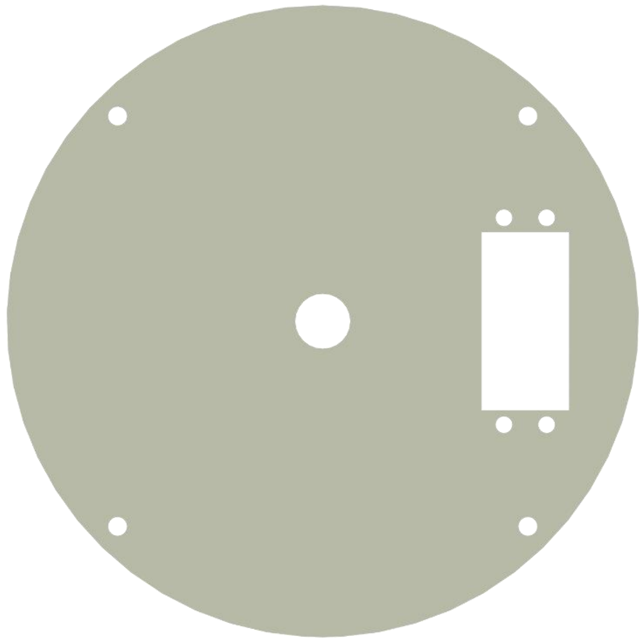
10.3.13 Fin

10.3.14 Guide Plate



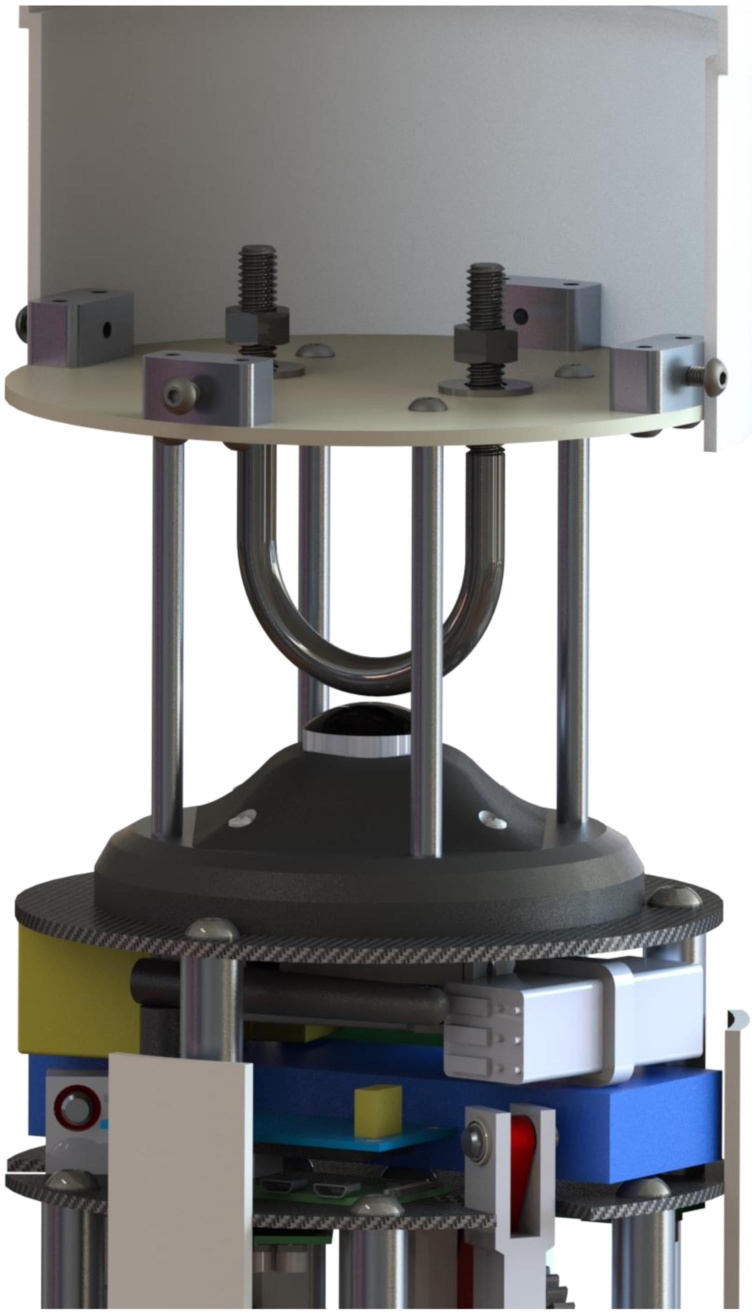
10.3.15 Lower Airframe

10.3.16 Motor Plate



10.3.17 Nosecone

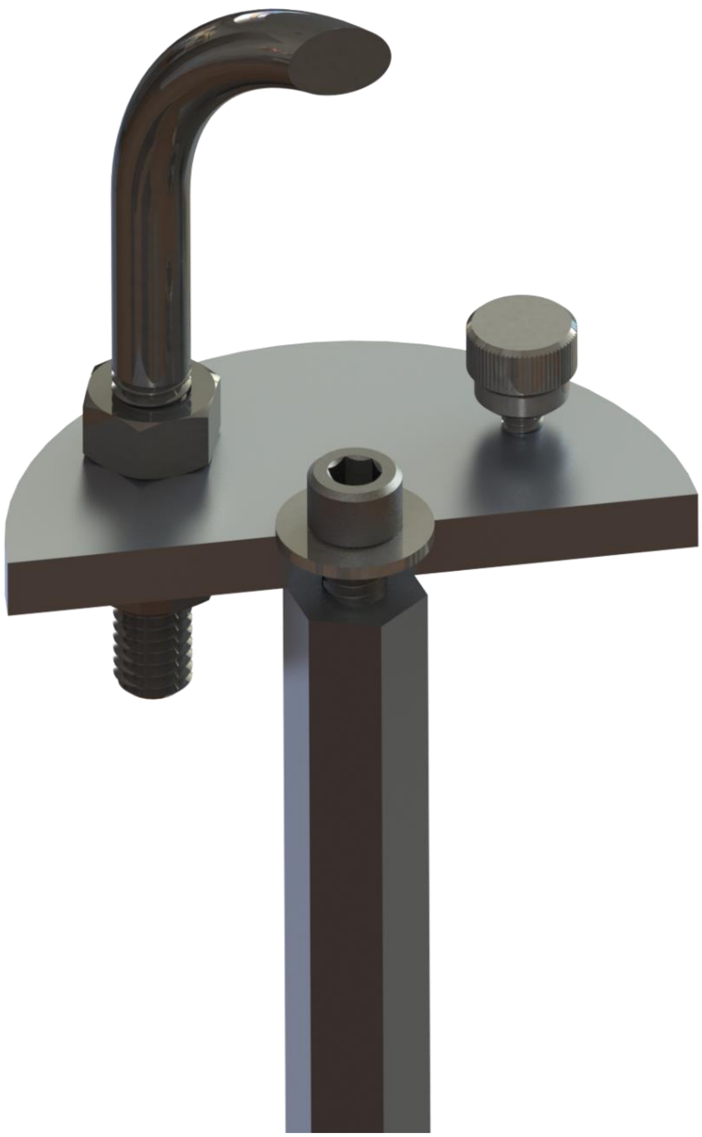
10.3.18 Payload Interface



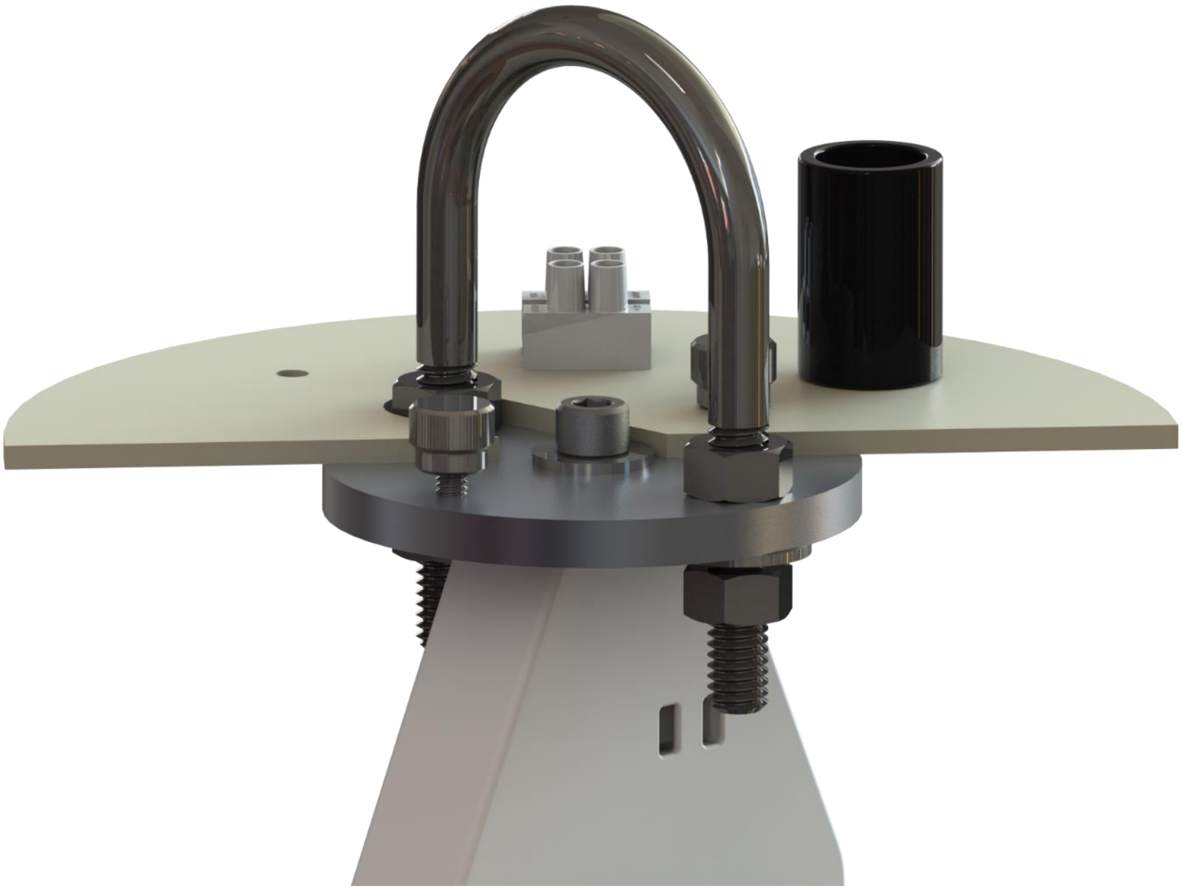
10.3.19 Piston

10.3.20 Recovery Bay Sled

10.3.21 Recovery Bay Spine

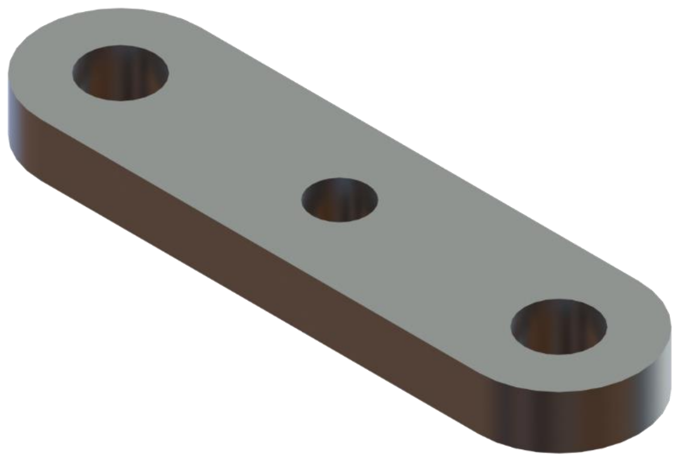


10.3.22 Recovery Bay Upper Bulkhead

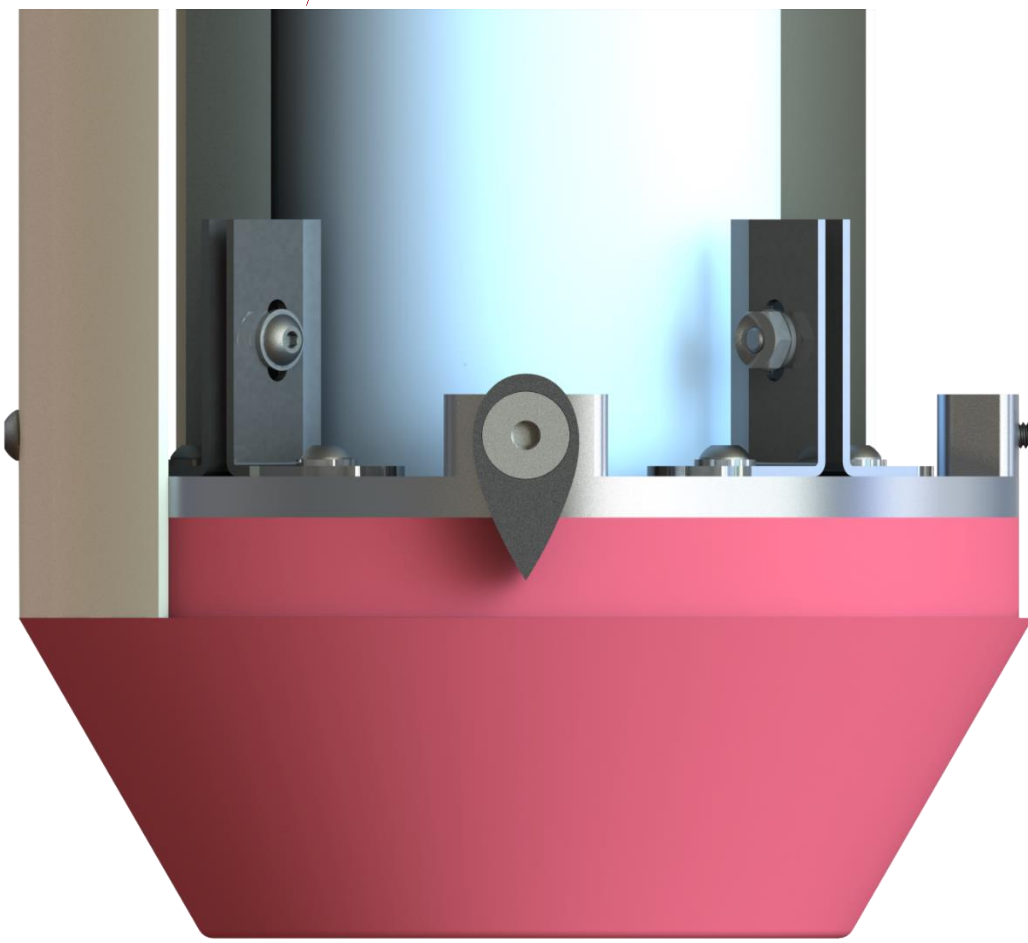


10.3.23 Recovery Bay

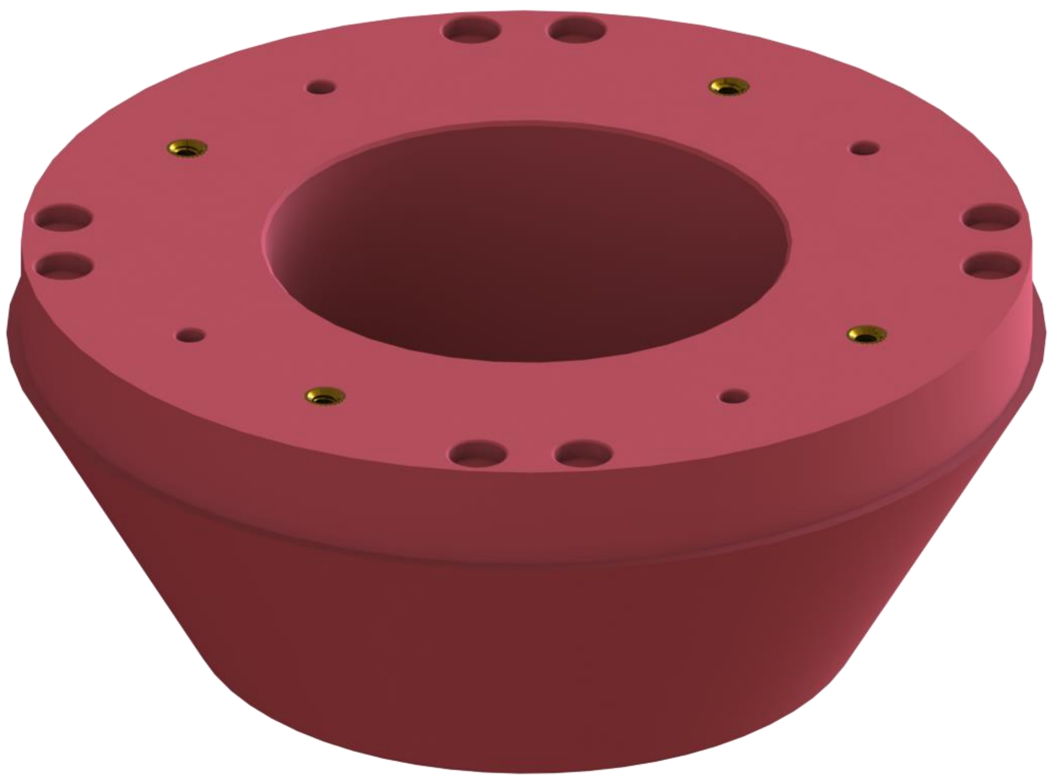
10.3.24 Spine Adapter



10.3.25 Tailcone Assembly



10.3.26 Tailcone



10.3.27 Vehicle Assembly

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