Project Voss

Critical Design Review

Purdue University 2021

500 Allison Road
West Lafayette, IN 47906

Purdue Space Program - Student Launch

Table of Contents

Table of Acronyms

Acronym or Abbreviation	Definition
PSP-SL	Purdue Space Program: Student Launch
ASL	Aerospace Sciences Laboratory
BIDC	Bechtel Innovation Design Center
GCS	Ground Control Station
FEA	Finite Element Analysis
CAD	Computer Aided Design
CFD	Computational Fluid Dynamics
FDM	Fused Deposition Modeling (3D Printing)
PLS	Planetary Lander System (Lander Team)
ABCS	Aero-Braking Control System (Airbrakes Team)
SOS	Self Orientation System
LCS	Lander Control System
PICS	Panoramic Image Capture Subsystem
R&D	Retention and Deployment Subsystem
D&L	Descent and Landing Subsystem
AGL	Above Ground Level
MSL	Mean Sea Level
COTS	Commercial Off-The-Shelf
NAR	National Association of Rocketry
PPE	Personal Protection Equipment
CFR	Code of Federal Regulations
APCP	Ammonium Perchlorate Composite Propellant
OEW	Operating Empty Weight
R&VP	Requirements and Verification Plans
MFSS	Motor and Fin Support Structure
IFVR	In-Flight Video Recording System
CTI	Cesaroni Technology Inc.
VDF	Vehicle Demonstration Flight
FOS	Factor Of Safety
AGL	Above Ground Level
CoM	Center of mass

1 CDR Report Overview

1.1 Team Summary

Team Name	Purdue Space Program – NASA Student Launch (PSP-SL)
Team Address	500 Allison Road, West Lafayette, IN 47906
Team Mentor Name	Christopher Nilsen
Team Mentor Email	cnilsen@purdue.edu
Team Mentor Cell Phone	(813)-442-0891
Team Mentor TRA/NAR Certifications	TRA 12041, Level 3 Certified
Hours Spent ON CDR	386 Person Hours

Table 1.1: PSP-SL Team Summary

1.2 Launch Vehicle Summary

The following section will explain in-depth the launch vehicle that will be constructed for the 2020-2021 Project Voss.

Vehicle Name	All Gas, All Brakes
Target Altitude	4100' AGL
Motor Selection	Cessaroni Technologies Inc. L1115-0
Vehicle Predicted Mass	52.1 lbm
Vehicle Outer Diameter	6.17"
Vehicle Length	87.2"
Vehicle Independent Sections	3
Vehicle Recovery System	Dual Deployment: Apogee and 900' AGL

Table 1.2: Launch Vehicle Summary

1.2.1 Target Altitude

As discussed in PDR, the target altitude for Project Voss is 4100'. This altitude was obtained through various simulations, incorporating both the launch conditions and mass margins. Additionally, the inclusion of the ABCS will allow the team more accurately hit the target apogee, and the possibility of ballast in the nose cone would increase the overall mass of the vehicle and reduce the expected apogee.

1.2.2 Final Motor Choice

After much discussion, the 2021 PSP-SL Project Voss will utilize the 4 grain CTI L1115 reload motor. In-depth analysis and justification for this motor choice is provided in section 3.4.3.

Cesaroni Technology Inc. L1115	
Fuel	Ammonium Perchlorate
Oxidizer	Atomized Aluminum
Thrust Profile	Regressive
Propellant Mass	83.79oz
Gross Mass	154.14oz

Table 1.3: Final Motor Choice

1.2.3 Size and Mass of Launch Vehicle

The 2021 launch vehicle, designed for Project Voss, is a 6" inner diameter launch vehicle. The entirety of the launch vehicle is 87.2" long from the tip of the nose cone to the aft of the swept fins. The launch vehicle consists of a 3" round nose cone (1" internal shoulder), 31" upper airframe, 1" switch band (mounting of 5" avionics bay), and a 50" lower airframe. The vehicle itself is primarily constructed with G12 filament-wound fiberglass, while the nose cone is 3D printed using chopped carbon fiber reinforced nylon, which proves to be much stronger than commonly used 3D printed materials. From tip to aft, the internal components are the primary payload, main parachute, avionics bay, drogue parachute, airbrakes, and MFSS. The predicted mass of the launch vehicle is 52.1lbm.

Project Voss Launch Vehicle	
Expected Mass	52.1 lbm
Length	87.2”
Outer Diameter	6.17”
Rail Size	144”

Table 1.4: Launch Vehicle Specifications

1.2.4 Recovery System

The recovery system ensures controlled descent and landing of the launch vehicle. Major components include two altimeters, a drogue parachute, a main parachute, and black powder charges used for ejection. The specifics on these design choices are outlined in the table below. The components will work to deploy the drogue parachute at apogee, main parachute above 500’, and land the launch vehicle within 2500’ of the launch position. The system uses multiple altimeters and associated ejection systems to ensure redundancy.

Primary Altimeter	Atlus Metrum TeleMetrum
Redundant Altimeter	PerfectFlite StratoLoggerCF
Main Parachute	Rocketman High Perfomance CD 2.2, 144” Diameter
Main Deployment	900’ (700’ for redundant)
Drogue Parachute	Fruity Chutes Classic Elliptical, 24” Diameter
Drogue Deployment	Apogee (Apogee+1s for redundant)
Ejection Charge Type	FFFFG black powder

Table 1.5: Recovery System Specifications

1.3 Payload Summary

PSP-SL's 2020–21 Payload experiment is titled “*Drag and Drop*,” owing to its systems’ central aerobraking and deployment concepts as well as the classic computing phrase.

The Payload team’s systems have been designed to satisfy the competition challenges, as well as challenges the team has imposed upon itself. The payload system comprises of a middle-of-descent deploying Planetary Landing System (PLS) and an apogee-adjusting AeroBraking Control System (ABCS). These payload experiments will remain completely contained within the launch vehicle until flight conditions are satisfied for them to become active. Both experiments have been designed to not interfere with the operation of the launch vehicle until their designated operation events are satisfied.

1.3.1 Primary Payload

The primary payload has been designed to meet the challenge requirements as outlined in the Handbook. The PLS will consist primarily of a deployable Lander Subsystem—or just “Lander”—and its associated Retention and Deployment Subsystem (R&D). The R&D contains the Lander within the Payload Bay until time of deployment, handing all associated flight loads which would otherwise be transferred through the Lander itself. When activated, the R&D ejects the Lander by mechanical means—without producing an additional independent section of the vehicle. The Lander will be fully deployed from the launch vehicle after the deployment of the vehicle’s main parachute, no lower than 500’ AGL. Afterward, the Lander will descend to the ground at a non-ballistic rate through the use of a parachute. Once grounded, the Lander will begin a coordinated orientation sequence, uprighting itself within the required bounds. Afterward, the Lander’s onboard Panoramic Image Capture Subsystem (PICS) cameras will be activated, take a picture of its surroundings, and transmit the data to the Payload Team’s Ground Control Station (GCS) for image processing and display.

1.3.2 Secondary Payload

The secondary payload has been designed to meet the additional technical requirements as outlined by PSP-SL. The ABCS consists of a mechanical apparatus capable of being integrated with the airframe of the vehicle. This device actuates linkages connected to sectioned plates in order to affect the aerodynamic cross-sectional area of the vehicle after the vehicle’s burn has completed, producing increased drag. An internal control system is being developed to monitor flight conditions, and through a closed-loop control system, the control system will actuate the mechanical device to produce the desired amount of drag. The control system will utilize flight conditions to predict the current amount of drag required for the vehicle to attain the desired apogee and will modulate the mechanical system to that end. This system is being implemented to increase the team’s apogee score and is something that the team has not done in previous years. Furthermore, due to the lack of experience with this form of control system on a high-powered rocket, the team has decided to dedicate much effort towards ensuring flight safety and stability.

2 Changes Made Since PDR

2.1 Changes Made to Vehicle Criteria

2.1.1 Vehicle

Since PDR, there have not been any changes made to the launch vehicle structurally. Changes in mass that occurred were solely due to measuring constructed systems, that weighed less than what was allocated to them.

2.1.2 Recovery

After the submission of PDR (particularly after the parachute drop test was conducted), the team chose not to utilize a slide ring on the main parachute to reduce shock loading during deployment. This decision was made because the team determined that it is of paramount importance that the main parachute opens as quickly as possible while still remaining within the margins of safety, specifically in order to grant the payload lander the maximum amount of time to deploy its own parachute. The parachute drop test confirmed that the main parachute opening distance after deployment would be close to the maximum required opening distance, so the slide ring would surely extend this parameter beyond the set limit. The team is confident the main parachute will be able to handle the shock loading of deployment on its own.

2.2 Changes Made to Payload Criteria

2.2.1 Primary Payload

The primary payload has received a great number of new features which assisted in the finalization of the design. While the R&D received a major overhaul in chosen components and structure, the Lander's internal design has since been implemented and iterated upon.

Production of the final design of the R&D required additional constraints on the side of the Lander in order to be resolved. The overall design of the R&D is intended to hold the Lander as securely as possible but must also be entirely out of its way during deployment. Failure to place components in proper locations will assure that the Lander never exits the vehicle. Once the team decided to transition the Lander from a mirrored design to a triple axially symmetric design, the R&D hardware was able to be reworked with a new form factor in mind. On the operational side of the R&D, initial prototyping illuminated potential areas of concern in relation to the mechanism's central method of actuation during flight. This concern has been addressed and discussed amongst the team. While the team believes that there is insufficient reason to expect a possible failure of the R&D system to retain the Lander, the team has worked to ensure that possible failure modes will not result in hazardous conditions for the ground team. Furthermore, confirmation of the mechanical design has enabled the electronic design of the R&D to unfold within the Payload Bay coupler section.

Since PDR, the Lander stand-in has been replaced with an entirely new assembly inspired by designs such as Apple's 2013 Mac Pro, allowing for high component modularity, access points, and 3D print manufacturability. As previously stated, this design is now triple axially symmetric, meaning that the device's central structure consists of three central plates configured differently for a particular subsystem. These include the Lander Control Subsystem, Descent and Landing Subsystem, and power distribution plate. These subsystems now share an internal space and interlock for assembly. Since the required panoramic image needs to be unobstructed, a top cupola section has been added, which contains the Panoramic Image Capture Subsystem's three cameras. Through preliminary testing, this design was deemed feasible by the electronics team. Meanwhile, the Lander's Self Orientation Subsystem's mechanical apparatus forms the base of the Lander at the other end of the body. The orientation method described in PDR has been prototyped and proven to operate as intended. At this point, the team believes that this concept has no reason to be modified. In the future, additional development into the Self Orientation Subsystem's control system will be required, but preliminary in-depth discussion has revealed promising approaches to the final methodology.

2.2.2 Secondary Payload

The secondary payload has finalized an essential electronics bay design. Meanwhile, the team has finished execution of vital aerodynamic analyses, developed further control system safety verification methods, and initiated a study into the fundamental formulae for control system design in this particular situation.

Initial prototyping of the mechanical system proved both promising and illuminating about the final manufacturing methodology to be used. The team continues to be satisfied by this design. On the other hand, much additional work needed to go towards providing mounting space for the ABCS's electronics systems. The final design incorporates the use of the Lower Bay's coupler section. While the team's large volume of components has necessitated iteration of this electronics bay to ensure all components can fit without falling victim to overheating, interference, and vibration, the final design has struck a balance between manufacturability and modularity.

While CAD work was being finished, the team's aerodynamics specialists were able to produce helpful results through CFD. These results appear to confirm the team's confidence in the ABCS's design and have allowed for further inquiry into the projected efficacy of the ABCS. Investigating multiple methods of altitude prediction, the team has formed a foundation for future development of the ABCS's control system as it progresses from burnout to apogee. Of course, always interested in safety, the team performed additional research into the ABCS's possible failure modes and procedures. The team has deliberated the system's options given the provided requirements as well as those imposed by the team itself. While the team has since determined not to pursue advanced methods like pressure-based stall detection, additional procedures will be implemented to eliminate the impact of the ABCS in the case of detected vehicle failure.

2.3 Project Plan Modifications

There have been no changes made to the project GANTT chart since PDR. Only minor updates have been made to the status of each team and NASA derived requirement.

2.3.1 Testing Procedures

Since PDR, the team has begun drafting testing procedures for each test the subteams perform. These testing procedures are included in sections 6.1 and 6.2. Each procedure discusses the purpose of the test, the setup and equipment required, the meaning of the possible results, and includes a full discussion of the results yielded once the test has been performed.

3 Vehicle Criteria

3.1 Design and Verification of Launch Vehicle

3.1.1 Mission Statement and Mission Success Criteria

Project Voss's goal is to construct a reusable launch vehicle capable of launching to a predetermined altitude of 4100'. This launch vehicle will contain a primary lander payload system, cameras that will record throughout the flight, and an aerobraking system. New members will be educated and have hands-on experience constructing this launch vehicle and gaining necessary knowledge on payload design and rocketry.

For the construction team, the mission will be deemed to be a success if the following criteria are met:

The Vehicle will maintain stability throughout the flight
The Vehicle will safely deploy all recovery systems
The Vehicle will remain as a single unit throughout the flight
The Vehicle will land with less than the maximum allowable kinetic energy
The Vehicle will actively control its apogee using an Aerobraking Control System
The Vehicle will be reusable without repairs or alterations
The Vehicle will deploy the payload at the designated altitude
The Team will abide by standard engineering and quality control practices during the design, construction, and launch of the Vehicle

Table 3.1 Mission Success Criteria

3.1.2 Vehicle Design

3.1.2.1 Chosen Design from PDR

The chosen design to fulfill the 2020-2021 Project Voss is a 6" inner diameter launch vehicle. The vehicle features a hemispherical nose cone, which houses the IFVR cameras to record the launch forward, side, and aft. The total length of the launch vehicle (tip to swept fin) is 87.2". The length consists of a 3" radius hemispherical nose cone, 31" upper airframe, 1" switch band, 50" lower airframe, and 2.2" from the sweep of the fins. The vehicle will utilize three trapezoidal fins, with a 12" root chord, a height of 6.5", tip chord of 6.2", and a sweep length of 7". The expected mass of the final launch vehicle is 52.1 lbm, which the team expects to be a slightly low estimate due to additional mass from components such as nuts and bolts, as well as paint. The vehicle also can include ballast in the nose cone to increase the vehicle's stability.

3.1.2.1.1 Nose Cone

Between PDR and CDR, the hemispherical nosecone choice has been confirmed based on promising results from the subscale flight. The subscale flight showed that the hemispherical design performed as predicted in simulations. The ease of manufacturing and simplicity of the design were important factors in this decision as well.

The final design is 6” in diameter and has a 0.787” lip protruding from the nose cone's aft section. This lip helps ensure radial stability between the nose cone and the payload bay. The lip is shorter than the typical coupler sections between airframes. Still, the decision was acceptable because the nose cone is not being jettisoned and would provide more room for both the camera bay and the payload bay.

The nose cone's main features include the six holes that line the wall of the nose cone. These holes are used for the attachment of the nosecone to the attachment plate. Because the design is 3D printed, the holes for attachment contain square nut slots to avoid threaded holes, which are difficult to 3D print. Furthermore, the nose cone also contains openings that will house the Inflight Video Recording System (IFVR). One camera will be pointed forward and one outward within the nosecone. Pieces will be cut from a 6” acrylic dome to cover these openings, which will be attached to a lip within the openings.

The aft-facing camera will be the OV5647 camera, chosen for its small profile and flexible PCB, which increases the number of possible orientations in the nose cone. The camera sensor will be secured from inside the nose cone with additional hardware, as the flexible PCB does not provide any reliable mounting points. The remaining two cameras chosen for the IFVR will both be the UC-365 Arducam cameras. These cameras feature a solid PCB with four holes of diameter 2.2mm. The nosecone will feature holes for both the forward and outward-facing cameras, which will allow them to be secured to the nosecone with M2 screws.

The housing for the battery, cameras, and computers will be attached to the four M3 bolt attachment points located on the forward opening within the nosecone.

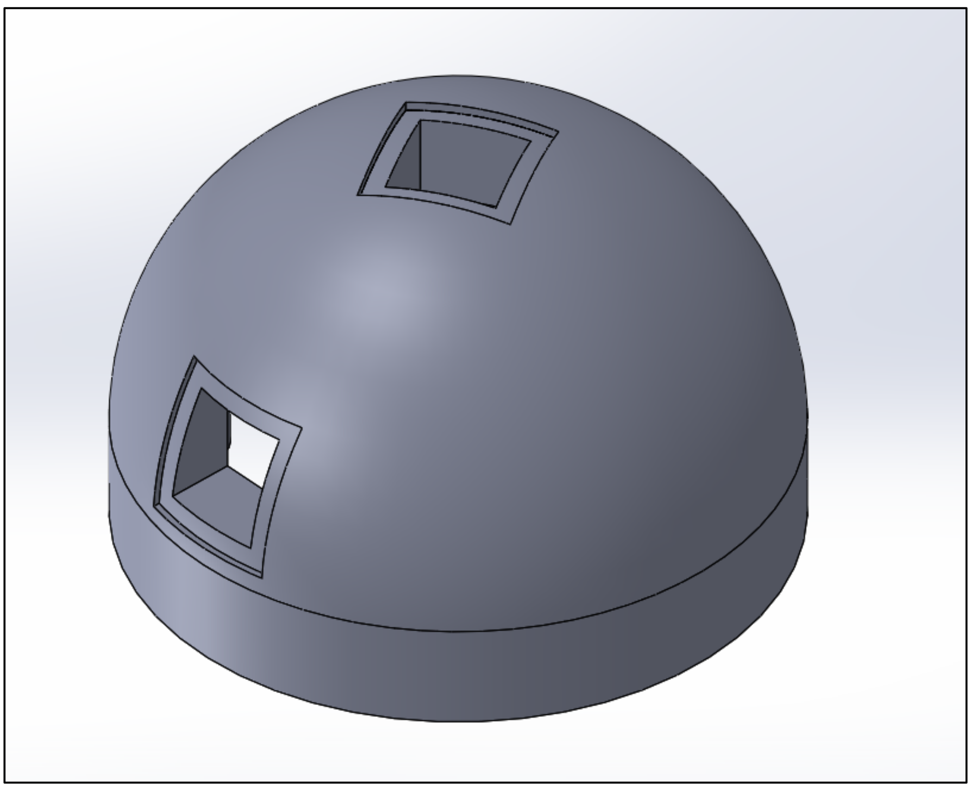


Figure 3.1: Side view of the camera openings in the nose cone

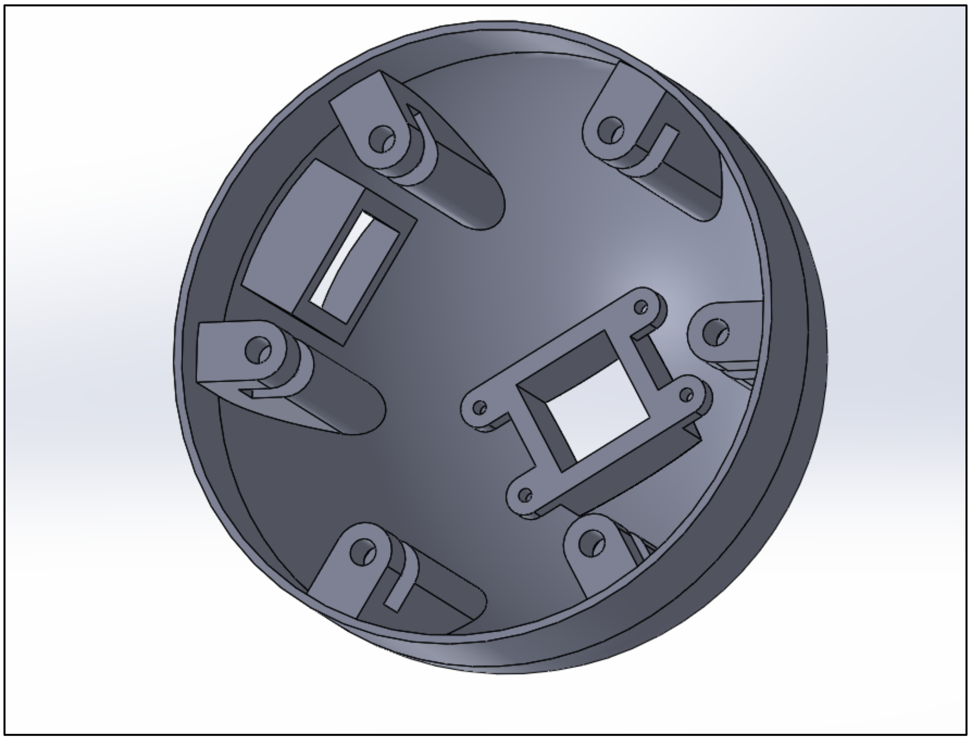


Figure 3.2: Inside view of the camera openings in the nose cone

3.1.2.1.2 Inflight Video Recording System (IFVR)

The Inflight Video Recording System (IFVR) consists of three Raspberry Pi Zero single board computers, two UC-365 Arducam cameras, and one OV5647 camera. The system will feature a camera facing forward, a camera facing axially outward, and a camera facing aft of the launch vehicle, all housed within the launch vehicle's nose cone. The cameras will be powered by one 18650 Li-Ion cell with a suitable charge protection circuit to prevent over-discharge, over-current, or any other harmful malfunction of the battery. The single cell was chosen as it has the capacity to power each camera for more than three hours, greatly exceeding the requirements for the IFVR.

The cameras will be activated using a Python script when the IFVR system is activated before the vehicle's launch. Once the flight is completed, a switch will be used to terminate the recordings so that they will save safely to the SD cards. The UC-365 Arducam cameras will record in 60fps 1080p, and the OV5647 camera will record in 48fps 1080p. These recording formats and resolutions were chosen through testing of both cameras to determine the maximum quality. This testing entailed determining at which settings the recordings looked the most visually appealing. Additionally, a 3-hour recording in 60fps 1080p required less than 23GB of SD card storage space, necessitating at least 32GB SD cards for a margin of safety.

The IFVR was tested in a cold environment at 0° F and in more temperate conditions at 50° F – 70° F to ensure camera function in all environments. Both cameras recorded their videos without issue despite any difference in conditions, validating the current hardware for use on the vehicle. It was also determined that battery capacity and current draw from the Raspberry Pi computers did not notably change throughout the different conditions. One Raspberry Pi computer draws around 80mA of current when idle and around 220mA of current when recording. Given that the battery is rated for 1880mAh of capacity and that there will be three cameras, this confirms the necessary battery capacity.

3.1.2.2 Justification for Chosen CDR Design

The team is confident that the current design can satisfy all the requirements set for Project Voss. Even with its relatively short length compared to the last competition, the distribution of mass allows the vehicle to obtain a stability of 2.94cal off the rail. This mass is mainly distributed in the forward portion of the launch vehicle, allowing for more complex systems in the primary payload.

Due to the success of last year's project, the team again decided to use an inner diameter of 6" for the launch vehicle. Although this does increase the vehicle's mass, the trade-off is the ability to methodically design the internal systems with the additional volume from the larger cross-sectional area. The team believes that having this freedom of design from the extra room is worth needing a more powerful motor to propel the additional mass.

The nose cone used this year is designed to be hemispherical, as opposed to a Von Karman design used in previous years. This decision was made for various reasons. Last year, the payload team used a large portion of the nose cone for their R&D, which the payload team decided early on in this project was not needed. Utilizing a similar nose cone would have resulted in dead weight, which is now allocated to various subsystems. Additionally, when traveling at the launch vehicle's speed, most of the drag induced on a blunt body is pressure drag. Since the vehicle will be traveling at about Mach 0.5, this pressure drag will not be very concerning due to the relatively slow speeds. Using a Von Karman nose cone would result in a much larger wetted surface and an undesirable amount of skin drag. Lastly, the current nose cone design allows the team to more effectively house the cameras capable of recording the forward, aft, and side view of the launch's ascent/descent.

The team again chose to include three fins, offset 120° from each other. By including three fins instead of four, the vehicle's mass decreases, and the drag induced by the booster section is greatly decreased.

The team chose to use the CTI L1115 as the motor to fulfill Project Voss. This choice was made for many reasons, which will be discussed in section 3.4.3.

3.1.3 **Dimensional Drawings**

3.1.3.1 Assembled Launch Vehicle

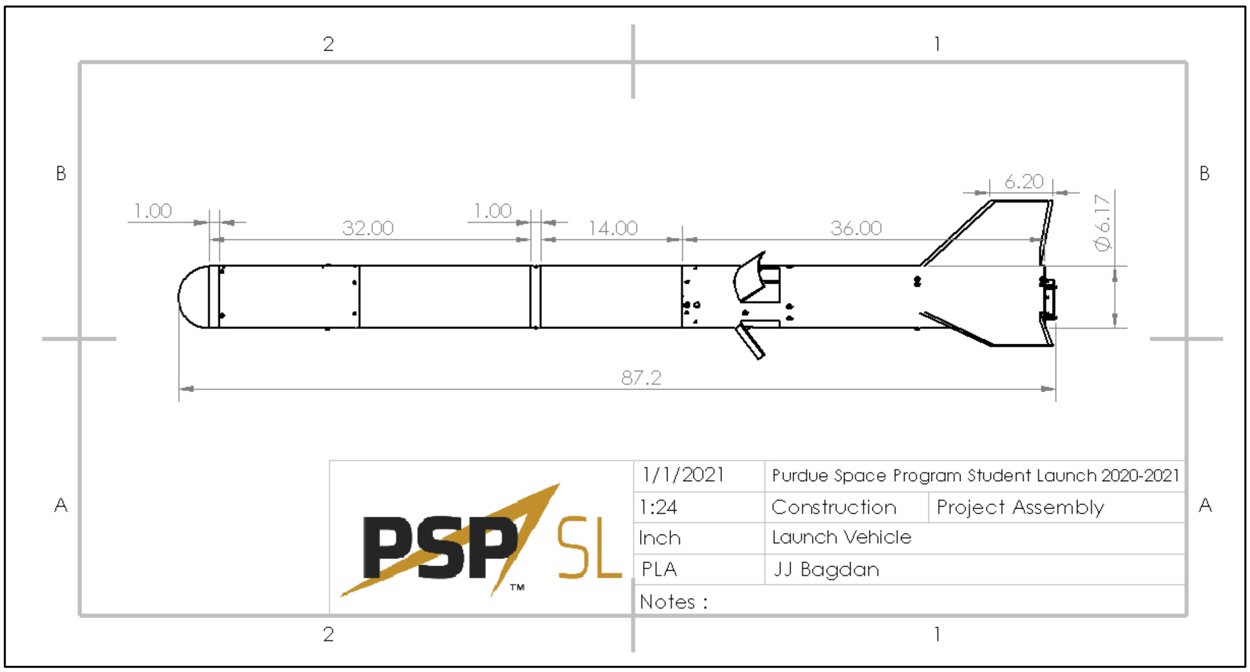


Figure 3.3 Complete Airframe Dimensional Drawing

The complete airframe shown above is constructed of G12 fiberglass and metal structural components. It consists of three main parts: the nose cone, the upper airframe, and the lower airframe. This drawing does not include the inner components of the launch vehicle, such as the parachutes or linkages. The total length of the launch vehicle is 87.2”, with an outer diameter of 6.17” and an inner diameter of 6”.

3.1.3.2 Lower Airframe Subsystem and Components

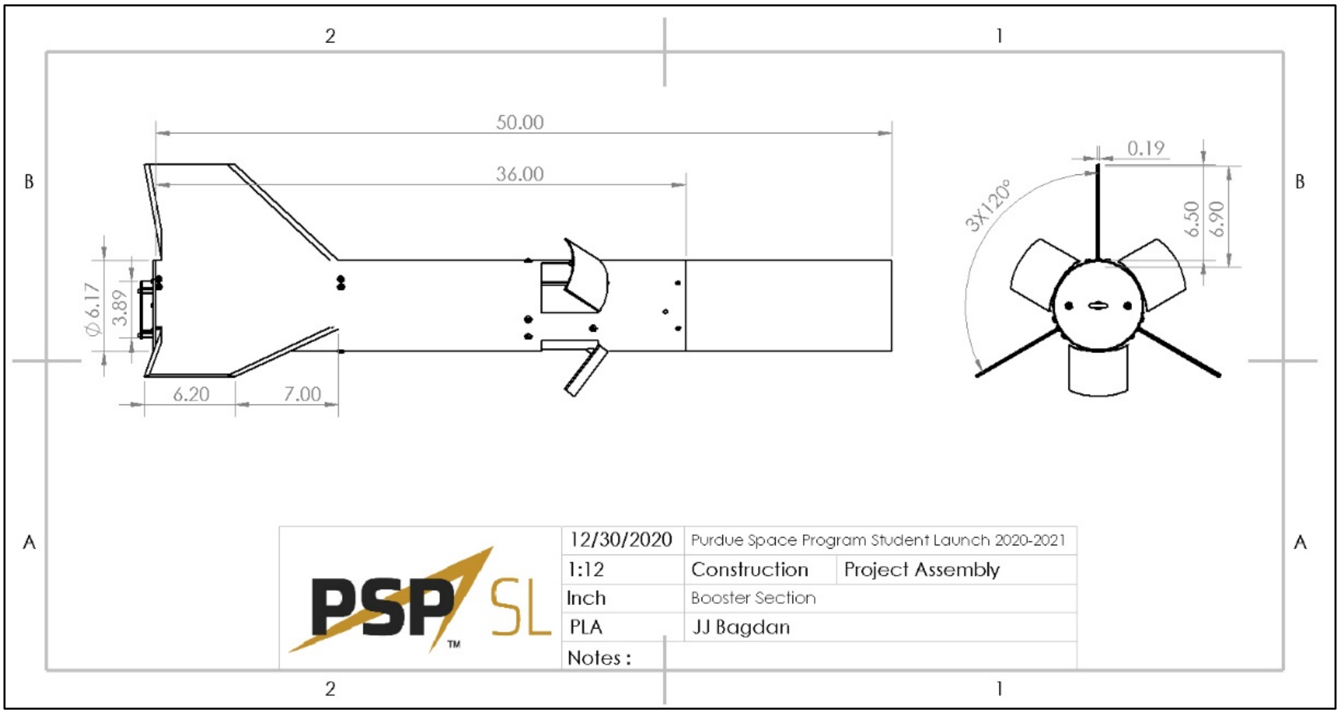


Figure 3.4: Lower Airframe Dimensional Drawing

As shown above, the lower airframe has an outer diameter of 6.17" and has a total length of 50". The fins are 6.5" wide and 12.4" tall, and they are mounted tri-angularly around the base using a special mounting structure, which is discussed below. The lower airframe's job is to house the motor, its related components, and the ABCS.

3.1.3.2.1 Motor & Fin Support Structure (MFSS)

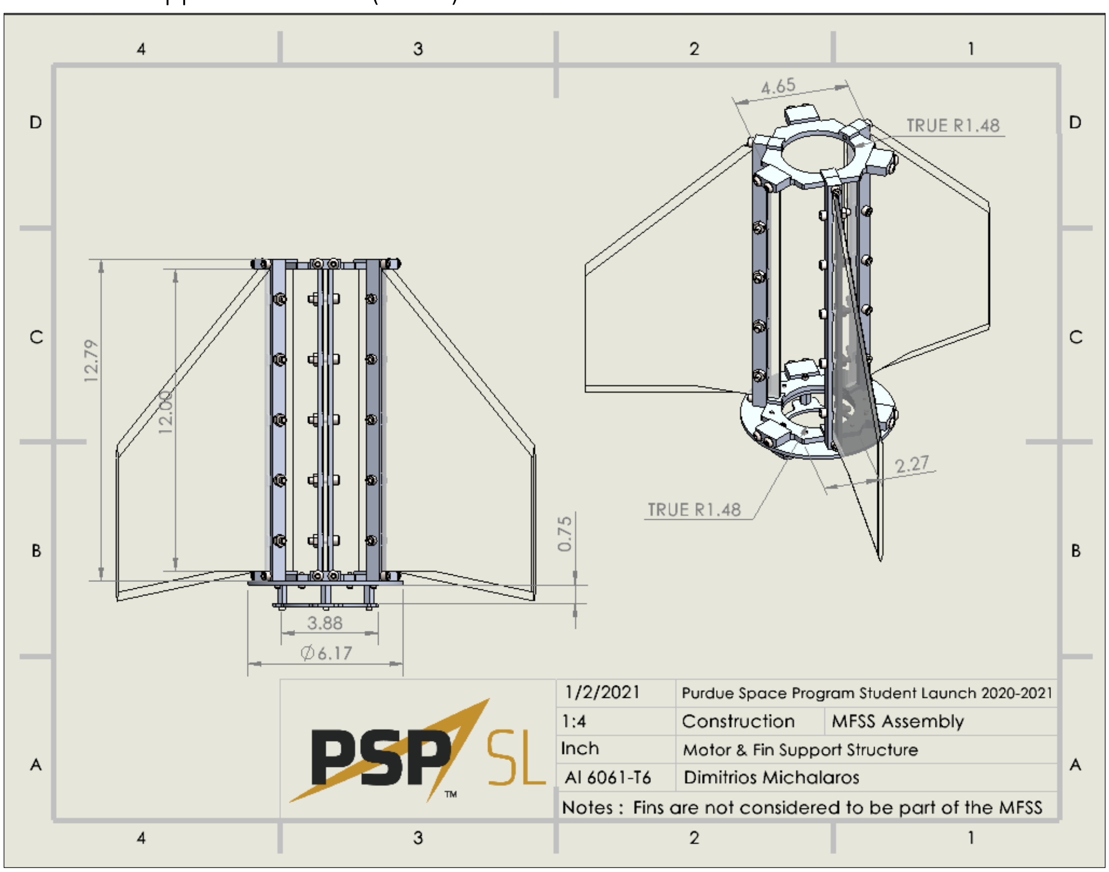


Figure 3.5: MFSS Assembly dimensional drawing.

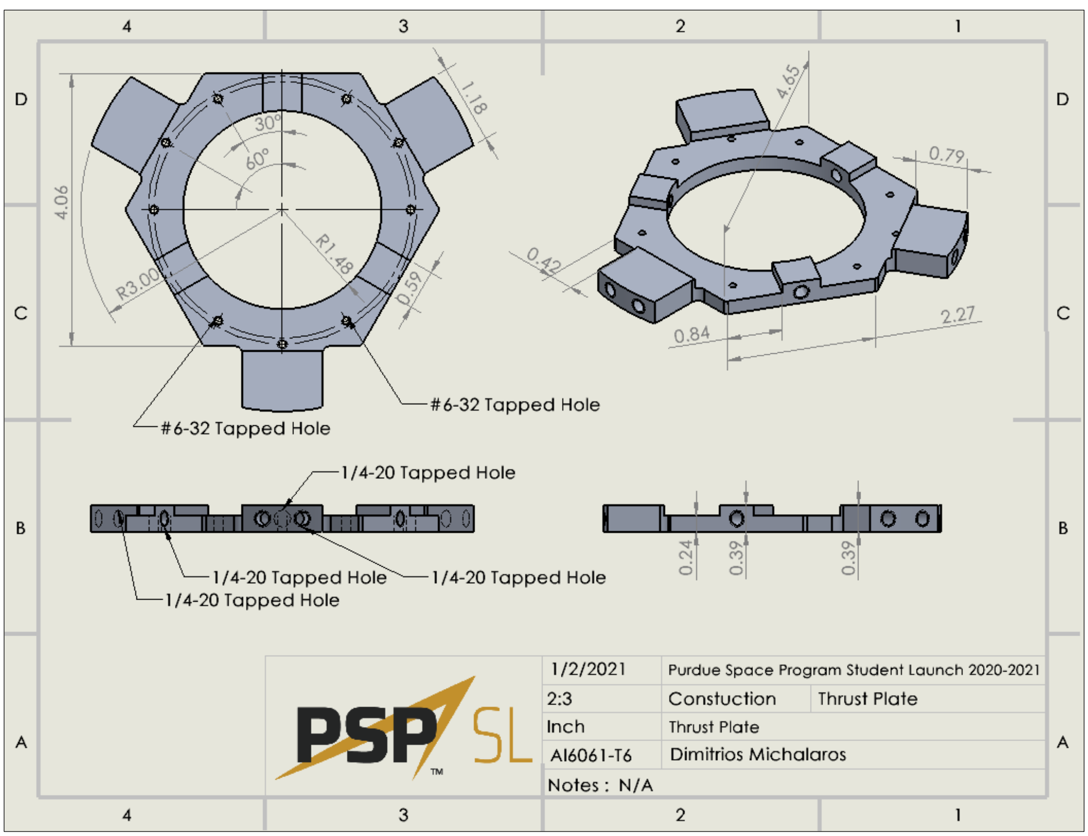
The Motor and Fin Support Structure (MFSS) has not changed since the Preliminary Design Review. The Bill of Materials for the MFSS is shown below.

Component	**Quantity**	**Material**	**Single Component Mass (lbm)**	**Total Mass (lbm)**	**Construction Method**
Thrust Plate	1	Aluminum 6061-T6	0.29	0.29	CNC Milled
Thrust Plate Flange	1	Aluminum 6061-T6	0.35	0.35	CNC Milled
Fin Support Spar	3	Aluminum 6061-T6	0.25	0.75	CNC Milled
Upper Centering Plate	1	Aluminum 6061-T6	0.29	0.29	CNC Milled
Female #6-32 standoff	3	Aluminum 6061-T6	0.0083	0.0249	Off-the-shelf
Motor Retainer Plate	1	Aluminum 6061-T6	0.045	0.045	Waterjet/Laser-cut
#6-32 ANSI Cap Head Screw 0.5" long	9	Steel (ρ = 0.278 lbm/in^3)	0.034	0.306	Off-the-shelf
#6-32 ANSI Cap Head Screw 0.75" long	3	Steel (ρ = 0.278 lbm/in^3)	0.0044	0.0132	Off-the-shelf
1/4-20 ANSI inch Button Screw 0.75" long	12	Steel (ρ = 0.278 lbm/in^3)	0.014	0.168	Off-the-shelf
1/4-20 ANSI inch Cap Head Screw 1.125" long	6	Steel (ρ = 0.278 lbm/in^3)	0.022	0.132	Off-the-shelf
1/4-20 ANSI inch Cap Head Screw 1" long	15	Steel (ρ = 0.278 lbm/in^3)	0.02	0.3	Off-the-shelf
1/4-20 ANSI inch Nut Hex	15	Steel (ρ = 0.278 lbm/in^3)	0.0071	0.1065	Off-the-shelf
Total Component Mass:				1.7499	
Total Fastener Mass:				1.0257	
Total Top-Level Assembly Mass				2.7756	

Table 3.2: Bill of Materials for the MFSS Assembly

Dimensional drawings of each of the custom-built components are shown in the following sections.

3.1.3.2.1.1 Thrust Plate



f

Figure 3.6: Thrust Plate Dimensional Drawing

3.1.3.2.1.2 Thrust Plate Flange

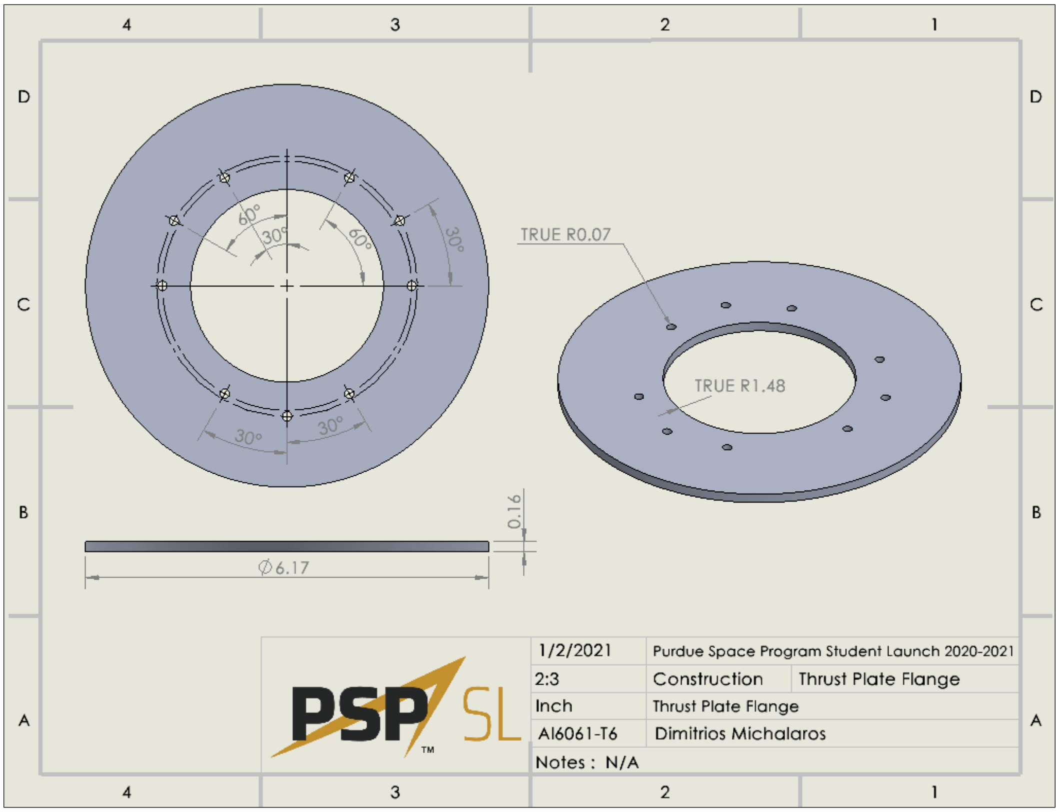


Figure 3.7: Thrust Plate Flange Dimensional Drawing

3.1.3.2.1.3 Centering Plate

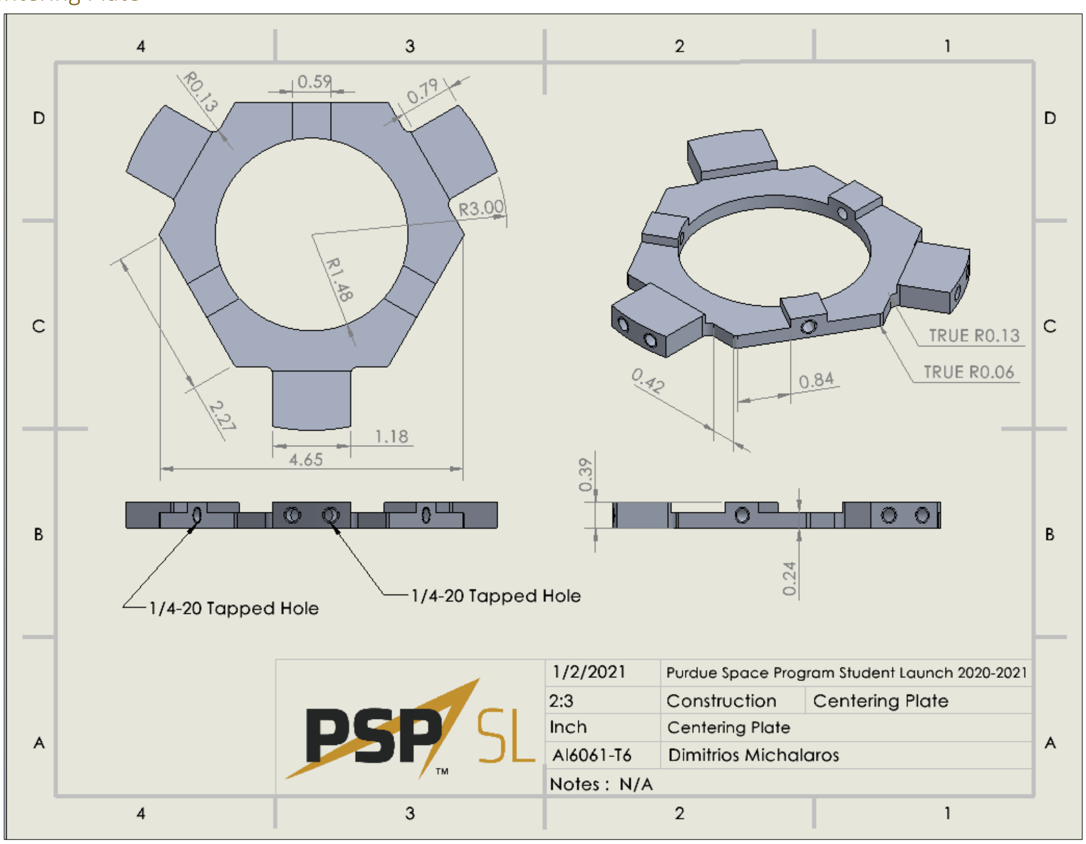


Figure 3.8: Centering Plate Dimensional Drawing

3.1.3.2.1.4 Fin Support Spar

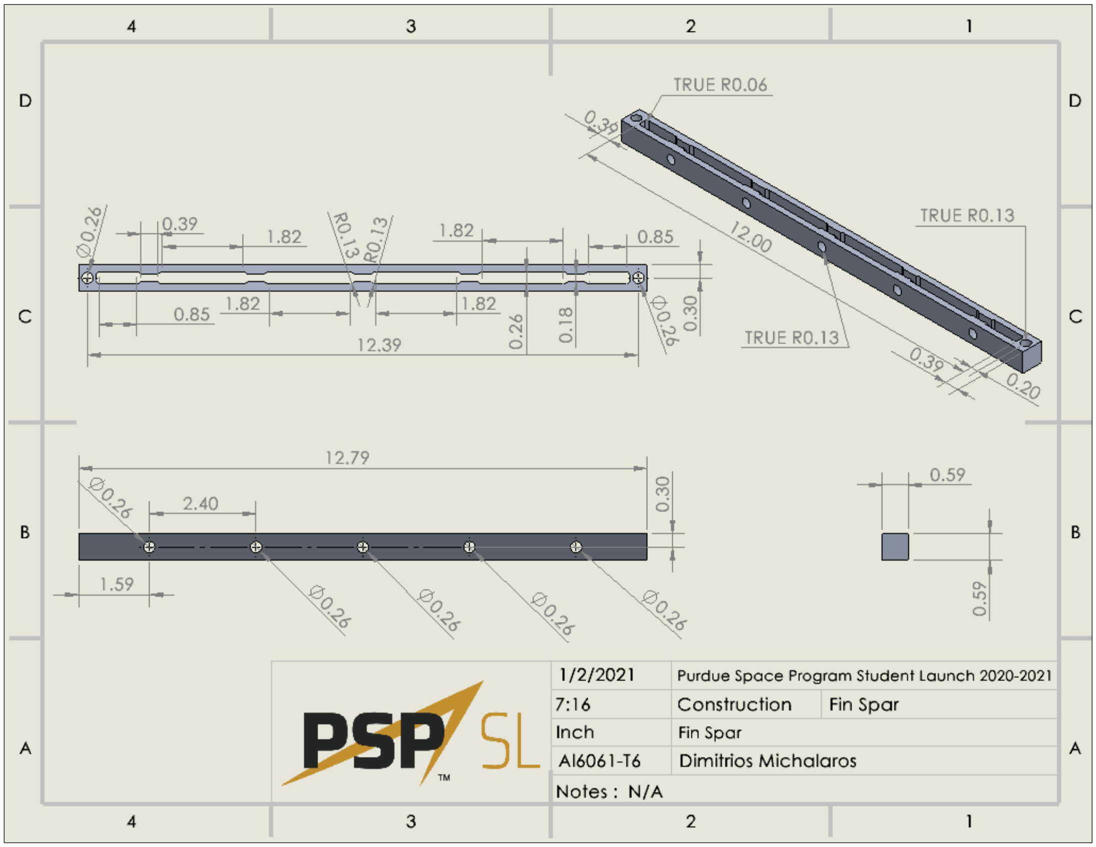


Figure 3.9: Fin Support Spar Dimensional Drawing

3.1.3.2.1.5 Motor Retainer Plate

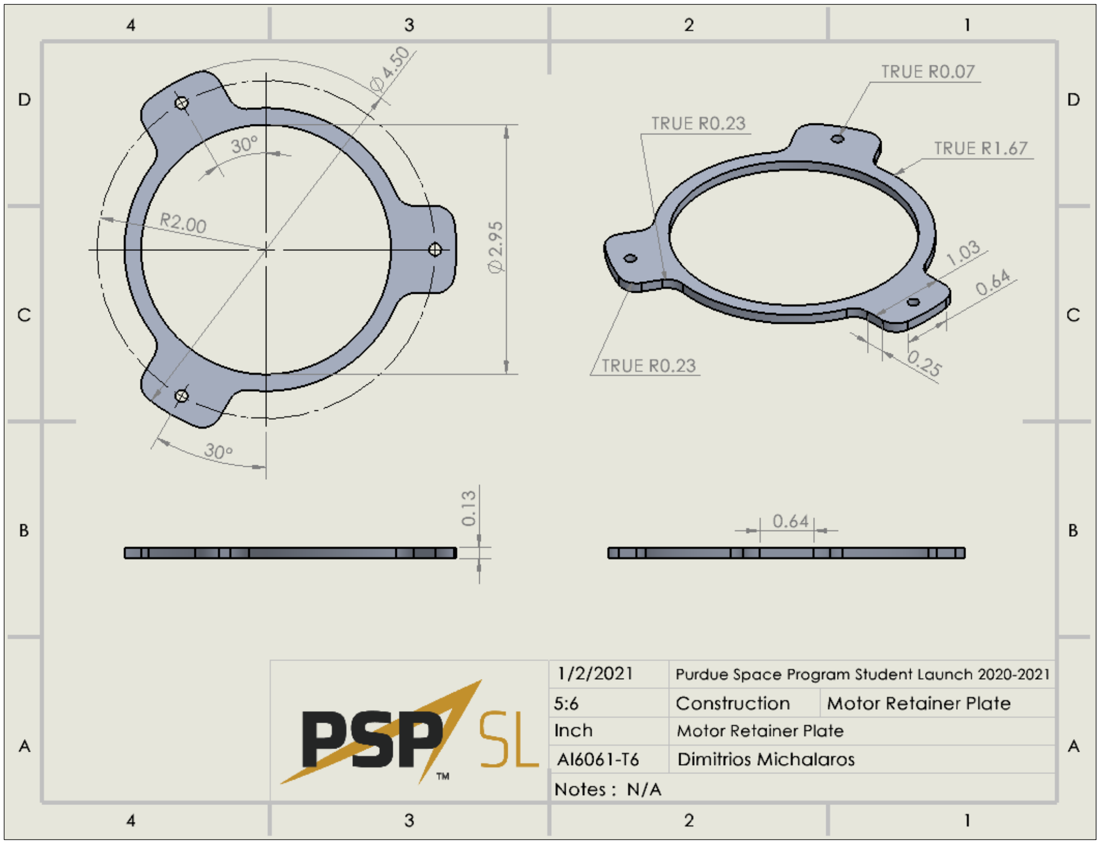


Figure 3.10: Motor Retainer Plate Dimensional Drawing

3.1.3.3 Avionics Bay Subsystem and Components

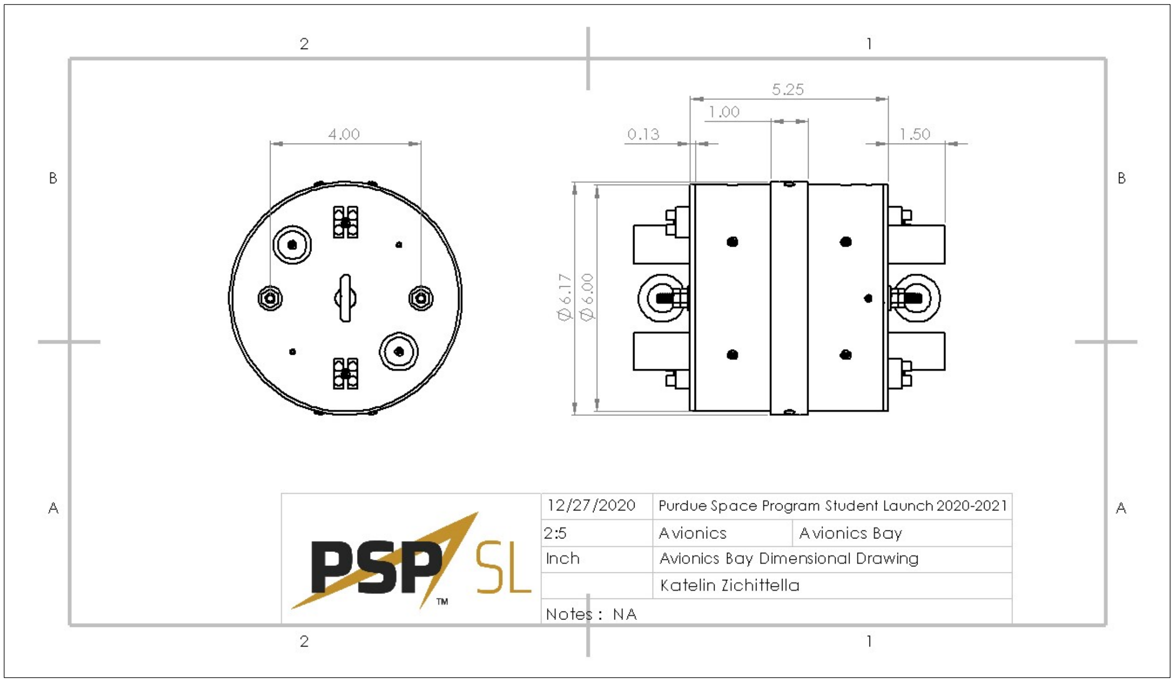


Figure 3.11: Avionics Bay Dimensional Drawing

The avionics bay has a coupler outer diameter of 6”, coupler length of 5”, and a total length of 8.25”, including the black powder canisters. It also has a switch band around the center with a width of 1”. The avionics bay's primary purpose is to house the primary and redundant altimeter/ejection systems and provide an attachment point for the drogue and main parachutes.

3.1.3.4 Upper Airframe Subsystem and Components

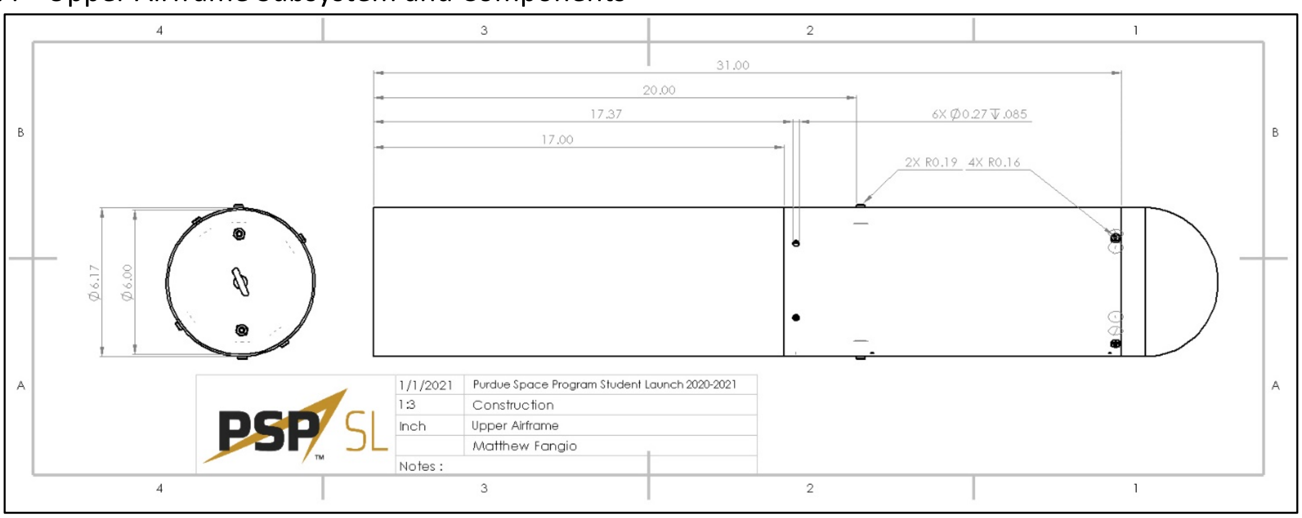


Figure 3.12: UpperAirframe Dimensional Drawing

The upper airframe has a diameter of 6.17" and a total length of 31”. The upper airframe houses all recovery gear and attaches to the avionics and payload bays.

3.1.3.5 Nose Cone and Payload Subsystem and Components

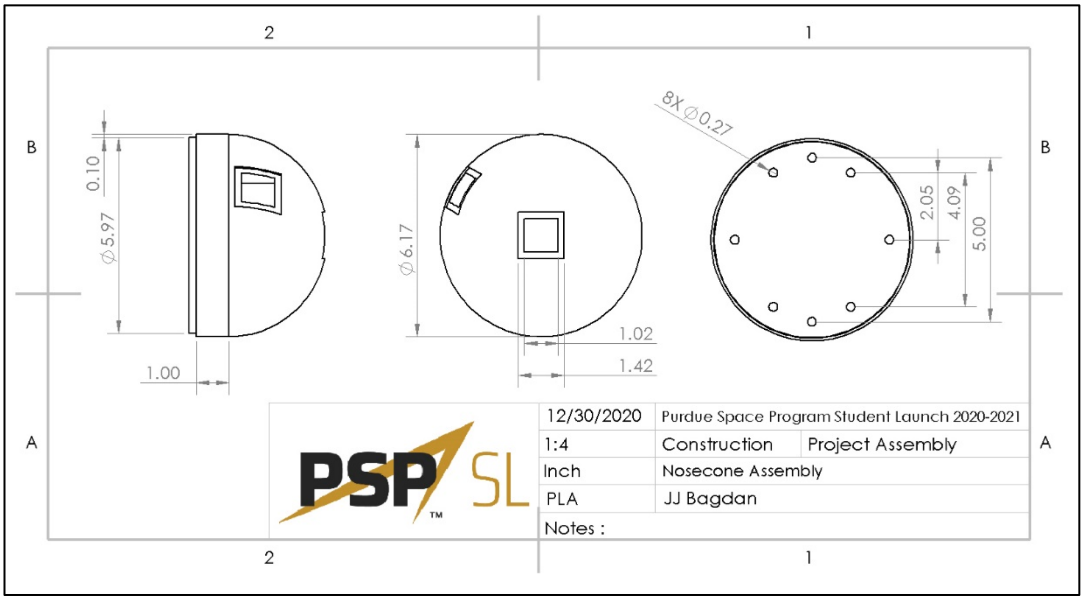


Figure 3.13. Nose Cone Dimensional Drawing

The nose cone's spherical design is meant to take advantage of the relatively low airspeeds the launch vehicle will encounter. It has a diameter of 3" and a shoulder-length of 1". It houses the cameras for the IFVR to capture video from the flight and includes viewports for each of the cameras.

3.1.4 **Locations of Separation Points and Energetic Materials**

The vehicle's separation points are located between the payload section and the upper recovery section, and between the booster section and the lower recovery section. The three overall vehicle sections are tethered together by the parachute shock cords. The ejection charges (energetic materials) are located on the exterior of the forward and aft avionics bay bulkheads. By placing the separation points and energetic materials at these locations, the ejection charges' detonation forces the drogue and main parachutes up and out the other ends of the recovery sections in an efficient canon-like deployment.

3.1.5 **Manufacturing Readiness**

Several parts have been bought from Madcow Rocketry, whereas several stock parts have either been purchased from McMaster-Carr or were taken for free from the Bechtel Innovation Design Center (BIDC) stockpile. More specifically, the stock used for the thrust and centering plate was taken from BIDC and is Aluminum 6061-T6. The stock block was split into two parts of roughly equal thickness with a vertical bandsaw, then had their thickness further reduced to provide an even finish to ease probing and CNC machining processes. The stock for the motor retainer plate and the thrust plate flange are also from the BIDC stockpile. The fin support spars' stock parts were purchased from McMaster-Carr because most of the already available stock at the BIDC did not have sufficient length. These purchased stock parts are Al 6061-T6.

The launch vehicle's three fins were outsourced and manufactured out of fiberglass. For the aluminum parts of MFSS, the team is conducting the manufacturing process utilizing machines available at the BIDC and the Aerospace Sciences Laboratory at Purdue Airport. Currently, a timeline for developing the MFSS has been prioritized as they require more complex machines and thus training to operate. The machining process for the MFSS contains 7 independent parts: the thrust plate, the centering plate, the motor retainer, the thrust plate flange, and three fin spars. The thrust plate and centering plate first require CNC milling on a 3-axis machine to create dovetails to clamp the parts onto a 5-axis work-holding jaw, since the parts themselves require CNC milling on a 5-axis enabled machine. This is due to the six holes that have been placed radially on the spokes of each plate. Both the thrust and the centering plate use the same CAM setup, except for the nine vertical #6-32 holes that only exist as an operation on the thrust plate file only. The last operation for the thrust/centering plates will be done on a 3-axis CNC mill with a chuck type work-holding jaw that will clamp onto the inner hole at the center of the plate and then the dovetail and leftover stock at the bottom will be faced down in a single operation.

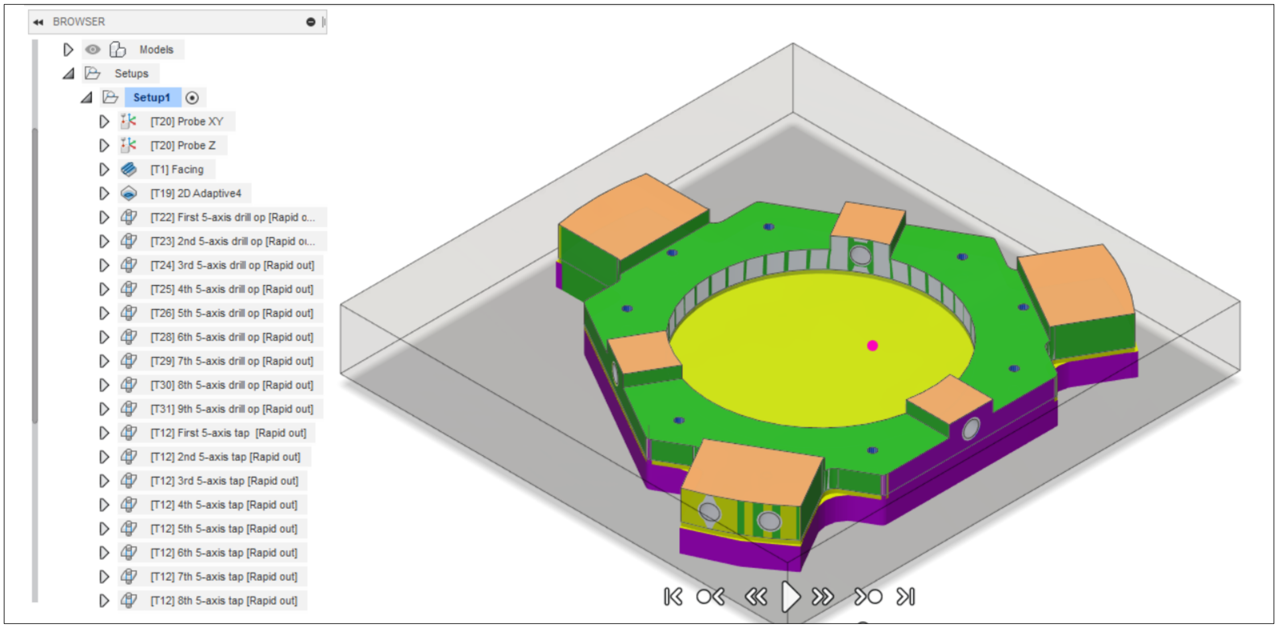


Figure 3.14: Thrust plate CAM simulation in Autodesk Fusion360 of the remaining stock before the last facing down operation. Dovetail is still attached but not visible. Work-holding jaw and tool are hidden.

The motor retainer will be machined via a water jet cutter out of sheet metal stock that has been reserved in BIDC. The thrust plate flange will also be cut using a water jet-following face mill operation to reduce the material to the proper thickness. Finally, each spar will be CNC milled on a 3-axis mill out of the Al 6061-T6 bar stock parts, which have been cut to the proper length using a band saw.

The team is confident in its ability to develop these parts with the proper resources efficiently. Most of the components have been ordered and are being prepared for the machining processes.

3.1.6 **Design Integrity**

3.1.6.1 Suitability of Shape and Fin Style for Mission

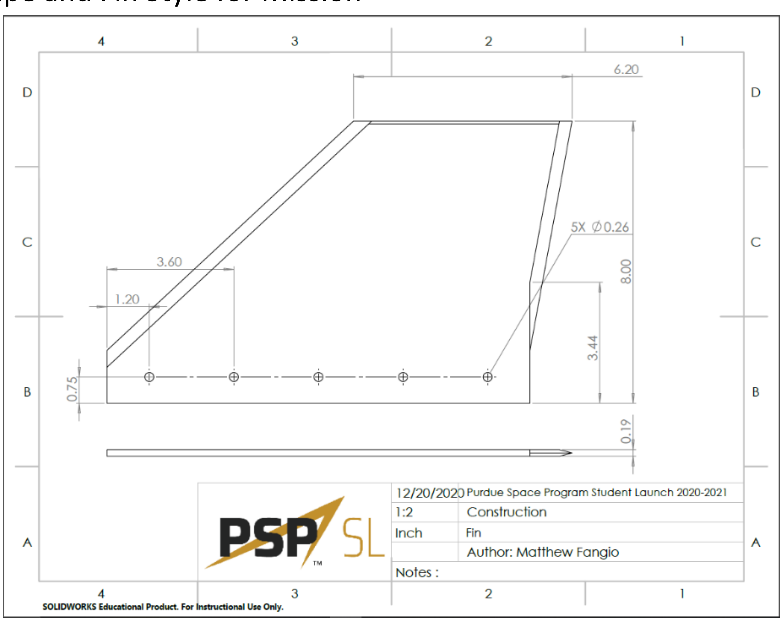


Figure 1.15: Fin Dimensional Drawing

This design's fin set consists of three trapezoidal fins placed symmetrically around the launch vehicle base and constructed of G10 fiberglass. This arrangement will create the best balance between effective surface area and additional weight, allowing the launch vehicle to reach its highest altitude with an acceptable amount of stability. The entire structure is estimated to weigh 2.775 lbm, which is less than last year, as the current launch vehicle has a dedicated fin mounting structure rather than using epoxy for each fin.

3.1.6.2 Proper Use of Materials

As mentioned previously, the components of the MFSS are manufactured using Al 6061-T6. This selection was made after a trade study where Al 6061-T6 was compared to two other commonly used alloys in constructing structural parts. Greater weight was given to the availability of stock at the BIDC and cost of purchase of a stock part that has the dimension needed to make the thrust or the centering plate from McMaster-Carr since the BIDC has a long process from the start of CAM to completion of the part. Availability at BIDC is based on abundance and capability to identify the type of stock used. The team often found stock parts close to the necessary dimensions clearly labeled as Al 6061-T6, but no stock parts labeled as Al 7075 in the stockpile of BIDC.

DECISION CRITERIA		MFSS MATERIAL CONFIGURATION OPTIONS								
		AL 6061 - T6			AL 7075			SAE304 STAINLESS STEEL		
Subteam Requirements:		**Predicted Status:**		**Cleared?**	**Predicted Status:**		**Cleared?**	**Predicted Status:**		**Cleared?**
High Yield and Tensile Strength		High		Y	High		Y	High		Y
Low Mass Density		Low		Y	Low		Y	High		N
Construction Wants:	**Weight:**	**Value:**	**Unit:**	**Score:**	**Value:**	**Unit:**	**Score:**	**Value:**	**Unit:**	**Score:**
Mass Density		0.10	lb/(in^3)	5	0.10	lb/(in^3)	5	0.29	lb/(in^3)	2
Yield Strength		35,000.00	psi	3	73,000	psi	5	29,733	psi	2
Yield Strength / Mass Density Ratio	0.3	358,813.57	in	3	715,686.27	in	5	103,960.62	in	1
Availability at BIDC*	0.35	High		5	Low		2	Very Low		1
Cost for a 6"x6" plate, 1 ± 0.125" thick	0.3	$37.53	USD	5	$70.92	USD	2	$63.18	USD	2
Manufacturability	0.05	Good		4	Excellent		5	Poor		1
Total Merit:		**4**			**3**			**1**		
SELECTED CONFIGURATION:		**X**			_			_		

Weight Sum:	1	Must be 1

Grade Scale:	Reference Value:
Excellent	5
High /Good	4
Medium	3
Low	2
Very Low / Poor	1

Table 3.3: MFSS Material Trade Study

The material used for the lower airframe and the fins is G10 fiberglass, which is a material that has a lower mass density than aluminum (0.065 lb/in^3 for G10 vs. 0.1 lb/in^3 for Al 6061-T6 and Al 7075). G10 fiberglass also has a comparable tensile strength with the yield and tensile strength of Al 6061-T6. Specifically, G10 fiberglass has a tensile strength between 38,000 psi and 45,000 psi, depending on the direction (crosswise and lengthwise application of stress, respectively).

3.1.6.2.1 Fins

This year's fin design is similar to last year's, but it has been optimized for reaching a higher altitude. Three fiberglass trapezoidal fins were chosen, with each fin having a root chord of 12", a tip chord of 6.2", a height of 6.5", a sweep length of 7", a sweep angle of 47.1 degrees, and a thickness of .187". The significant change from last year is the fin mounting structure, designed to avoid the need for overweight epoxy and the imprecise construction process of creating internal and external epoxy fillets.

The team's first decision was whether to keep the trapezoidal fins from last year or use new elliptical fins. When tested side by side in OpenRocket, it was found that elliptical fins provided no altitude advantage without compromising stability. Swept trapezoidal fins were also determined to be simpler to manufacture and were chosen as the winning design.

The next decision to make was height, tip chord, and root chord, ultimately determining the fins' surface area. Surface area is critical in determining the aerodynamic lift force for fins as well as the parasitic drag created, so it must be delicately balanced ($L = C_L P A$, where *L* is the aerodynamic lift force, Cl is coefficient of lift, *P* is dynamic pressure $P = \frac{1}{2}\rho V^2$ and *A* is the surface area of the fin). ($D = \frac{1}{2} C_d \rho V^2 A$, where D is the drag force, ρ is density, C_d is coefficient of drag, A is the surface area of the fin, and V is the velocity of the fin). The lift force is what the fins use to fix the launch vehicle's trajectory if it strays from its path, so the team wants to maximize that force. On the other hand, excessive drag lowers the launch vehicle's altitude potential, so the team wants to minimize that force. OpenRocket was used to test combinations of height ranging from 5" to 8" in increments of .75", combinations of tip chord ranging from 4.5" to 7.75" in increments of .75", and combinations of root chord ranging from 10.5" to 13" in increments of

.75". The results showed that the best combination for maximum altitude with acceptable stability was a height of 6.5", and tip chord of 6.2", and a root chord of 12."

Once height, tip chord, and root chord were chosen, the sweep angle and sweep length naturally followed as 47.1 degrees and 7" respectively. This arrangement provided an optimized balance between aerodynamic lift and drag, allowing the launch vehicle to reach its greatest altitude.

The team continues to use G10 fiberglass as its high tensile strength and mass density ratio, combined with its manufacturability, makes for an ideal material. Consequently, the team will also continue to use a fin thickness of .187," as it has been proven to be optimal for fins made out of G10 fiberglass.

The significant change the team made to the fin design this year is our dedicated fin mounting structure. Last year's construction with epoxy was messy, time-consuming, imprecise, and lead to a heavy rocket. To avoid these complications, the team designed spars to hold each fin. Each fin and spar have five holes that line up once the fin has been inserted so screws can be used to secure the two together. The slot in the spars where the fins are inserted is .1 mm thinner than the fins, ensuring a snug fit and widening via sanding if necessary. Each spar is then connected to the upper centering plate and the thrust plate to transfer the stresses of flight from the fin onto the launch vehicle's main structure. This design reduces the overall launch vehicle weight by eliminating the need for epoxy.

FEA Fin Analysis

All FEA work on the fins was done through the SolidWorks simulation program. To simulate the lift force on the fin, 50 N was applied to an area of .42 in^2 marked out along the tip chord and .0625 into the fin, represented by purple arrows. This force results in 26.63 psi being applied to the target area, roughly twice the standard atmospheric pressure. Anchor points were placed inside each of the five attachment points to represent the screws and spar, which will hold the fins, represented by green arrows. This load force is greater than the forces the fin will face during an actual launch and should provide more than enough of a safety factor for the test.

To model the force of drag on the fin, 800 N was applied along a 3.59 in^2 area running along the beveled leading edge of the fin, represented by purple arrows. This force was determined using the equation for dynamic drag force, $F = \frac{1}{2} C_d \rho v^2 A$, where F is the drag force, Cd is the drag coefficient, ρ is the air density, v is velocity, and A is the fin's area. OpenRocket provided the values for the C_d of .5 and the maximum velocity of 524 ft/s. Using the equation above gives a drag force of 21.17 N, which is increased to a test force of 800 N to achieve a large safety factor. The anchor points were placed inside the five attachment holes, as done in the lift FEA, and are represented by green arrows.

FEA Lift Analysis

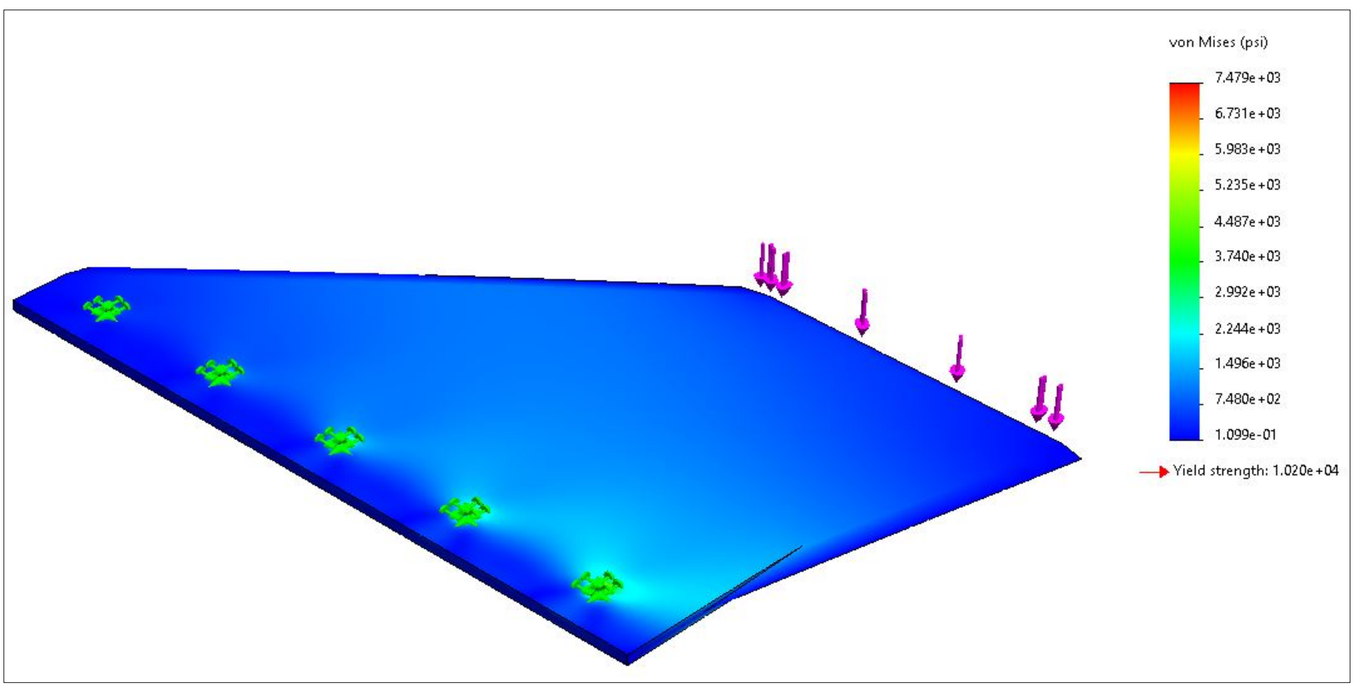


Figure 3.16: Fin Lift Force FEA for Von Mises Stress

The figure above shows the Von Mises stress test done on the fin representing the flight's lift force. This test is presented as a contour plot, where dark blue colors show low stress and brighter red colors show high stress. As shown in the figure, the fin did very well, showing fin stress having a minimum of 1.099e-01 psi and a maximum 2.244e+03 psi around the attachment points, far below the minimum tensile strength of 38,000 psi. This test shows the great advantage of the new fin mounting spars and validates the design's integrity.

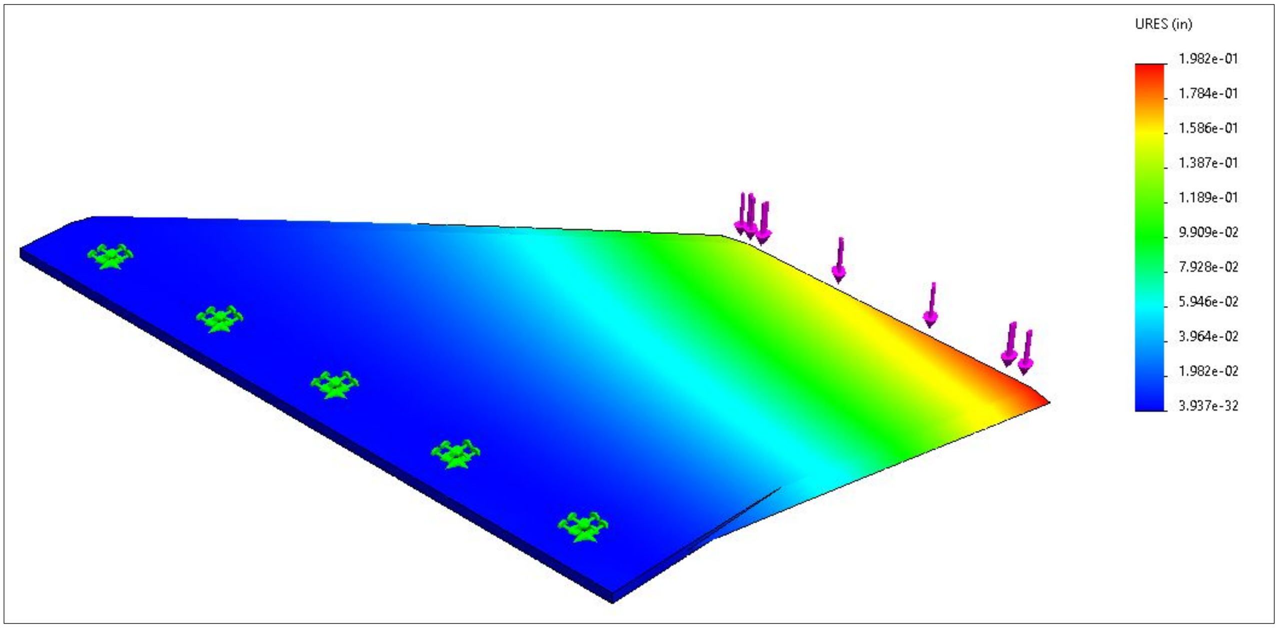


Figure 3.17: Fin Lift Force FEA for URES Displacement

The figure above shows the URES displacement test done on the fin to represent the flight's lift force. This test is also presented as a contour plot with dark blue indicating low displacement and red indicating high displacement. The displacement is applied as expected, with very low displacement at the attachment points and steadily increasing until it reaches its highest displacement at the bottom corner of the tip chord. That being said, the minimum fin displacement was 1.937e-32in, and the maximum fin displacement was 1.982e-1in. Thus, the fin shows no sign of being irreparably damaged by the flight.

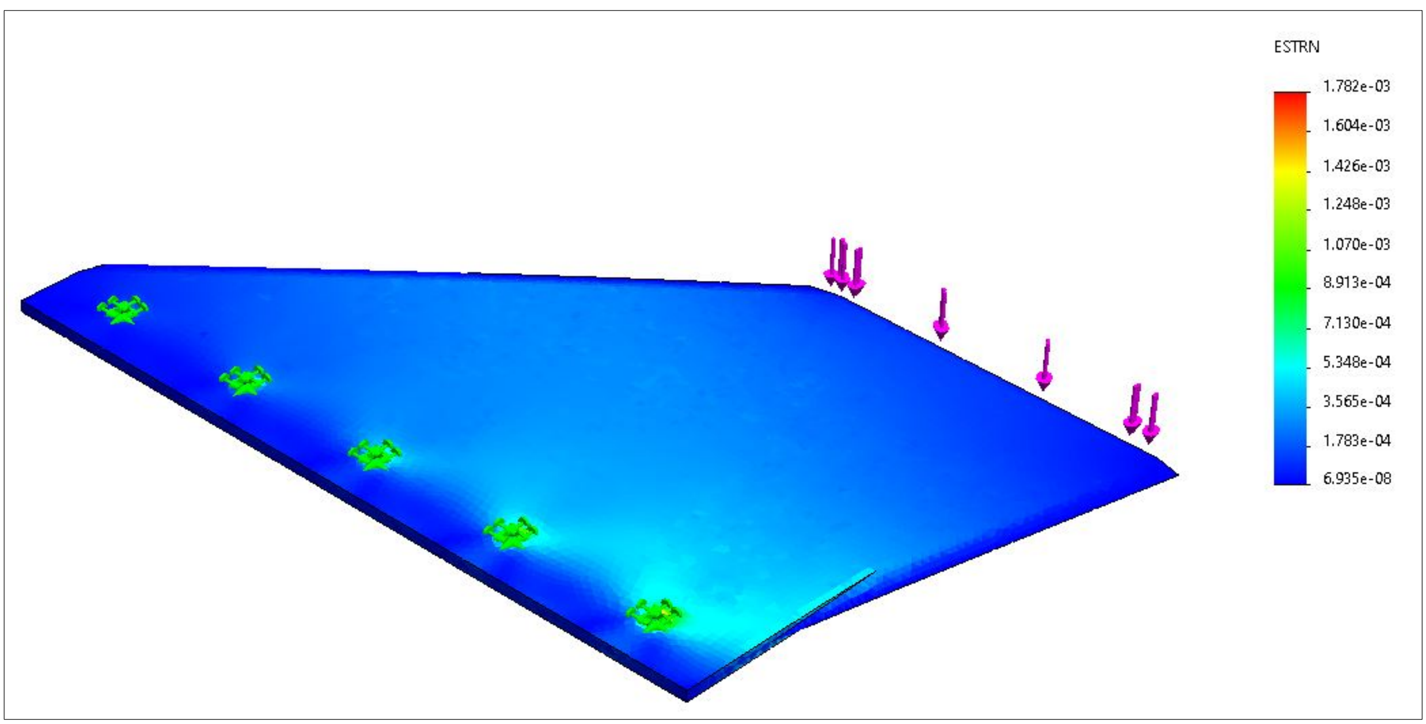


Figure 3.18 Fin Lift Force FEA for ESTRN Strain

Finally, the figure above shows the ESTRN strain test done on the fin representing the flight's lift force. This test is presented as a contour plot with dark blue representing the lowest strain and bright red representing the highest strain. Like the stress test, the fin design performed very well, with most of the fin only experiencing a strain of 6.935e-8. The bottom attachment point saw the highest amount of strain, but it was still low, only 5.348e-4. This test demonstrates that the fin will have no problem withstanding the lift forces generated during flight.

FEA Drag Analysis

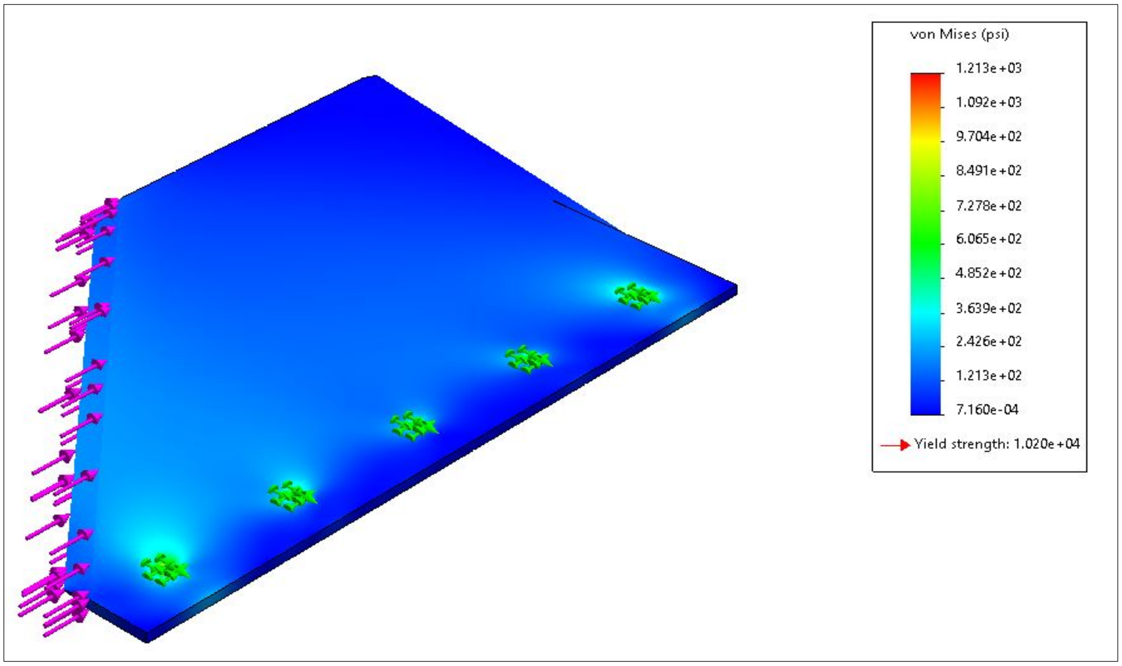


Figure 3.19 Fin Drag Force FEA for Von Mises Stress

The figure above shows the Von Mises stress test applied to the fin representing the flight's drag force. These test results are presented as a contour plot where dark blue is the lowest stress, and bright red is the highest stress. As shown above, the test was very successful, with most of the fin only experiencing 7.16e-4 psi and a slightly higher 3.639e+2 psi experienced around the attachment points, which is far below the minimum tensile strength of 38,000 psi. This test's results clearly show the integrity of this fin design and the benefit of using spars to mount the fins.

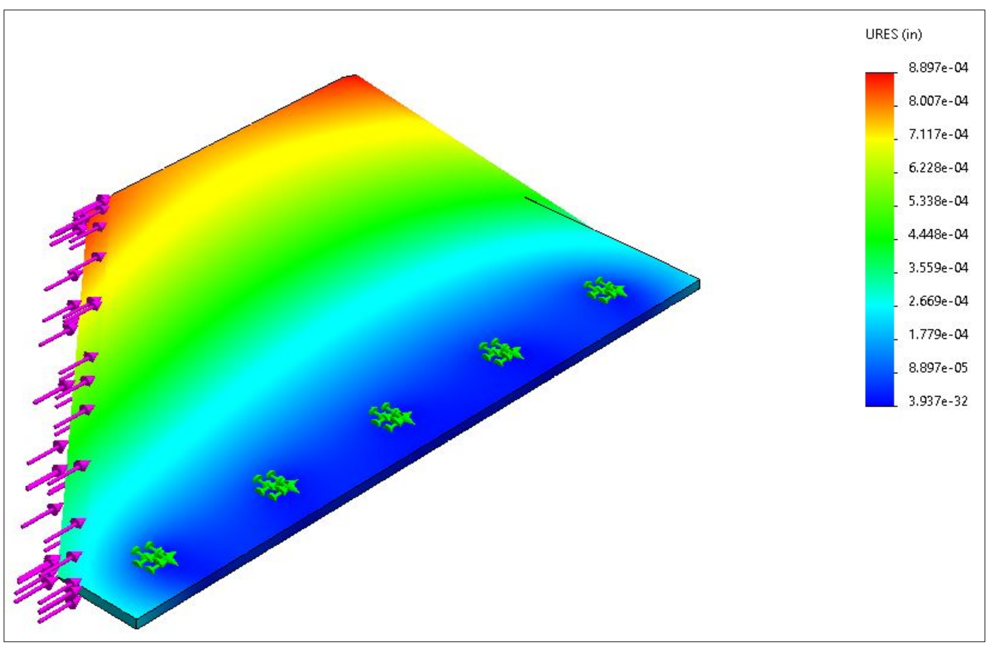


Figure 3.20: Fin Drag Force FEA for URES Displacement

The figure above shows the URES displacement test applied to the fin to represent the flight's drag force. These test results are displayed as a contour plot where dark blue represents the lowest displacement, and brighter colors represent increasing displacement until it reaches red, the maximum displacement. This test behaved as expected, with the lowest displacement, 3.907e-32 in, being around the attachment points and slowly increasing until it reaches its maximum displacement, 8.897e-4 in, at the top and bottom of the tip chord. This test's maximum displacement is still well below the critical displacement for this design and confirms its structural integrity.

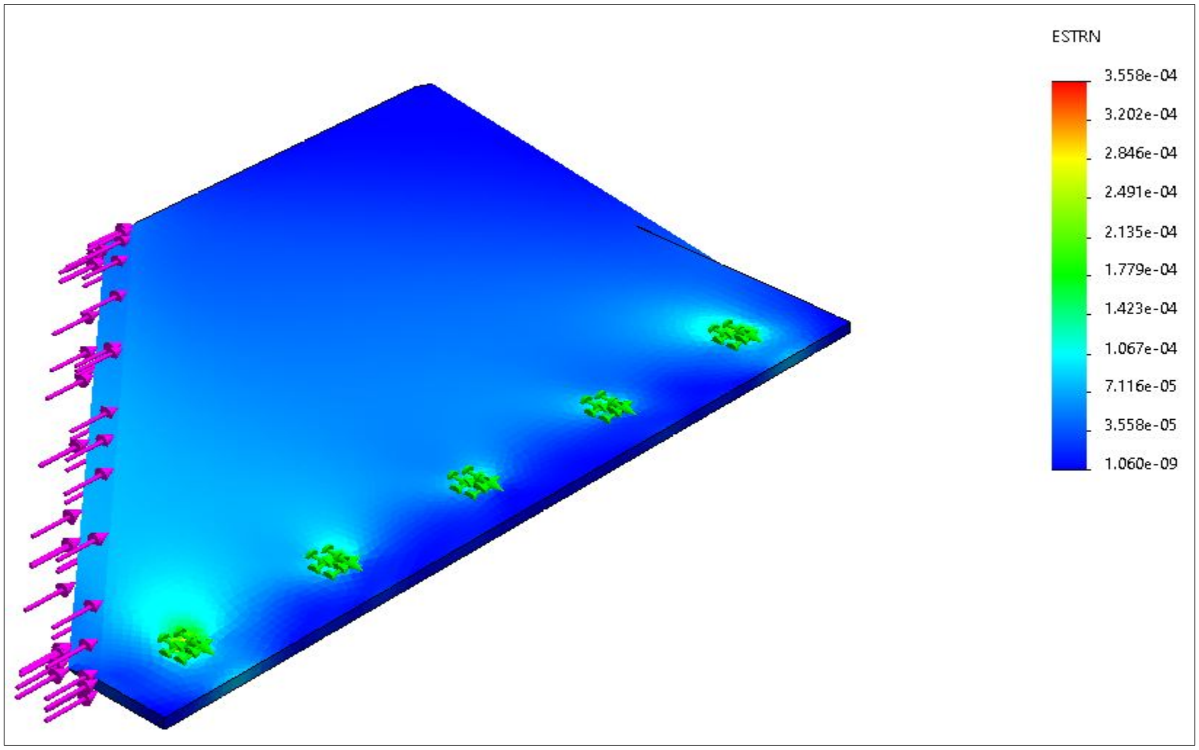


Figure 3.21 Fin Drag Force FEA for ESTRN Strain

The final FEA test representing the drag force of flight is the ESTRN strain test, shown in the figure above. These simulation results are displayed as a contour plot with dark blue representing the lowest strain and lighter colors expressing more significant strains. This test was also very successful, as most of the fin shows the lowest amount of strain, 1.06e-9, with only slightly higher strain levels seen around some of the attachment's points, reaching 1.067e-4. This final test confirms the fin's design integrity and proves it will withstand the forces of flight.

3.1.6.2.2 Bulkheads

3.1.6.2.2.1 MFSS FEA Static Studies Summary

The launch vehicle contains the Motor and Fin Support Structure (MFSS), which has two significant structural bulkheads, the thrust plate sub-assembly, which contains the thrust plate and the thrust plate flange, and the centering plate. Certain Finite Element Analysis Static Studies were conducted utilizing SolidWorks Simulation Static Study on an assembly containing the whole MFSS sub-assembly, the lower airframe, the three fins, the motor nozzle, and the motor casing aft closure. The setup was assumed to be fixed on the top of the lower airframe. Contact sets were created as bonded, and the parts were assigned their respective material, with the steel or rigid option (no deformation) used for the nozzle and the aft closure of the motor casing, since the focus is mainly on the MFSS and the two bulkheads. The following forces were applied, each constituting a separate static study, with the only difference between each study being the force applied:

- 50 N normal to the lifting area on each fin tip (fin tip lift load simulation). This force is the same force that was applied to the fin over the same area in the fin tip lift load simulation presented in the previous section.
- 800 N on the fin leading-edge cross-sectional area on each fin (fin leading edge drag simulation). This force is the same force that was applied to the fin over the same area in the fin leading edge drag simulation presented in the previous section.
- 1800 N (direction is vertical upwards) on the motor nozzle, which represents the maximum thrust of the motor (motor thrust simulation).

- 250 N (direction is vertical downwards) on the aft closure of the motor casing, to simulate stress on the retainer from the motor being accelerated vertically downwards (retainment simulation).
- 1600 N on the trailing edge cross-sectional area of a single fin, intended to simulate a load from landing if the launch vehicle impacts the ground on one side (fin trailing edge landing load simulation).
- 1600 N normal to the fin tip in the inward radial direction of the launch vehicle, which is intended to simulate a load from landing if the launch vehicle impacts the ground on one side (fin tip landing load simulation). This is one of the most unlikely and extreme impact scenarios since the launch vehicle is not expected to land in this horizontal orientation with the tip of the fin hitting the ground first, due to the main parachute force.
- 1600 N (direction is vertical upwards) normal to the motor retainer, which is intended to simulate a load from landing in case the launch vehicle impacts the ground vertically with the motor retainer (motor retainer landing load simulation). This is a more likely impact scenario.

In each of these cases, the lowest tensile strength of G10 fiberglass (38,000 psi) and the yield stress of Al6061-T6 (35,000 psi) is not surpassed. More specifically:

- The fin tip lift load simulation has a minimum Factor of Safety (FOS) of 2.89 (maximum stress is 12,110 psi).
- The fin leading edge drag simulation has a minimum FOS of 4.3 (maximum stress is 4,696 psi).
- The motor thrust simulation has a minimum FOS of 5.2 (maximum stress is 4,525 psi).
- The retainment simulation has a minimum FOS of 41.9 (maximum stress is 565.3 psi).
- The fin trailing edge landing load simulation has a minimum FOS of 1.2 (maximum stress is 10,820 psi).
- The fin tip landing load simulation has a minimum FOS of 0.29 (maximum stress is 34,860 psi).
- The motor retainer landing load simulation has a minimum FOS of 6.5 (maximum stress is 3,557 psi).

The relatively high minimum factors of safety in the simulations above prove the design's integrity under extremely high stresses on key structural parts (mainly the MFSS and the lower airframe). Detailed contour plots with stress distribution follow in each case, to identify stress load paths and stress concentration areas where mitigation techniques have been applied. Certain cases also include displacement and factor of safety contour plots. These cases focus on external components, such as the lower airframe, as SolidWorks does not provide a section clipping tool for the displacement and FOS contour plots, as is the case with the stress contour plot.

3.1.6.2.2.2 Thrust Plate of MFSS

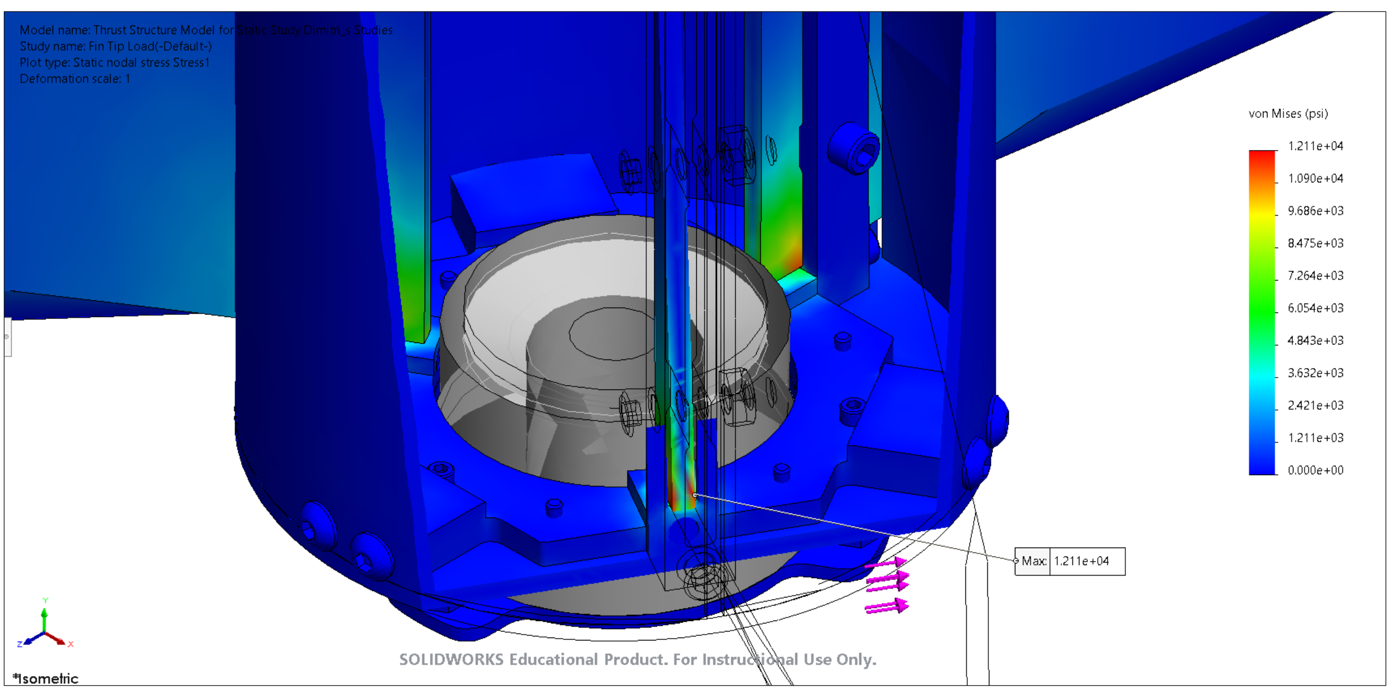


Figure 3.22: Von Mises stress contour plot on thrust plate from the fin tip lift load simulation

In the fin tip lift load simulation, the thrust plate takes a load in the green range (between 4843 psi and 8475 psi) near the connection point with the spar that supports each fin. This load is gradually dissipated as it moves away from the contact point and the plate's thicker part towards the thinner part. The largest parts of the thrust plate experience loads below 1211 psi. The minimum FOS here is 2.89, and the thrust plate itself does not sustain any load that comes close to the yield strength of Al 6061-T6, which is 35,000 psi, since the maximum stress sustained is on the fin itself is 12,110 psi.

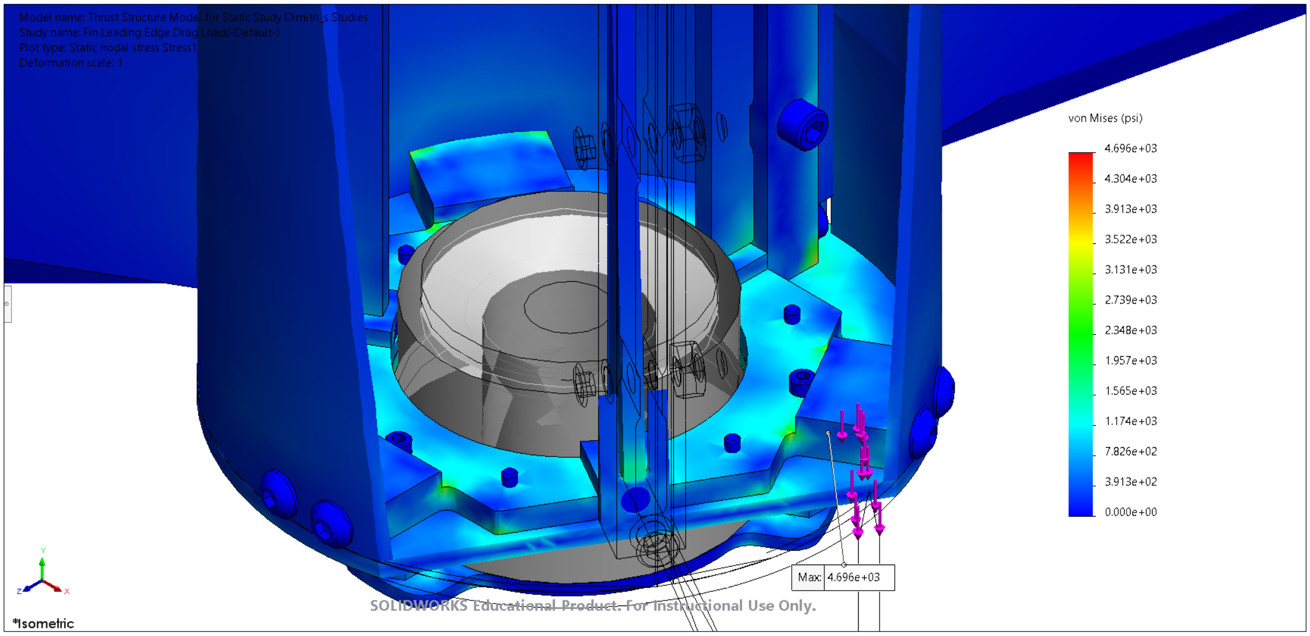


Figure 3.23: Von Mises stress contour plot on thrust plate from fin leading edge drag simulation.

In the fin leading edge drag simulation, the thrust plate sustains a well-distributed load transferred from the spars to the attachment points. The stress is in the cyan region (1174-1565 psi) near the contact point and most regions of the thin hexagonal area, with stress increasing where the load changes directions, in the corners of the thrust plate. The stress transitions into the green (1565-3131 psi) and even the yellow and orange regions (3552-3913 psi) at corner points. Larger stress concentrations at those points are avoided with the utilization of fillets and rounds, which make the distribution of stress smoother. The stress load is eventually transferred to the lower airframe through the spokes and the fasteners. There is an increase from the dark blue region (0-391.3 psi) to the light blue (782.6-1174 psi) and then to the cyan and to the green regions as stress transfers from the thrust plate spoke to the wall of the lower airframe. The minimum FOS, in this case, is 4.3, and the thrust plate itself does not sustain any load that comes close to the yield strength of Al 6061-T6, which is 35,000 psi, since the maximum stress sustained is on the standoff (which is hidden here) at 4696 psi.

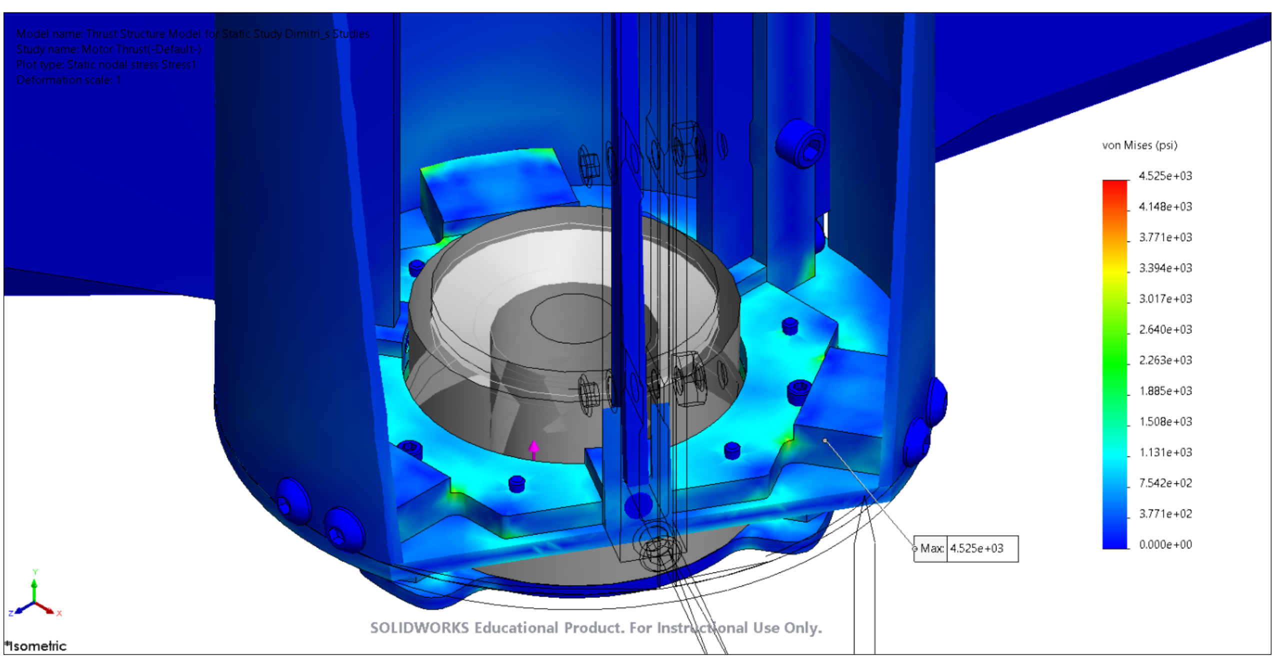


Figure 3.24: Von Mises stress contour plot on thrust plate from motor thrust simulation.

In the motor thrust simulation, stress is transferred from the standoffs to the thrust plate flange and then from the thrust plate flange to the thrust plate. The stress is again well distributed along the hexagonal region, with stress being primarily in the cyan region (1131-1508 psi) near the top surface of the hexagonal region, with stress increasing near the corners at the point of contact with the thrust plate flange, approaching the green region (1508-3017psi), where the rounds help with distributing stress and preventing large stress concentrations near the corner points of the hexagon. Stress increases to the green and approaches the yellow-orange region (3394-3771 psi) near the spoke corner, where the fillet dissipates it. Stress is eventually transferred to the lower airframe through the spokes, transitioning from the dark blue region (0-377.1 psi) to cyan and then green, on the point of contact. The minimum FOS, in this case, is 5.2, and the thrust plate itself does not sustain any load that comes close to the yield strength of Al 6061-T6, which is 35,000 psi, since the maximum stress sustained, 4525 psi, is on the standoff, which is hidden here.

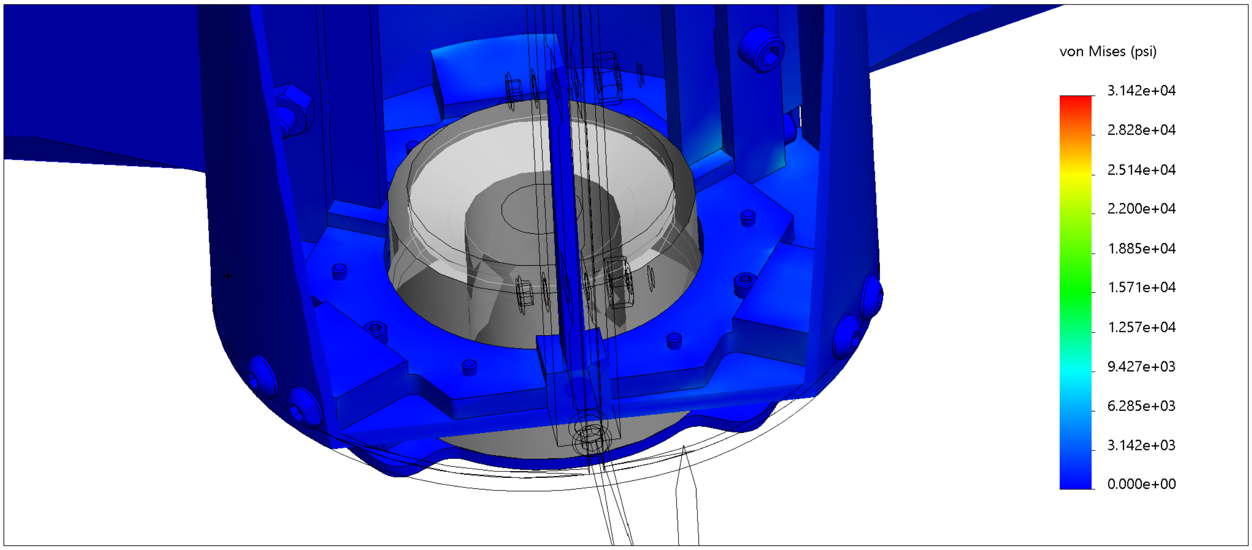


Figure 3.25: Von Mises stress contour plot on thrust plate the fin tip landing load simulation.

In the fin tip landing load simulation, the thrust plate sustains a nearly uniformly distributed stress load in the dark blue region (0-3142 psi), with higher stress concentration at the contact point with the fin spar, where stress approaches the cyan region (6285-9427 psi). There is larger stress near the thrust plate's contact point spoke with the lower airframe, where stress approaches the cyan region again, transitioning from the dark blue region, as it is transferred from the thrust plate to the lower airframe. The minimum FOS here is 0.29, and the thrust plate itself does not sustain any load that comes close to the yield strength of Al 6061-T6, which is 35,000 psi, since the maximum stress is 31,420 psi, sustained near the top of the booster tube.

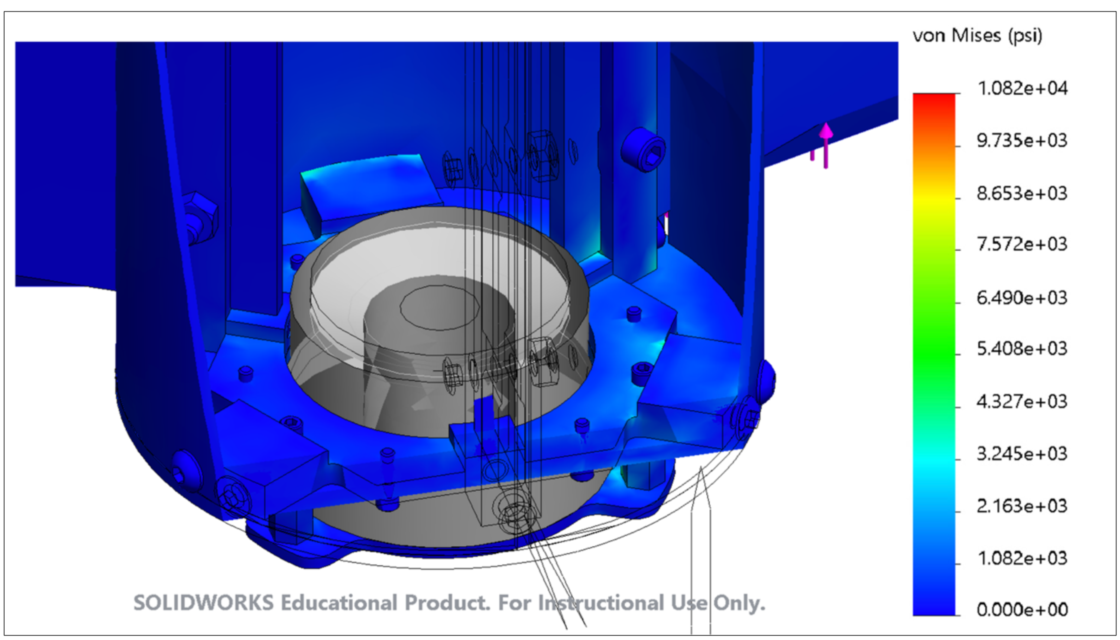


Figure 3.26: Von Mises stress contour plot on thrust plate landing load on fin trailing-edge simulation.

In the fin trailing-edge landing load simulation, the thrust plate again sustains uniformly distributed stress load in the dark blue region (0-1082 psi), with higher stress concentration at the contact point with the fin spar, where stress approaches the cyan region (2163-3245 psi). Stress near the contact point of the thrust plate spoke with the lower airframe is also in the cyan region as it is transferred from the thrust plate to the lower airframe.

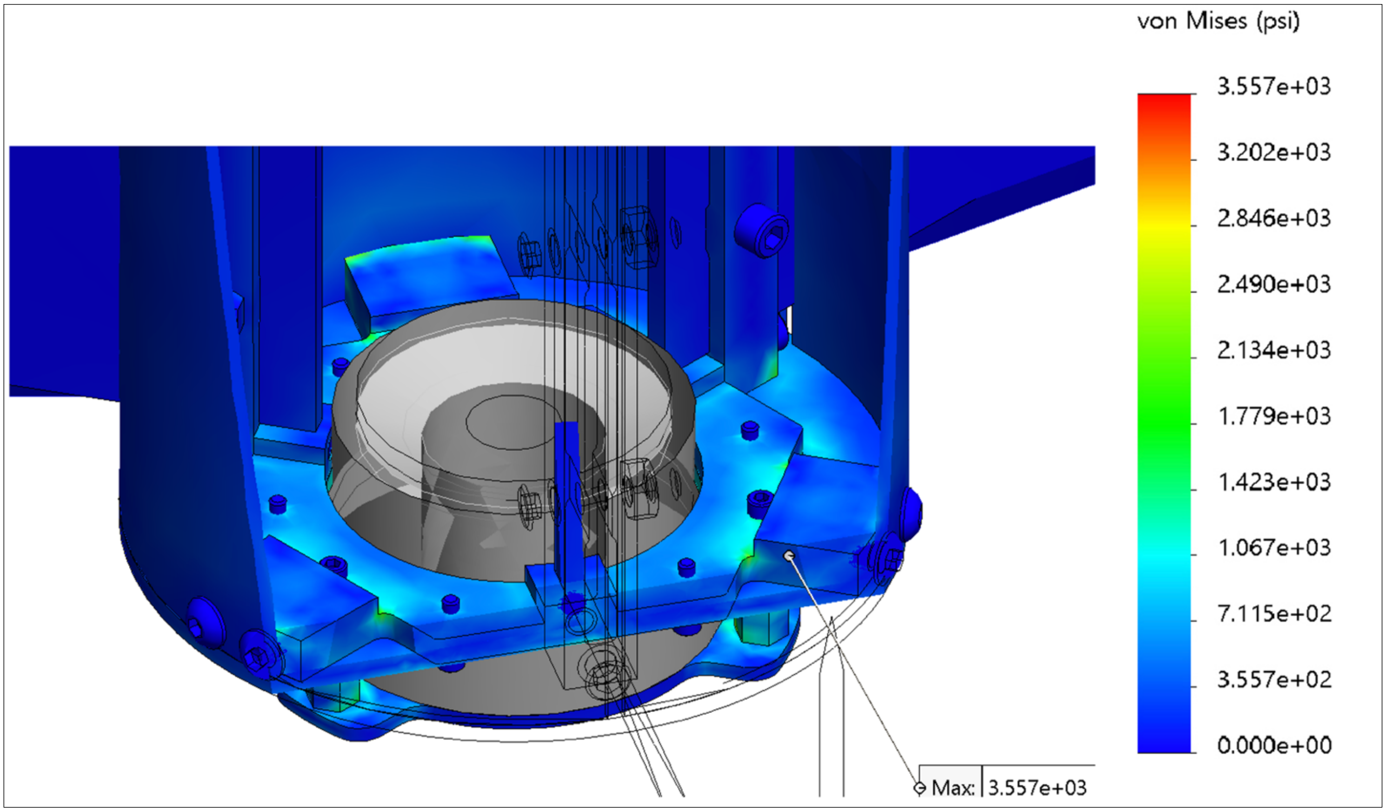


Figure 3.27: Von Mises stress contour plot on thrust plate from landing load on motor retainer simulation.

In the landing load on the motor retainer simulation, stress is transferred similarly from the standoffs to the thrust plate flange and then from the thrust plate flange to the thrust plate (similar to the motor thrust simulation). The stress is still well distributed along the plate, with stress in the cyan region (711.5-1067 psi) near the top surface of the hexagonal region, and increasing near the corners at the spokes. The stress here increases to the green region (1423-1779 psi), and the filleted edges help distribute stress and prevent large concentrations near the corner points. Stress approaches the yellow-orange region (2490-3202 psi) near the spoke corner, and is gradually dissipated. Stress is eventually transferred to the lower airframe through the spokes after an increase at the point of contact of the spoke with the wall, transitioning back to the dark blue region (0 – 355.7 psi).

3.1.6.2.2.3 Thrust Plate Flange of MFSS

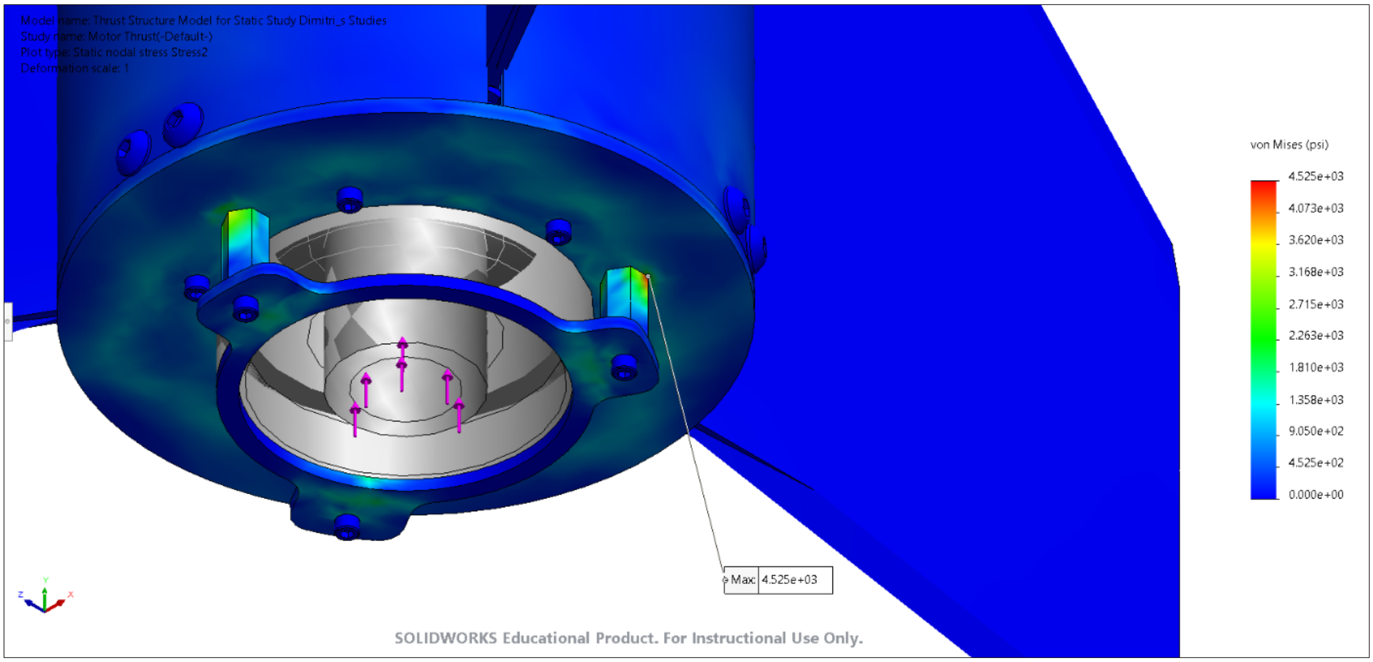


Figure 3.28: Von Mises stress contour plot on thrust plate flange from motor thrust simulation.

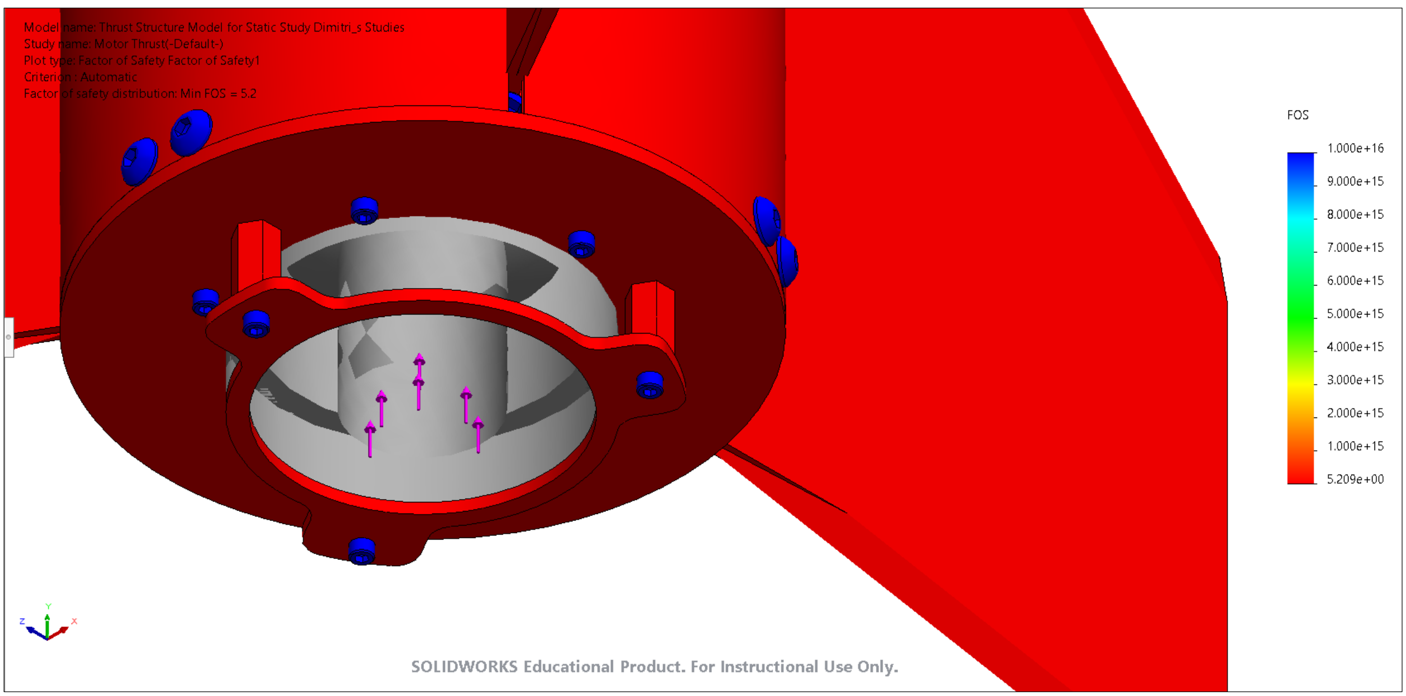


Figure 3.29: Factor of safety contour plot on thrust plate flange from motor thrust simulation.

In the motor thrust simulation, the thrust plate flange sustains a relatively larger stress load at the contact points with the standoffs and the motor case aft closure (shown in semi-transparent grey, approximated as rigid in this study). The region around the contact points is in the green region (1810-3620 psi) and the load travels towards the contact points, where the stress is eventually transferred. The minimum FOS here is 5.2 and the thrust plate flange itself does not sustain any load that comes close to the yield strength of Al 6061-T6, which is 35,000 psi, since the maximum stress sustained is 4525 psi on the standoff.

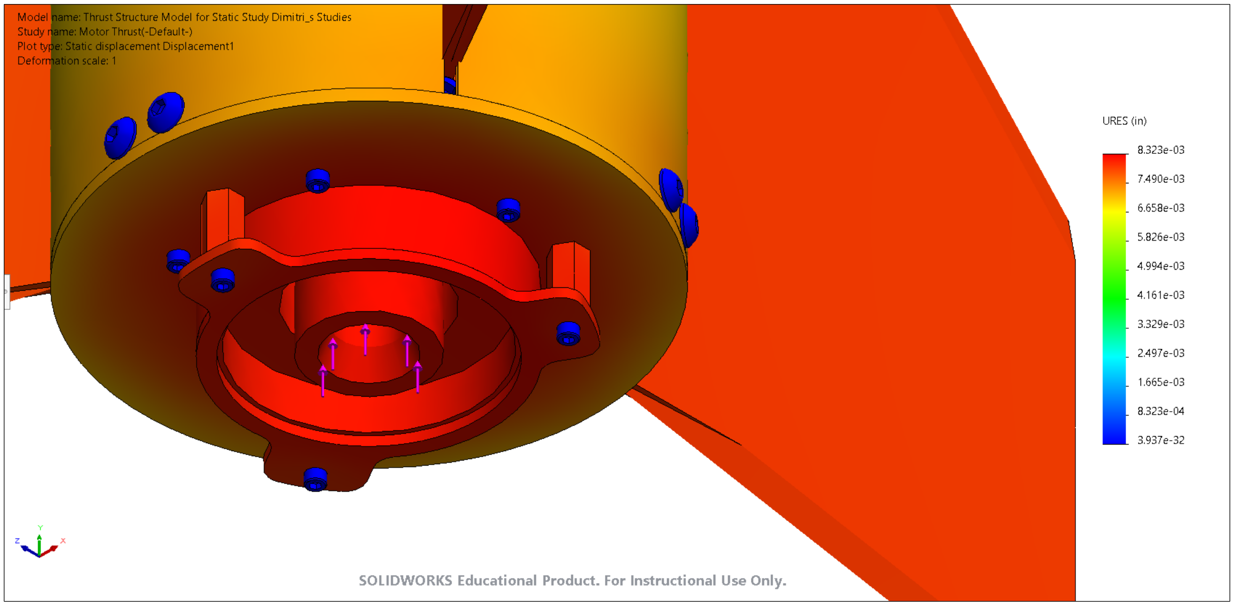


Figure 3.30: Displacement contour plot on thrust plate flange from motor thrust simulation.

The displacement of the thrust plate flange in the motor thrust simulation is larger close to the center of the flange, where the force is applied, and smaller towards the point of contact with the lower airframe. This is expected since the stress is largely applied on the inner hole edge and the points of contact with the standoffs.

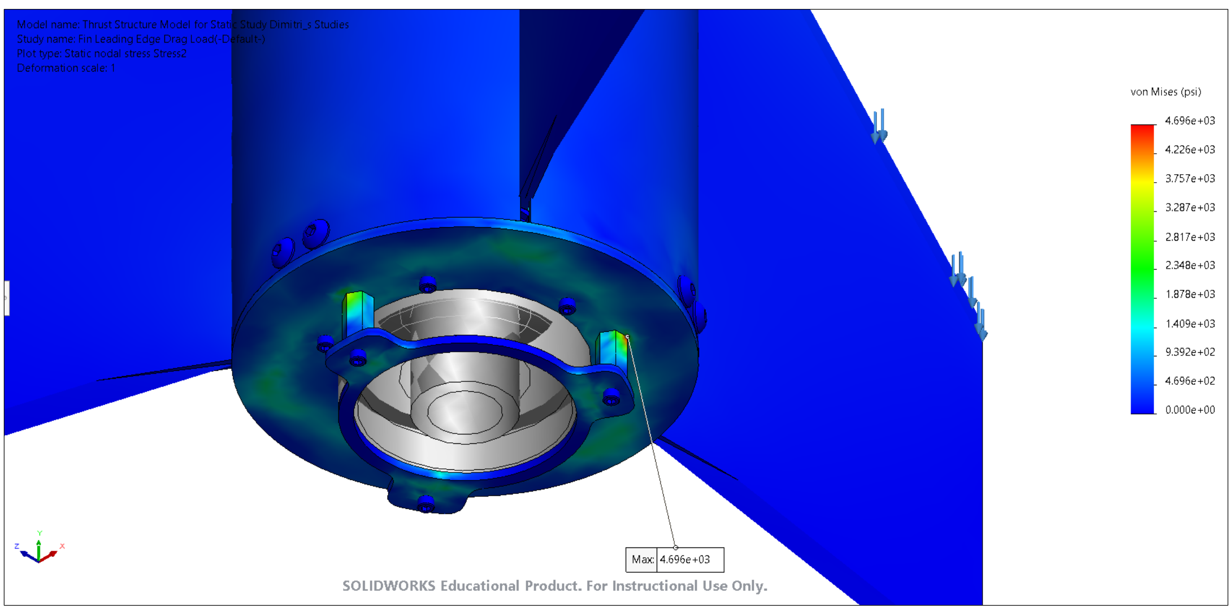


Figure 3.31: Von Mises stress contour plot on thrust plate flange from fin leading edge drag simulation

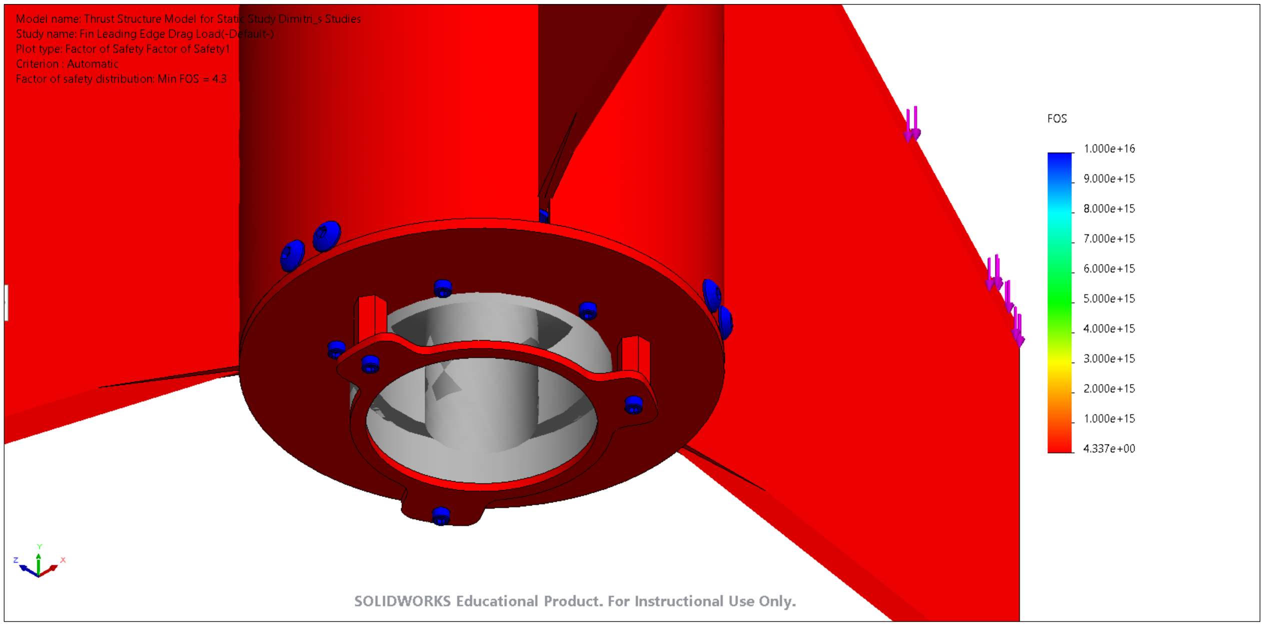


Figure 3.32: Factor of safety contour plot on thrust plate flange from fin leading edge drag simulation.

In the fin leading edge drag simulation, the thrust plate flange sustains a relatively larger stress load at the contact points with the standoffs and the motor case aft closure (shown in semi-transparent grey, approximated as rigid in this study). The region around the contact points is shown in the green region (1878-3757 psi) and the load travels, demonstrated by the cyan-green paths, toward the contact points with the standoffs, coming from the lower airframe bottom end. The stress is eventually transferred to the standoffs. The minimum FOS here is 4.3 and the thrust plate flange itself does not sustain any load that comes close to the yield strength of Al 6061-T6, which is 35,000 psi, since the maximum stress sustained is 4696 psi on the standoff.

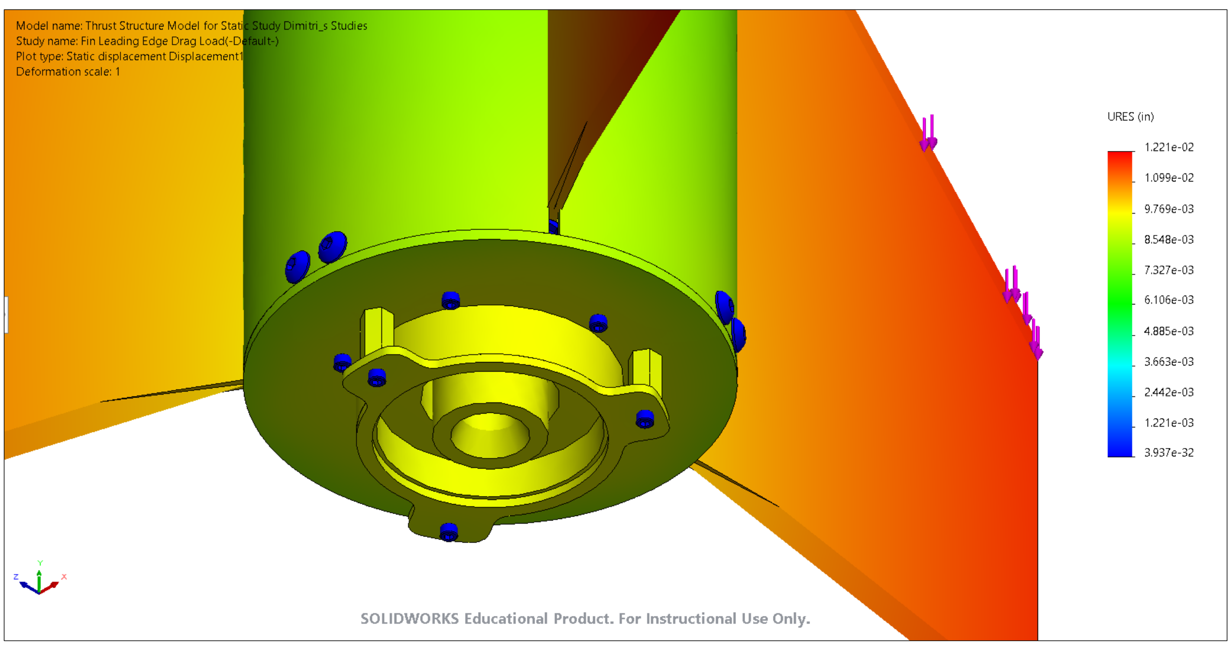


Figure 3.33: Displacement contour plot on thrust plate flange from fin leading edge drag simulation.

The displacement of the thrust plate flange in the fin leading edge drag simulation is larger close to the center of the flange, where the force is applied, and smaller towards the point of contact with the lower airframe. This is expected since the stress is largely applied around the inner hole area, which is the larger area of the point of contact with the thrust plate, which carries the load from the fin spars.

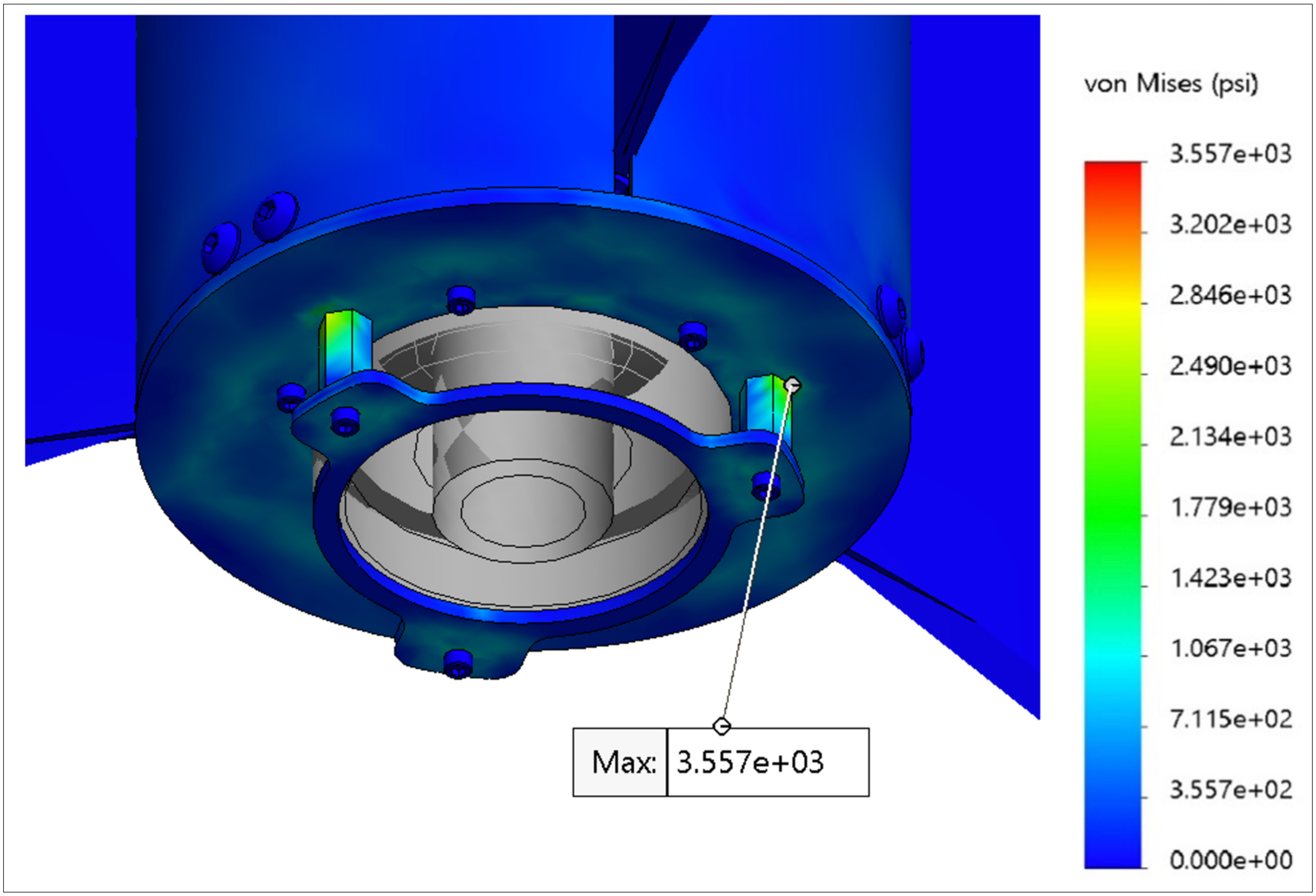


Figure 3.34: Von Mises stress contour plot on thrust plate flange from landing load on motor retainer simulation.

In the landing load on the motor retainer simulation, The points where motor retainer connects to the standoffs indicates stress levels in the green region (1423-2134 psi) which is then transferred into the standoffs themselves. Nearing the top of the standoffs where they connect with the flange, the stress begins to peak into the yellow region (2846-3202 psi) indicating the stress peaks at the top of these standoffs. The stress then is uniformly distributed on the thrust plate flange and mixes between the blue through green regions (0-2490 psi) and is then transferred to the thrust plate and lower tube to drop into only the blue region.

3.1.6.2.2.4 Centering Plate of MFSS

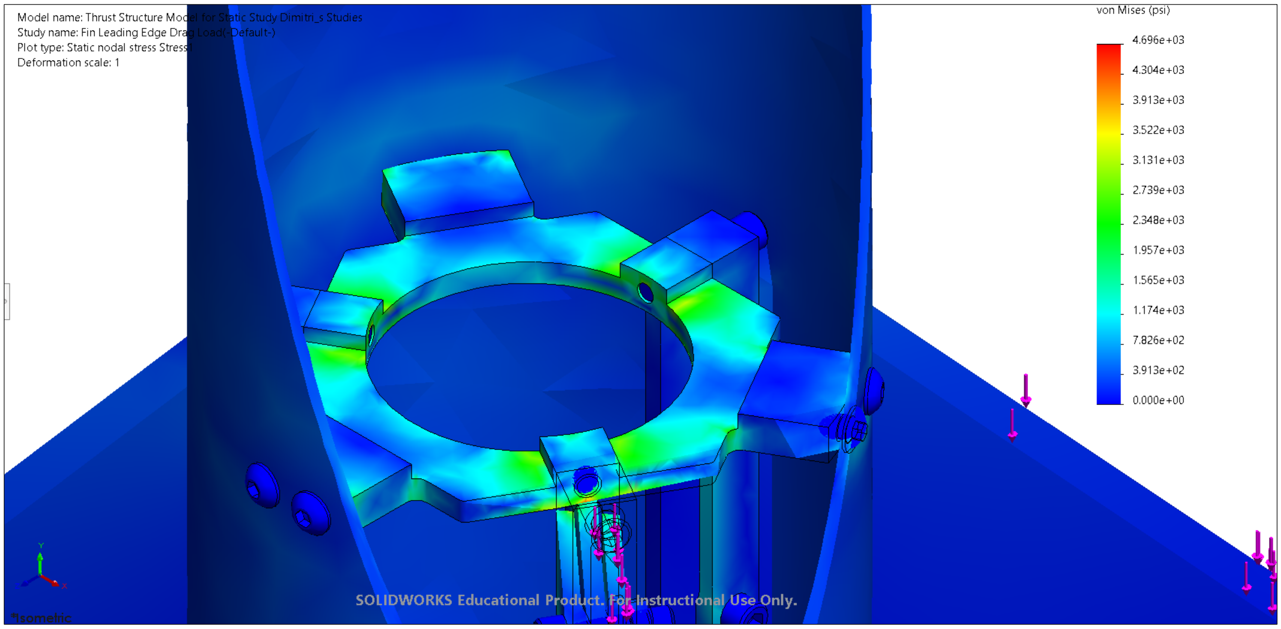


Figure 3.35: Von Mises stress contour plot on centering plate from fin leading edge drag simulation.

In the fin leading edge drag simulation, the centering plate sustains a well-distributed load transferred from the spars to the plate's attachment points. The stress is in the cyan region (1174-1565 psi) near the contact point and in certain regions of the thin hexagonal area close to the spokes. Stress increases where the load changes directions, primarily in the corners, especially from the thicker attachment point of the spar and outwards toward the spokes, on the thinner region. The stress transitions into the green (1565-3131 psi) and even the yellow and orange regions (3552-3913 psi) near the spar attachment points. The corners of the spars also have green regions shown. Larger stress concentrations at those points are avoided with the utilization of fillets and rounds, making the distribution of stress more uniform. The stress load is eventually transferred to the lower airframe through the spokes and the fasteners. There is an increase from the dark blue region (0-391.3 psi) to the light blue (782.6-1174 psi) and then to the cyan and to the green regions as stress transfers from the thrust plate spoke to the wall of the lower airframe. The minimum FOS here is 4.3, and the centering plate itself does not sustain any load that comes close to the yield strength of Al 6061-T6, which is 35,000 psi, since the maximum stress sustained is on the standoff beneath the thrust plate flange (which is hidden here) at 4696 psi.

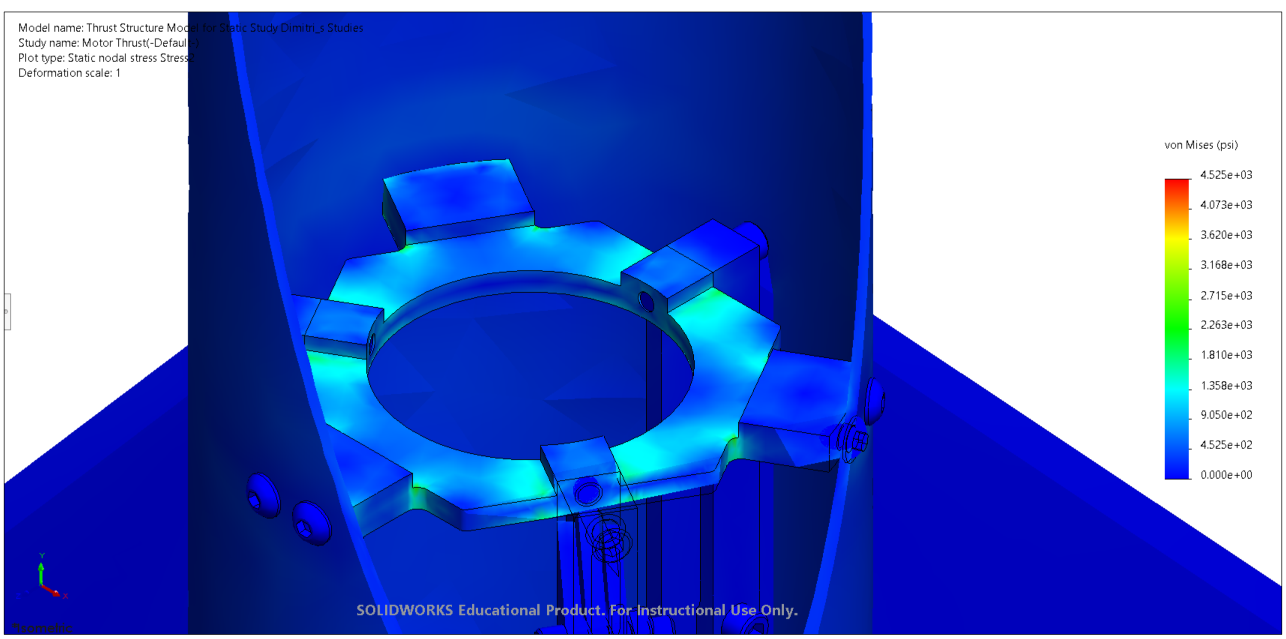


Figure 3.36: Von Mises stress contour plot on centering plate from motor thrust simulation.

In the motor thrust simulation, stress is transferred from the thrust plate to the spars and then from the spars to the centering plate. The stress is distributed along the hexagonal region, with stress being primarily in the cyan region (1131-1508 psi) near the top surface of the hexagonal region, with stress increasing close to the spar attachment points, approaching the green region (1508-3017 psi). Stress also increases to the green region near the spoke corners, where it is dissipated and distributed smoothly by the fillet. Stress is eventually transferred to the airframe through the spokes, transitioning from the dark blue region (0-377.1 psi) to cyan, on the point of contact. The minimum FOS here is 5.2 and the centering plate itself does not sustain any load that comes close to the yield strength of Al 6061-T6, which is 35,000 psi, since the maximum stress sustained is 4525 psi on the standoff, which is hidden here.

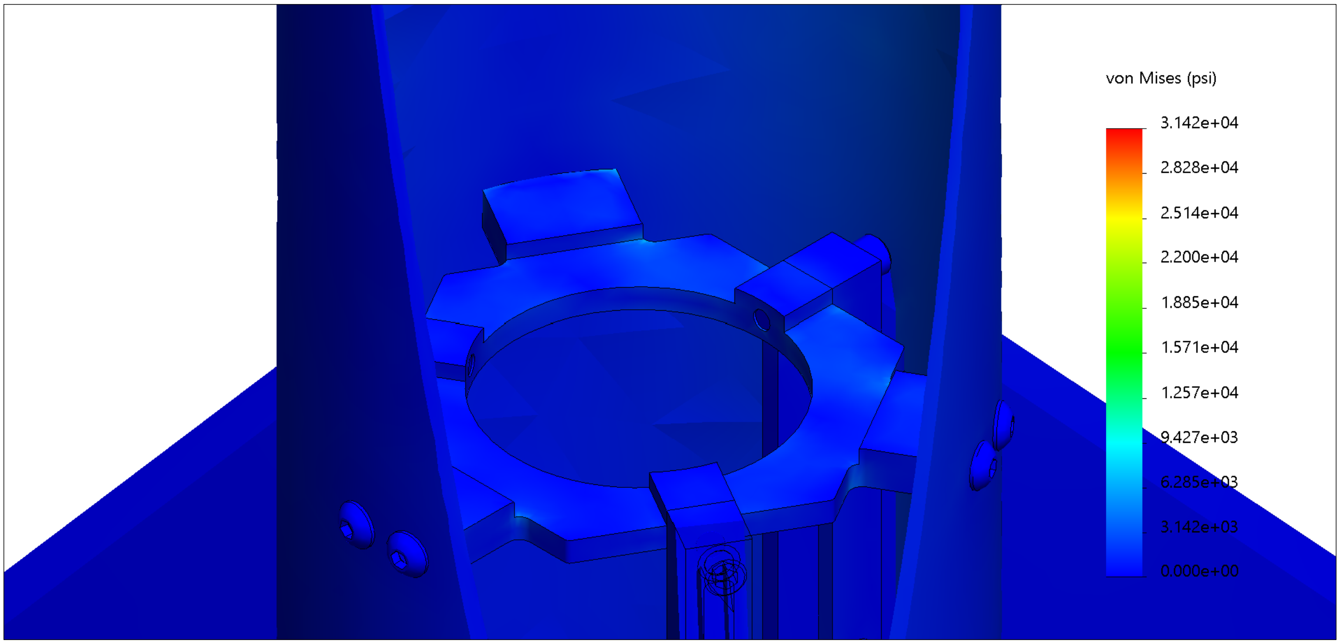


Figure 3.37: Von Mises stress contour plot on centering plate from landing load on fin tip simulation.

In the landing load on the fin tip simulation, the centering plate stress remains uniform in the blue region (0-6285 psi). Near the rounded edges where the spokes mold with the hexagonal geometry, the stress reaches the cyan region (6285-9427 psi) and then is transferred entirely to the spoke and then the lower body tube.

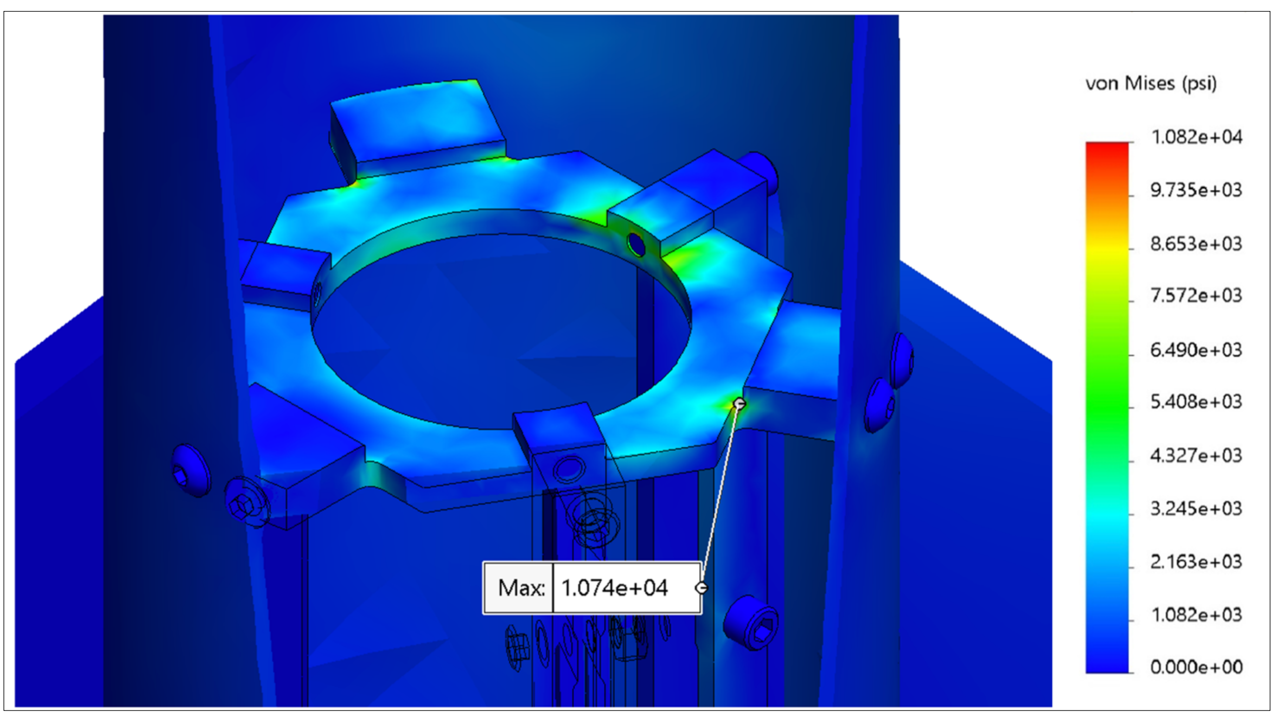


Figure 3.38: Von Mises stress contour plot on centering plate from landing load on fin trailing-edge simulation.

The landing load on the fin trailing-edge simulation indicates the centering plate sustains a larger load due to the fin spar of the corresponding fin which the force was applied to (top right corner). The stress at these edges reaches the yellow region of the plot (~8653 psi). The stress is then distributed across the plate where cyan regions (2163 – 3245 psi) are found. These regions are especially pronounced near edges and rounded corners of the plate's geometry. The stress is then transferred into the lower tube where it indicates falling in the blue region (0 – 2163 psi).

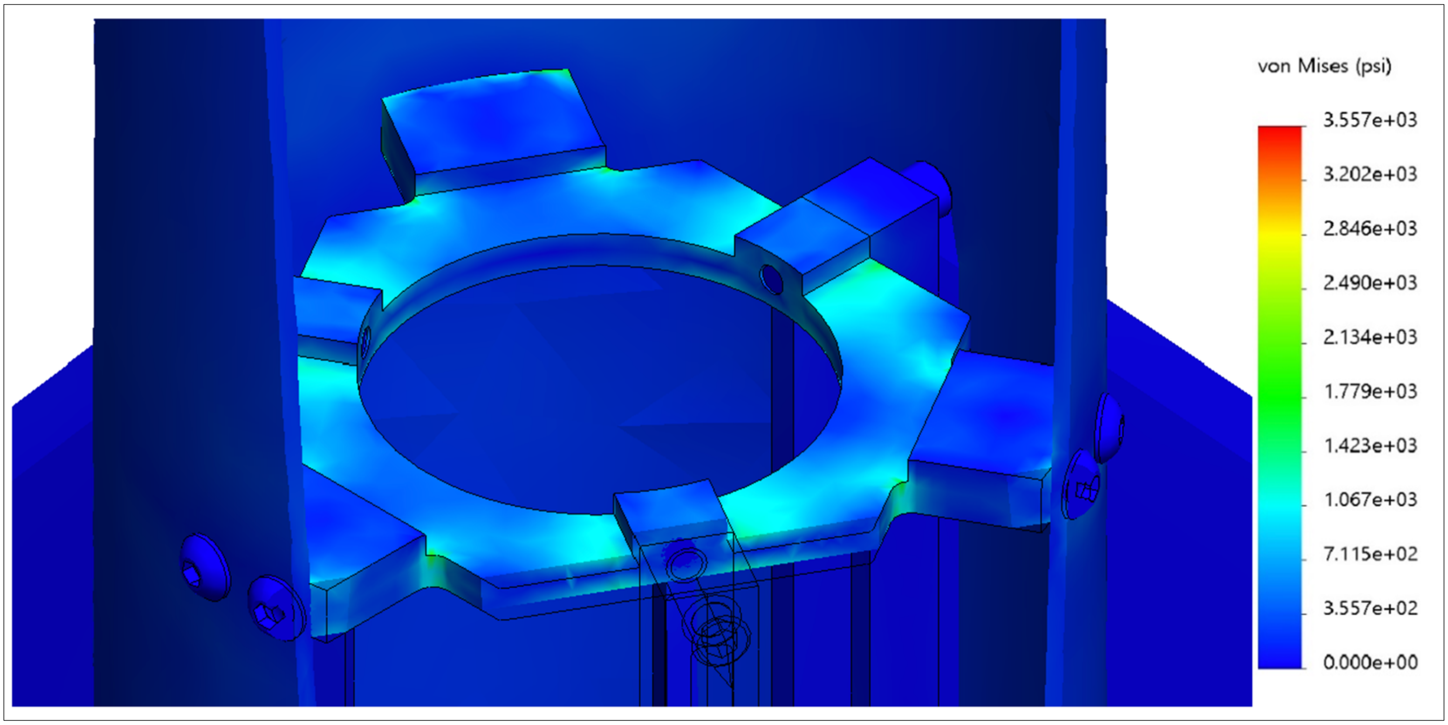


Figure 3.39: Von Mises stress contour plot on centering plate from landing load on motor retainer simulation.

Similar to the landing load on the fin trailing-edge simulation, in the landing load on motor retainer simulation, the stress is mainly concentrated near the edges of the centering plate, reaching the cyan region (711.5-1067 psi). Primarily, the edges where the fin spars connect to the plate, and the rounded where the spokes of the plate connect to the hexagonal geometry. From the edges, the stress is transferred into the spokes and the lower tube where it falls into the blue region (0-355.7 psi).

3.1.6.2.2.5 Avionics Bay Bulkheads & Payload Coupler Bulkheads

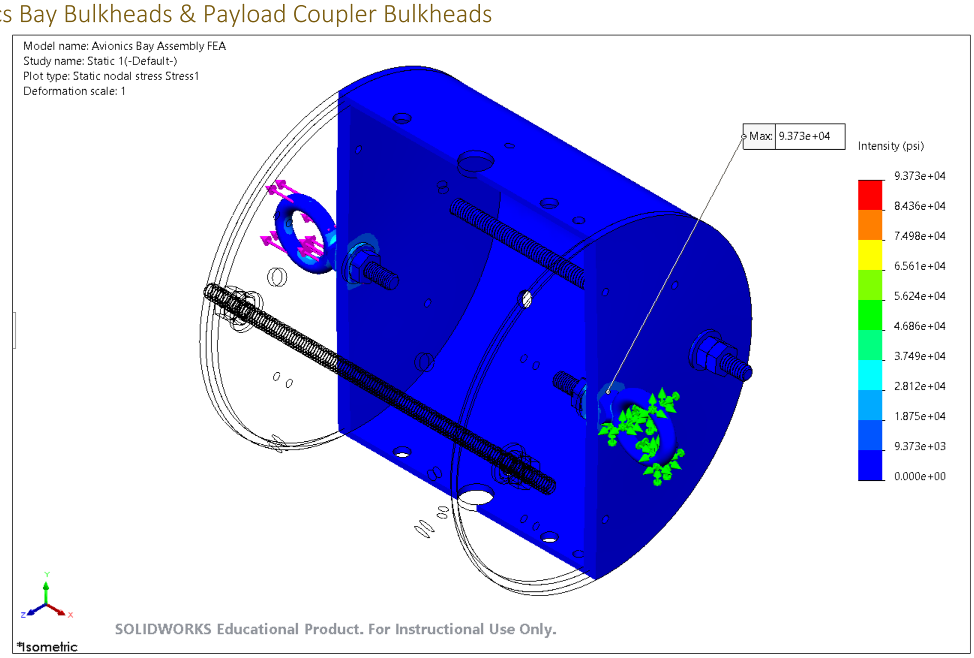


Figure 3.40: Avionics bay bulkhead stress intensity contour plot from main parachute deployment simulation.

The main parachute deployment simulation static study focused on the avionics bay assembly bulkheads, assuming one eye bolt is fixed while the other eye bolt is under a a force of 500 lbf, which is the maximum force rating for the eye bolts. This force is part of the tension force from the main parachute deployment. The intensity stress does not overcome the minimum tensile strength of 38,000 psi for G10 fiberglass. It reaches a maximum near the cyan range on the bulkheads themselves, at the contact points with the eye bolt and the other fasteners (28120-37490 psi). This value is very close to the tensile strength value of G10 fiberglass, though in this case, the eye bolts are expected to fail first. The rest of the assembly remains below the value of 18,750 psi, in the darker blue ranges. The team is confident that the bulkhead and the eye bolts will not fail, as the design has been proven to be functional and safe from the two successful full-scale launches that were conducted last year, where no damage was observed in the bulkheads. Note that the avionics bay shares the load from the deployment of the main parachute with the upper coupler bulkheads in the payload section of the upper airframe. These bulkheads are nearly identical, and their assembly is also very similar to the avionics bay. The simulation of the payload coupler bulkhead is very similar, with higher stress appearing mostly on the point of contact and the larger part of the bulkhead remaining under 24,280 psi, in the dark blue region.

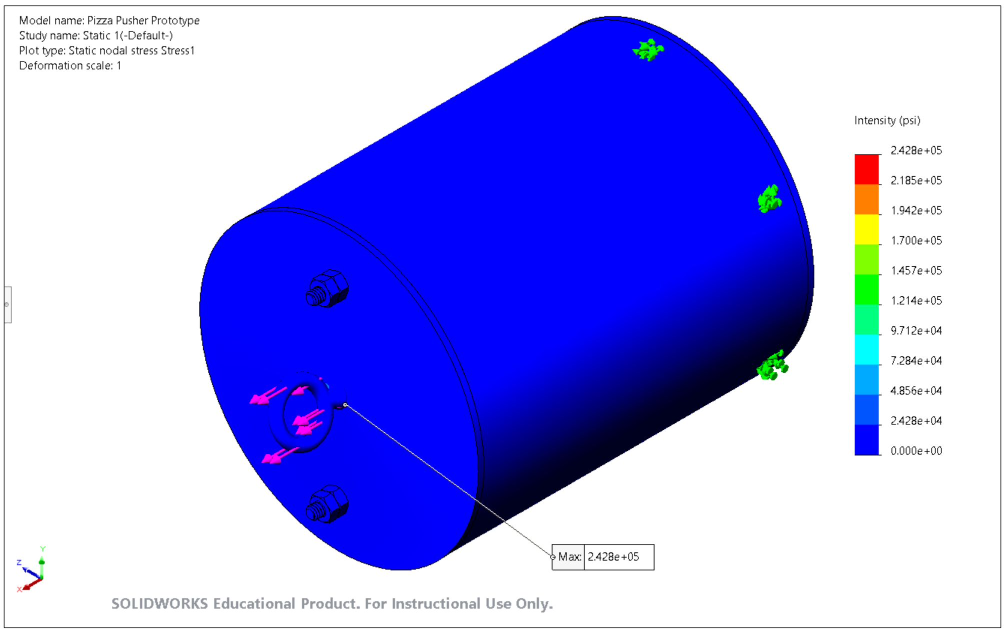


Figure 3.41: Payload coupler bulkhead stress intensity contour plot from main parachute deployment simulation. Here the fixation points (green arrows) are six radial holes which connect to the upper airframe.

3.1.6.2.3 Structural Elements

3.1.6.2.3.1 Fin Spars of MFSS

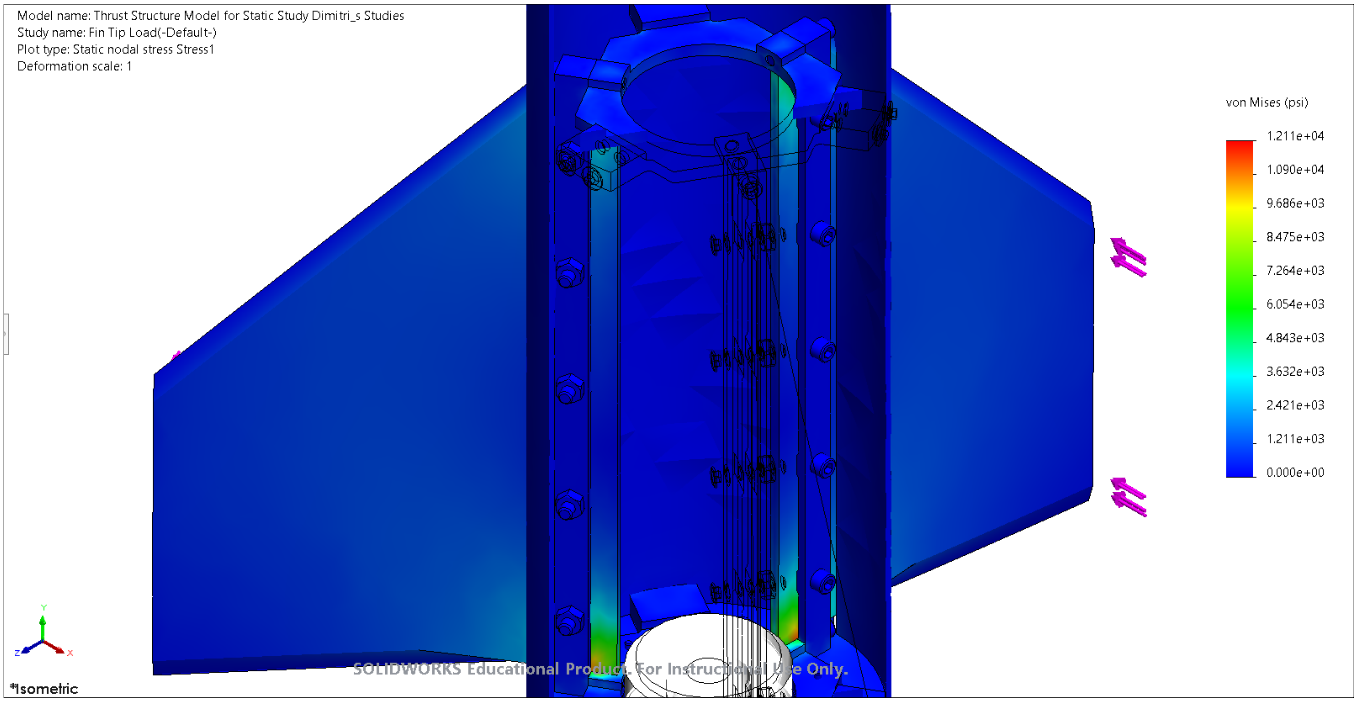


Figure 3.42: Von Mises contour plot on fin spar from fin tip lift load

In the fin tip lift load simulation, it is observed that the fin spar experiences a nearly uniform stress load below 1211 psi (dark blue region). Most of the load is on the fin itself and the thrust plate, and at interference points of the spar with the fin. The minimum FOS here is 2.89 and the fin spar itself does not sustain any load that comes close to the yield strength of Al 6061-T6, which is 35,000 psi, since the maximum stress sustained is 12,110 psi on the fin itself.

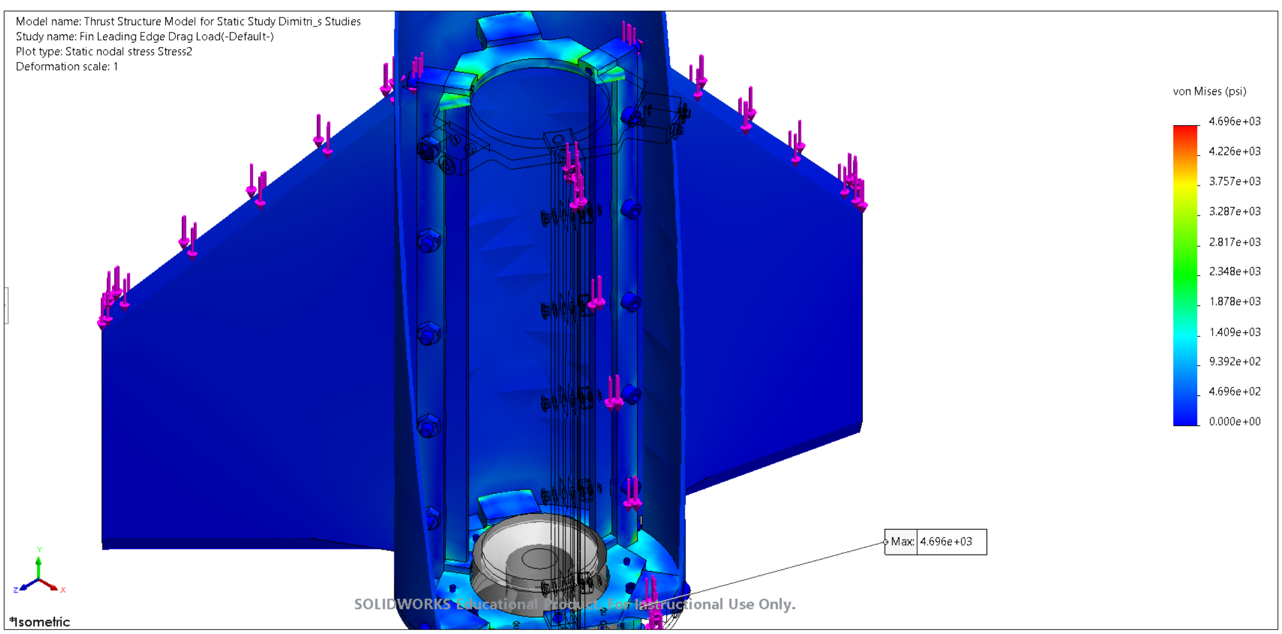


Figure 3.43: Von Mises stress on fin spars from fin leading edge drag simulation.

In the fin tip leading edge drag simulation, it is observed that the fin spar experiences a nearly uniform stress load below 939.2 psi (dark blue region), with areas near the top having an increased amount of stress, shown to be in the green region (1878-3757 psi). Most of the load is on the thrust plate, the thrust plate flange, the centering plate, and the spar's interference points with the fin. There is a load transfer from the fin spar to the thrust plate flange and the two plates at the attachment points, where stress gradually increases and reaches the cyan (939.2-1409 psi), then green, and even the yellow-orange region (3757-4226 psi). The minimum FOS here is 4.3, and the fin spar itself does not sustain any load that comes close to the yield strength of Al 6061-T6, which is 35,000 psi, since the maximum stress sustained is on the standoff beneath the thrust plate flange (which is partially hidden here) at 4696 psi.

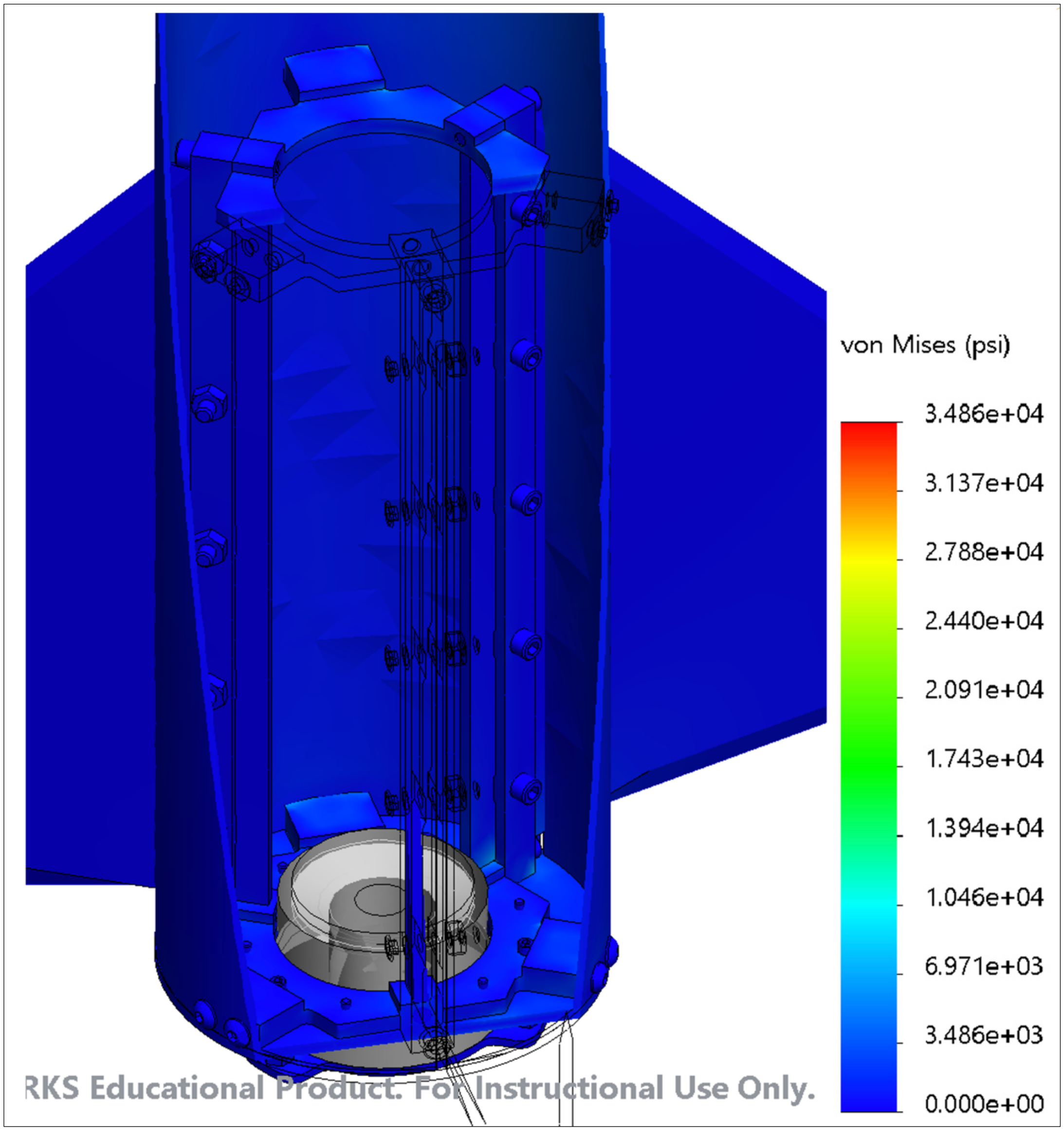


Figure 3.44: Von Mises stress contour plot on fin spars from landing load on fin tip simulation.

The landing load on the fin tip simulation largely indicates the fin spars remain in the blue region of stress levels (0-3486 psi). Stress is transferred from the fin spars to the thrust plate and the centering plate indicating the stress falls in the cyan region (6971 – 10,046 psi). The stress is transferred uniformly across the plates and into the tube, showing cyan regions around the plate spokes and blue into the tube.

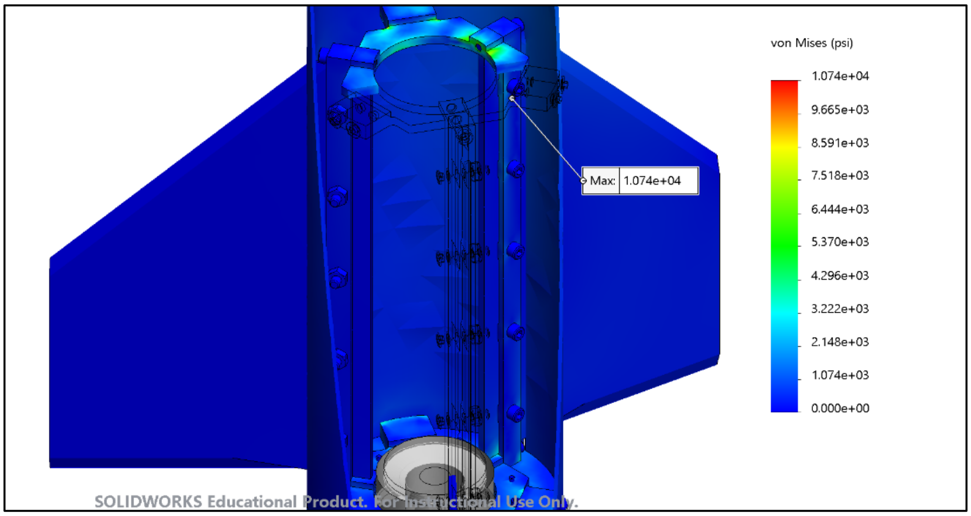


Figure 3.45: Von Mises stress contour plot on fin spars from landing load on fin trailing-edge simulation.

The landing load on the fin trailing edge simulation shows a uniform distribution of stress across the fin spar with maximum stress reaching the cyan region (2148-3222 psi) at contact points. As mentioned previously, the stress is transferred to the centering plate, thrust plate and thrust plate flange.

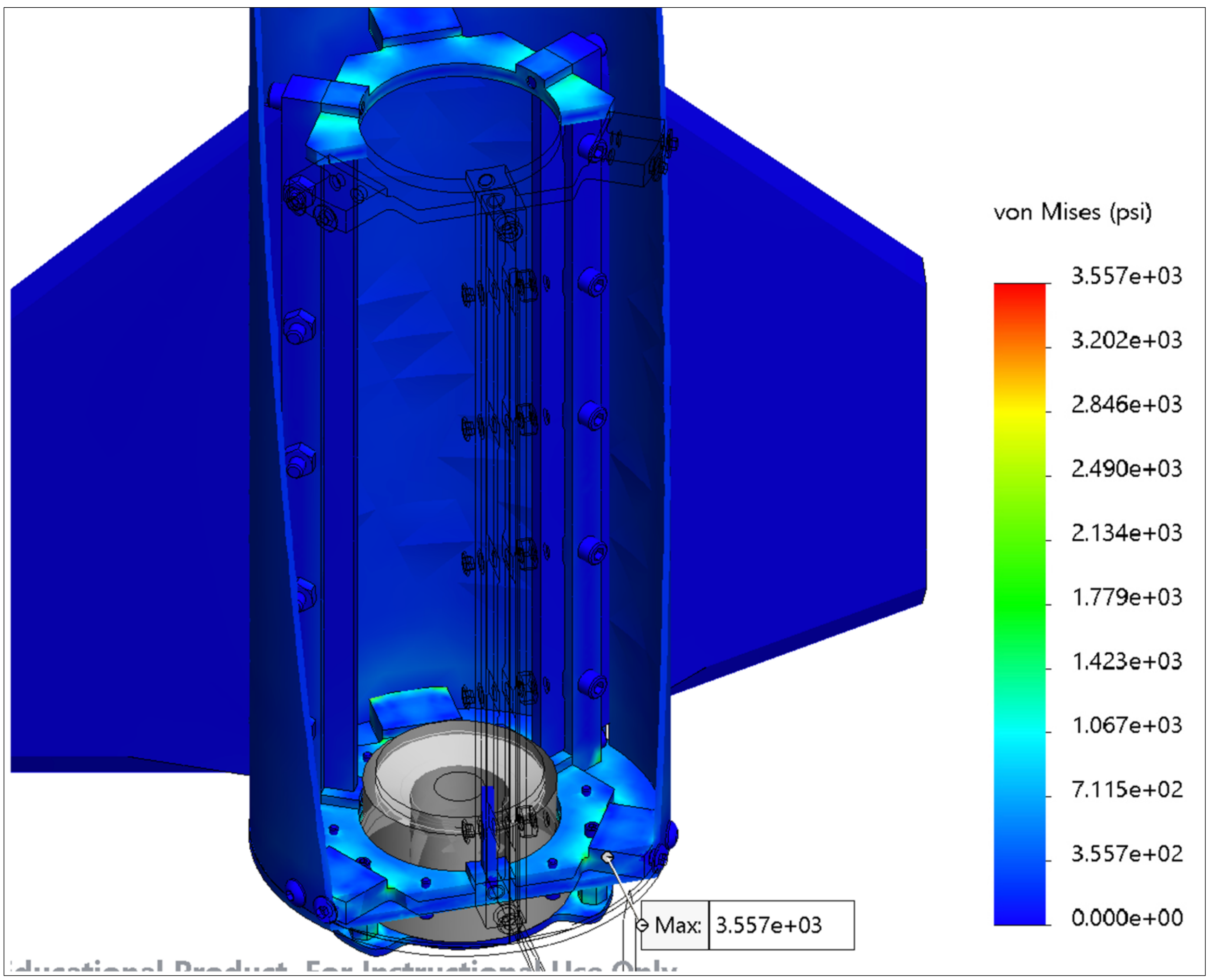


Figure 3.46: Von Mises stress contour plot on fin spars from landing load on motor retainer simulation.

The landing load on the motor retainer simulation shows a uniform distribution of stress across the fin spar with stress reaching the cyan region (711.5-1067 psi) and the green region(1067-2846 psi) at the contact point edge with the thrust plate flange. Again, the stress is transferred to the centering plate, thrust plate and thrust plate flange.

3.1.6.2.3.2 Lower Airframe

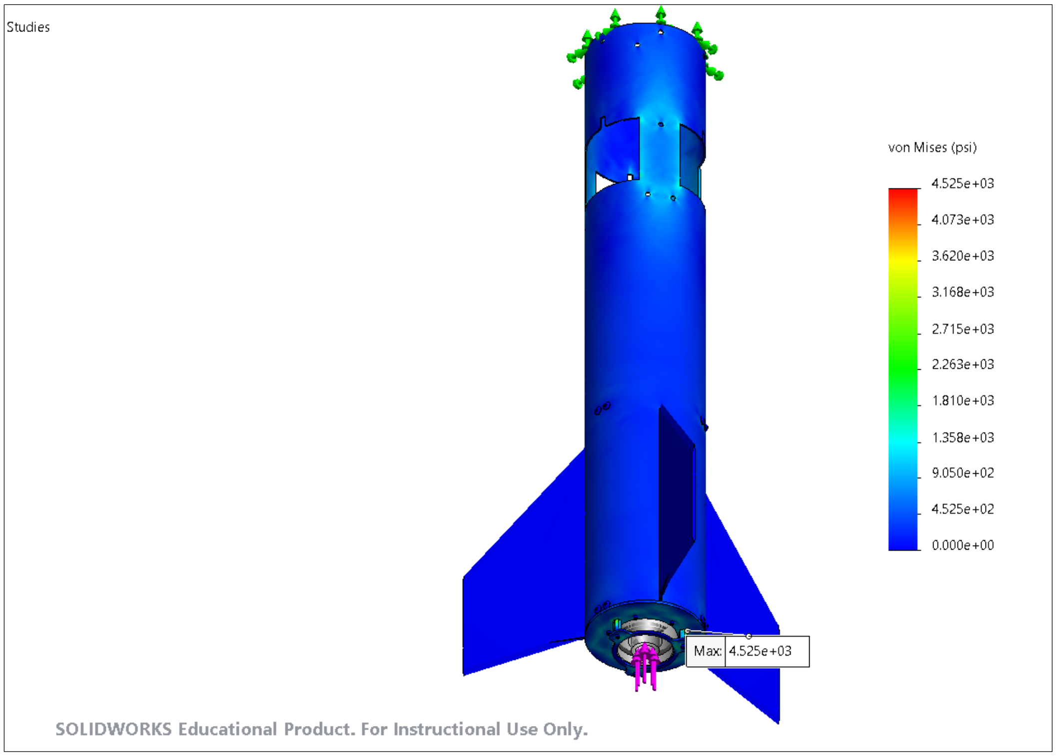


Figure 3.47: Von Mises stress contour plot on lower airframe from motor thrust simulation.

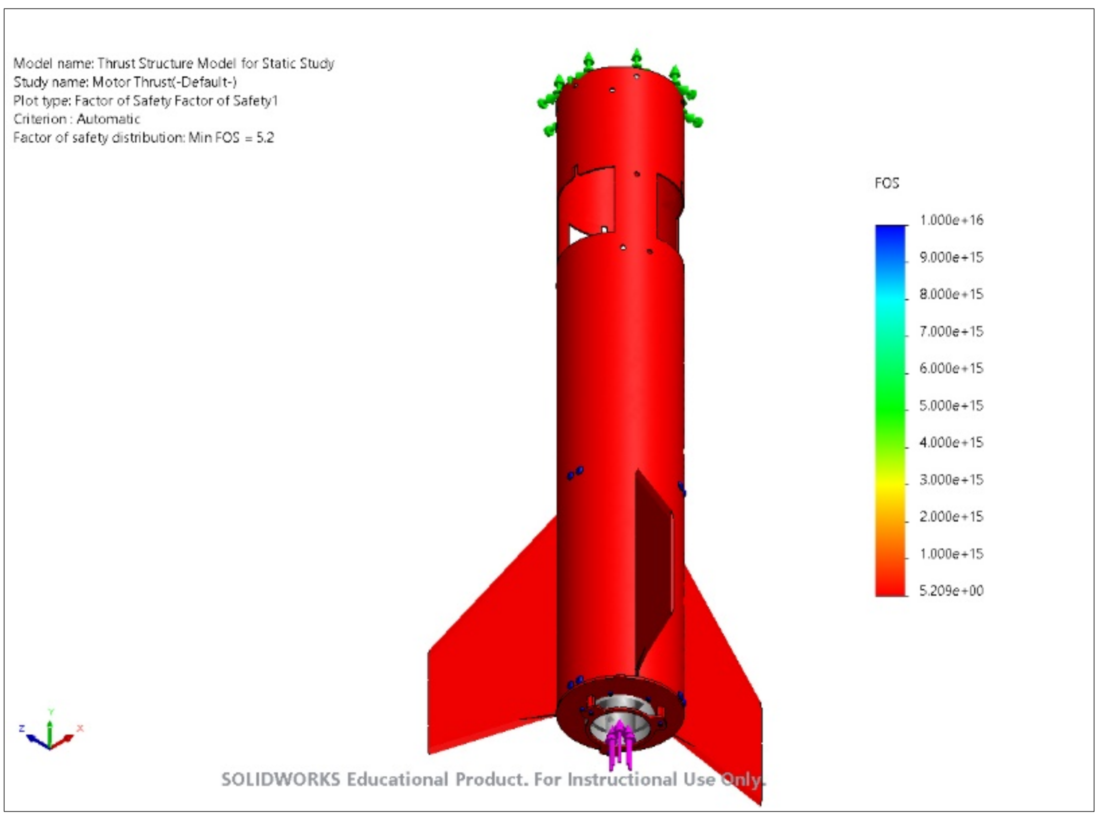


Figure 3.48: Factor of safety contour plot on lower airframe from motor thrust simulation.

In the motor thrust simulation, the lower airframe sustains a relatively larger stress load at the contact points with the thrust and centering plate spokes, near the fasteners, and at the contact point with the thrust plate flange. The region around the contact points transitions from dark blue (0-905.0 psi), to light blue (905.0-1358 psi), to cyan (1358-1810 psi). The stress loads then travel toward the fixed points (indicated by the green arrows), with stress concentrating at the point where the path becomes narrower, near the airbrakes holes. There, the stress load transitions to cyan and then to the green region (1810-3620 psi) and the load dissipates toward the top and away from the "choke point", where the stress is eventually transferred to the fixed points. The minimum FOS here is 5.2, and the lower airframe itself does not sustain any load that comes close to the minimum compressive strength of G10/G12 fiberglass, which is 65,000 psi, since the maximum stress sustained is 4525 psi on the standoff.

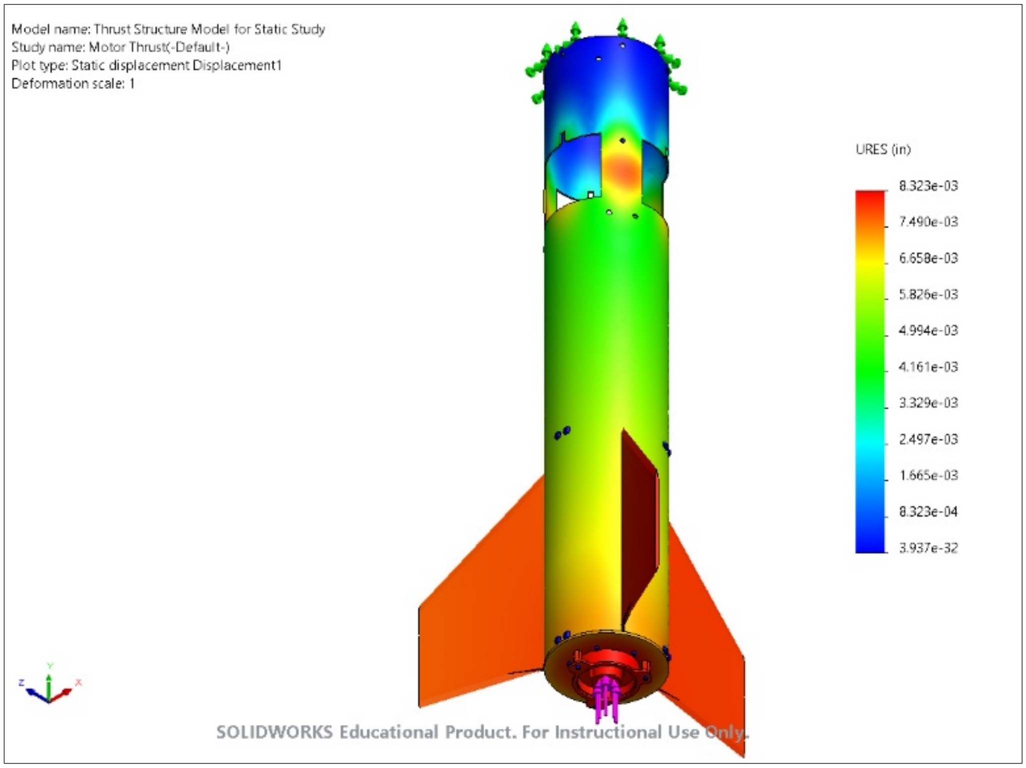


Figure 3.50: Displacement contour plot of lower airframe from motor thrust simulation.

The displacement of the lower airframe in the motor thrust simulation is larger close to the center of the lower airframe and initially towards the bottom, where the force is applied, and smaller towards the top, which is expected, since the stress is dissipating towards the top. However, the displacement becomes very large again where the stress increases, at the "choke points" and then becomes smaller again near the top, reaching the minimum values at the fixed points.

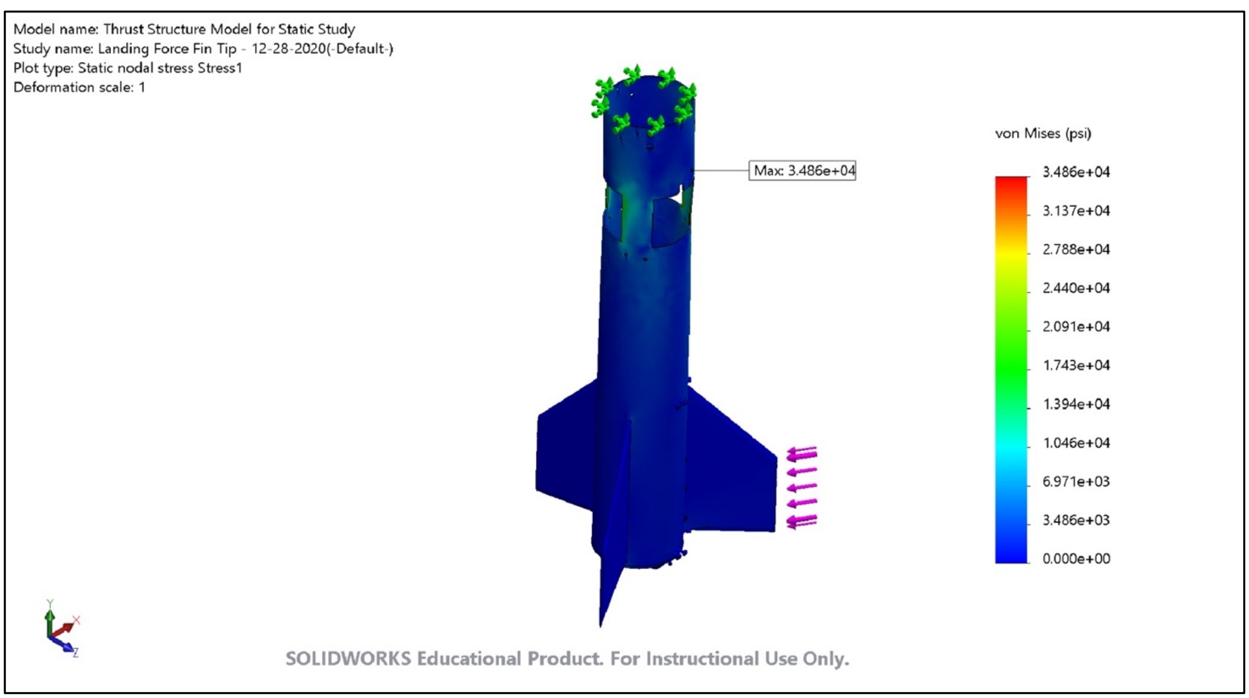


Figure 3.51: Von Mises stress contour plot on lower airframe from landing load on fin tip simulation.

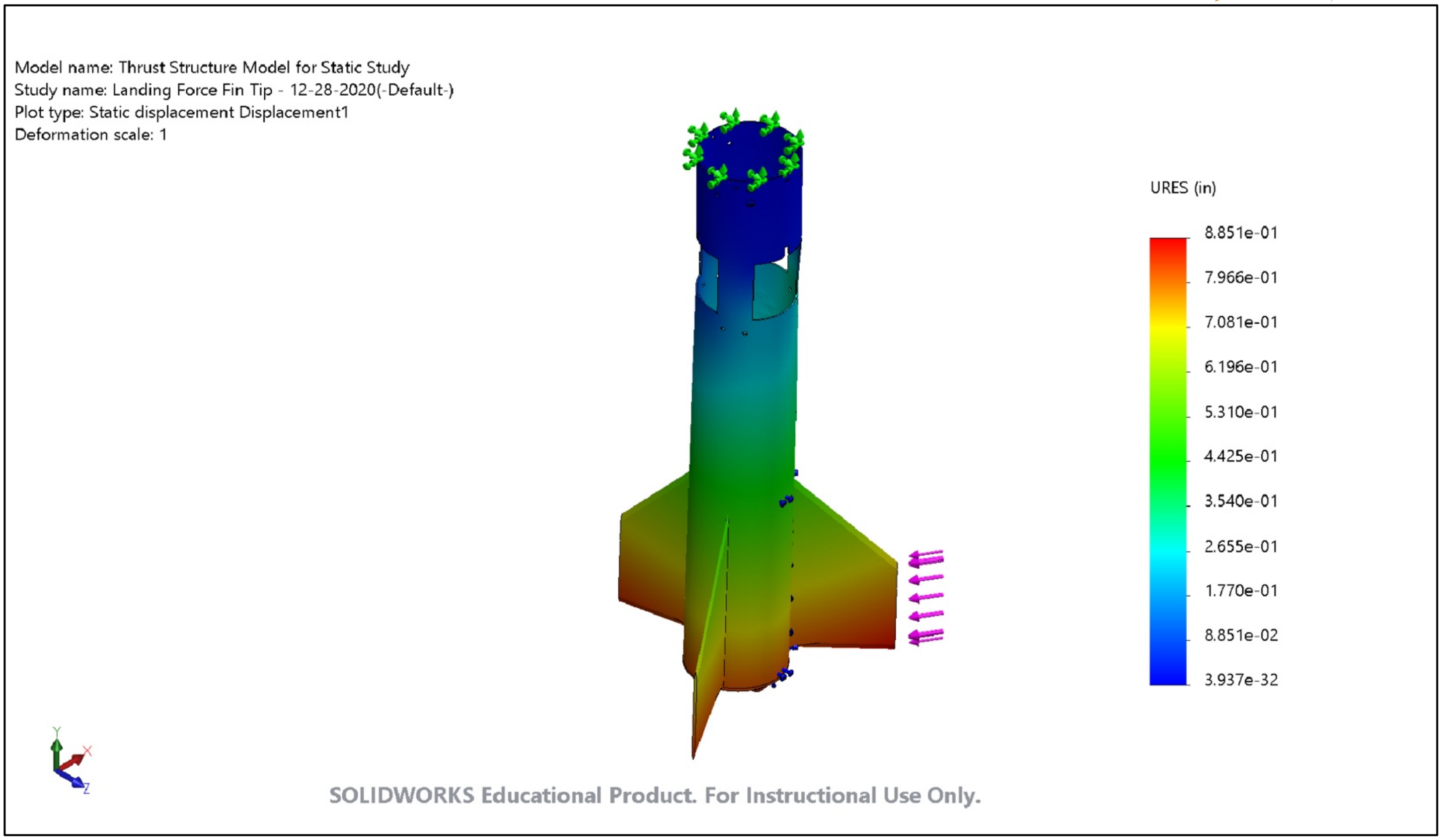


Figure 3.52: Displacement contour plot on lower airframe from landing load on fin tip simulation.

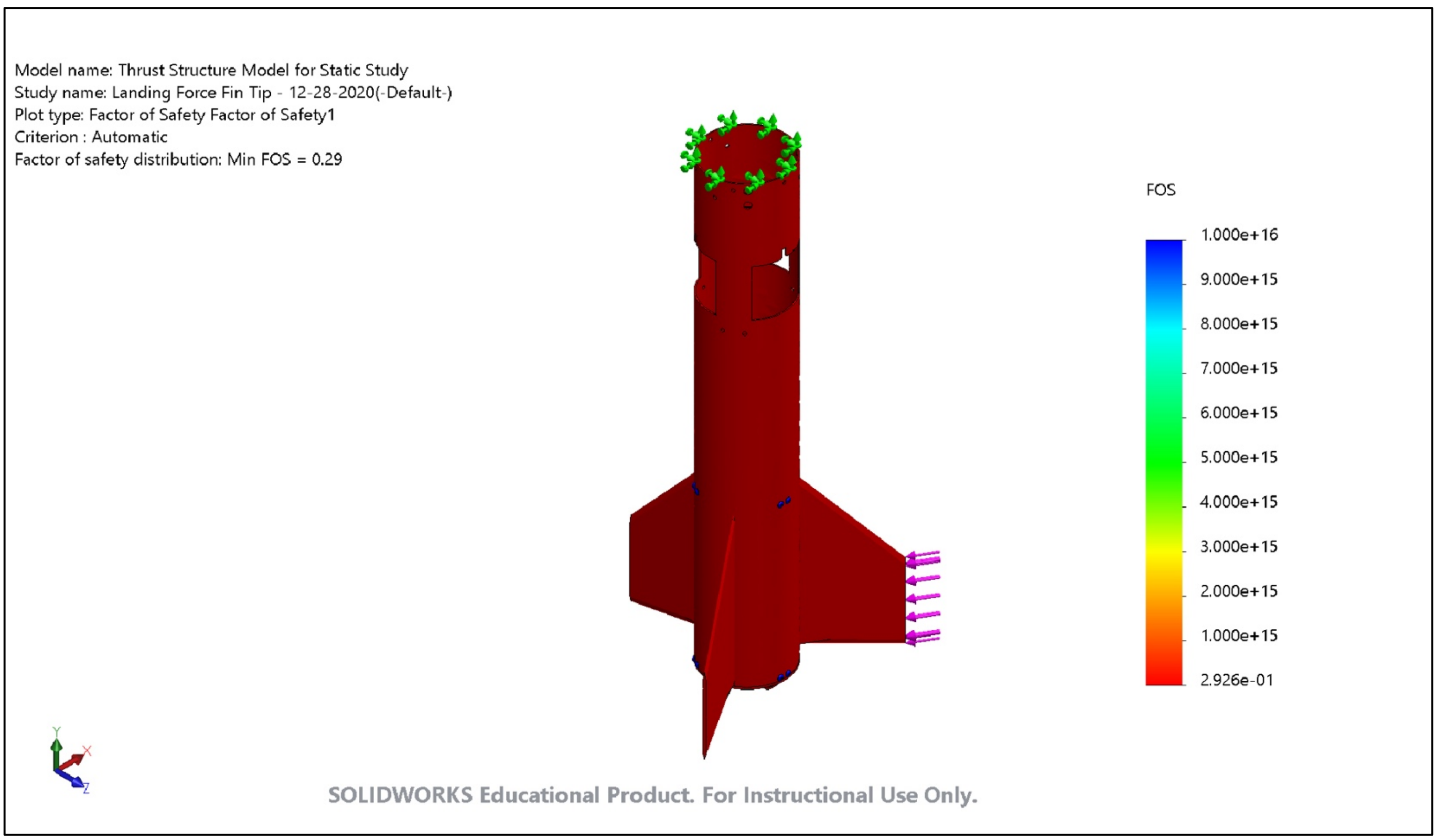


Figure 3.53: Factor of safety contour plot on lower airframe from landing load on fin tip simulation.

The landing load on the fin tip simulation indicates the maximum stress on the lower airframe is near the top of the tube. The stress (Figure 3.53) is uniform across the majority of the tube and is then sitting in the blue region (0-3486 psi) and near the top slots where the airbrake system will be placed, the stress moves to a higher cyan through yellow region (13,940 – 27,880 psi). It is also shown the minimum FOS is 0.29.

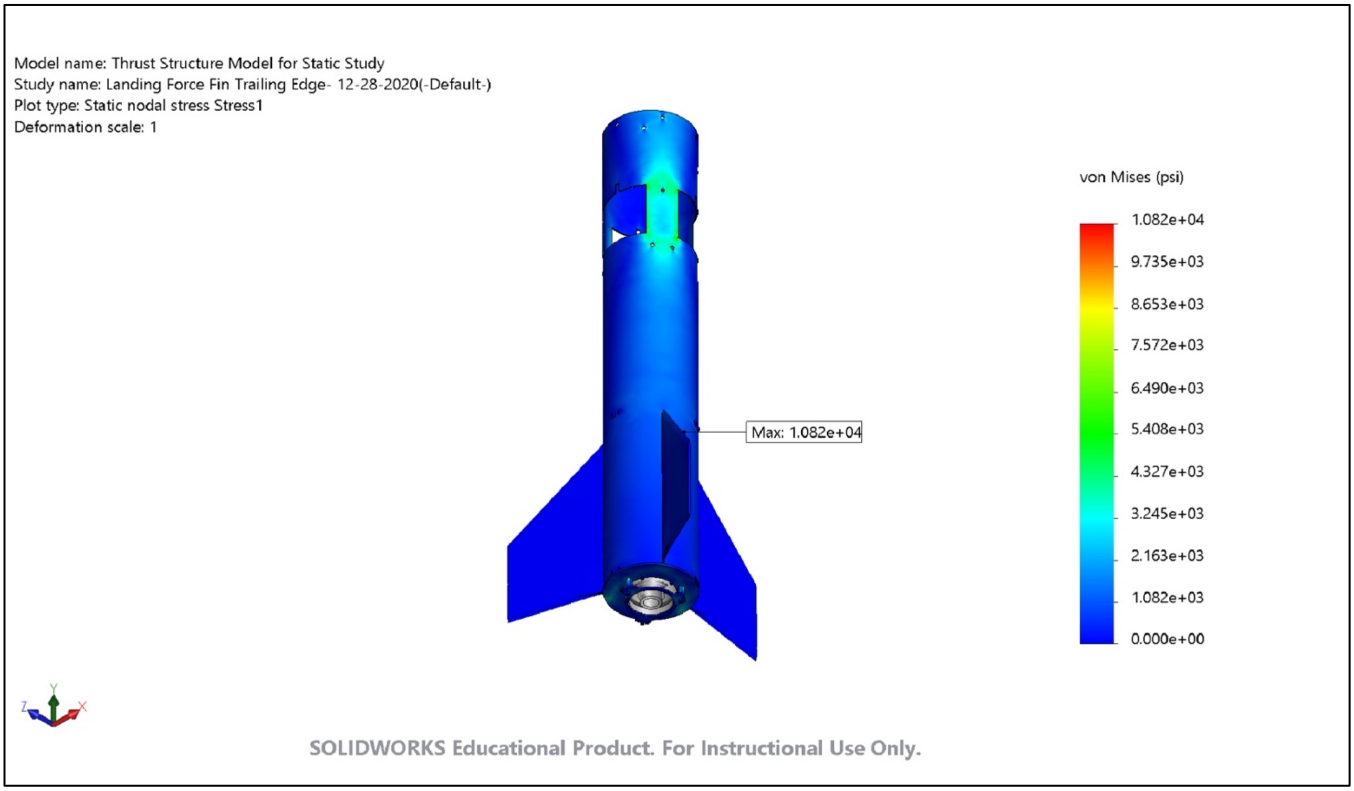


Figure 3.54: Von Mises stress contour plot on lower airframe from landing load on fin trailing-edge simulation.

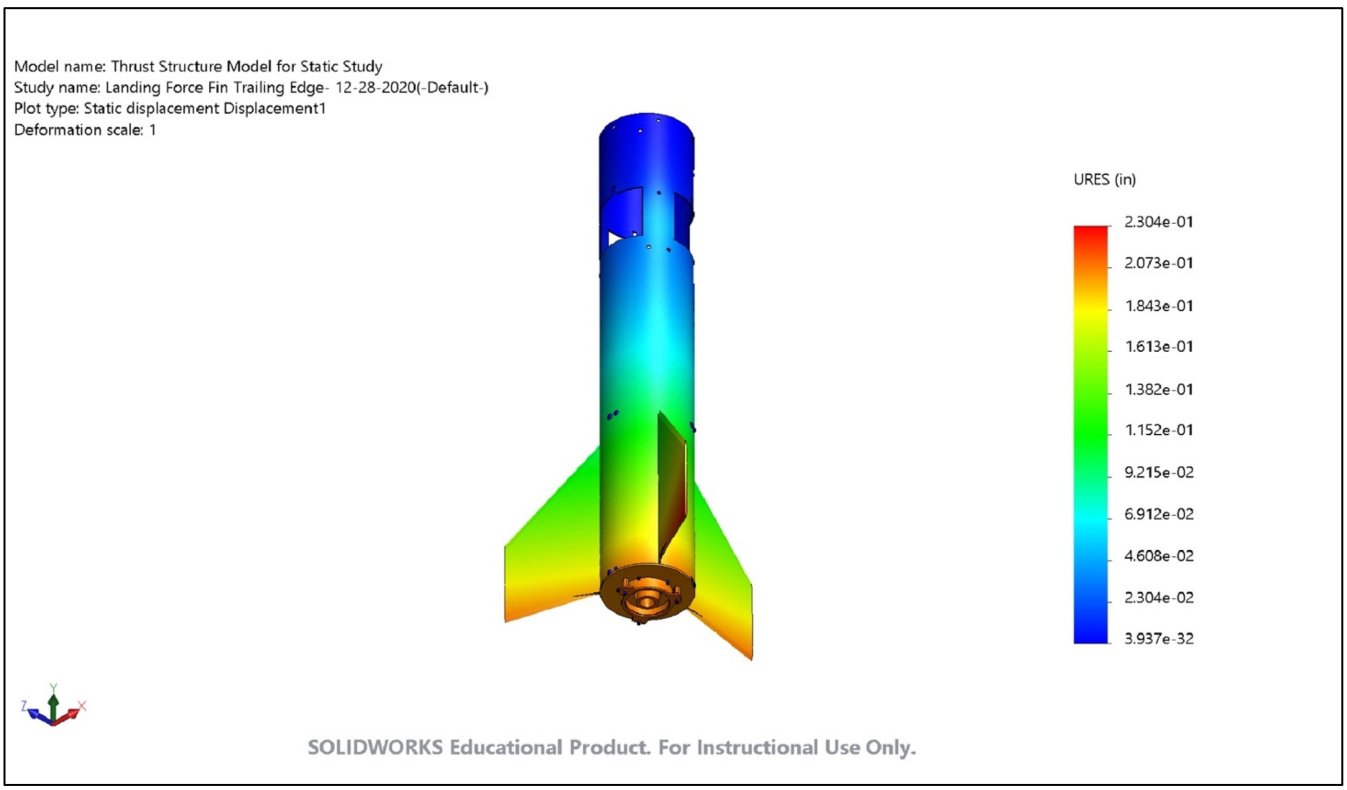


Figure 3.55: Displacement contour plot on lower airframe from landing load on fin trailing-edge simulation.

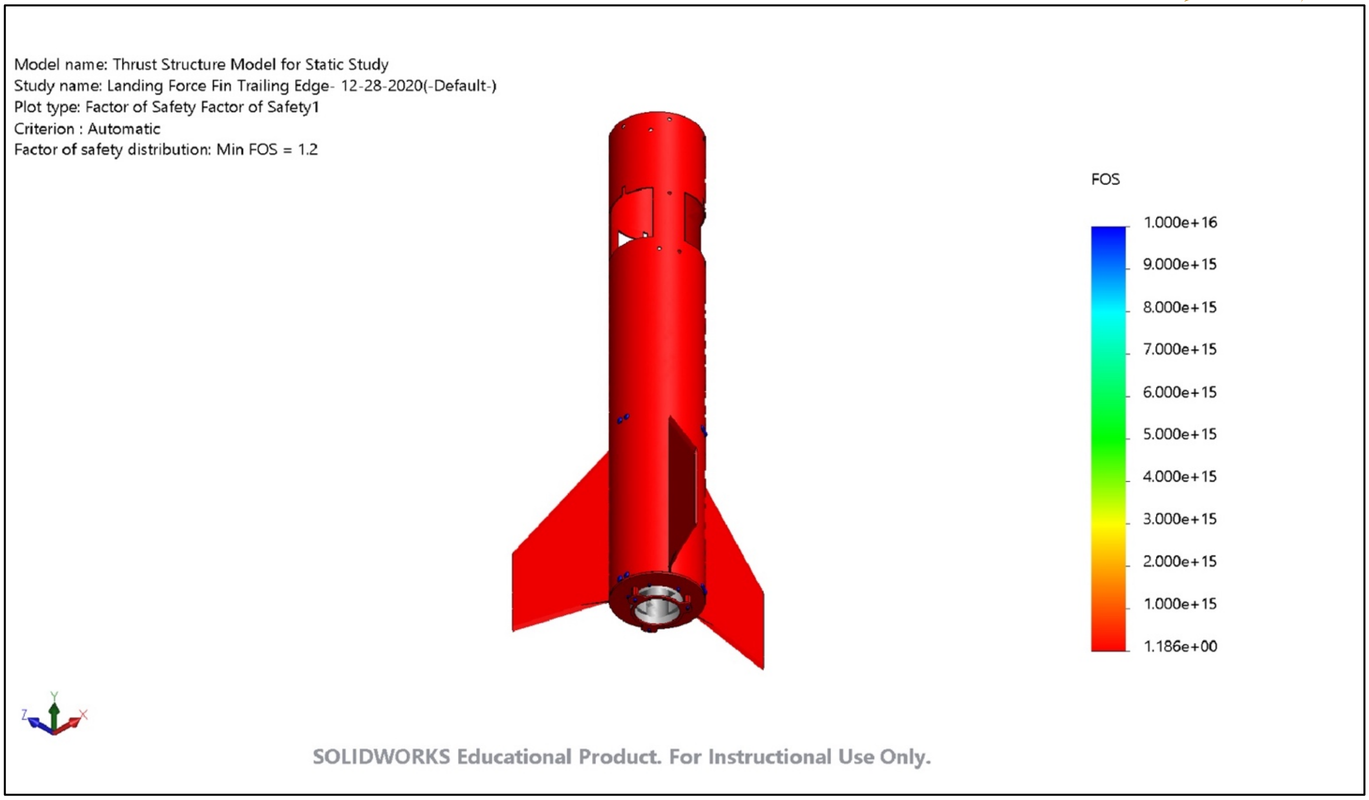


Figure 3.56: Factor of Safety contour plot on lower airframe from landing load on fin trailing-edge simulation.

In Figure 3.54, the landing load on fin trailing edge simulation shows mostly a uniform distribution of stress until reaching the top of the airframe tube. There the stress is between the cyan and green regions (2163 – 7572 psi) indicating most of the stress is concentrated around those points near the airbrake system. The minimum factor of safety distribution of this study was about 1.2.

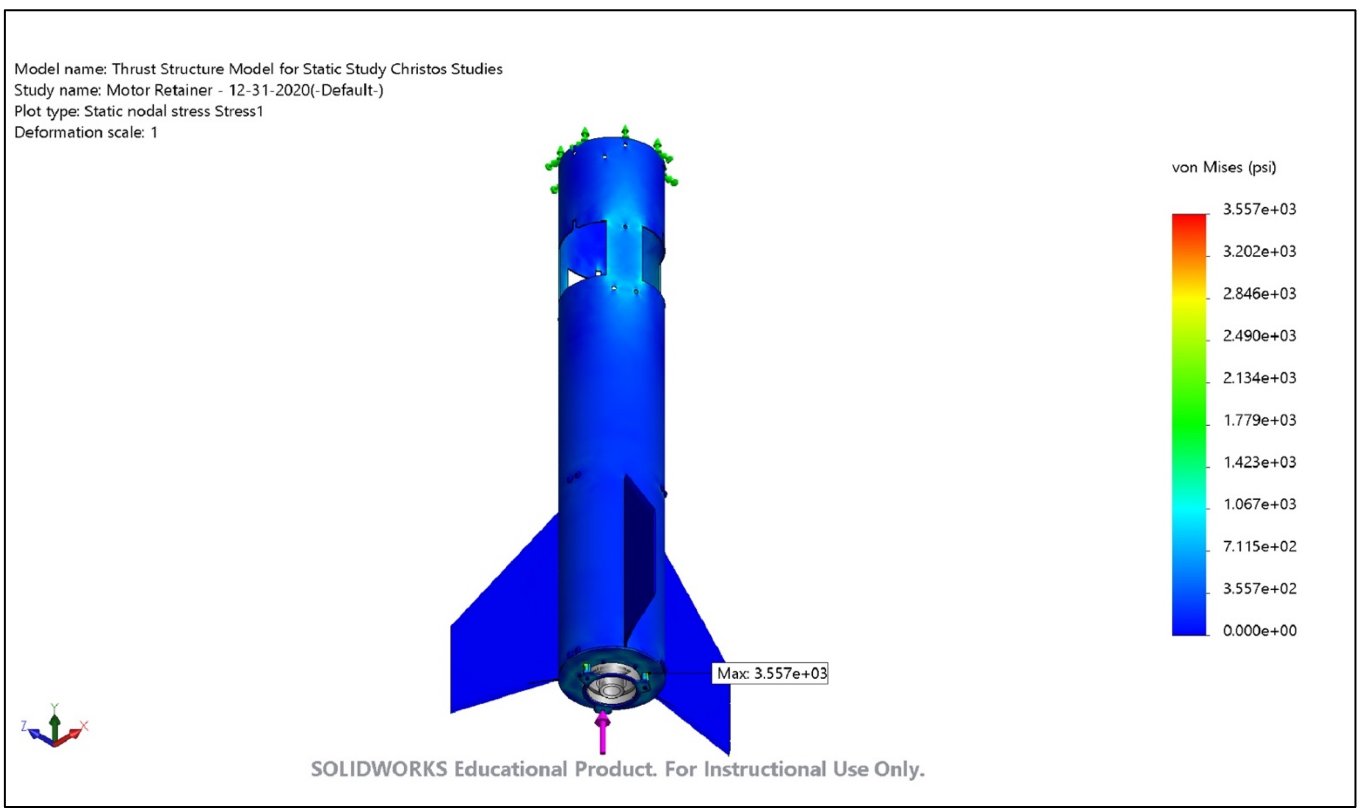


Figure 3.57: Von Mises stress contour plot on lower airframe from landing load on motor retainer simulation.

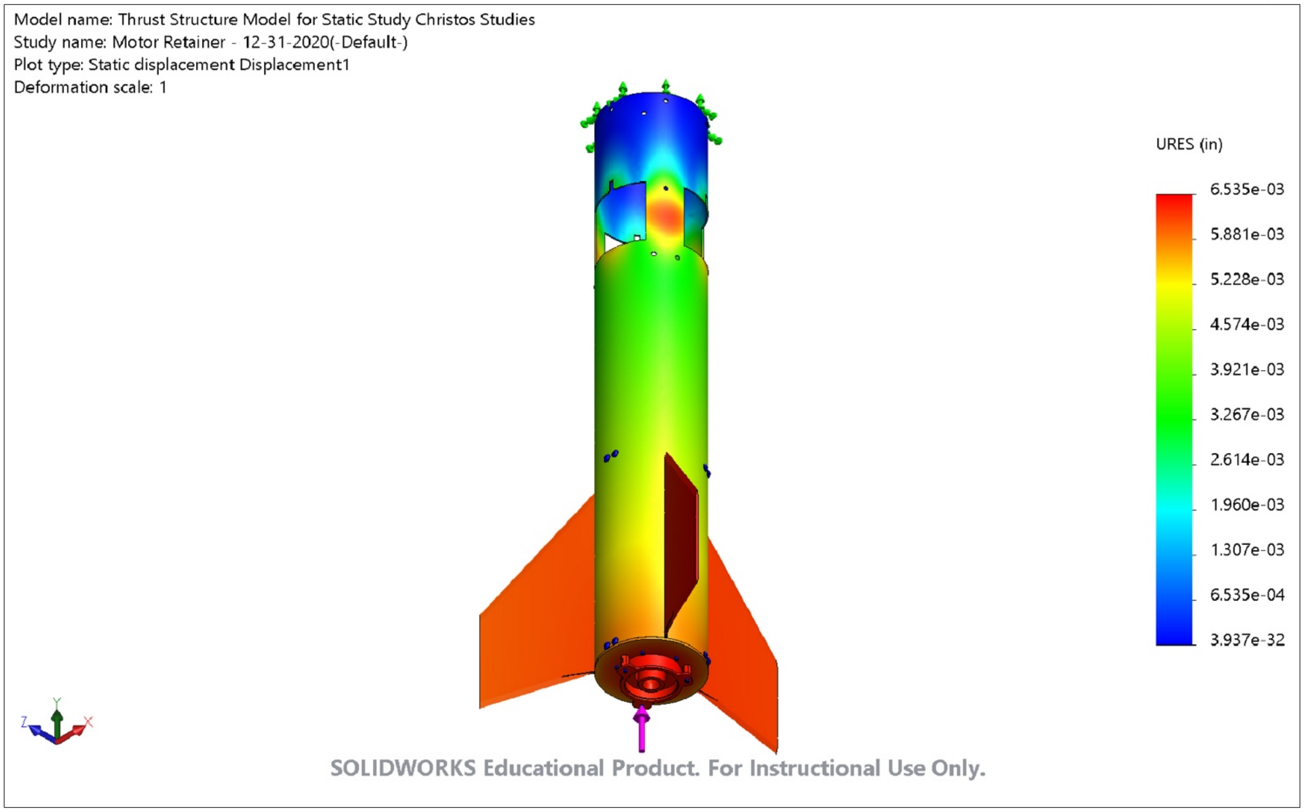


Figure 3.58: Displacement contour plot on lower airframe from landing load on motor retainer simulation.

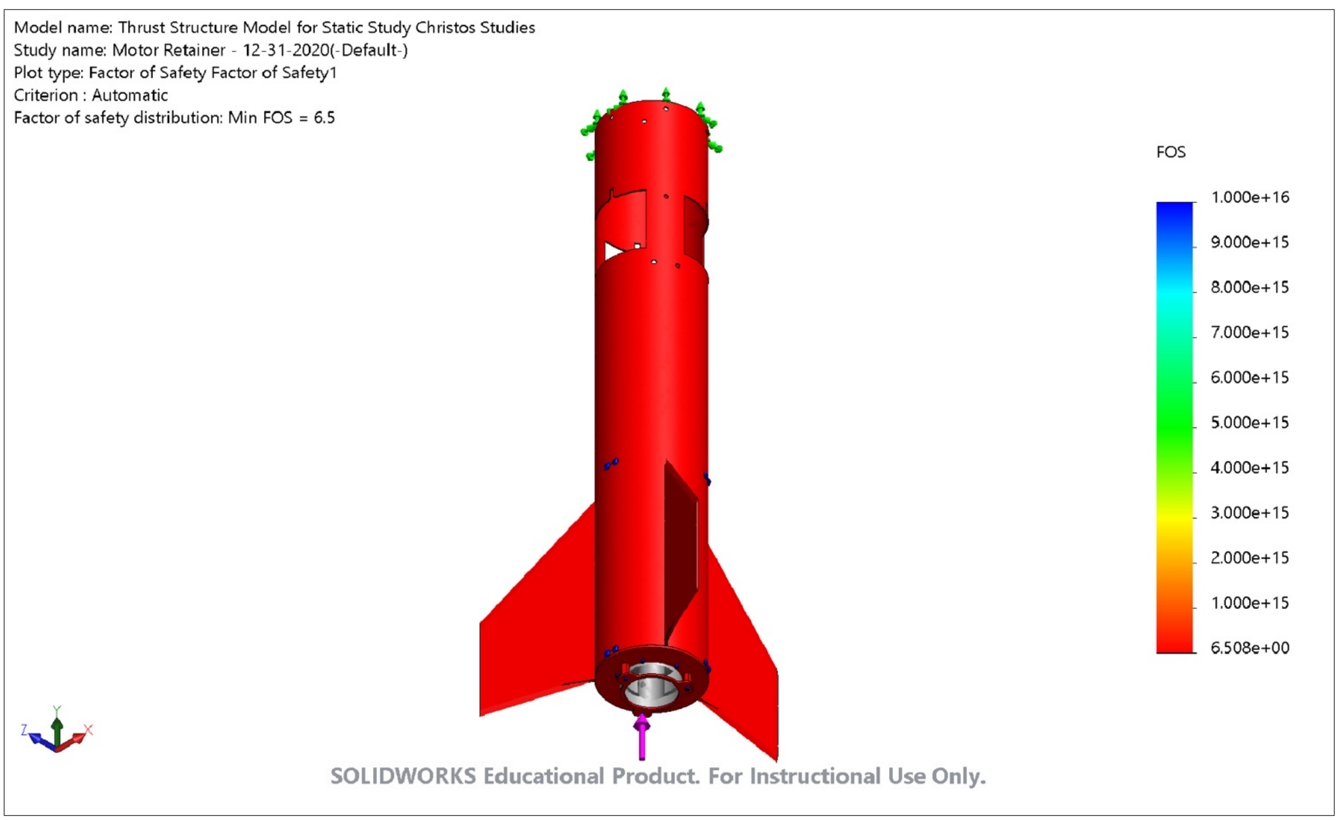


Figure 3.59: Factor of Safety contour plot on lower airframe from landing load on motor retainer simulation.

Similar to the landing load on fin trailing edge simulation, the landing load on the motor retainer shows mostly a uniform distribution of stress up to the top of the airframe tube where the airbrake system will be installed. The stress is in the cyan region (711.5 – 1067 psi) indicating most of the stress is transferred up there. The minimum factor of safety distribution of this study was about 6.5.

3.1.6.3 Motor Mounting and Retention

3.1.6.3.1.1 Motor Retainer Plate of MFSS & standoffs

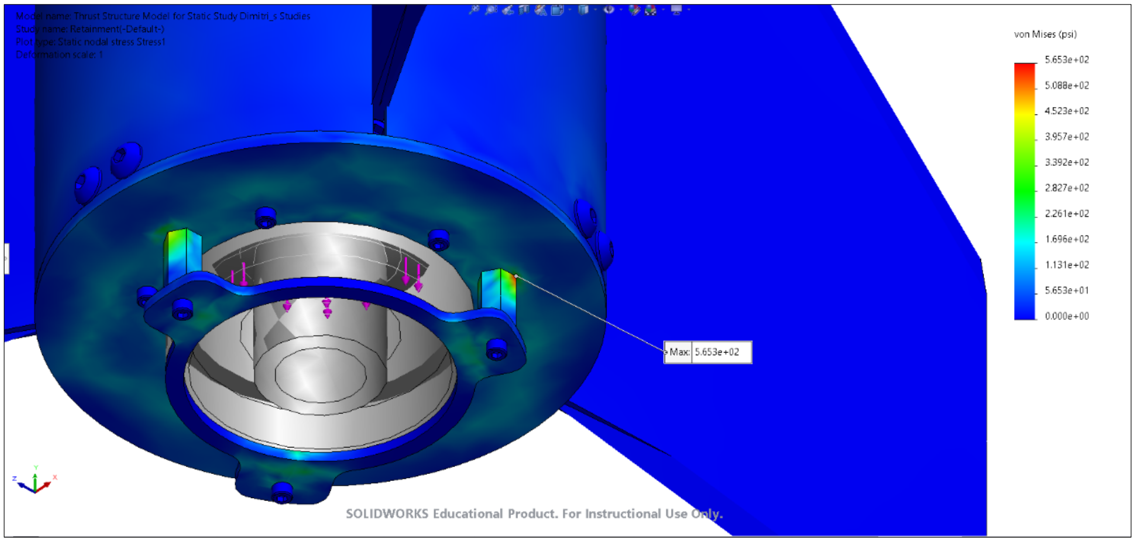


Figure 3.60: Von Mises stress contour plot on motor retainer from retainment simulation.

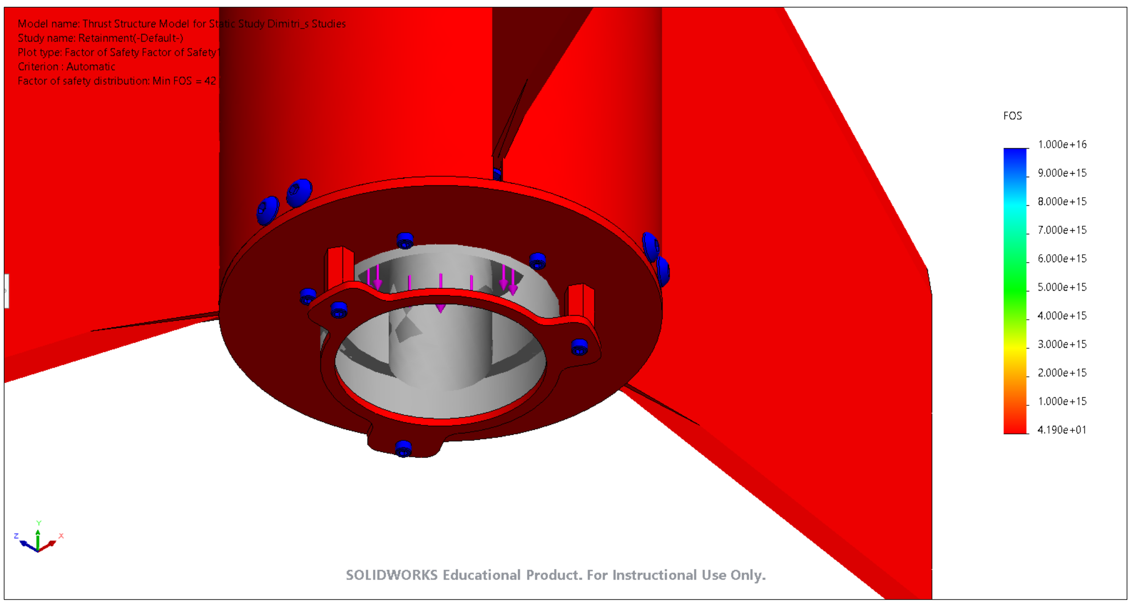


Figure 3.61: Factor of safety contour plot on motor retainer from retainment simulation. Notice that the minimum FOS is 41 times the approximate weight of the launch vehicle.

In the motor retainment simulation, the stress is transferred from the motor case aft closure (displayed in grey and approximated as rigid here) to the motor retainer plate, with stress being higher near the standoff attachment points, as stress is transferred from the ring to the standoffs and eventually to the thrust plate flange and the thrust plate. The minimum factor of safety in this simulation is 41 times the force applied, which is roughly equivalent to the launch vehicle's weight, not just the motor weight.

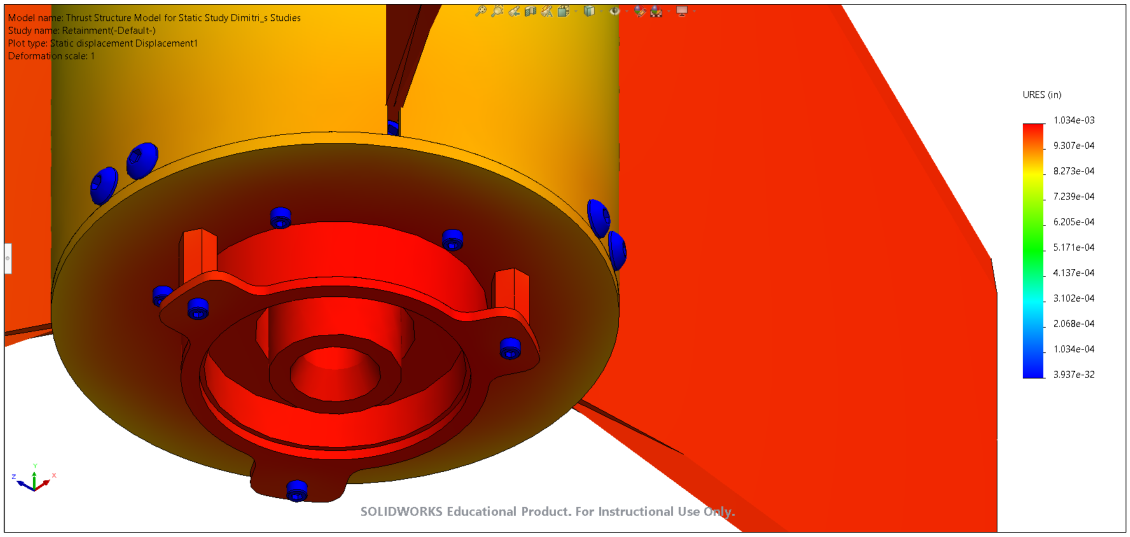


Figure 3.62: Displacement contour plot on motor retainer from retainment simulation

The displacement is largest near the motor retainer plate's inner hole, where the ring of the plate is in contact with the motor case aft closure that transfers the load. Displacement is similar for the fins as well, which are fixed onto the MFSS, meaning that the whole MFSS responds to the load on the retainer, which is why the retainer assembly with the motor retainer plate and the three standoffs can withstand large forces, as demonstrated by the minimum FOS.

3.1.6.4 Final Mass of Vehicle and Subsystems

The team decided to perform a mass calculation of the final vehicle and subsystems so far to ensure that the different components and subsystems being created are not overweight. This is a crucial step in the design review process because if the mass of the launch vehicle is similar to that of the vehicle mass in OpenRocket, then the team knows that all the calculations and predictions are done regarding trajectory and apogee calculations are accurate. The team did an in-depth mass calculation for the entire launch vehicle incorporating all the different parts and subsystems of the launch vehicle to calculate the launch vehicle's final mass. The table below shows the breakdown of the different sub-components of the launch vehicle and each component's mass.

Airframe Component	Internal Component		Internal component Mass (oz)	Airframe Component Mass (oz)	Material
Nose Cone	Nose Cone		17.31	0	Carbon Fiber reinforced Nylon
Nose Cone Shoulder	Adjustable Ballast		0	1.61	N/A
	Tube Coupler		8.4		G12 Fiberglass
Payload Section	Payload Coupler	Bulkhead	3.94	38.739	G10 Fiberglass
		Bulkhead	3.91		G10 Fiberglass
		Payload Tracker	8		
	Payload		144		N/A
Upper Recovery Section	Main		32	28.64	Nylon
Avionics bay	Avionics		85.12	1.41	Glass Fiber
	Tube Coupler		36.07		Glass Fiber
Lower Recovery Section	Launch lug		0.123	23.68	Aluminum
	Drogue		2.19		Nylon
Booster Section	Airbrakes		63.61	71.59	Onyx Carbon Fiber/ Stainless Steel
	Booster Coupler	Bulkhead	3.91		G10 Fiberglass
		Optional lower ABCS Bulkhead	3.75		G10 Fiberglass
		Booster Tracker	8		
	Trapezoidal fin set		48.64		G10 Fiberglass
	Thrust/fin structure		44.8		Aluminium 6061-T6
	Launch lug				
Final Mass of Launch Vehicle without motor (lbm)			42.465		
L1115-0 Motor mass (lbm)			9.63		
Final Mass of Launch Vehicle with motor (lbm)			52.1		

Table 3.4: In-depth overview and calculation of the mass of the launch vehicle.

The final mass of the launch vehicle is 52.1 lbm. This is approximately the launch vehicle mass estimated by OpenRocket. However, the team expects an increase in vehicle mass once all the material properties are applied to the different parts, as the different components and sub-components of the launch vehicle do not have their material characteristics applied as of the submission of CDR. It is also important to note that the final mass of the launch vehicle shown above does not incorporate the mass of the camera. Also, the mass does not include miscellaneous components (nuts, screws, epoxy, paint, etc.) that will increase the final mass. The team also can add ballast to the nose cone, so it is expected that the final mass will be around 53 lbm.

3.2 Subscale Flight Results

3.2.1 Scaling Factors

3.2.1.1 Constant Factors

- The outer diameter of the subscale launch vehicle was constructed following a 0.51:1 ratio from subscale to full scale
- The nose cone's length and the lower recovery section were scaled using a 0.5:1 ratio from subscale to full scale.
- The inner diameters of the payload, avionics bay, lower recovery and booster sections were scaled using a 0.5:1 ratio.
- The trapezoidal fin set followed a 0.5:1 ratio

3.2.1.2 Variable Factors

- The motor's diameter followed a 0.51:1 ratio while its length followed a 0.25:1 ratio. This was because the motor chosen for the subscale launch vehicle had to abide the permissions the team had for the launch field and the allowable altitude given to the team by Indiana Rocketry Incorporated and the commercially available motors. This caused the motor tube and retainer to be variable factors as well.
- The wall thickness of the payload, avionics bay, lower recovery and booster section was scaled to a 0.74:1 ratio. This was due to a manufacturing limitation. The only 3" G12 fiberglass that the team had access to happened to have a thickness in the 0.74:1 ratio. Otherwise, the team would have had to manufacture their own fiberglass. Similarly, the fin set thickness was scaled to a 0.27:1 ratio because of the limited G10 fiberglass access.
- The payload length was scaled to a 1.11:1 ratio. Since the subscale launch vehicle only used the drogue parachute, the team didn't need space for the main parachute. However, the team had to make the payload length longer to maintain a similar stability as the full-scale launch vehicle.

3.2.2 Launch Day Conditions

The 2021 PSP-SL team conducted a subscale launch of the intended launch vehicle in order to test and verify preliminary design decisions and systems. The subscale vehicle was designed, debated, constructed, and launched in the two months leading up to November 7th. With the test launch on this date, the team wanted to compare anticipated apogee, drift distance, and velocity data with flight data, as well as verify the validity of the redesigned hemispherical nose cone.

The subscale launch on the 7th of November was held in a field located at the Purdue Dairy Farm, near West Lafayette, IN. The team mentor Christopher Nilsen obtained the necessary permissions to launch at this location. At the time of launch, roughly 11 am EST, the weather was sunny. Temperature ranged from 65-70 °F, wind was 6mph from SSW, and there was no precipitation.

3.2.3 Flight Analysis

3.2.3.1 Predicted Vs. Recorded Flight Data

For the subscale vehicle, the team chose to use the same motor as last year's competition. The Aerotech H148R is a reloadable redline propellant motor of 1.5 in diameter, 5.98 in length, 4.3 oz propellant weight, and 10.9 oz loaded weight. The motor provides an average thrust of 33.27 lbf over its 1.4 s regressive burn time. This motor in conjunction with the subscale vehicle were launched and the altimeter recorded the following information shown compared to the predicted Simulink simulation data.

	Predicted (Simulink)	**Recorded**
Wind (mph)	10	6
Inclination (degrees)	10	15
Apogee (ft)	625	620
Max Velocity (ft/s)	182	202.4
Drift Distance (ft)	234	207
Ascent Time (s)	6.8	6.9

Table 3.5: Predicted vs. Recorded Flight Data of Subscale Launch

As seen in the table above, the prediction data holds when compared to the measured parameters. The launch weather fared below the wind used for calculation, while the inclination angle was higher. In addition to the inclination angle, the subscale vehicle was observed to tilt further to a greater angle from the vertical after clearing the launch rail, before stabilizing a few seconds later to proceed on a straight path. In other words, the vehicle began twisting off the end of the rail, reducing its angle of attack from vertical to a slightly horizontal angle until the moment was neutralized and the vehicle stabilized. The team concluded this twist was a product of low stability margin and an offset center of mass in its construction.

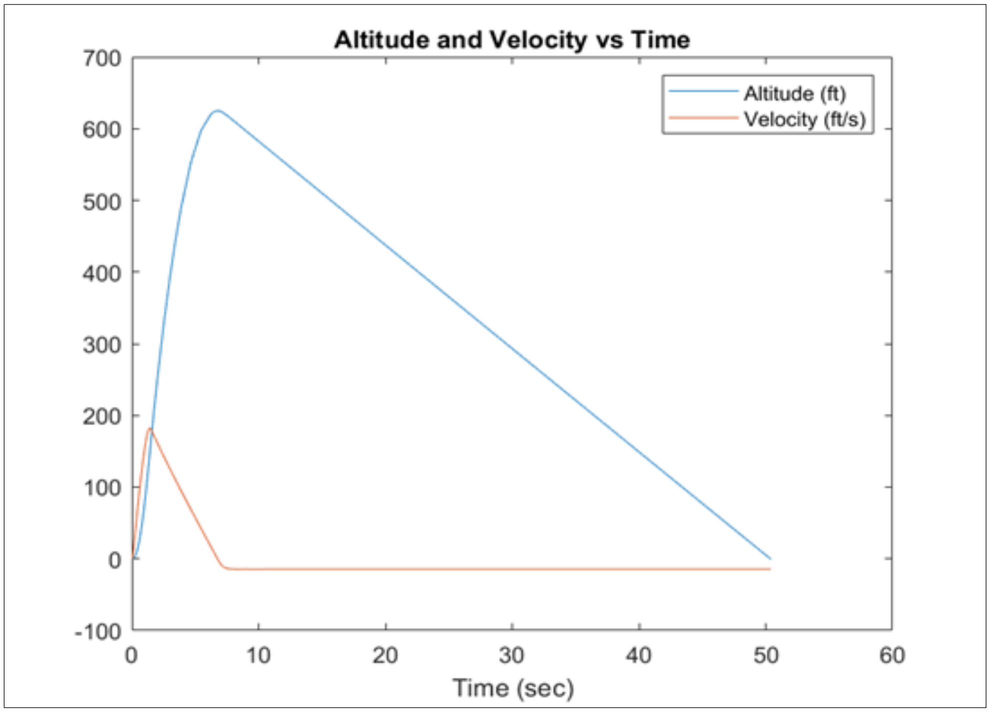


Figure 3.63: Simulated Altitude and Velocity of Subscale Launch from Simulink

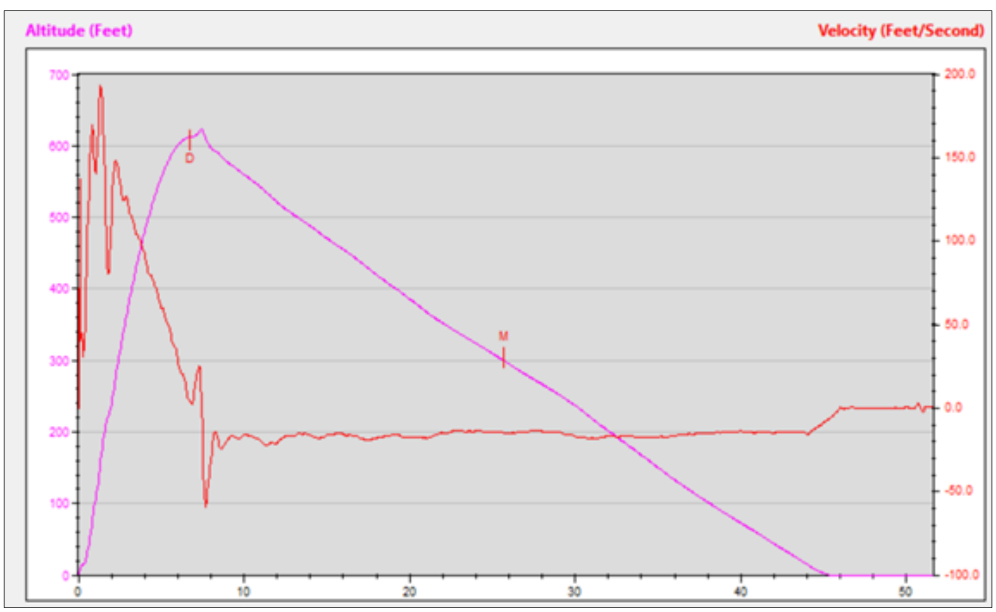


Figure 3.64: Recorded Altitude and Velocity of Subscale Launch from Altimeter Data

Two of the most important factors, altitude and velocity, were extremely close between simulation and practice. Actual altitude varied by a mere 5’ from prediction (lower recorded), while maximum velocity varied by 20.4 fps during ascent (lower predicted). Similarly, the drift distance varied by 27’ (predicted 234’ vs. actual 207’). The team is satisfied with the accuracy of these results.

3.2.3.2 Errors and Discontinuities

Each cause for error in the calculation was appropriately identified. The team determined the lower apogee was a product of the unexpected launch angle tilt from both the rod and the induced moment. Additionally, the higher measured velocity was due to mass discrepancy. The lower drift distance is likely attributed to slower wind speeds on launch day than anticipated in this set of simulations. The similarities speak to the validity of the team's Simulink program in estimating critical flight parameters.

3.2.3.3 Estimated Full-Scale Drag Coefficient

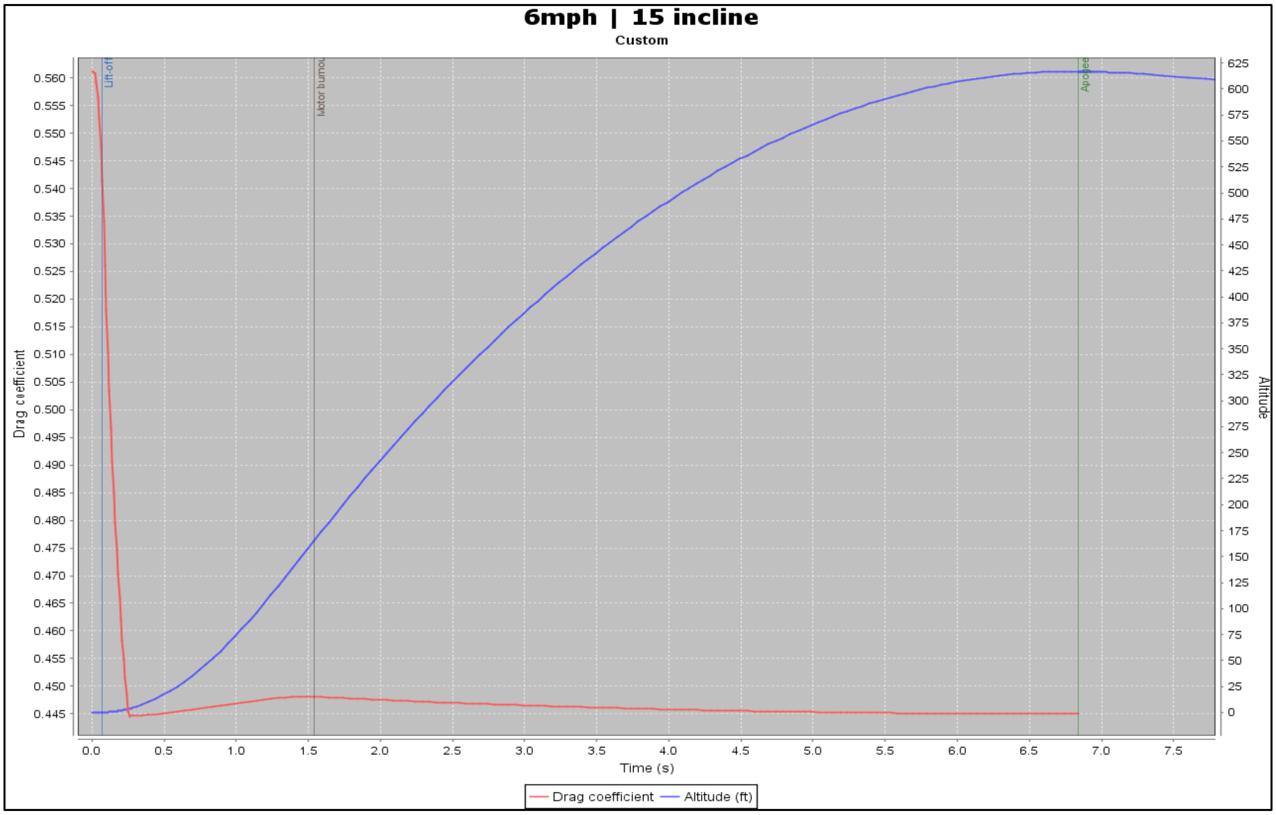


Figure 3.65: Drag Coefficient of Subscale Vehicle in Launch Day Conditions

The team used OpenRocket to simulate a subscale vehicle flight using the launch day conditions (6 mph wind and 15° inclined launch rail). OpenRocket was used extensively for simulation purposes. The program produced similar results to the Simulink simulation for subscale and has been confidently used in prior competitions, verifying its validity. In this simulation, the vehicle reached an apogee of 616.8' AGL (actual was 620') with a maximum drag coefficient of 0.56 at launch (0 s) and 0.45 in flight during ascent (1.44 s). Throughout the flight, the vehicle has an average drag coefficient of 0.455. Due to the subscale model's dynamic similarity and full-scale vehicle exteriors, the drag coefficient will be the same. The nose cone, airframe, and fin structure account for all exterior features of the vehicle and are of constant scale between subscale and full-scale. As a result, the full-scale vehicle has an estimated drag coefficient of 0.455.

The ABCS creates a significant deviation from the estimated coefficient of drag. The airbrakes will extend flaps from the exterior body outward, increasing the vehicle's wetted area and inducing significant skin friction and pressure drag on the vehicle. This action will greatly increase drag and its coefficient while active, slowing the vehicle. The estimated drag coefficient does not account for this system in flight.

3.2.4 **Impact to Full-Scale Design**

Following the subscale launch's success, the team deemed the current full-scale design optimal and safe to pursue. The main factor discussed after the flight was the feasibility of the hemispherical nose cone. The subscale data results proved that this was indeed not only a safe system, but that it would work as intended. The subscale flight data verification proves that the full-scale design is the team's best chance for a successful Project Voss.

3.3 Recovery Subsystem

3.3.1 Avionics & Recovery CDR Design & Justification

3.3.1.1 Coupler

In order to connect the upper and lower recovery sections of the vehicle and house all avionics components, a coupler with an outer diameter of 5.998", an inner diameter of 5.775", and an overall length of 5" is used. The coupler is made of the same material as the rest of the airframe (G12 fiberglass), and its dimensions were chosen to provide the required volume for all the necessary avionics components. Additionally, the coupler has two bulkheads on the forward and aft ends to provide support for the rest of the recovery components.

The coupler includes two ½” holes for access to the key switches used for the altimeters, four corresponding holes used to secure the switch holders to the coupler with ½” 6-32 screws and hex nuts, twelve ¼” holes for attaching the coupler to the upper and lower recovery sections of the vehicle, and four #8 static port holes on the forward end of the coupler to allow the altimeters to determine the current altitude.

3.3.1.2 Switch Band

The switch band has an inner diameter of 6”, an outer diameter of 6.17” (allowing it to slide over the coupler), and a length of 1” (allowing it to accommodate the key switch holes while adding as little additional length to the vehicle as possible). Likewise, it has ½” holes for accessing the key switches and four holes for the 6-32 screws and hex nuts. The switch band is epoxied around the center of the coupler and is also made of G12 fiberglass.

3.3.1.3 Bulkheads

One G10 fiberglass bulkhead with the same outer and inner diameters as the coupler and an overall thickness of 0.25” seals each end of the avionics bay. Each bulkhead has one ¼” hole in the center to fit an eye bolt, two ¼” holes located 2” away on either side to fit the threaded rods, and six additional #4 holes spaced evenly about the same circumference as the threaded rod holes. These holes are used to secure two 8g capacity black powder canisters and two terminal blocks, which are placed on opposite sides from each other on the bulkhead. The remaining two holes are used to feed the lighter connection wires from the interior of the coupler to the terminal blocks.

3.3.1.4 Primary and Redundant Altimeters and Batteries

The primary altimeter is the Altus Metrum TeleMetrum, which operates with a 3.7 V LiPo battery. This altimeter was chosen because of its high reliability in many past launches as well as its GPS/live telemetry capabilities. The redundant altimeter is the PerfectFlite StratoLoggerCF, which differs from the Missile Works RRC3+ Sport used in past years. This change was made due to configuration changes in the vehicle that necessitated a relatively short avionics bay (5” in length), so the RRC3+ Sport would be a tight fit. The StratoLoggerCF is much shorter (only 2”) and has all of the capabilities of the RRC3+ Sport, so it was chosen instead to serve as the redundant altimeter. The team prioritizes utilizing two altimeters of different makes/models in order to increase the likelihood that if a failure occurs in the primary system, the same one will not also occur in the redundant system, resulting in catastrophic failure.

Figure 3.66: TeleMetrum Altimeter

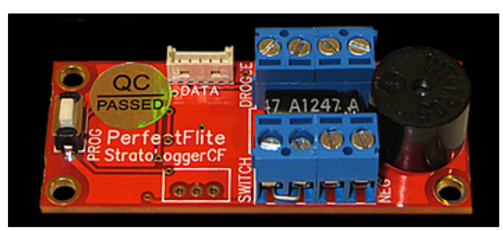


Figure 3.67: StratoLoggerCF Altimeter

3.3.1.5 Ejection Charges

The ejection charge type that is used is FFFFg black powder stored in black powder canisters on the bulkheads of the avionics bay. Each bulkhead supports the ejection canisters for either the drogue or the main parachute and has both a primary and a redundant charge in separate canisters. The forward charges eject the main parachute, and the aft charges eject the drogue parachute. These charges have been sized based on the interior volume of the airframe on either side of the avionics bay. The redundant charges contain 1g more of black powder to ensure ejection occurs at the expected times in flight. Black powder was chosen as the ejection charge because it is relatively lightweight, quite reliable, has been used successfully in many past launches, and avoids the use of highly regulated high-pressure gas. The team avoids the use of pressurized gas ejection systems due to the regulatory complexity around their use. The ideal calculated amounts of black powder to use are 3g for the primary charge for the main parachute, 4g for the redundant charge for the main parachute, 2g for the primary charge for the drogue parachute, and 3g for the redundant charge for the drogue parachute.

3.3.1.6 Switches

Key switches are mounted on radially opposite sides of the coupler and allow for the activation of the avionics systems. Key switches were selected because they cannot be deactivated without a key, preventing accidental in-flight disarmament. The switches are located entirely within the coupler in switch holders to reduce their aerodynamic effect, keep them protected from external forces and debris, and prevent damage upon landing. These key switches replace previously used rocker switches, which had a high risk of in-flight disarmament.

Figure 3.68: Key Switch

3.3.1.7 Switch Holder

In order to secure the key switches to the interior of the coupler, a custom-designed 3D printed switch holder with a curved face that matches that of the inner diameter of the coupler is used for each one. Each has two screw holes to allow it to be attached and a recessed area to house the switch and prevent any accidental disarmament. Previously, the switch holder was epoxied to the coupler to secure the switch in place. In order to maintain a more accurate estimate of weight and to permit the exchange of switch holders, the epoxy was replaced with screws.

3.3.1.8 Altimeter Sled

The altimeter sled is a custom-designed 3D printed part that serves as a secure mount for the altimeters and their batteries within the avionics bay. It is designed to mount between the two threaded rods that run axially down the avionics bay and has built-in mounting posts for the altimeters and compartments on the opposite sides for their respective batteries. One primary goal with this CDR altimeter sled design was to take advantage of the specific strengths of additive manufacturing. This meant avoiding overhangs that could not be bridged and ensuring the weakest axis is not in the direction of any loads.

There were multiple iterations of the altimeter sled which involved different techniques to facilitate more efficient manufacturing. The final design includes integrated mounting posts (a change from the PDR altimeter sled design) which can be tapped to allow the altimeters to be secured to the sled with minimal additional hardware.

3.3.1.9 Battery Guard

The battery guard is an additional custom-designed 3D printed part that ensures the batteries are housed safely and securely throughout the flight. It has a similar profile as the altimeter sled and acts as a cap for the battery compartments. Throughout the design process, multiple iterations of the battery guard were evaluated to make loading the batteries into the vehicle more efficient and increase the security of the batteries once inserted. This included analyzing using Velcro straps post-PDR as a method to secure the batteries. However, due to the orientation of the 3D printed parts and the already lengthy process of disassembling the avionics bay, this alternative option to the battery guard was not chosen. Instead, the batteries are housed in the altimeter sled and secured on the end with the battery guard.

3.3.2 **Parachutes and Attachment Hardware**

3.3.2.1 Parachute Choices

The selected drogue parachute is a 24” diameter Fruity Chutes Classic Elliptical parachute. This parachute was chosen because it is especially compact and lightweight, and it has a relatively high drag coefficient for its size (1.55). It was also used last year with great success. The selected main parachute is a 144” diameter Rocketman High-Performance CD 2.2 parachute, which differs from the 120” SkyAngle CERT-3 XXL parachute used last year. The reason this change was made was that it was retroactively determined that the SkyAngle parachute was undersized for the vehicle last year, and the team is sizing the vehicle similarly this year. Therefore, a search was made for a larger main parachute that was not excessively expensive. Considering factors such as cost, diameter, and maximum vehicle weight, the search was narrowed down to the aforementioned Rocketman parachute. This parachute can support a vehicle with a maximum weight of around 54lbm, is also quite compact and lightweight, and has a strong listed drag coefficient (2.2). Both main and drogue parachutes are made of 1.1 oz ripstop nylon.

With the current designed vehicle weight, the chosen 144" diameter Rocketman High-Performance parachute was verified with the Simulink simulation to balance the maximum landing kinetic energy and maximum descent time requirements. Also, deploying the main parachute specifically at 900' AGL balances the need for the payload system to have enough time to separate from the vehicle and the requirement that the descent time is under 90 seconds.

Figure 3.69: Drogue Parachute

Figure 3.70: Main Parachute

3.3.2.2 Attachment Hardware and Heat Shielding

The drogue parachute will be attached to a 30' long, 3/8" wide tubular Kevlar shock cord, while the main parachute will be attached to a 60' long, 3/8" wide tubular Kevlar shock cord. The shock cords will be attached to the parachutes and ¼" stainless steel bulkhead eyebolts via ¼" stainless steel quick links. To protect the parachutes from hot ejection charge gases, an 18" to a side square Nomex blanket will wrap around each parachute while they are packed inside the airframe sections.

3.3.3 **Electrical System and Schematics**

3.3.3.1 Electrical Components and Redundancy

Full redundancy is achieved in the avionics electrical components by utilizing completely independent circuits for the primary and redundant deployment systems. These circuits include wiring from each altimeter to its dedicated switch, battery, terminal block and black powder canister for drogue parachute deployment, and terminal block and black powder canister for main parachute deployment. Two altimeters of different makes/models are used in order to increase the likelihood that if a failure occurs in the primary system, the same one will not also occur in the redundant system, resulting in catastrophic failure. Additionally, the redundant main charge is programmed to go off 200' lower than the primary main ejection charge, the redundant drogue ejection charge is programmed to go off 1s after the primary drogue ejection charge, and the redundant ejection charges for both parachutes contain 1g more of black powder than the primary ejection charges. These factors all work together to contribute to overall reliability and success in the processes of separation and deployment as prescribed in the mission requirements.

3.3.3.2 Wiring Diagram (Schematic)

The wiring diagram below displays the avionics electrical components in a visual layout. The TeleMetrum altimeter is powered by a 3.7V LiPo battery and an external key switch and sends signals to the primary main and drogue parachute ejection charges. The

StratoLoggerCF altimeter is powered by a 9V alkaline battery and another external key switch and sends signals to the redundant main and drogue parachute ejection charges.

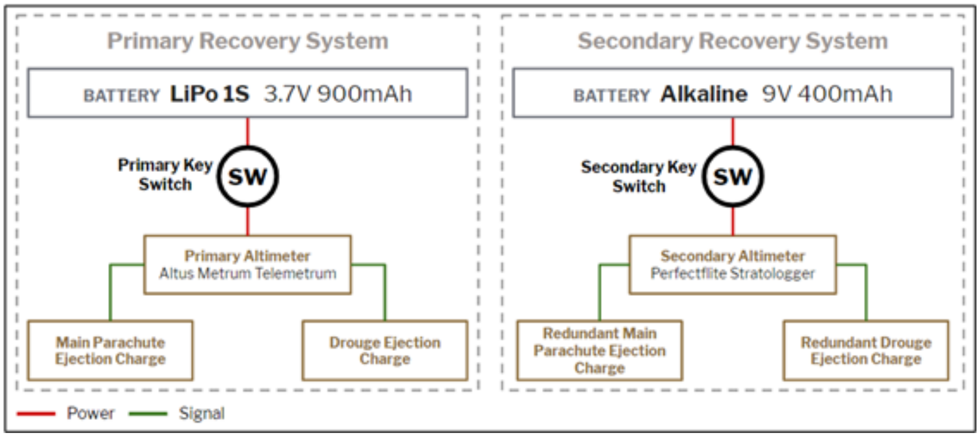


Figure 3.71: Avionics Wiring Diagram

3.3.4 **CAD and Dimensional Drawings**

3.3.4.1 Avionics Bay Assembly and Sub-Assemblies (CAD)

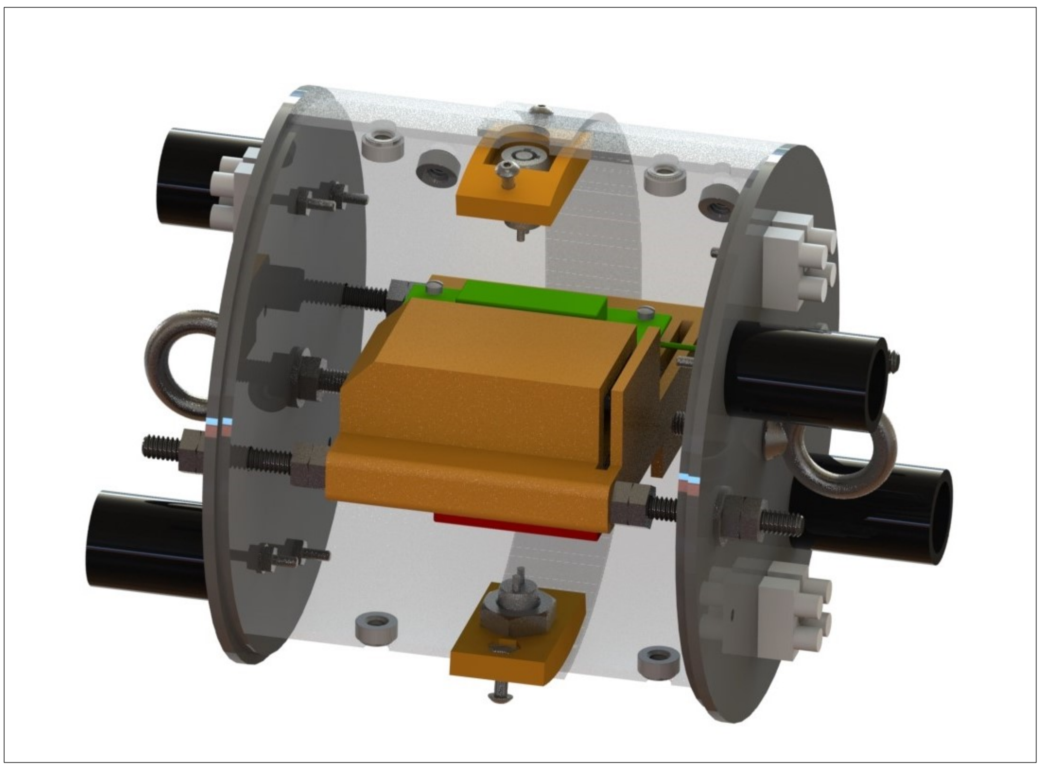


Figure 3.72: Section View of Avionics Bay Assembly

Figure 3.73: Avionics Coupler Sub-Assembly

Figure 3.74: Avionics Bulkhead Sub-Assembly

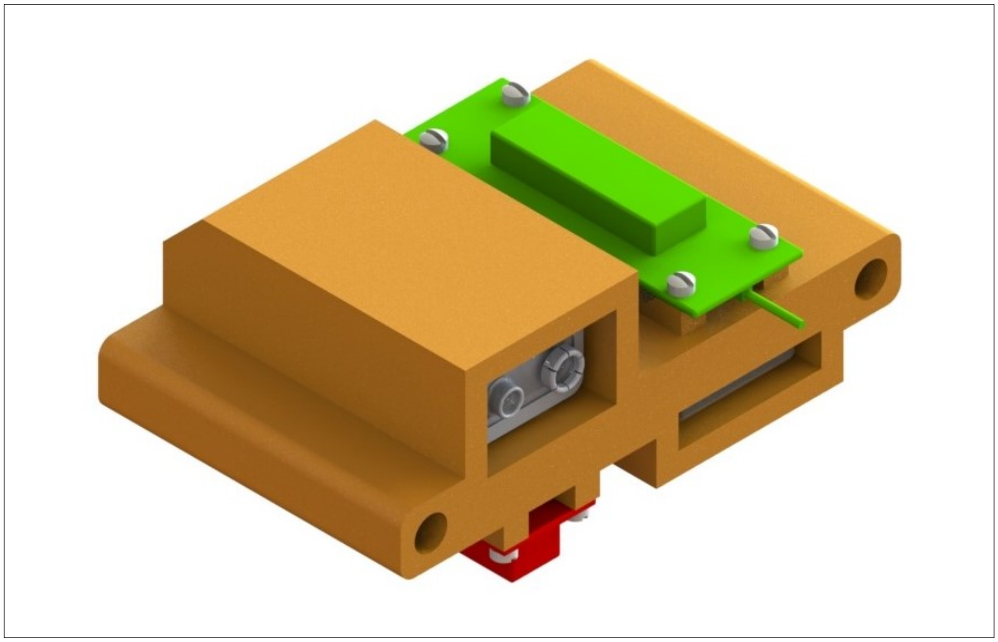


Figure 3.75: Altimeter Sled Sub-Assembly

3.3.4.2 Custom-Designed and 3D Printed Parts (Dimensional Drawings)

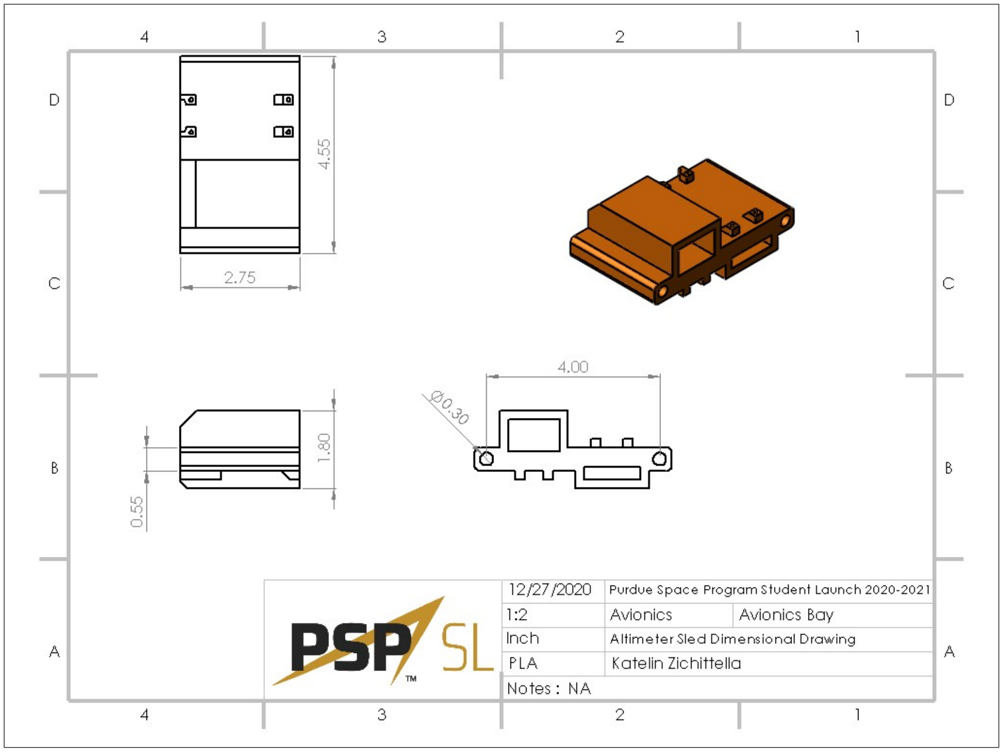


Figure 3.76: Altimeter Sled Dimensional Drawing

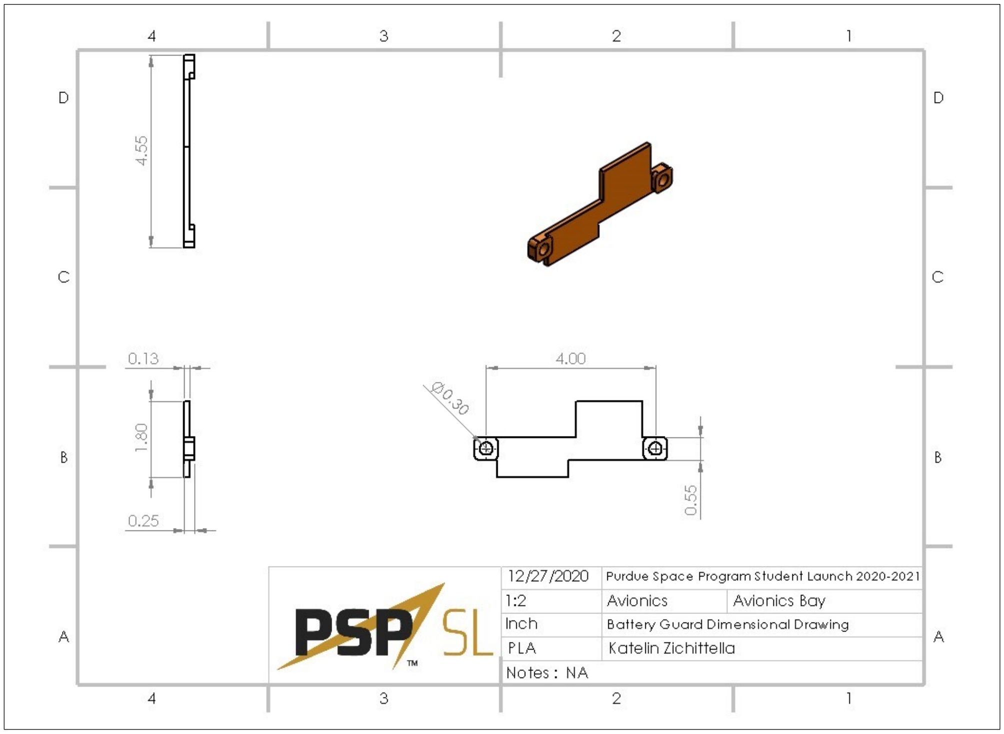


Figure 3.77: Battery Guard Dimensional Drawing

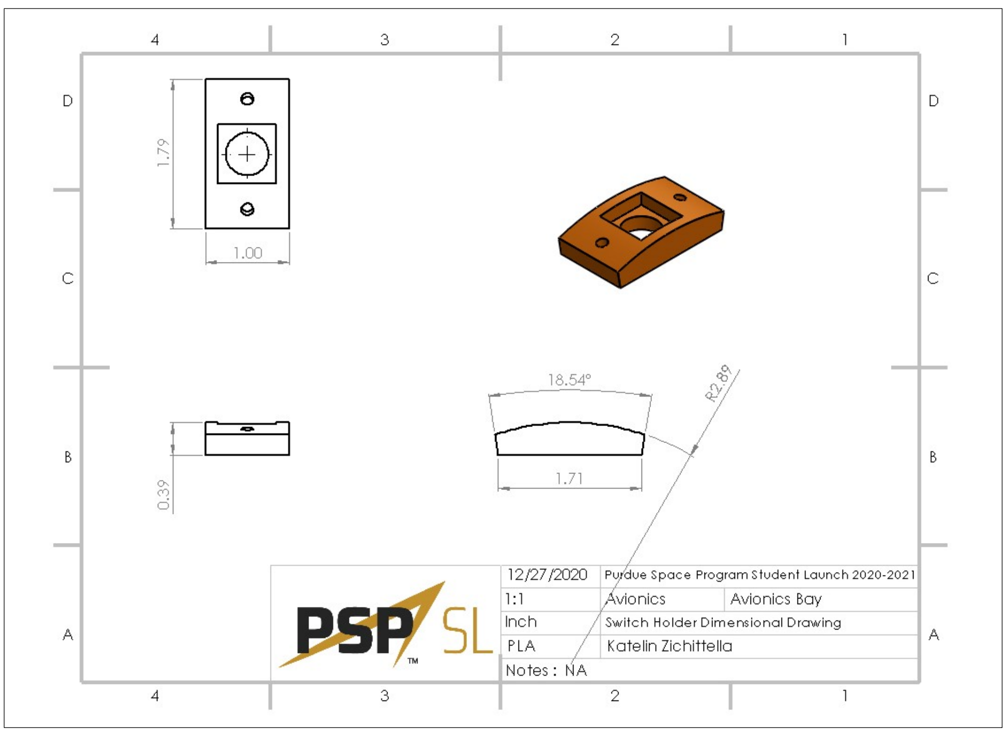


Figure 2.78: Switch Holder Dimensional Drawing

3.3.5 Ejection Charge Sizing and Airframe Pressurization

Both parachutes are deployed via black powder charges initiated by redundant altimeters. The primary drogue charge ignites at apogee with the redundant at apogee plus one second, and the primary main charge ignites at 900' AGL with the redundant at 700' AGL. The primary main charge contains 3g of FFFFg black powder, and the redundant main charge contains 4g of black powder (3g + 1g) in order to absolutely ensure complete separation. Similarly, the primary drogue charge contains 2g of black powder and the redundant drogue charge contains 3g of black powder.

By calculating the cross-sectional area of a single 4-40 shear pin and multiplying it by the shear strength of nylon, it is possible to calculate the force necessary to shear a single shear pin.

$$Area_{Pin} = \pi R^2$$

$$Area_{Pin} = 3.1415 \cdot (0.056in)^2 = 0.009852in^2$$

$$Force_{Pin,Failure} = Area_{Pin} \cdot \tau_{Nylon}$$

$$Force_{Pin,Failure} = 0.009852in^2 \cdot 10000psi = 98.52lbf$$

From there, one can determine how much force is required to shear four pins and use that to calculate how much pressure is necessary on a 6” diameter bulkhead to sufficiently shear all four pins.

$$4 \cdot Force_{Pin,Failure} = 394.1lbf$$

$$Area_{Bulkhead} = \pi R^2$$

$$Area_{Bulkhead} = 3.1415 \cdot (3in)^2 = 28.27in^2$$

$$P_{Bulkhead} = \frac{4 \cdot Force_{Pin,Failure}}{Area_{Bulkhead}} = \frac{394.1lbf}{28.27in^2} = 13.94psi$$

By using the equation below (where 0.006 is the pressure coefficient corresponding to a desired pressure on the bulkhead of 13.94psi, D is the diameter of the airframe, L is the length of the airframe section, and G is the mass of black powder in each canister in grams), the amount of black powder needed to sufficiently shear all the nylon shear pins can be calculated. The final value is multiplied by 1.2 and always rounded up as a safety factor.

$$G = Mass_{BP} = C_P \cdot D^2 \cdot L \cdot 1.2$$

Upper Recovery Section Side (Main) Primary Ejection Charge

$$G = Mass_{BP} = 0.006 \cdot (6in)^2 \cdot 9.25 \cdot 1.2 \approx 3 \ grams \ of \ black \ powder$$

Upper Recovery Section Side (Main) Redundant Ejection Charge

$$G = 3g + 1g = 4 \ grams \ of \ black \ powder$$

Lower Recovery Section Side (Drogue) Primary Ejection Charge

$$G = Mass_{BP} = 0.006 \cdot (6in)^2 \cdot 6.25 \cdot 1.2 \approx 2 \ grams \ of \ black \ powder$$

Lower Recovery Section Side (Drogue) Redundant Ejection Charge

$$G = 2g + 1g = 3 \ grams \ of \ black \ powder$$

3.3.6 Tracking Devices

The primary tracking device of the launch vehicle is the TeleMetrum altimeter, which contains a 70 cm ham-band transceiver for telemetry downlink as well as an on-board, integrated GPS receiver. The output power of the RF transceiver is 40 mW, and the specific frequency used by the team will be 434.55 MHz. From past experience, it is known that the transmitter on the TeleMetrum has a range of at least one mile and is very reliable in establishing and maintaining a connection to the ground station during flight. This connection (to a standard laptop) is made using a TeleDongle and Yagi Arrow 3 Element antenna.

All major vehicle sections (tethered or otherwise) are equipped with active GPS tracker/transmitters. These provide constant position information for the entire vehicle during flight, easing recovery in the event of an accident. In the previous year’s project, the team temporarily lost a section of the launch vehicle due to lower-than-expected cloud cover and shock cord failure. While the section was recovered a month later, the team has decided that to avoid any risk of section loss, COTS GPS tracking modules will be added to each independent vehicle section (in this case, this meant adding trackers to the payload and booster sections). The team has selected the EggTimer Rocketry EggFinder system, as it provides long-range tracking, is low in weight, and is low in power consumption. The team has created a 3D printed housing that contains the GPS module, battery, and a key switch. The selected battery is expected to provide enough power for more than 4 hours of tracking, enough for 2 hours of pad time and 2 hours of vehicle location time. There will be two of these modules in the final vehicle, one in each of the breakpoint couplers, where they will not interfere with the other vehicle systems. One of the assembled tracking modules can be seen below in figure 3.79.

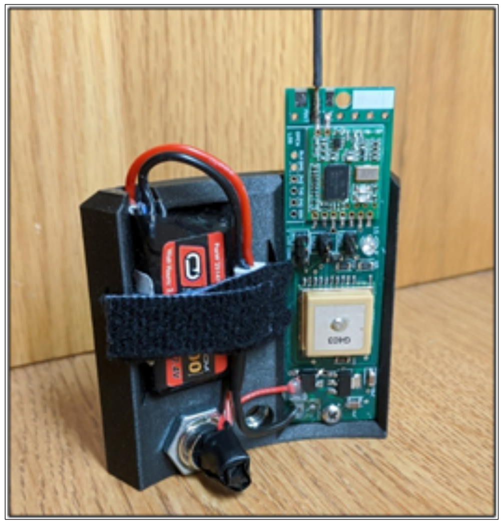


Figure 3.79: Complete GPS Tracking Module

3.4 Mission Performance Predictions

3.4.1 Trajectory Analysis

The first tool the team used to analyze launch vehicle trajectory was OpenRocket. From the competition last year, the team placed emphasis on analyzing trajectory at varied windspeeds and launch rail inclinations. The team conducted simulations at windspeeds beginning at 0 mph up to 15 mph in increments of 5 mph, and rail inclinations (in degrees from vertical) beginning at 0 degrees up to 15 degrees in increments of 5 degrees. The team determined this to be an ample amount of data that would cover a wide base of potential launch conditions from which the best- and worst-case scenarios could be chosen on the criteria of apogee. A most realistic scenario was selected by comparing the average windspeed in Huntsville, AL during April and launch day rail guidelines to those simulated. Huntsville has an average windspeed ranging from 6-8 mph due south in April. The most realistic speed was then chosen as 10 mph in order to allow a margin of error, as wind is likely to increase as the vehicle climbs. Windspeeds above 15 mph are unheard of in April non-inclement weather. Wind speeds of this magnitude would also warrant launch day to be pushed back. For these two reasons, the team chose 15 mph as the upper windspeed bound. Per NASA SL guidelines, the launch rail is expected to be canted 5-10 degrees away from crowds depending on wind conditions. To again allow for a margin of error, the most realistic angle was chosen as 10 degrees. In all simulations, there was no consideration given to atmospheric temperature, pressure, or humidity as these factors were found to greatly vary at this time of year in Huntsville and have miniscule impact on simulation results.

3.4.1.1 OpenRocket Simulations

To generate simulation data, the most up-to-date full-scale launch vehicle body was input to OpenRocket along with important launch location characteristics such as latitude of 34.6°N, longitude of -86.7°E, altitude of 600 ft, and launch rail length of 144”. Two locations were used for all simulations: Huntsville, Alabama and Purdue Dairy, Indiana.

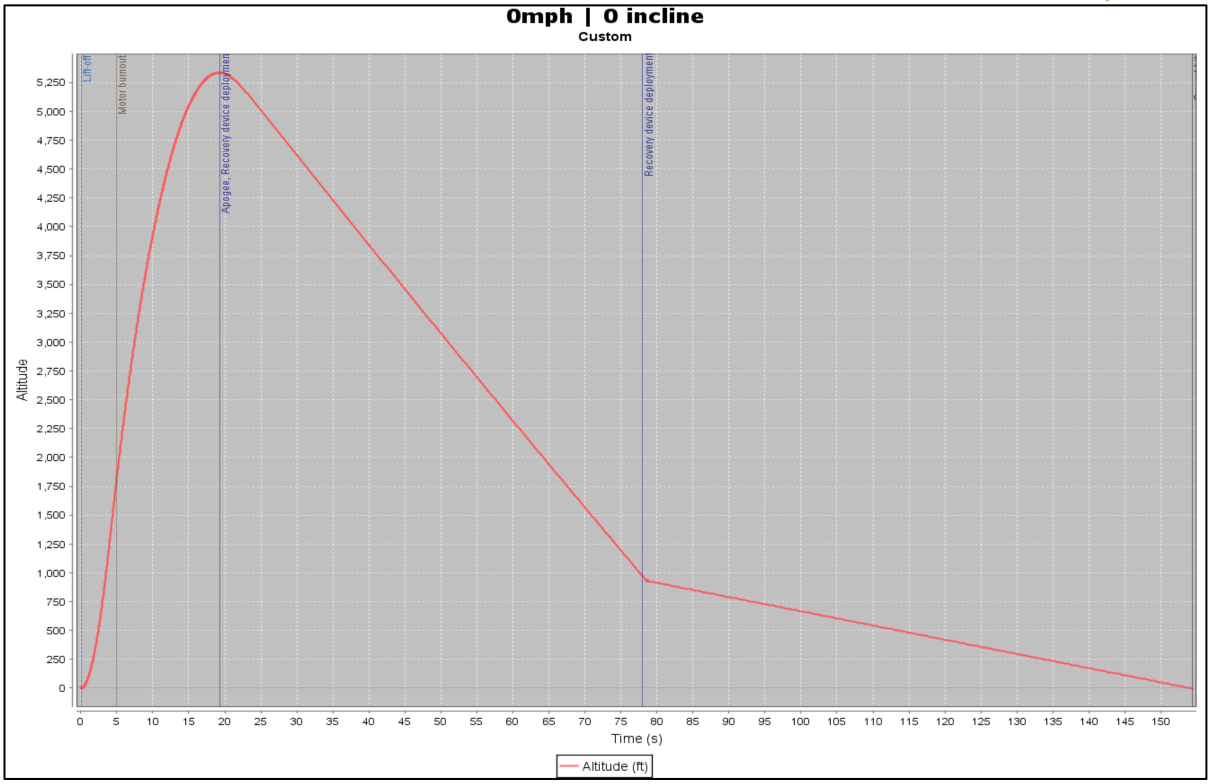


Figure 3.80: OpenRocket Altitude vs. Time of Ideal Launch Conditions

From the plot above, the vehicle is anticipated to reach a maximum altitude of 5335.7’ AGL in ideal launch conditions of no wind and a directly vertical launch. This apogee is far above the target altitude of 4100’. The team has prepared for potential overshoot of the target altitude scenarios through the airbrake and adjustable nose cone ballast systems, both of which can be used to slow the vehicle to approach the desired apogee. This simulation provided the maximum potential altitude achievable by the current design of the launch vehicle and is the best-case scenario. As mentioned, this scenario is unlikely to occur as the launch rail will be canted by some degree of tilt based on realistic non-deal wind conditions.

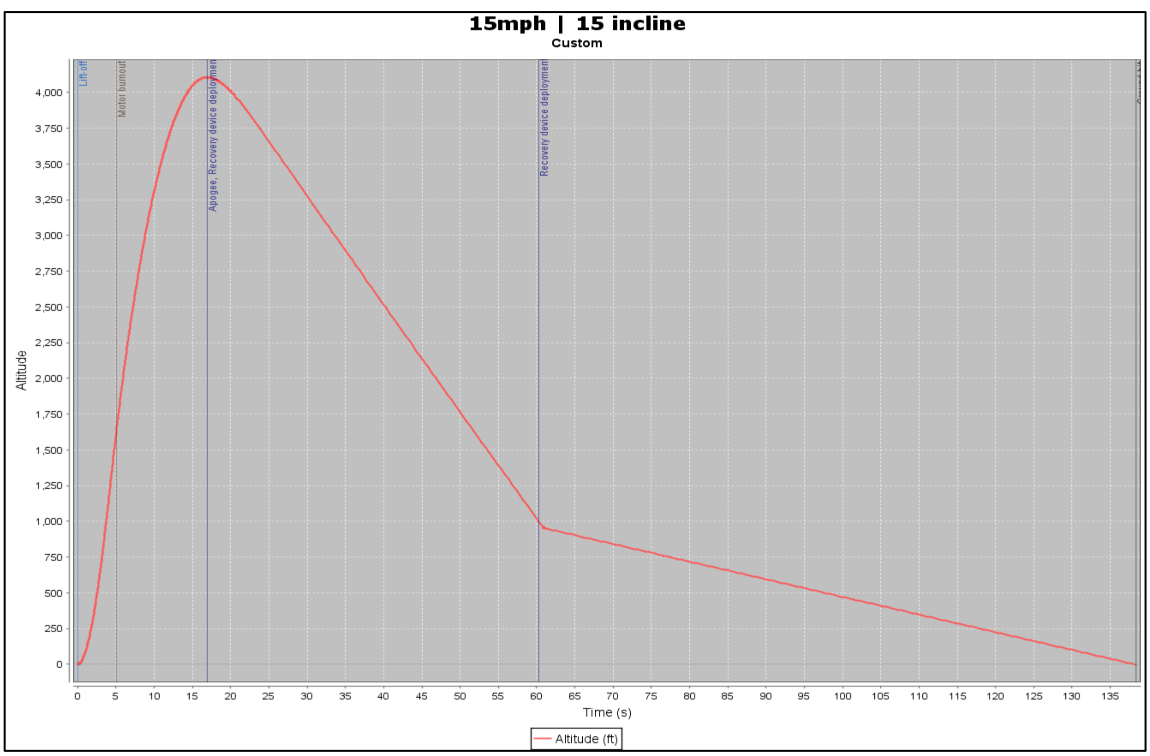


Figure 3.81: OpenRocket Altitude vs. Time of Worst-Case Launch Conditions

The team decided that the upper bound of 15 mph wind and 15° inclination is the largest possible magnitude either parameter could experience on launch day. These values represent the worst-case scenario launch conditions for the vehicle, achieving an apogee of 4105.8' AGL according to OpenRocket. This altitude is precisely the team's target apogee, however without the use of airbrakes or the adjustable nose cone ballast. Similar to the ideal scenario, these conditions are unlikely to occur on launch day. Average wind conditions at Huntsville do not exceed 12 mph, and the SL handbook cites rail angles ranging from 5-10 degrees, no higher. The best and worst-case scenario conditions provide a reasonable basis to measure the range of altitude operations the current launch vehicle is capable. This provides the team with a general sense of what to expect on launch day.

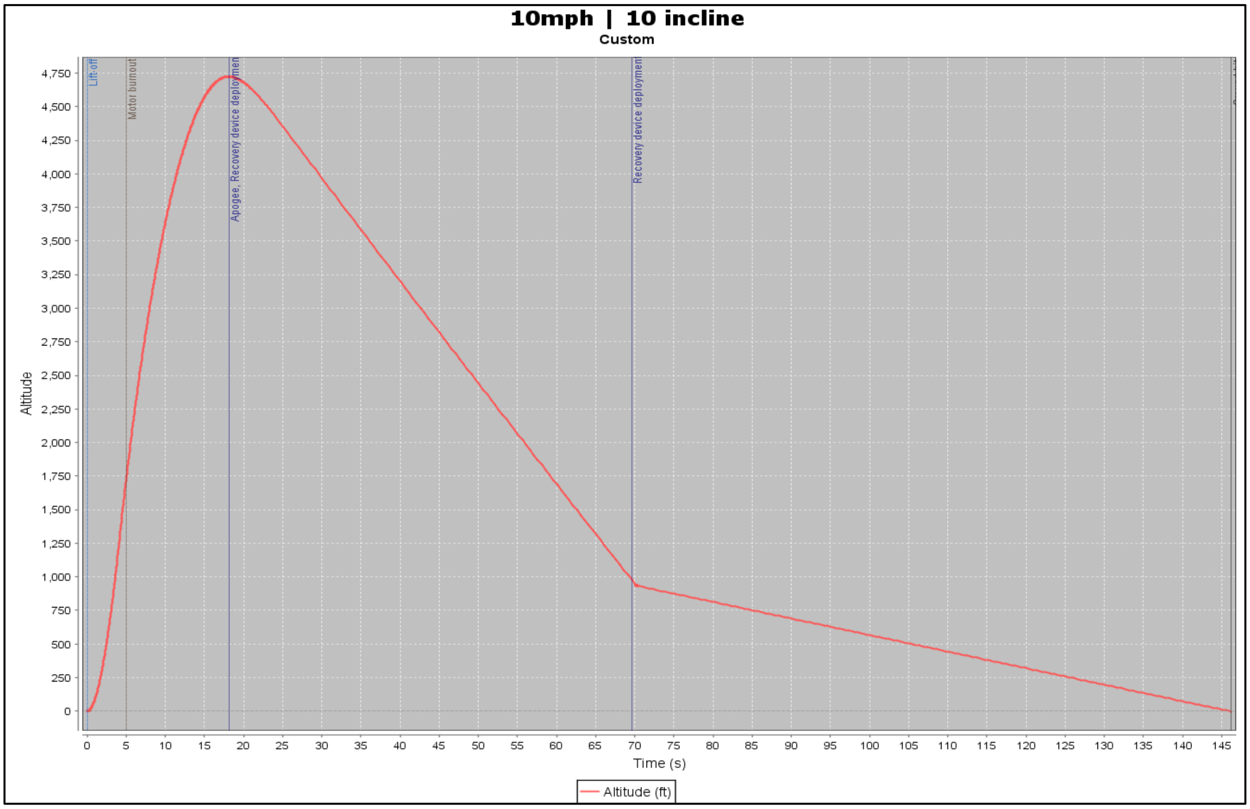


Figure 3.82: OpenRocket Altitude vs. Time of Realistic Launch Conditions

The third simulation, shown above, is the most realistic launch day conditions expected by the team. The vehicle reaches an expected apogee of 4724.9', higher than the target altitude of 4100'. As vehicle construction begins, the vehicle weight is expected to increase which in turn would lower the predicted apogee of this simulation closer to the target altitude. The current OpenRocket model does not account for all future mass of miscellaneous parts such as nuts and screws as well as final masses for subsystems such as the airbrakes which will also lower apogee. These would all increase future weight of the vehicle. In prior competitions, the team knows that epoxy used to attach components adds weight that cannot be accounted for in this stage of the process. Additionally, with the verification of a round nose cone from the subscale launch, there is extra room in the nose cone for ballast weight or camera/avionics systems where weight can be added, if necessary, to lower the altitude.

Angle (deg)	**Windspeed (mph)**			
	0	5	10	15
0	5335.7	5310.6	5237.1	5121.3
5	5270.2	5170.7	5031.3	4859.8
10	5076.6	4915	4724.9	4513.5
15	4771.5	4564.4	4340.4	4105.8

Table 3.6: All OpenRocket Apogee Simulation Potential Scenarios by Windspeed and Inclination Angle

All simulations in this section were complete using OpenRocket 15.03. The simulations used extended Barrowman calculation method and six degrees of freedom Runge-Kutta 4 panel method. Altitude calculations were completed in a 0.5 second time step and the shape of Earth was approximated as spherical.

3.4.1.2 RAS Aero II Verification

The team also performed simulations in RAS Aero II, something that was not done during the preliminary design review, to approximate the final apogee value of the launch vehicle. The team ran the simulations in RAS Aero II to confirm that the values predicted in OpenRocket are accurate. The differences between RAS Aero II and OpenRocket are minimal. However, there are a few cases with minor differences, though the team decided that these differences are not significant enough to be considered when predicting final launch apogees of the launch vehicle. The simulations were run using the most recent information of the launch vehicle like mass and size. The values received from the simulations are displayed below in table 3.7.

Angle (deg)	**Windspeed (mph)**			
	0	5	10	15
0	5411	5388	5320	5197
5	5301	5240	5102	4905
10	4989	4959	4764	4521
15	4525	4571	4338	4085

Table 3.7: Apogee values for each respected launch angle were calculated using RAS Aero II simulations

3.4.1.3 Simulink

The team has also developed a two degrees-of-freedom custom vehicle trajectory simulation in Simulink format to understand how different possible vehicle configurations affect flight and to inform the selection of the new main parachute. Developing a custom simulation parallel to OpenRocket and RAS Aero II offers a greater range of control of different parameters to achieve as much accuracy as possible, acts as verification of the OpenRocket and RAS Aero II simulations, and increases the team's knowledge of and experience with flight dynamics.

The simulation includes a multitude of useful features. Various vehicle and vehicle component characteristics such as mass, size, drag coefficient, and motor thrust, as well as environmental characteristics such as launch rail angle and wind speed, can be input and modified via a MATLAB script. The Simulink model itself utilizes these parameters and established motion equations to simulate the powered ascent, coast, descent under the drogue parachute, and descent under the main parachute phases of flight. Altitude, drift distance, vertical velocity, and horizontal velocity over the flight time are then returned to MATLAB to be plotted and analyzed.

The four critical requirements that this simulation verifies are descent time, drift distance, rail exit velocity, and landing kinetic energy of the heaviest section of the vehicle. These values are calculated from the simulation results and compared to the numerical requirements. Pass/fail results are returned to the user to provide a very quick and simple verification.

For all the Simulink sections, the simulation was run with a launch rail angle inclined at 10° from vertical, the horizontal wind speed set at 10mph, and no additional mass. The vehicle was launched into the wind.

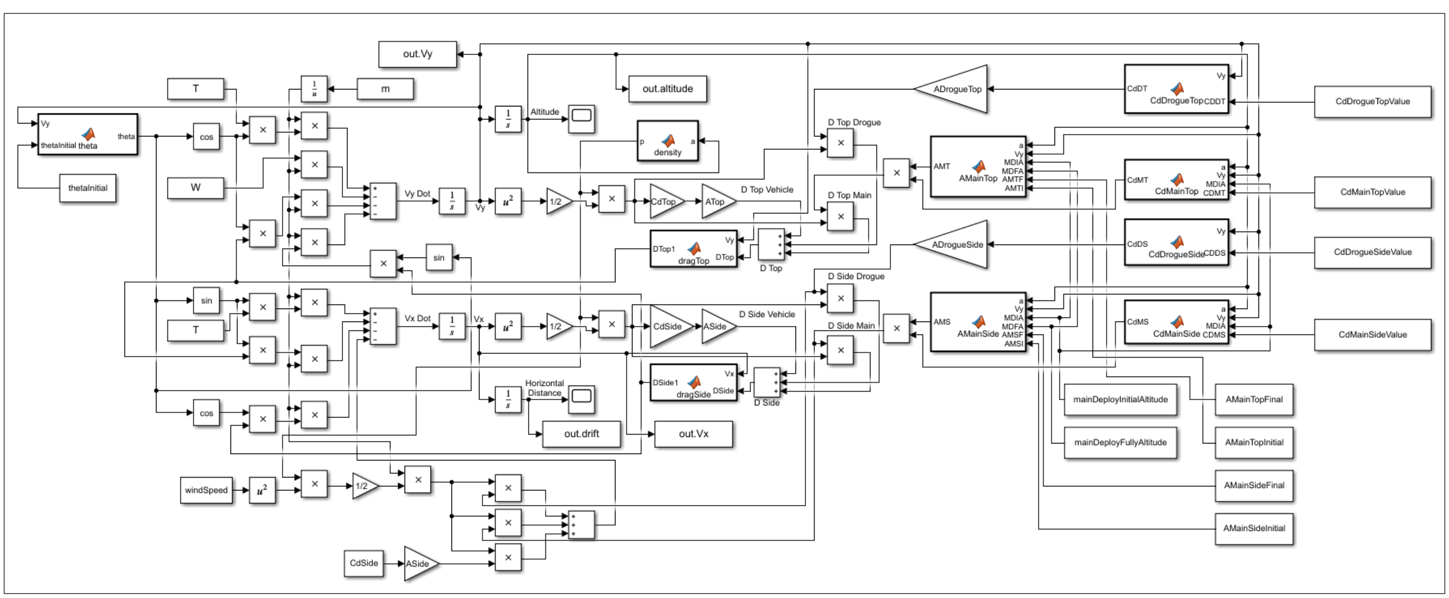


Figure 3.83: Simulink Model

The figure above shows the visual Simulink model.

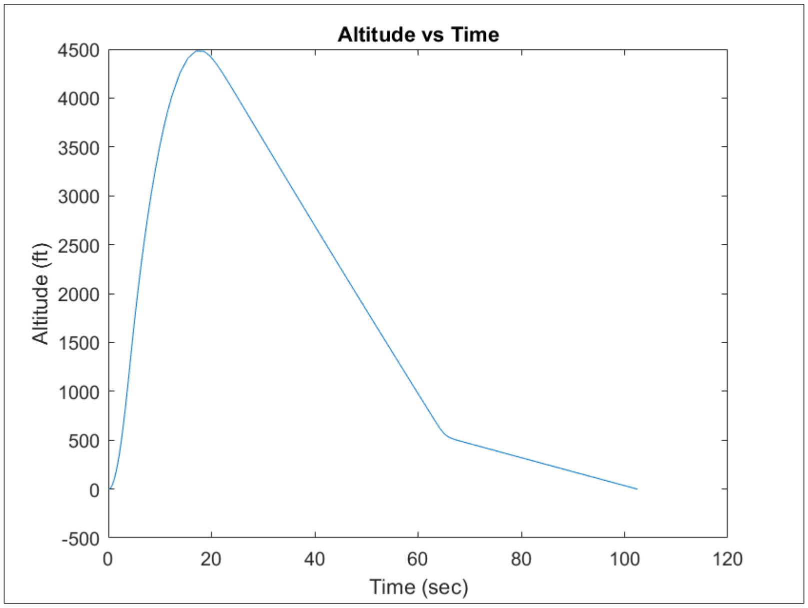


Figure 3.84: Simulink Altitude vs. Time Plot

The figure above shows the altitude vs. time plot generated from the simulation results. Most notable is the 85.5s descent time from apogee to landing, which is below the maximum requirement of 90s. One interesting thing to note is that a few of the parachute parameters were adjusted based on empirical observations from last year. The coefficient of drag of the drogue parachute was increased from the supplier reported value, the coefficient of drag of the main parachute was decreased from the supplier reported value, the main parachute deploys slightly lower than the set altitude, and the main parachute opens a little less than fully.

Parameter	**Value**	**Pass/Fail**
Apogee	4482'	N/A
Ascent Time	17.0s	N/A
Drogue Descent Velocity	87.7ft/s	N/A
Landing Velocity	14.3ft/s	N/A
Descent Time	85.5s	Pass
Drift Distance	521'	Pass
Rail Exit Velocity	62.2ft/s	Pass
Landing Kinetic Energy of the Heaviest Section	74.8ft-lbf	Pass

Table 2.8: Simulink Important Returned Parameter Values

In the table above are some important returned parameter values from the Simulink simulation. These were generated under ideal launch conditions to provide a quick and high-level evaluation of the chosen design parameters. The most significant values to note are the four critical requirements, which are descent time, drift distance, rail exit velocity, and landing kinetic energy of the heaviest section. These are highlighted in gold in the table above. As can be seen, with the current vehicle design all four critical requirements are passed.

3.4.2 **Vehicle Characteristics**

3.4.2.1 Stability Versus Time

3.4.2.1.1 OpenRocket

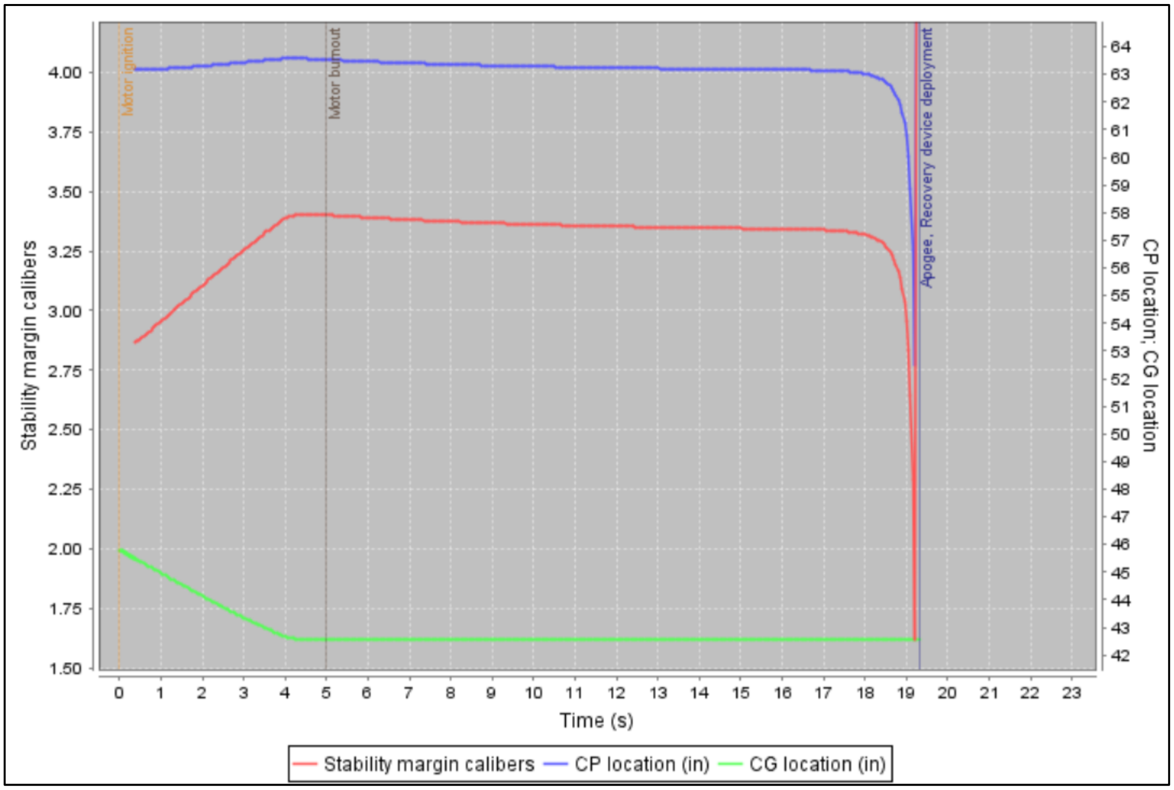


Figure 3.85: OpenRocket Stability vs Time Simulation of Ideal Case (0 Deg Incline, 0mph in Huntsville, Alabama)

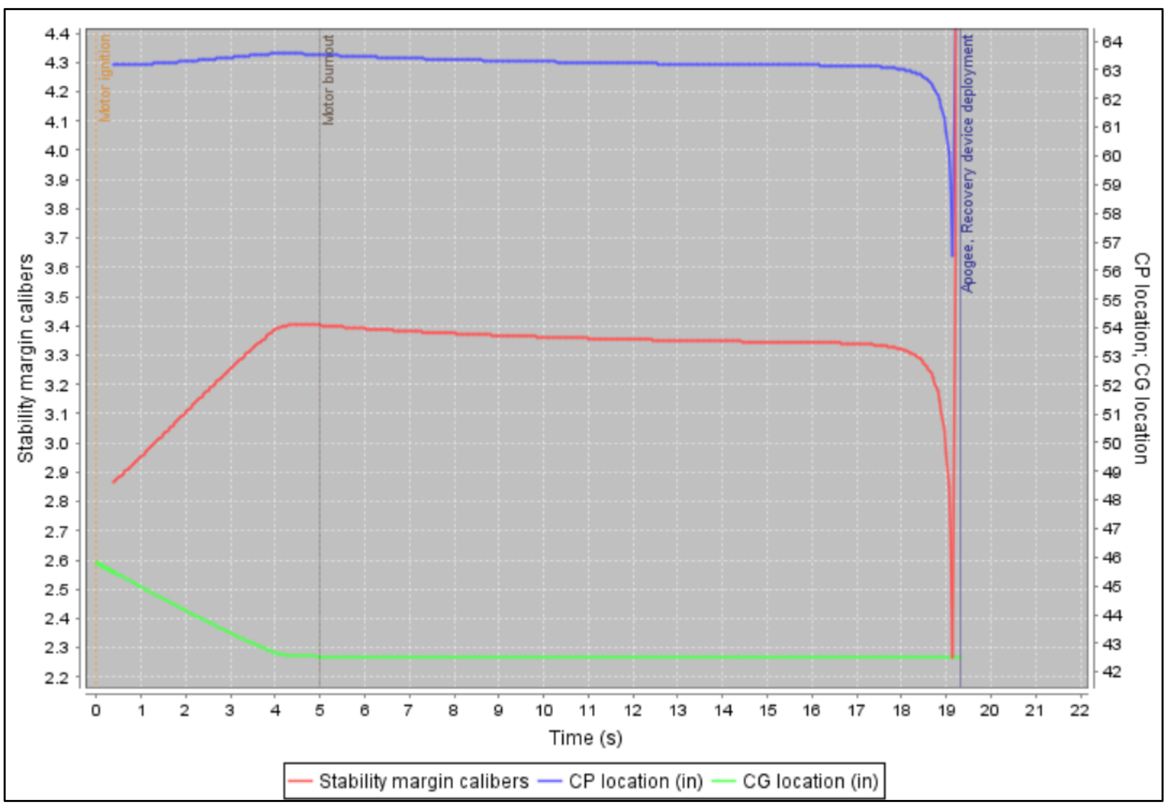


Figure 3.86: OpenRocket Stability vs Time Simulation of Ideal Case (0 Deg Incline, 0 mph in Purdue Dairy Farm, Indiana)

After running simulations on OpenRocket, it can be seen from figure 3.85 above that the launch vehicle leaves the 144” launch rail at Huntsville, AL with a stability margin of 2.867 cal. This meets the minimum requirement of two calibers. Likewise, as shown above in figure 3.86, the launch vehicle leaves the launch rail at Purdue Dairy, IN with a stability margin of 2.867 cal as well. In fact, the difference between the simulations conducted for the two locations is minimal. While in the ascent phase, the launch vehicle does not experience a noticeable stability drop. However, once the launch vehicle starts to slow down due to gravity and drag, the low velocity prevents the fins from maintaining aerodynamic stability. This is when the launch vehicle starts to arc as it approaches the launch apogee. Here, the OpenRocket simulated stability starts to fall significantly. Regardless of this, the launch vehicle comfortably maintains a stability margin over 3.0 cal for almost the entirety of the boost and coast phase.

The center of pressure (CP) is the average location of a pressure field acting on a body. In other words, it is the point where the total sum of all pressures acts on the vehicle. At launch, the CP of the launch vehicle at both locations is 63.168” from the datum, which is considered the tip of the nose cone. Likewise, the center of gravity (CG) is the average location of the weight of a body. In other words, it is the point where the weight of the body may be considered to act. At launch, the CG of the launch vehicle at both locations is 45.805” from the datum, which places it 17.363” above the initial center of pressure. As the motor burns out, the CG gradually moves higher at a constant rate to about 42.543” from the datum at Huntsville, AL and 42.542” at Purdue Dairy, IN. This happens due to the linear solid propellant burn rate. The launch vehicle here experiences a total shift of around 3.262" at both locations.

3.4.2.1.2 RAS Aero II

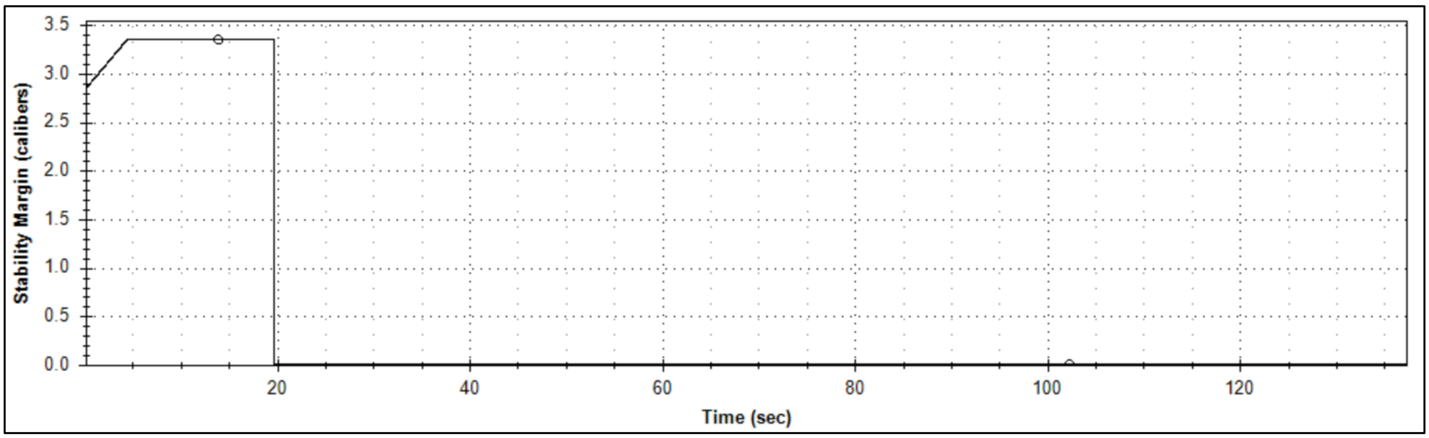


Figure 3.87: RAS Aero II Stability vs Time Simulation of Ideal Case (0 Deg Incline, 0mph in Huntsville, Alabama)

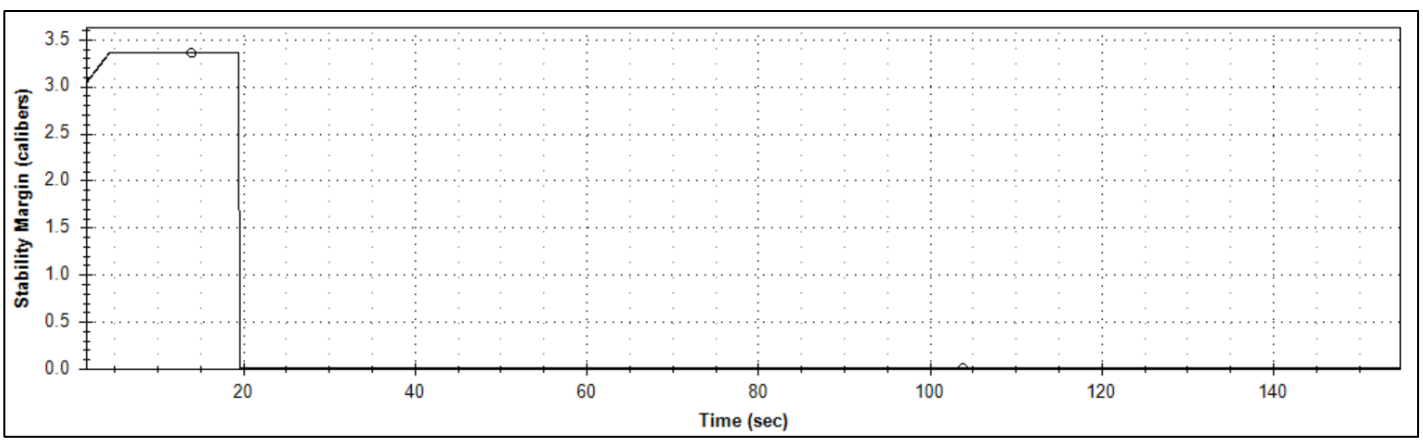


Figure 3.88: RAS Aero II Stability vs Time Simulation of Ideal Case (0 Deg Incline, 0mph Purdue Dairy Farm, Indiana)

The RAS Aero II simulations of the ideal case shows that the launch vehicle is never below the required 2.0 cal of stability required off the launch rail. According to RAS Aero II, the vehicle starts at a stability of around 2.83 cal at both locations. This implies a slight differential between its results and the results from the OpenRocket simulation. Nevertheless, both simulations show that the stability margin requirement off the launch rail is met.

Over the course of the flight, the stability margin eventually increases to a value of about 3.36 cal at both locations. To achieve this simulation, RAS Aero II needed an input value for the center of gravity and the team took this value from the OpenRocket model of the launch vehicle – introducing some error. However, the CP was calculated within RAS Aero II to ensure accuracy in its own means.

RAS Aero II calculated the CP at launch to be 63.25” from the tip of the nose cone while the CG was inputted to be 45.81” (45.805 rounded up) from the tip of the nose cone. According to RAS Aero II, this places the CG 17.44” above the CP. Here, a slight differential between the two software be seen again. While RAS Aero II calculates the CP at launch to be 63.25” away from the nose cone, OpenRocket calculates it to be 63.168” away. This slight difference is a good example to explain the differential in the starting stability mentioned earlier. However, the initial difference between the CP and the CG at launch is relatively uniform for both software – RAS Aero II calculates the CP and CG of the launch vehicle to be 17.44” apart from each other at launch, while OpenRocket calculates them to be 17.363” apart.

3.4.2.2 Drag versus Time

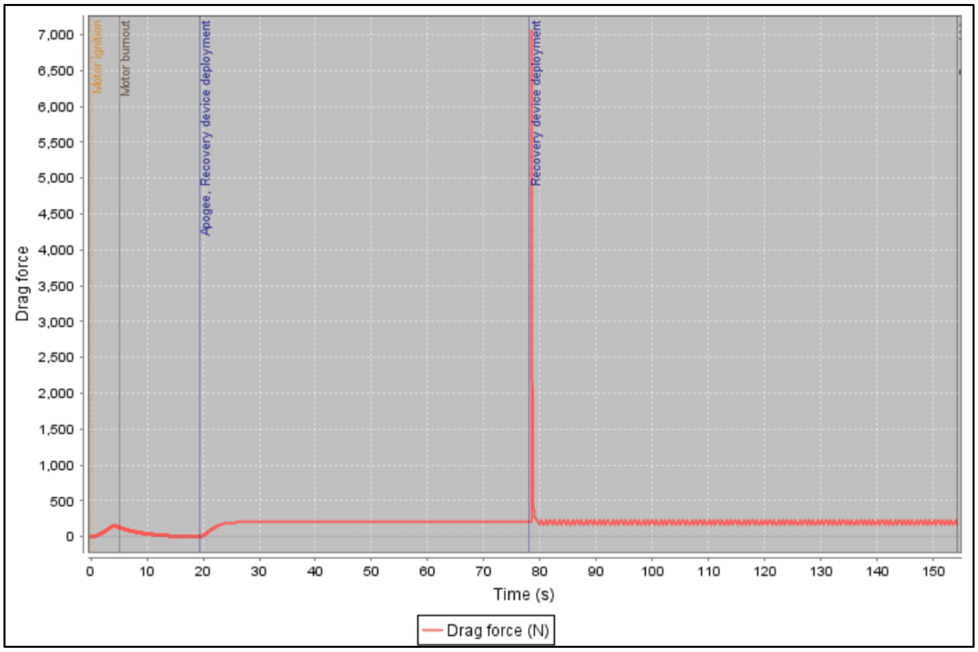


Figure 3.89: OpenRocket Drag Vs Time Simulation of The Ideal Case (0 Deg Incline, 0mph Huntsville, Alabama)

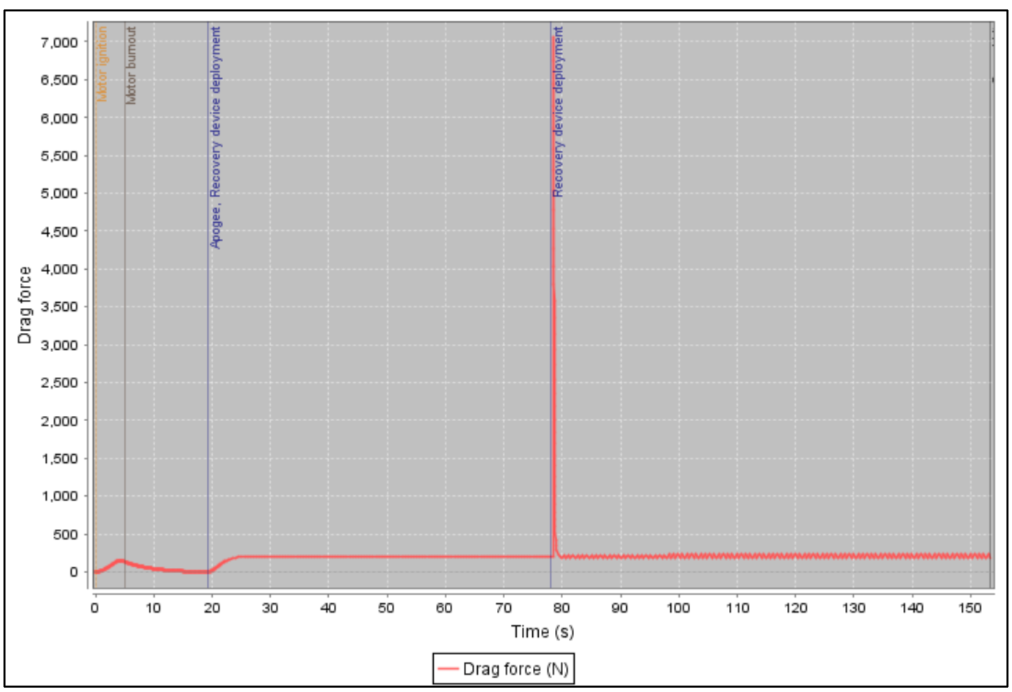


Figure 3.90: OpenRocket Drag Vs Time Simulation of The Ideal Case (0 Deg Incline, 0mph Purdue Dairy Farm, Indiana)

As can be seen above, for the ideal case of 0 mph wind speed and 0-degree incline, the drag force stays at around 201.5 N at Huntsville, AL and 201.6 N at Purdue Dairy, IN during most of the flight time. According to the OpenRocket simulations, the case is similar for different wind speeds and inclinations because the wind speed of the launch vehicle is considerably higher than the max possible wind speed. Therefore, according to the simulations, different wind speeds and inclinations have little impact on the drag force. Additionally, the drag spikes seen once the parachute is deployed are expected given the purpose of a parachute.

3.4.2.3 Drift Distance Estimations & Hand Calculations

To look at drift distance, the team used OpenRocket to simulate the drift distance at different wind speeds with a 0-degree incline and a potentially more representative 10-degree incline, 10 mph wind speed condition. Furthermore, the team used the equation drift distance equals wind speed multiplied by the descent time, assuming the wind blows in only one direction during descent.

3.4.2.3.1 OpenRocket Drift Calculation

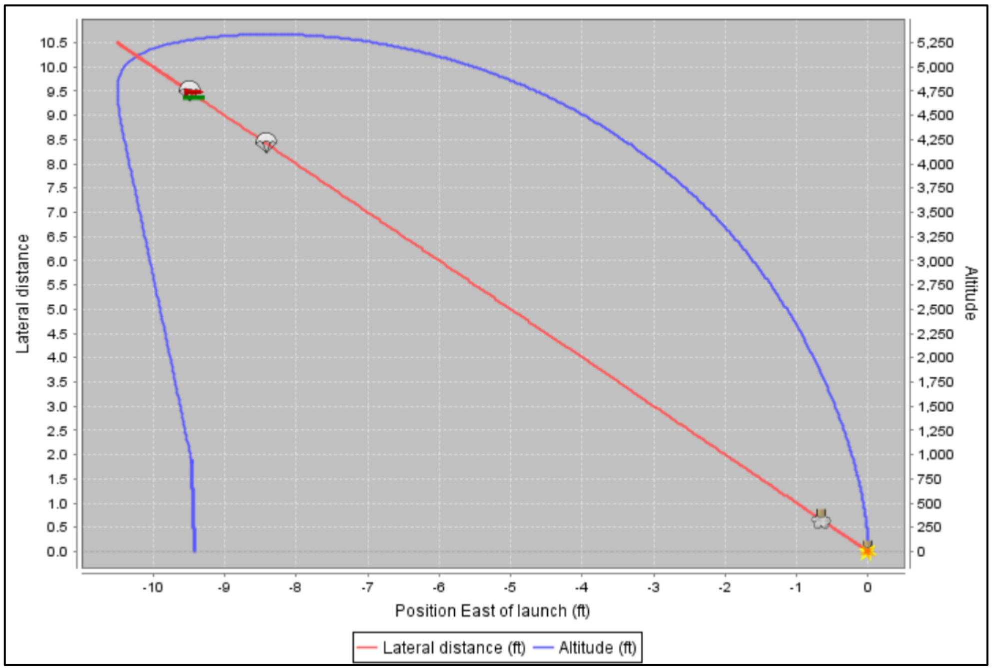


Figure 3.91: OpenRocket Drift Distance Simulation of The Ideal Case (0 Deg Incline, 0mph Huntsville, Alabama)

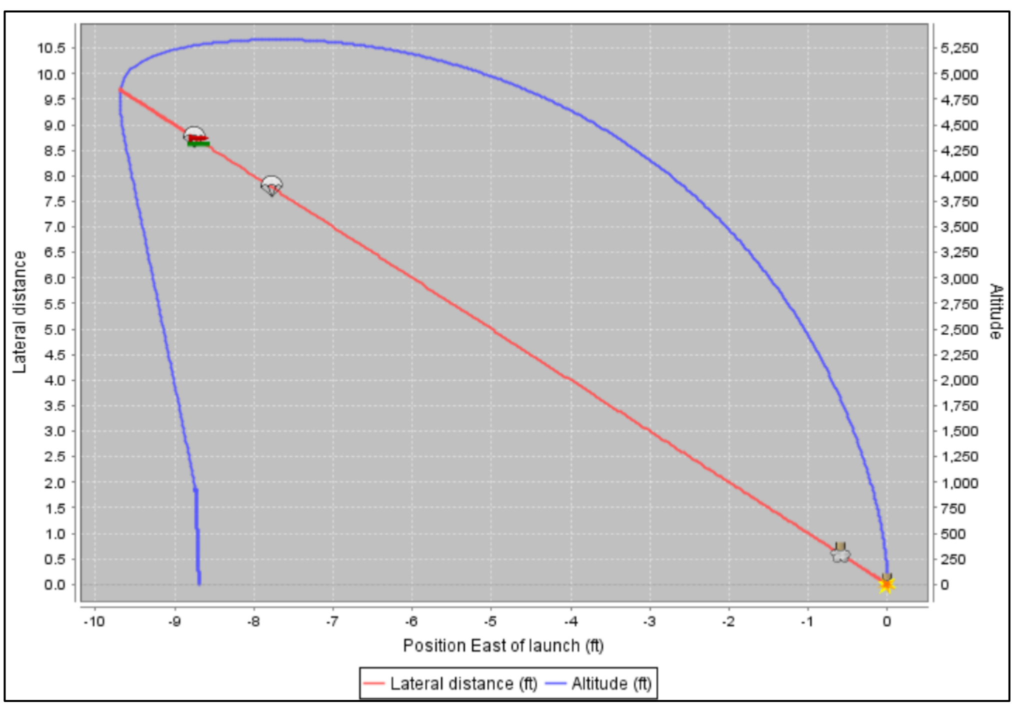


Figure 3.92: OpenRocket Drift Distance Simulation of The Ideal Case (0 Deg Incline, 0mph Purdue Dairy Farm, Indiana)

The simulation for an average wind speed of 0 mph shows that the maximum drift distance at Huntsville, Alabama, is roughly around 10.5'. As the launch vehicle tilts into the wind, it is simulated to travel 9.4' west of the launch rail. At Purdue Dairy, Indiana, the maximum drift distance is around 9.7'. As the launch vehicle then tilts into the wind, it is simulated to travel 8.7' west of the launch rail.

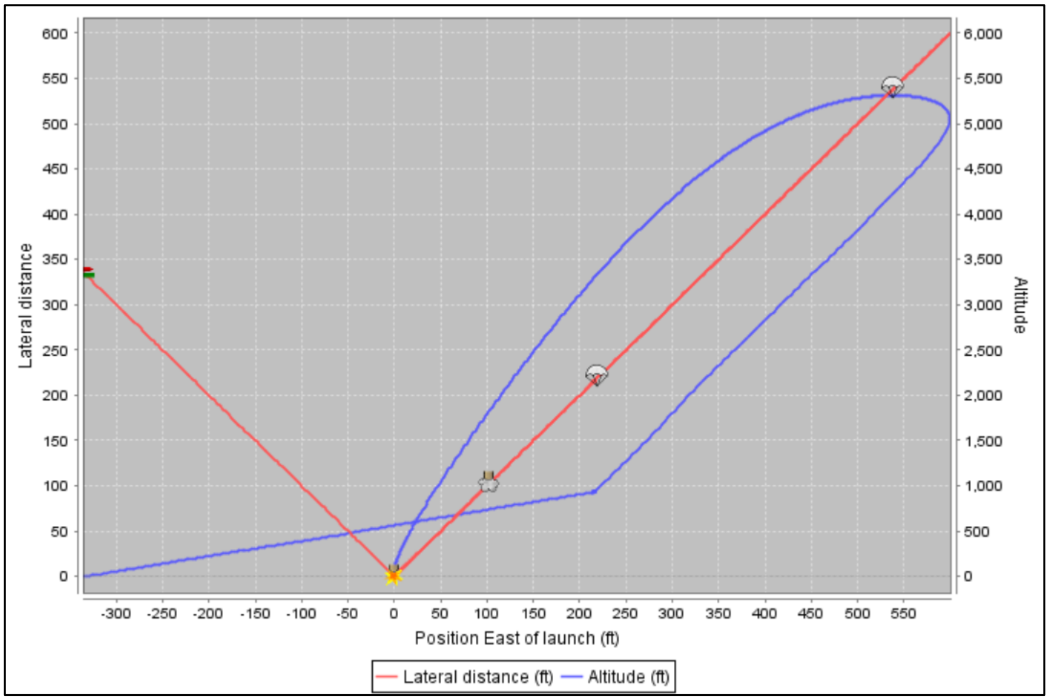


Figure 3.93: OpenRocket Drift Distance Simulation for 0-degree incline and a 5mph wind speed (Huntsville, Alabama)

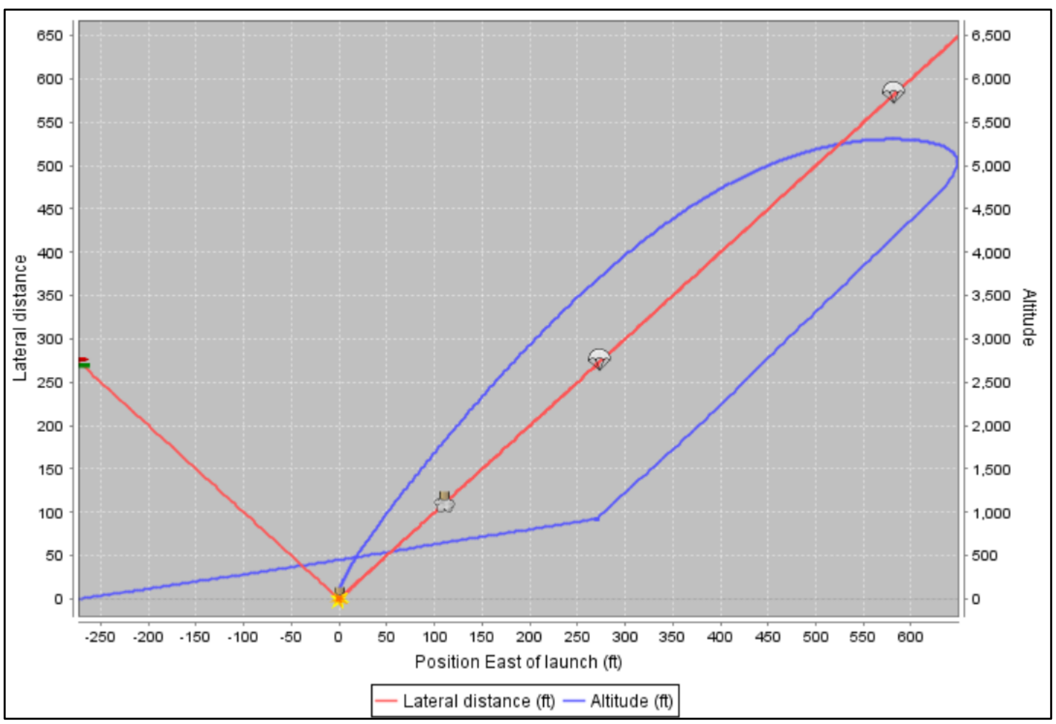


Figure 3.94: OpenRocket Drift Distance Simulation for 0-degree incline and a 5mph wind speed (Purdue Dairy, Indiana)

The simulation for an average wind speed of 5 mph with a standard deviation of 0.5 mph and a 10% turbulence intensity shows that the maximum drift distance at Huntsville, Alabama is approximately 600'. As the launch vehicle tilts into the wind, it is simulated to travel 600' east of the launch rail before drifting back over the launch position during the recovery stage, touching down 335.4' west of the launch position. At Purdue Dairy, Indiana, the maximum drift distance is approximately 648.4'. As the launch vehicle tilts into the wind, it is simulated to travel 648.4’ east of the launch before experiencing touchdown around 273.2’ west of the launch position.

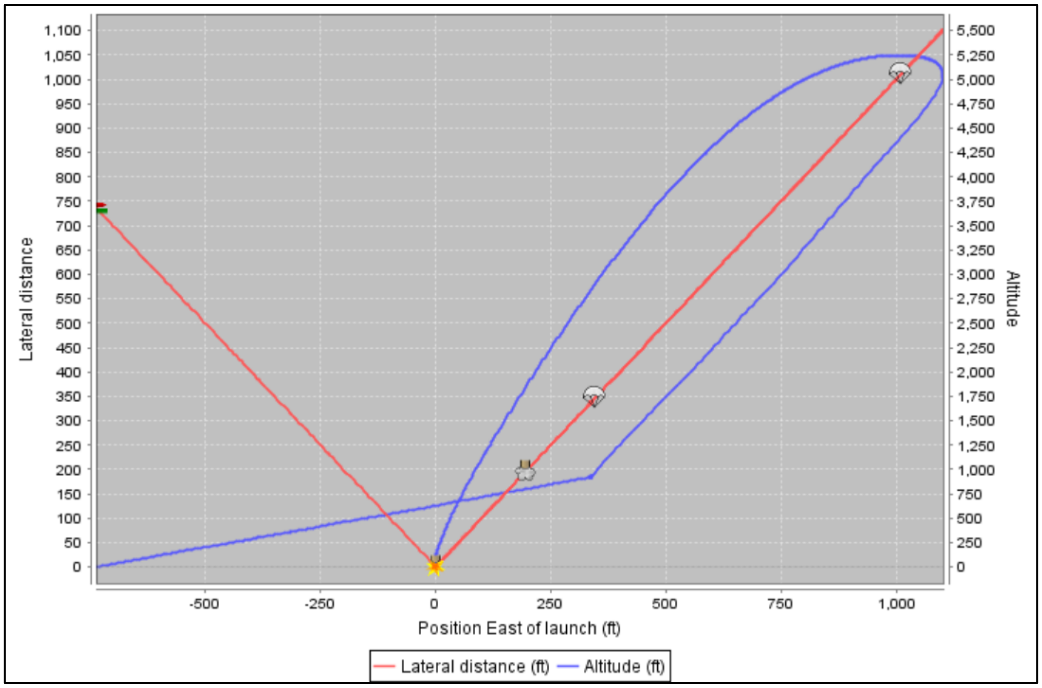


Figure 3.95: OpenRocket Drift Distance Simulation for 0-degree incline and a 10mph wind speed (Purdue Dairy, Indiana)

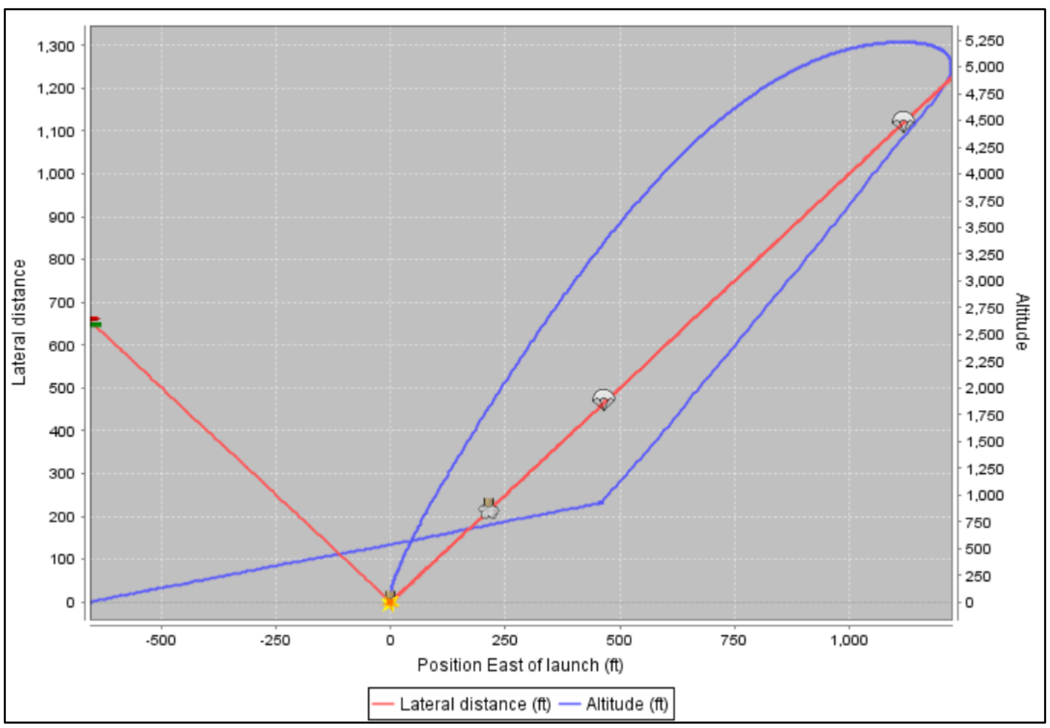


Figure 3.96: OpenRocket Drift Distance Simulation for 0-degree incline and a 10mph wind speed (Purdue Dairy, Indiana)

The simulation for an average wind speed of 10 mph with a standard deviation of 1 mph and a 10% turbulence intensity shows that the maximum drift distance at Huntsville, Alabama, is roughly around 1101’. As the launch vehicle tilts into the wind, it is simulated to travel 1101’ east of the launch before drifting back over the launch position during the recovery stage, experiencing touchdown 736.3’ west of the launch position. At Purdue Dairy, Indiana, the maximum drift distance is around 1223’. As the launch vehicle tilts into the wind, it is simulated to travel 1223’ east of the launch before drifting back over the launch position and experiencing touchdown 656.2’ west of the launch position.

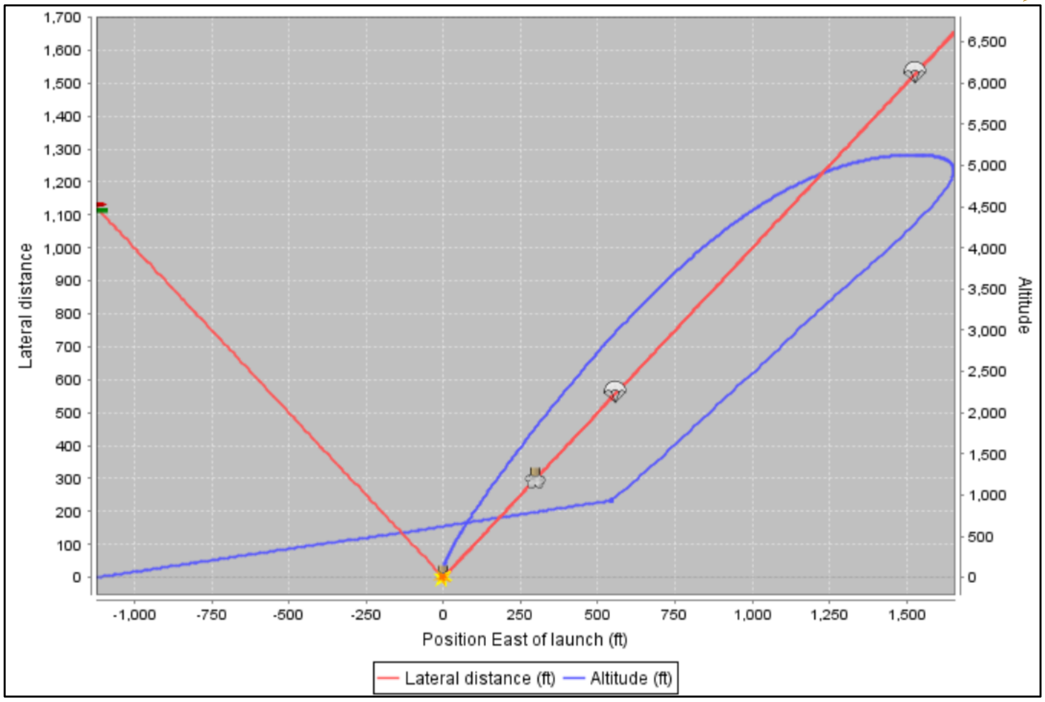


Figure 3.97: OpenRocket Drift Distance Simulation for 0-degree incline and a 15mph wind speed (Huntsville, Alabama)

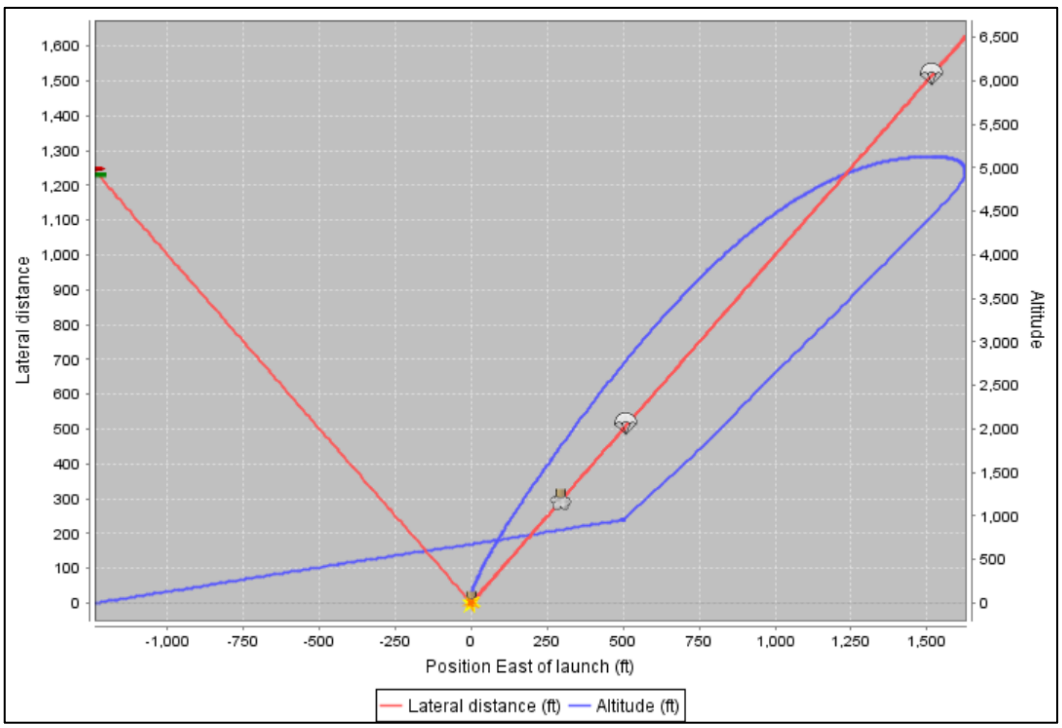


Figure 3.98: OpenRocket Drift Distance Simulation for 0-degree incline and a 15mph wind speed (Purdue Dairy, Indiana)

The simulation for an average wind speed of 15 mph with a standard deviation of 1.5 mph and a 10% turbulence intensity shows that the maximum drift distance at Huntsville, Alabama, is roughly around 1653'. As the launch vehicle tilts into the wind, it is simulated to travel 1653' east of the launch before drifting back over the launch position during the recovery stage, experiencing touchdown around 1123' west of the launch position. At Purdue Dairy, Indiana, the maximum drift distance is around 1625'. As the launch vehicle tilts into the wind, it is simulated to travel 1625' east of the launch before drifting back over the launch position and experiencing touchdown around 1240' west of the launch position.

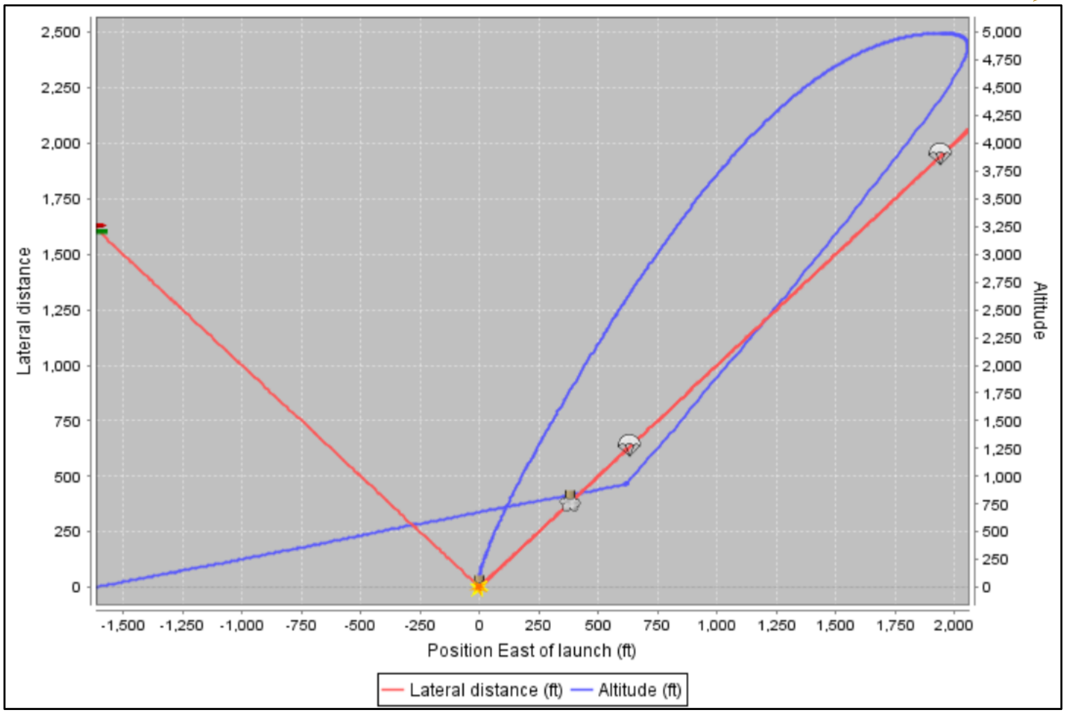


Figure 3.99: OpenRocket Drift Distance Simulation for 0-degree incline and a 20mph wind speed (Huntsville, Alabama)

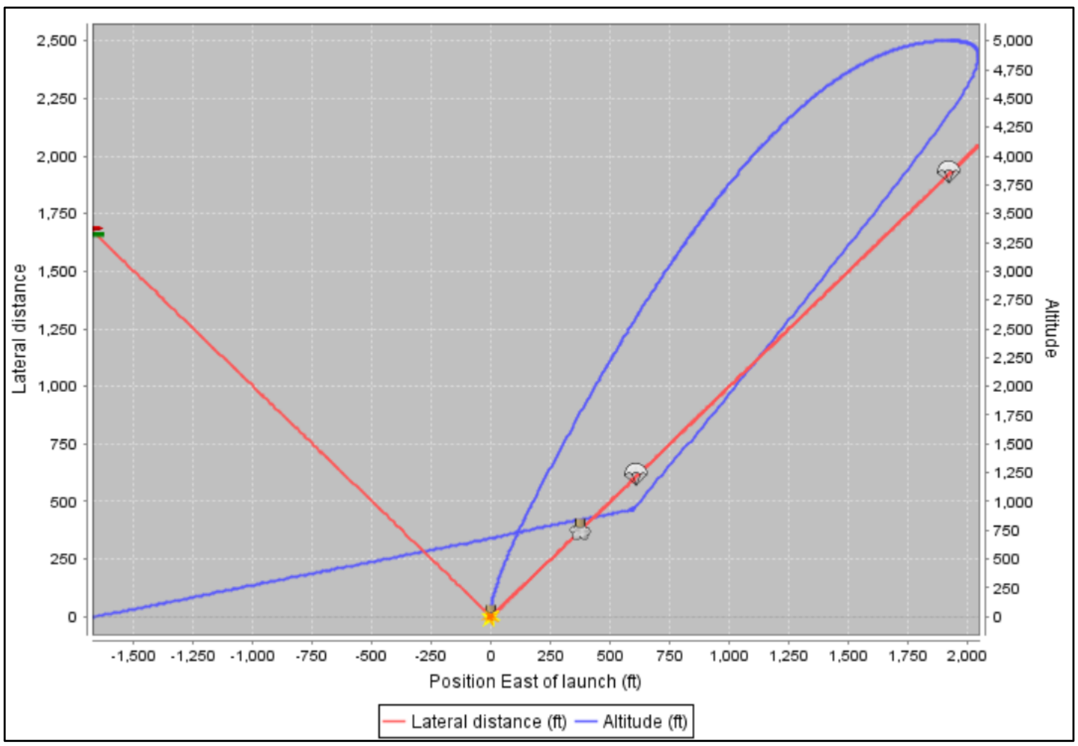


Figure 3.100: OpenRocket Drift Distance Simulation for 0-degree incline and a 20mph wind speed (Purdue Dairy, Indiana)

The simulation for an average wind speed of 20 mph with a standard deviation of 2 mph and a 10% turbulence intensity shows that the maximum drift distance at Huntsville, Alabama, is roughly around 2057'. As the launch vehicle tilts into the wind, it is simulated to travel 2057' east of the launch before drifting back over the launch position during the recovery stage, experiencing touchdown around 1614' west of the launch position. At Purdue Dairy, Indiana, the maximum drift distance is around 2045'. As the launch vehicle tilts into the wind, it is simulated to travel 2045' east of the launch before drifting back over the launch position and experiencing touchdown around 1674' west of the launch position.

3.4.2.3.2 Simulink

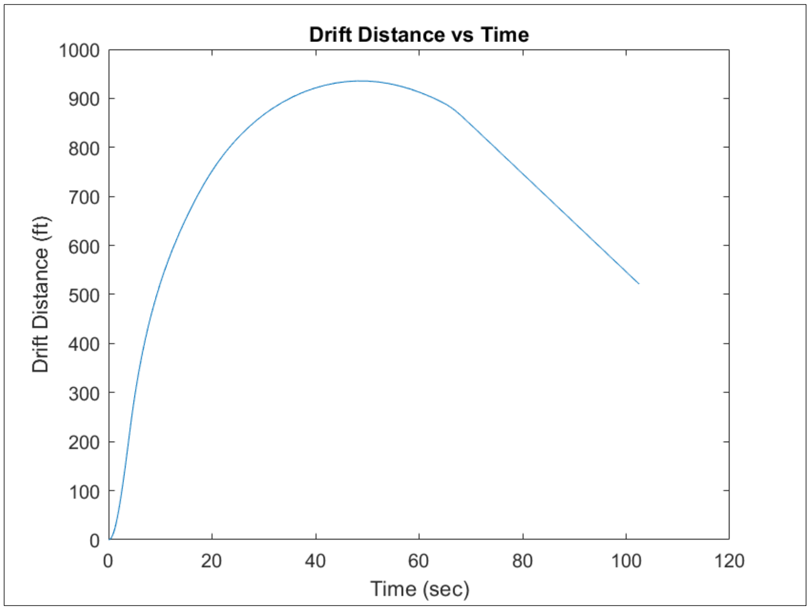


Figure 3.101: Simulink Drift Distance vs Time Plot

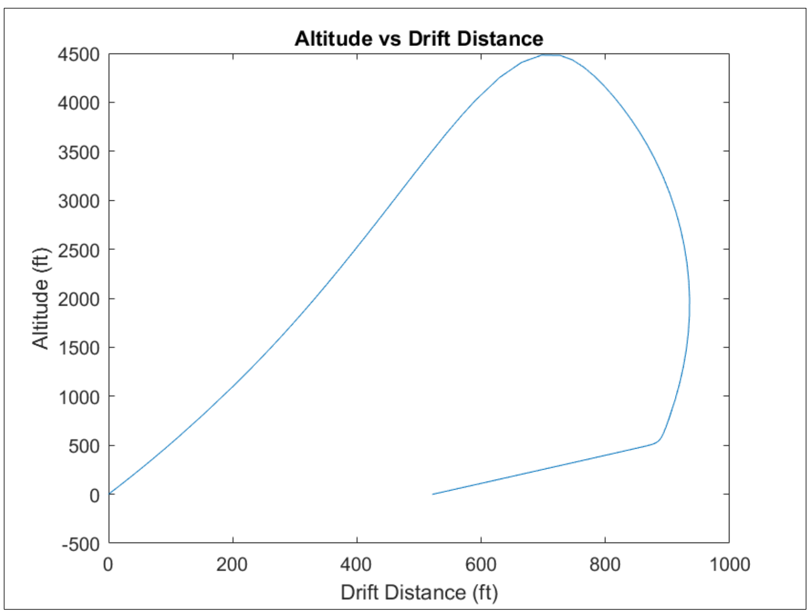


Figure 3.102: Simulink Altitude vs Drift Distance Plot

The Simulink simulation predicts a 521' drift distance upon landing from the starting location of the vehicle, which is well under the maximum requirement of 2500'. One interesting thing to note is that once the main parachute deploys, the side drag becomes large enough that the wind force causes the vehicle to reverse its horizontal direction, decreasing the overall drift. This fact reinforces the team's decision to launch into the wind.

3.4.2.3.3 Hand-Calculations

To calculate the drift distance for a launch, the team utilized the equation: $D = t * V_\infty * \sin(\theta)$. In the equation, D represents drift distance, t represents flight time, V_∞ represents the windspeed and $\sin(\theta)$ represents the horizontal component of the windspeed referencing the launch angle θ. Using a flight time of 84.9 s and windspeed of 29.33 ft/s (20 mph) with a launch angle of 90° (relative to the ground), the resulting equation is:

$$D = (84.9s)(29.33ft/s) * \sin(90)$$

This results in a drift distance of 2490.4'. Even in this most extreme case, the drift distance is still below the maximum of 2500'. Although this equation is a good estimation of the maximum drift distance, it does result in a larger drift distance then expected from simulation. For example, the below figure depicts a launch in 20 mph and 0 degree launch angle, which portrays a maximum drift distance of 2200'. Again, in this most severe launch scenario, the drift distance is below the maximum allowed.

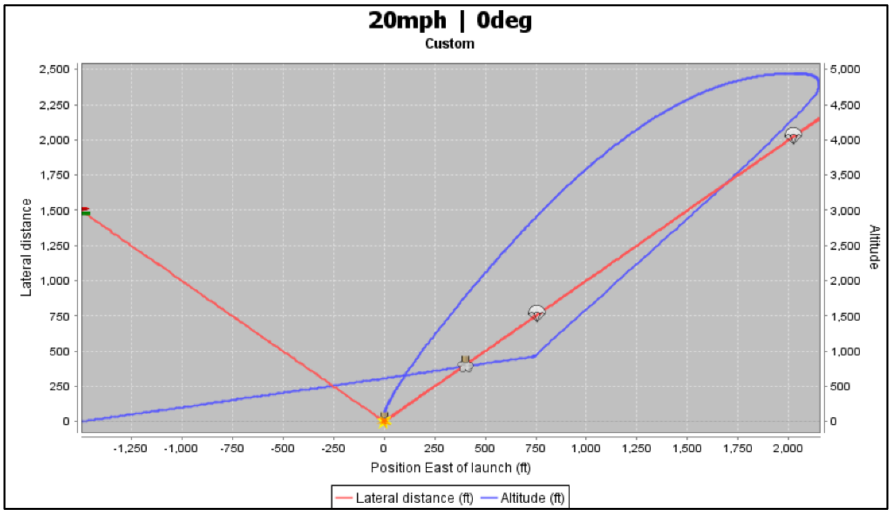


Figure 3.103 Example Graph of Drift Distance

3.4.3 Motor Characteristics

The motor chosen to fulfill Project Voss' propulsion system is the CTI L1115. The L1115 is one of the most powerful 4-grain L-type SRM, with a total impulse of 1128.38 lb-s over its 4.48 s burn time. Thrust provided during the flight is 327 lbf at rail exit, 385.48 lbf at its maximum, and an average of 251.78 lbf. The curve below is based off the motor data given, where the motor depicts the values of time and thrust throughout the burn time.

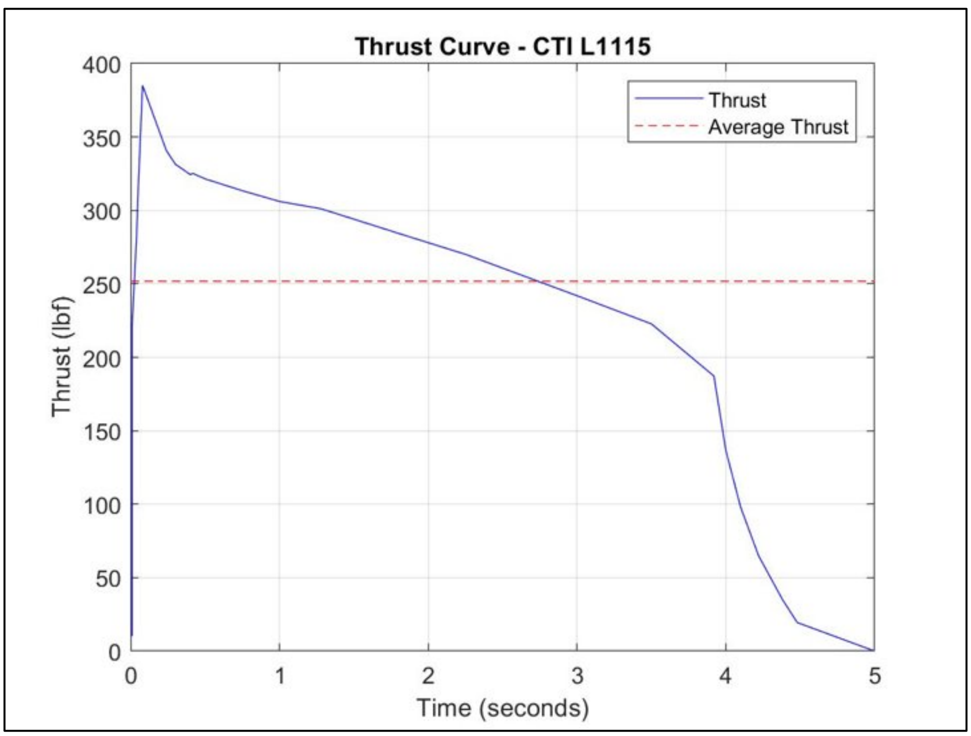


Figure 3.104: Thrust vs time for CTI L115

The CTI L1115 burns classic propellant, consisting of Ammonium Perchlorate as the oxidizer and Atomized Aluminum as the fuel. The figure below depicts a cutaway of a similar SRM to better visualize the inner workings of the motor.

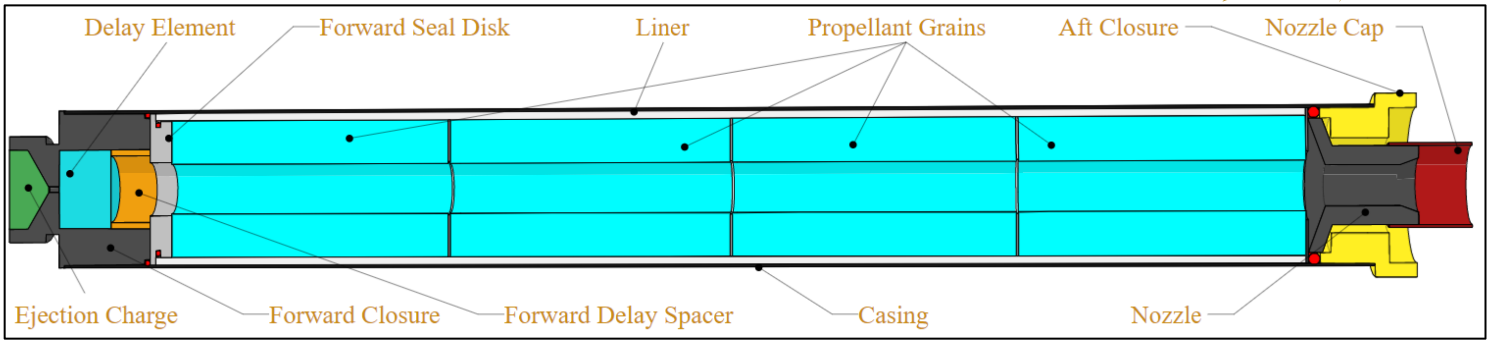


Figure 3.105: Inner working of a motor

3.4.3.1 Thrust-to-Weight Verification:

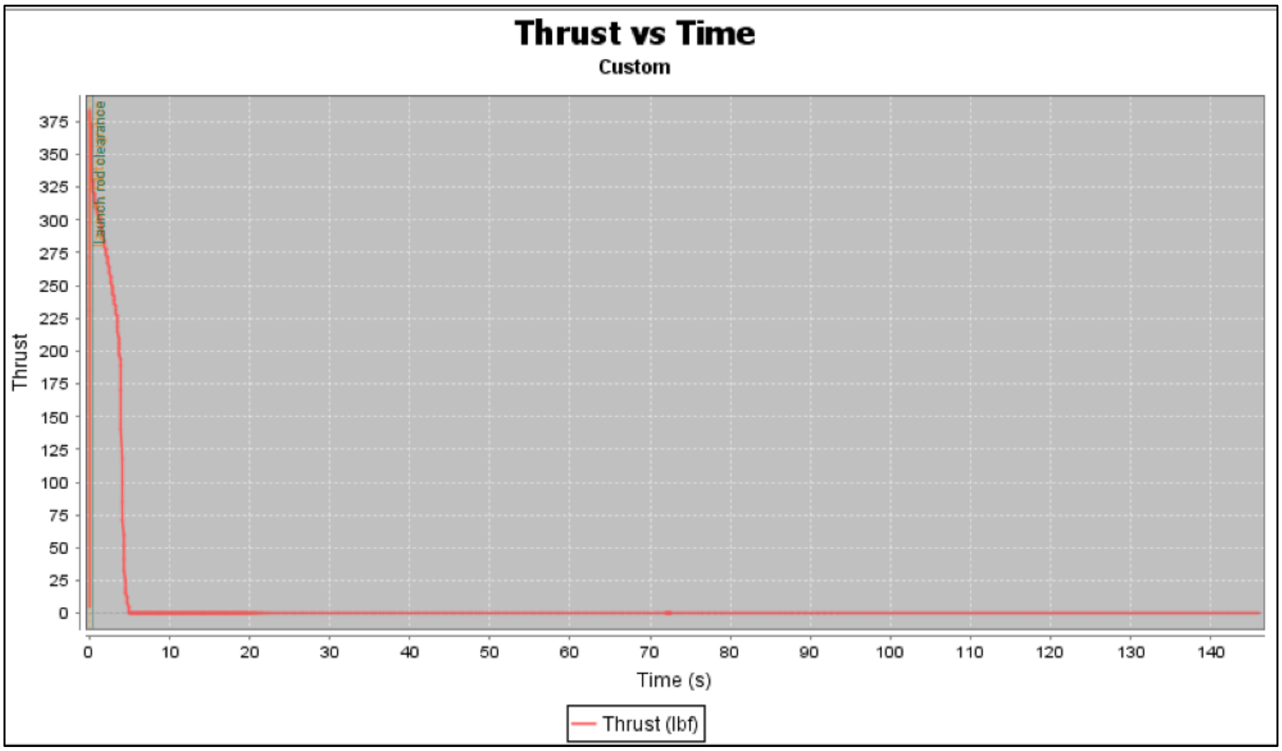


Figure 3.106 OpenRocket Simulation of Thrust vs Time

According to OpenRocket simulation, the launch vehicle clears the rod at 0.36 seconds into flight. Based on the provided thrust data from thrustcurve.org, the thrust provided at this time in the flight is 327 lbf. This was calculated with linear interpolation, with the data presented below. Solving for the variable x determines the thrust off the rail, which is the critical thrust to weight ratio, as the vehicle is at its maximum weight. With our current design, the launch vehicle has a Gross Lift-off Weight of at most 54 lbm after ballast, which gives a thrust to weight ratio of 327/54 = 6.1 minimum, which satisfies the necessary 5:1 ratio.

Time (sec)	Thrust (lbf)
0.3	331.5513
0.36	x
0.4	324.4784173

Table 3.9: Thrust and Time Linear Interpolation

3.4.4 Kinetic Landing Energy

Descent Under	Descent Velocity (ft/s)
Drogue	87.7
Main	14.3

Table 3.10: Simulink Vehicle Descent Velocities

Vehicle Section	Landing Kinetic Energy (ft-lbf)
Upper Section	43.8
Middle Section	32.5
Lower Section (Dry)	74.8
Total Launch Vehicle (Dry)	151.1

Table 3.11: Simulink Landing Kinetic Energies

The most important value to note from these tables is the landing kinetic energy of the heaviest section of the vehicle (the lower section), which the Simulink simulation predicts to be 74.8 ft-lbf. This value (as well as the landing kinetic energies of the other independent sections) is under the maximum requirement of 75 ft-lbf.

4 Payload System

4.1 Payload Criteria

4.1.1 Mission Statement

4.1.1.1 Planetary Landing System

The mission of the Planetary Landing System (PLS) is to capture a level, 360° panoramic photograph of the landing site of the launch vehicle after being safely deployed from the vehicle during main parachute descent. The Lander Subsystem will be actively retained by the Retention and Deployment subsystem (R&D) during flight and after deployment of the vehicle's main parachute. The operation of the PLS will be designed to prevent interference with the launch vehicle after deployment. To ensure the safety of personnel on the ground, the descent design of the Lander must be optimized for reliability.

4.1.1.2 AeroBraking Control System

The mission of the AeroBraking Control System (ABCS) is to improve the team's apogee score by increasing the altitude precision and accuracy of the launch vehicle towards the desired altitude. The ABCS will employ an autonomous control system designed to predict the vehicle's current apogee error during coast and adjust the vehicle's drag state to reduce this error. This system will require operation in a high-speed compressible aerodynamic regime, requiring utmost care for mechanical design, material usage, and assurance of stability; the ABCS will be designed with fail-safes to ensure proper deactivation given the possibility of loss of control authority by the vehicle's fins.

4.1.2 Mission Success Criteria

4.1.2.1 Planetary Landing System

This mission is divided into seven essential phases, each with a defining event leading into the action of the next. The system's mission will be considered complete if its status proceeds to the final phase without failure. In the case of a critical failure, additional contingency plans have been considered. The phases of the PLS mission are shown as follows:

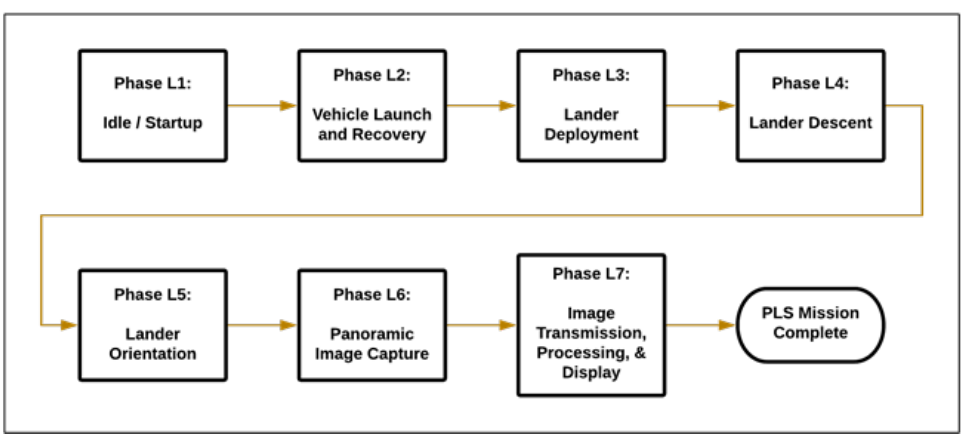


Figure 4.1: Planetary Landing System Mission Overview

L1: Idle / Startup:

The first phase of the mission includes the time between launch vehicle activation and the time of launch. During this time, the R&D will be in an idle but active state in preparation for launch. The Lander will also be in a state of sleep, awaiting indication from R&D to awaken and activate. Given that the vehicle is ready to launch, the mission may proceed to Phase L2.

L2: Vehicle Launch and Recovery

While the vehicle is in its ascent phase, the R&D will be ensuring that flight loads are transferred around the body of the Lander contained inside. The Lander will not be able to exit the vehicle during this stage due to a mechanical lock, which will not unlock without action from the onboard R&D controller. When the vehicle begins its downwards descent, the load state of the R&D will tend to flip, but the system will still be designed to handle the required loads. The R&D will be capable of withstanding jerk from both the drogue parachute and the main parachute before continuing to Phase L3.

L3: Lander Deployment

After the completed deployment of the vehicle's main parachute by approximately 700' AGL, the R&D will begin to deploy the Lander. The R&D will begin to separate the lock vertically constraining the nosecone section of the Payload Bay, allowing the Lander to slide downwards with gravity. Once fully released, the nose cone section will slide downward before hitting stops. The Lander will then be unconstrained in one lateral dimension, allowing it to fall to the side and begin descent. During this time, the Lander's electronic systems will be awoken by the R&D, allowing it to proceed to Phase L4. If the R&D is unable to deploy the Lander for any reason, the Lander will remain sleeping and must be able to be deactivated on the ground with the permission of an RSO. With the Lander having exited the vehicle, the R&D system will remain in an open configuration until touchdown, retaining the nose cone of the vehicle.

L4: Lander Descent

The Lander should be clear of the launch vehicle by 500' AGL, by which time a parachute delay method will have disconnected itself from the Lander, allowing the Lander's parachute to open. It should be noted that even if the Lander does not properly awaken, the parachute will still be designed to deploy, ensuring the safety of the Lander and personnel on the ground. The parachute will bring the terminal velocity of the Lander to a speed greater than the speed of the launch vehicle to ensure no interference occurs. The Lander will then reach the ground, landing in any orientation. If the awoken Lander detects a successful return to Earth, it will detach its parachute and begin Phase L5.

L5: Lander Orientation

Now grounded, but with no certainty of landing orientation, the Lander will begin to self-upright through the usage of a motorized Self Orientation Subsystem (SOS). By increasing the Lander's effective support area, the Lander can be assured to slowly bring itself into a stable upright configuration. The SOS will attempt to adjust its final state to ensure that the onboard sensors confirm orientation within 5° of the local gravitational acceleration vector. Once the control system has completed this task, the Lander will proceed to Phase L6. If the Lander is unable to complete this phase after a predetermined amount of time, it will deactivate itself to prevent injury to the ground team.

L6: Panoramic Image Capture

Leveled within the desired tolerance, the Lander should now be clear of debris and should have an elevated view of the launch field. The onboard Panoramic Image Capture Subsystem (PICS) cameras will activate and proceed to capture an image of the field. The image will be stored locally until it is ready to be sent to the team's Ground Control Station (GCS), moving the PLS to the final Phase L7.

L7: Image Transmission, Processing, & Display

Once a communication channel is secured between the GCS and the Lander, the PICS will begin to transfer the image data to the GCS via a radio transmitter. Once received, the GCS will store the image data. The Lander Subsystem has now completed its purpose and may be recovered. The GCS will then proceed to process the received image data and convert it into a viewable format. If the PICS utilizes a multi-image capture system, the GCS will need to stitch them together to view at one time. Once converted, the image will be displayed on the GCS's display screen for confirmation by the team, an RSO, and other NASA personnel. Having produced an image, the PLS has completed its mission and may be shut down for recovery by the ground team.

PLS Mission Completion

By the end of its mission, the PLS should have produced an unobscured image of the launch vehicle landing site. The system must overcome the challenge of being jettisoned from a high-power rocket during descent and master the elaborate dance of up-righting upon unknown terrain. At the same time, the system must satisfy every functional requirement set forth by the team to be considered ready to fly. If the system can produce these results without succumbing to material failure, inadvertent deployment, or a blocked camera subsystem, then the team will consider the mission a complete success.

4.1.2.2 AeroBraking Control System

This mission is divided into five essential phases, each with a defining event leading into the action of the next. The system's mission will be considered complete if its status proceeds to the final phase without failure. In the case of a critical failure, additional contingency plans have been considered. The phases of the PLS mission are shown as follows:

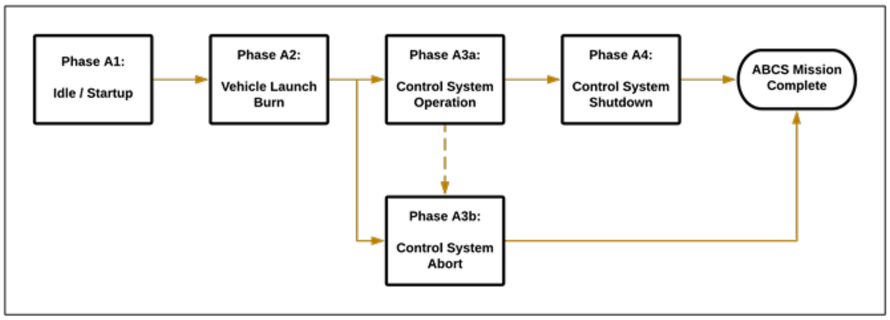


Figure 4.2: AeroBraking Control System Mission Overview

A1: Idle / Startup

The first phase of the mission includes the time between system power-on until the time of launch. During this time, the ABCS is in an idle state waiting for the detection of launch. Once the launch burn is detected by ABCS accelerometers, the ABCS will transition into phase A2.

A2: Vehicle Launch Burn

The ABCS control system will need to detect the acceleration state of the launch vehicle at the start of its burn before it begins its operational stage. The ABCS mechanical system will not be allowed to actuate until the system detects that the launch vehicle's burn has completed, and the vehicle has reached sufficient altitude, confirming a successful burn. With the ABCS remaining inactive during the boost phase, the vehicle's passive stabilizing fins can operate properly without possible loss of control authority. Once the criteria are met, the ABCS main drag control system loop may be activated, leading to Phase A3a. If the launch burn is not detected to have completed properly, then the control system will proceed to Phase A3b.

A3a: Control System Operation

Given that the flight conditions are acceptable for the operation of the Airbrakes, the ABCS drag control system loop will begin operation. This control system will employ a suite of sensors for detecting the current state of the vehicle; important inputs include the vehicle's linear and angular velocity, acceleration, air pressure, and altitude. A constant set by the team will be the desired final apogee of the launch vehicle. The control system will first predict the current apogee error of the vehicle relative to the desired apogee and then will determine how much drag would currently be required to achieve this final apogee. The ABCS will accordingly actuate its Airbrakes, producing this additional drag. An important initial condition of this phase is that the launch vehicle will, without intervention, achieve a final apogee greater than the desired altitude; without this condition, the ABCS would be unequipped to provide any additional required velocity—it can only act to remove mechanical energy from the system. If at any time during the active phase the control system detects that the launch vehicle has exceeded attitude state or acceleration bounds determined by the team to be within an acceptable range, the control system will immediately switch to Phase A3b to avoid possible loss of stability. If the ABCS continues operation up until apogee, it will transition to Phase A4.

A3b: Control System Abort

The ABCS will be designed to cease its functionality if it detects that it could potentially cause instability of the launch vehicle during ascent. If the ABCS reaches Phase A3b at any time from the beginning to the end of its designated operation time, it will follow a shutdown sequence to ensure that the system does not incur additional change in attitude or velocity. This shutdown sequence will involve immediately ending the apogee optimization drag control system loop and completely retracting the Airbrakes Subsystem's drag plates. This contingency plan is essential to allow the launch vehicle's stabilizing fins to return the attitude of the vehicle to an acceptable state, avoiding an induced tumble. With all external surfaces now inactive, the ABCS control system will remain inactive for the remainder of descent and touchdown. While the control system was unable to fully optimize and complete its final calculations and adjustments, the team will still consider this contingency plan as successful as the overall vehicle will still complete its mission.

A4: Control System Shutdown

When the vehicle reaches apogee according to flight sensors, the ABCS will begin a deactivation sequence to reduce the possibility of damage to components. This deactivation sequence will involve immediately ending the apogee optimization drag control system loop and completely retracting the Airbrakes Subsystem's drag plates. With all external surfaces now inactive, the ABCS control system's mission will be considered complete and will deactivate for the remainder of descent and touchdown.

ABCS Mission Completion

By the end of its mission, the ABCS will have hopefully reduced the error between the vehicle's actual apogee and the desired apogee relative to previous years' vehicles. The system must overcome the challenge of compressible regime aerodynamic loads while also deftly making complex trajectory calculations and executing upon them. At the same time, the system must satisfy every functional requirement set forth by the team to be considered ready to fly. If the system can produce these results without succumbing to material failure, having an uncontrolled deployment, or exasperate a hazardous vehicle tumble, then the team will consider the mission a complete success.

4.2 Planetary Landing System

4.2.1 System Overview

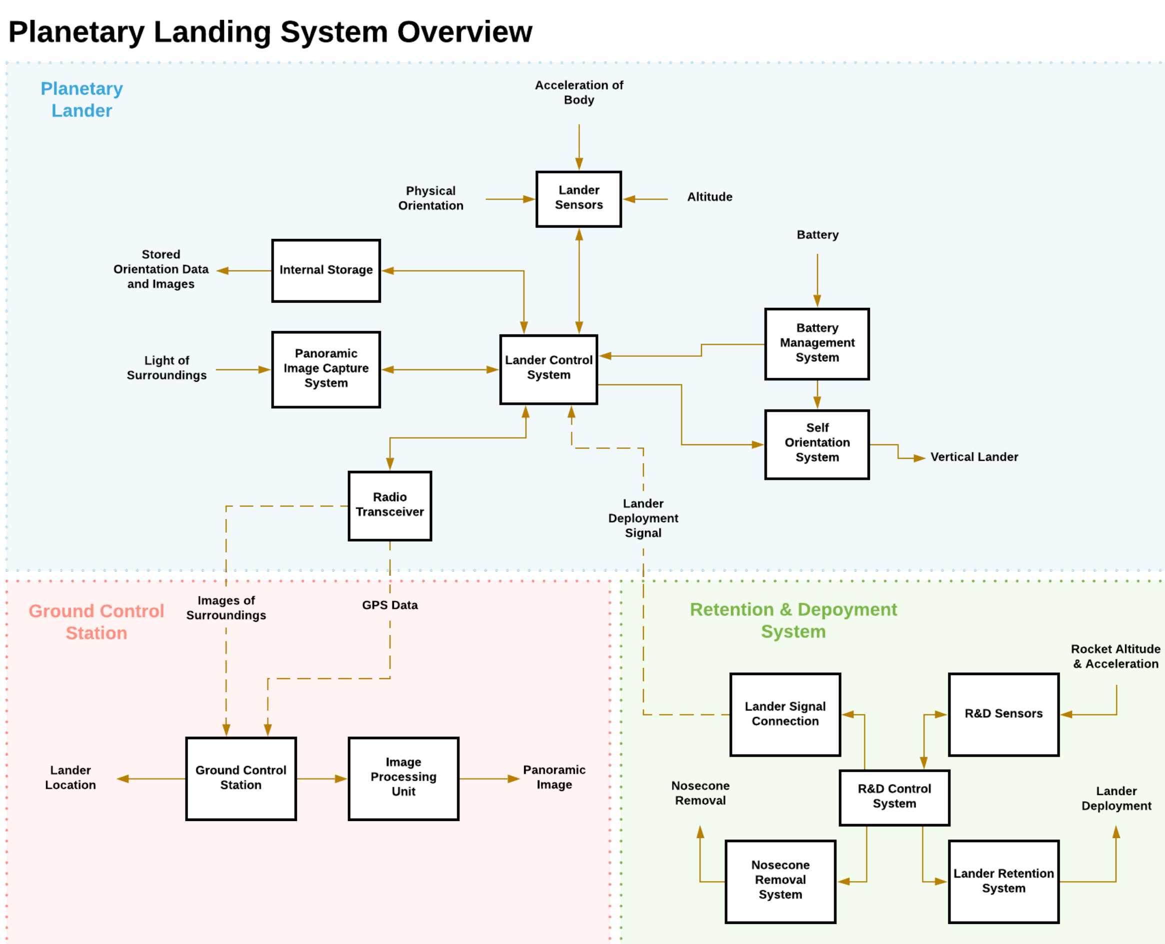


Figure 4.3: Planetary Landing System Functional Block Diagram

The Planetary Landing System (PLS) includes the Lander and its associated Retention and Deployment Subsystem (R&D). The Lander itself can be further broken down into the Self Orientation Subsystem (SOS), the Panoramic Image Capture Subsystem (PICS), the Lander Control Subsystem (LCS), and the in-flight Descent and Landing Subsystem (D&L). Once the launch vehicle has deployed its main parachute and its altitude has dropped below 700' AGL, the R&D system will release the Lander from the nose section of the launch vehicle. The Lander will then deploy its own parachute after falling a sufficient distance from the launch vehicle, ensured by the D&L. The Lander will strike the ground at a sufficiently low speed deemed by the design team to be acceptable for the continued operation of the Lander's subsystems. Once the Lander detects that it has stopped moving, the SOS will begin to orient the Lander within 5 degrees of vertical. Once the SOS has completed self-orientation, the PICS will take a photograph of the surrounding area from each of its 3 static cameras. The photos will then be stored in digital storage located in the Lander to await radio transfer to the GCS. Once a stable connection has been established, the photos will be transferred to the GCS. An image processing unit in the GCS will then combine all three images into one panoramic image. The full panoramic image will then be displayed on screen for inspection.

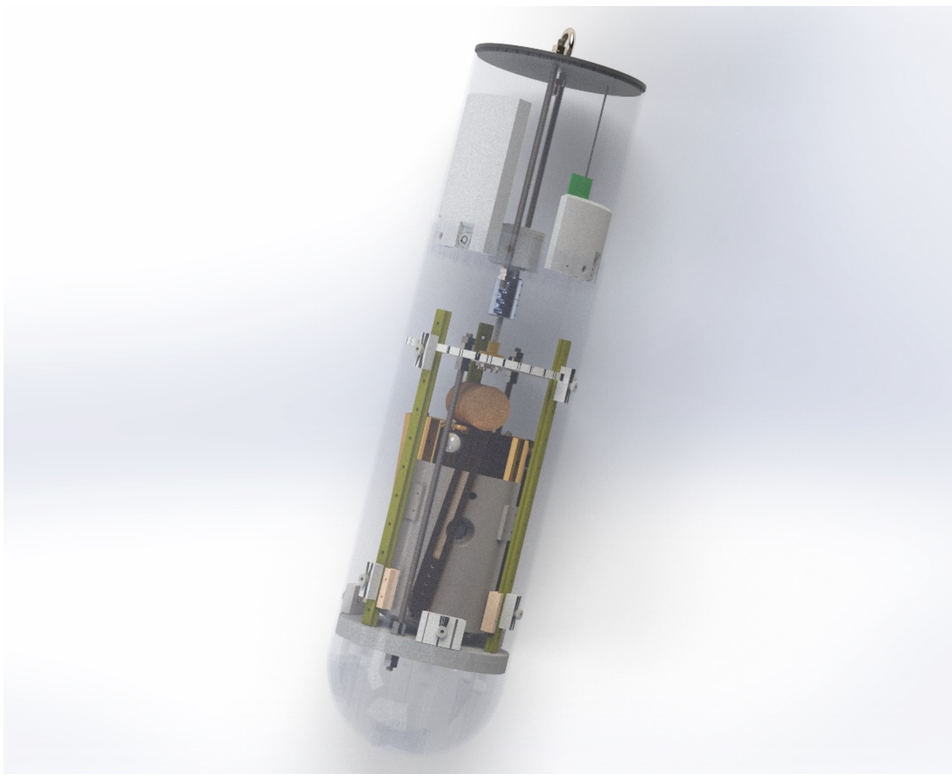


Figure 4.4: Overall Planetary Landing System Render (Closed Configuration)

The entire PLS faces the all-new challenge of retention and deployment during descent, which is not something PSP-SL has ever attempted in the past. In this unusual terrain—or lack thereof—the typical rules of static forces and loading are complicated by a lack of a solid sink of momentum and energy—otherwise just being the Earth itself. Therefore, the final design determined by the team utilizes the unique orientation and force state of the launch vehicle as it ascends under power and descends under drag forces. The final design of the R&D should be able to transfer all associated loads of flight while maintaining the ability to quickly and reliably deploy the Lander in a manner that can be tested on or near the Earth's surface. Given many new challenges to design, the team decided to focus efforts on simplicity of execution; overcomplicating the functionality of a design meant to perform in such unusual circumstances would likely lead to minute points of failure which could compound towards the destruction of the launch vehicle or injury of the ground team. Therefore, the PLS has been designed to utilize mostly open-air space within the Payload Bay. Please note that from hereon in, the Payload Bay refers to all internal space in the Payload section of the launch vehicle, excluding the internal space of the nose cone. In the above figure, it can be observed that the final design revolves around a simple concept: The actuation of a central lead screw releases the frontal section of the launch vehicle along with the Lander. By avoiding overcomplication of design in ways initially considered, such as by explosively ejecting the nosecone and Lander or releasing the nosecone and lander separately, the team has produced a more reliable system without as many dangerously compounding variables.

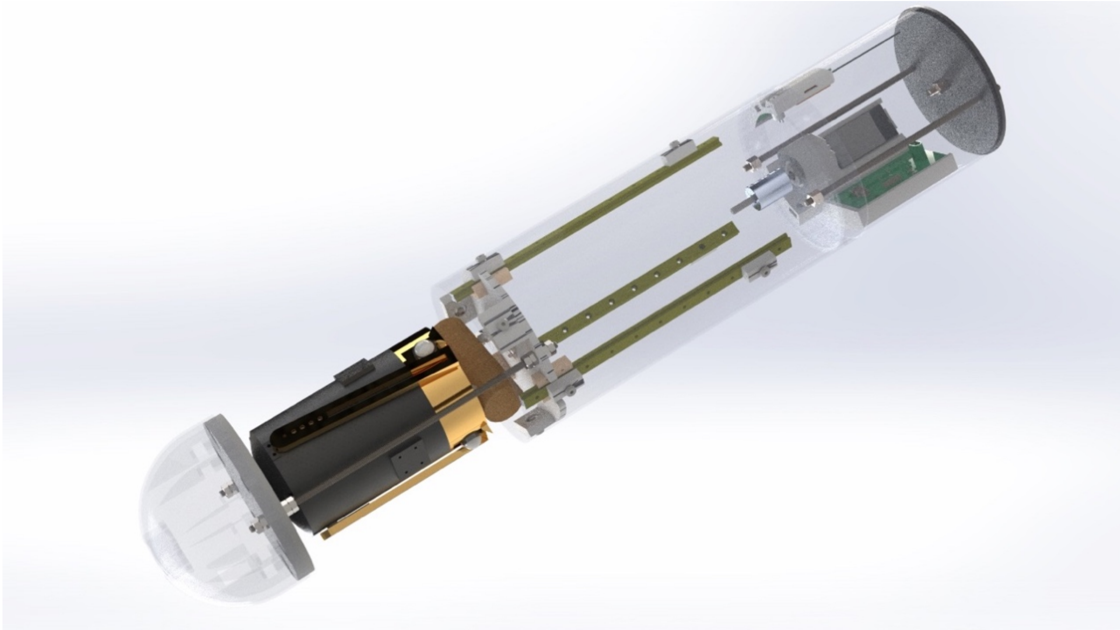


Figure 4.5 Overall Planetary Landing System Render (Open Configuration)

This particular design of the PLS has continued mostly in the same form since PDR, however many finer details have since been hammered out, allowing the previously abstract CAD designs to take full form. As above, it can be noted that the two railed internal structure of the R&D during PDR has been since modified to a three railed structure. Changes like this came after decisions made across many subteams of the Payload team. The following excerpts will aim to provide an in-depth discussion of many design changes and how they eventually brought the team's final designs to the point they are at now. Meanwhile, since PDR, solidification of the systems' central concepts has enabled entire new construction of electrical apparatus and associated electrical diagrams.

4.2.2 **PLS Detailed Design**

4.2.2.1 Retention and Deployment Subsystem

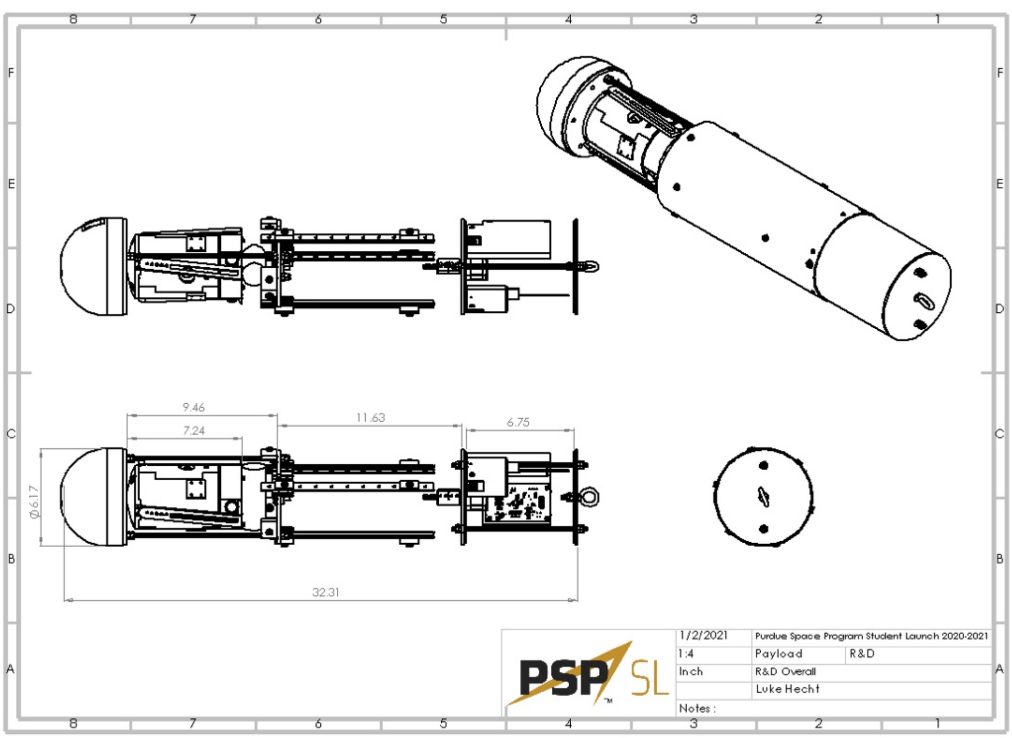


Figure 3.6 Overall R&D Drawing

The primary objective of the Retention and Deployment Subsystem (R&D) is to retain the Lander within the confines of the Payload Bay during flight until deployment time. Afterwards, the R&D must be able to release the Lander before the launch vehicle descends to the altitude of 500’ AGL in accordance with Subteam Requirement S.P.1.6. Due to its position near the nose of the launch vehicle, the R&D must also mechanically retain the launch vehicle’s nose cone for the entire duration of the flight.

Since PDR, the team has continued with a gravity-driven deployment scheme. The initially proposed designs included a mirrored, two guide-rail design that would act to constrain and guide the Lander during deployment. To coincide with the increasing complexity of the Lander’s design, the R&D has been redesigned to adopt a triple axially symmetric form factor. This new design has maintained a high level of compatibility with the Lander but increases the required number of components that must be installed into the Payload Bay. Furthermore, in order to mechanically retain and release the Lander, original designs included a high-torque DC motor which would act to unscrew the R&D’s central plate structure. Further analysis and prototyping have led to the team’s decision to replace the initial DC motor. The team decided to utilize a stepper motor to increase holding potential and ability to perform under the loads experienced during descent.

Important finalizations in design have allowed for the electrical and structural design of the R&D control system. This unit will be contained within the coupler between the main Payload Bay and the recovery section. By avoiding use of the main Payload Bay internal space, the R&D system has been able to be customized without the need to worry heavily about space constraints. The electronic design has matured significantly, utilizing prototype setups to inform the replacement of components like the motor driver to be more compatible with the chosen stepper motor. Satisfied with components, the team has also pursued production of an integrated PCB, allowing for simple design and testing of components which were previously connected independently. After final confirmation of the PCB design through testing, these parts will be ordered for use.

4.2.2.1.1 Mechanical Design

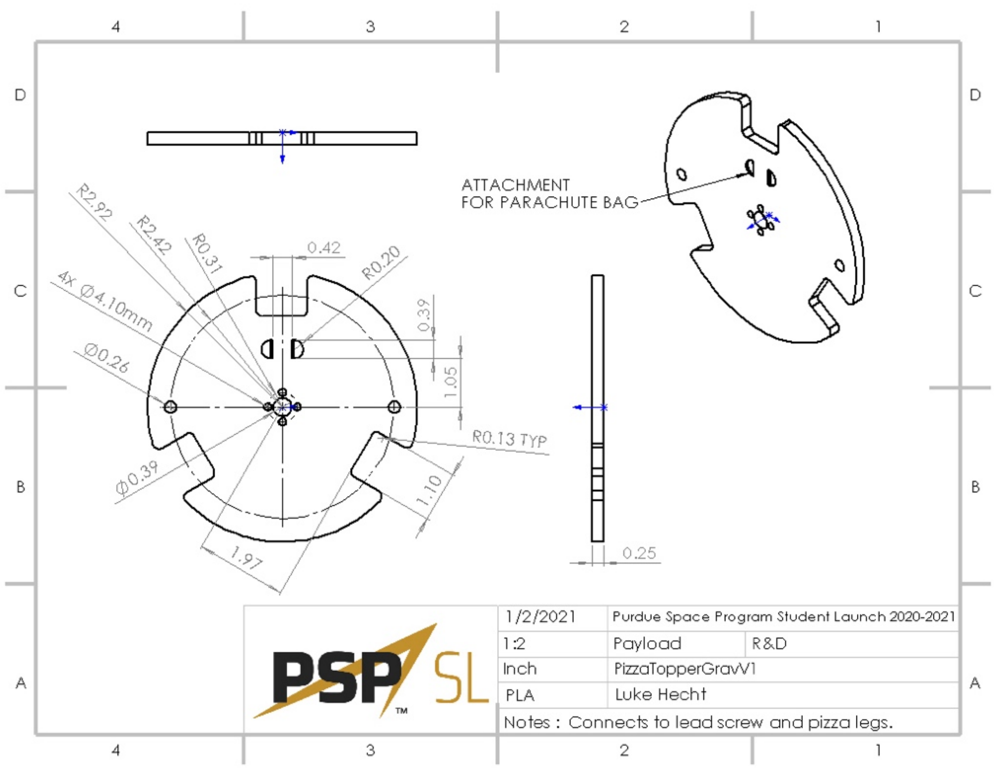


Figure 4.7 Pizza Table Topper Drawing

The R&D system uses a rail-secured sled referred to as the “Pizza Table” to open the nose cone, secure the Lander, and to transport the Lander so that it can be deployed. The Pizza Table consists of a thin plate with two, foot long ¼-20 threaded rods protruding from the top side. This shape was inspired by a pizza saver, a plastic device often found within pizza boxes to protect pizza from getting crushed. The Pizza Table’s legs wrap around the Lander and terminate at the vehicle’s nose cone. The two threaded rods replace previous designs’ 3D printed legs. This change was made for two reasons. The first reason was to make it simpler to connect to the launch vehicle’s nosecone, and the second reason was to add strength to help prevent potential bending when the launch vehicle hits the ground under main parachute descent. Since these two rods directly connect the Pizza Table to the nose cone, the apparatus allows for the nose cone's securement during flight and for ease of actuation when it comes time for deployment. Additionally, the Pizza Table has three slots cut out of it so that it may be centered within the Payload Bay by three rails. These three linear motion guide rails are mounted radially to the wall of the Payload Bay and serve to secure and guide both the Lander and the Pizza Table as mentioned. Finally, the Pizza Table has a lead screw nut mounted on its bottom side. This lead screw threads into the central lead screw of the Payload Bay which is attached to the stepper motor housed within the coupler between the Payload Bay and the recovery section.

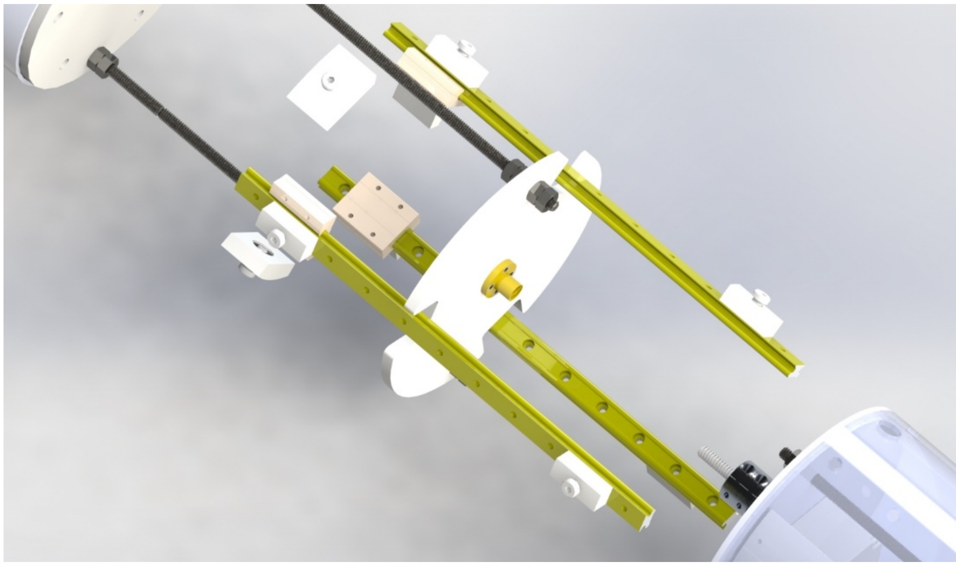


Figure 4.8: R&D "Pizza Table" Interweaving with Guide Rails (Vehicle Fuselage and Lander Removed)

During vehicle setup and flight, the Lander is secured on top and on bottom by being sandwiched between the nose cone attachment plate and the Pizza Table plate. Additionally, the Lander is centered within the bay by the three radially mounted guide rails on the payload bay wall. The Lander interfaces with these linear motion rails using the carriages that are sold in tandem with the rails. These carriages are screwed into the sides of the Lander so they line up with the positioning of the rails. (In the provided renders, the rail carriages will always be shown to be separate from the Lander, while the final assembly will include the permanent attachment of these carriages to the Lander. Unfortunately, the chosen guide rail models are unable to be modified. Please excuse this discrepancy. Any time the relevance of this issue presents itself, additional clarity will be provided.) Given these constraints, the Lander will be unable to move significantly within the Payload Bay without the activation of the deployment sequence. Below is the closed arrangement of the R&D, colored to provide clarity. In the following images, the yellow components are considered the Lander, the **green** components constitute the Pizza Table and the nosecone attachment plate, the **blue** components are structurally fastened to the launch vehicle's airframe, and the **magenta** components make up the R&D electronics subsystem.

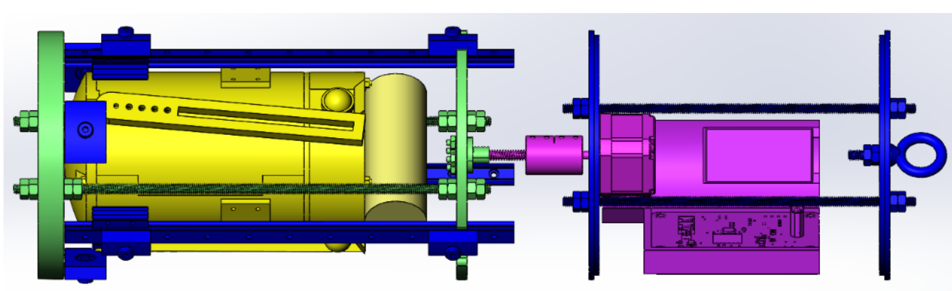


Figure 4.9: R&D Color Coded (Closed Configuration)

During descent under main parachute, the Payload Bay is oriented nose-down (left side of the above figure is the nose). When the rocket is at the appropriate deployment altitude, the R&D electronics subsystem will first signal for the stepper motor to unscrew the lead screw from the lead screw nut on the Pizza Table. This allows the Pizza Table, the Lander, and the nosecone to all fall along the length of the Payload Bay until the Pizza Table eventually collides with the three radially mounted slide stops on the Payload Bay walls. When the Pizza Table is at this position, the nosecone and Lander will be fully clear of the rocket airframe. The three rails do not extend past the top of the Payload Bay, so the Lander will be unconstrained on two sides. The bottom of the Lander contains convex geometry and will, without supports and minor perturbance, fall to one of its sides and away from the rocket, thus being deployed.

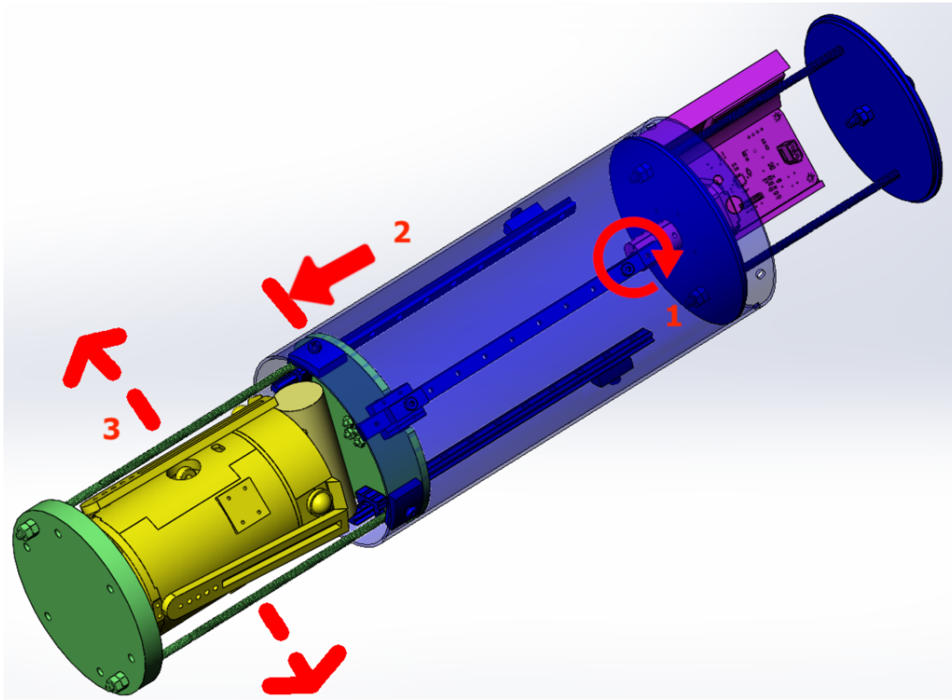


Figure 4.10: R&D Color Coded Deployment Sequence (Open Configuration)

Considering that the Lander has a parachute of its own, the team feared that this simple deployment sequence would result in immediate entanglement with the launch vehicle. To avoid such a failure mode, the team has researched a technology known as a parachute deployment bag. Traditionally, these deployment bags are loaded into the black-powder ejection sections of model rockets with the associated parachute inserted inside. Their aim is to retain the parachute until it has gained enough displacement from the rocket body. They are designed to easily release their parachute once held taut. That said, the team has decided to implement one of these devices into the R&D bay. Seen above the Lander's cupola is a cylindrical stand-in for the parachute within its bag. This bag will be fastened to the Pizza Table by means of small nylon rope. Upon deployment, the nylon rope will be long enough to release the parachute once the Lander has left the keep-out distance determined by Subteam Requirement S.P.1.2. Initial testing of this device suggests that the parachute bag provides very little resistance to the parachute itself once taut, however there is a slight probability that the parachute becomes stuck within this bag. In this case, the team has decided that the Lander could either descend alongside the launch vehicle or release itself near to the ground. While the former condition would fail Subteam Requirement S.P.1.2, it would follow that the Lander descends at a non-ballistic rate towards the ground, maintaining safety. In the latter case, the parachute would need to open and slow the Lander down quickly in order to avoid potential destruction or harm to the ground team. From initial testing, the chosen parachute will open from a standstill from heights as low at 10', without accruing very much speed, further ensuring that the Lander will safely make it to the ground in either scenario. For additional details about the chosen parachute and its associated parachute bag, see the Descent and Landing Subsystem section below.

4.2.2.1.2 Mechanical Testing

Preliminary tests of the R&D design have proven useful in several ways. Firstly, by applying both constant and impulse loads to a test assembly consisting of the Pizza Table, the lead screw, and the stepper motor, the team has determined back driving is highly unlikely to interfere with the system in a meaningful way. As when tested with constant loads approximating the flight conditions, no back driving was observed. When large impulse loads were tested, only trivial back driving occurred. The tests did reveal a potential issue, however, with a discrepancy between expected rpm and actual rpm with the stepper motor as the actual rpm observed was lower than expected, depending on the torque setting used. The team has accounted for this by testing a wide variety of speed and torque settings and measuring the actual rpm so that better predictions of rpm can be made. Additionally, the team determined that the lower-than-expected rpm wouldn't prove an issue as the unscrewing can simply be triggered earlier to compensate.

Final determination of the backdrive performance of the R&D will come from additional deployment testing. In order to more definitively address this loading uncertainty, the flight loads to be utilized in the final testing procedure will be sourced by calculations performed at the times of flight most likely to cause rapid deceleration of the Payload Bay, thus leading to backdrive. To get an estimate of these loads, the Payload Team looked for guidance from the Avionics team's Simulink model, as described in their Trajectory Analysis section. From physical intuition, the team determined that the points of greatest backdrive load would occur during launch vehicle drogue parachute deployment and main parachute deployment. The Simulink simulated acceleration experienced by the CoM of the launch vehicle can be seen below:

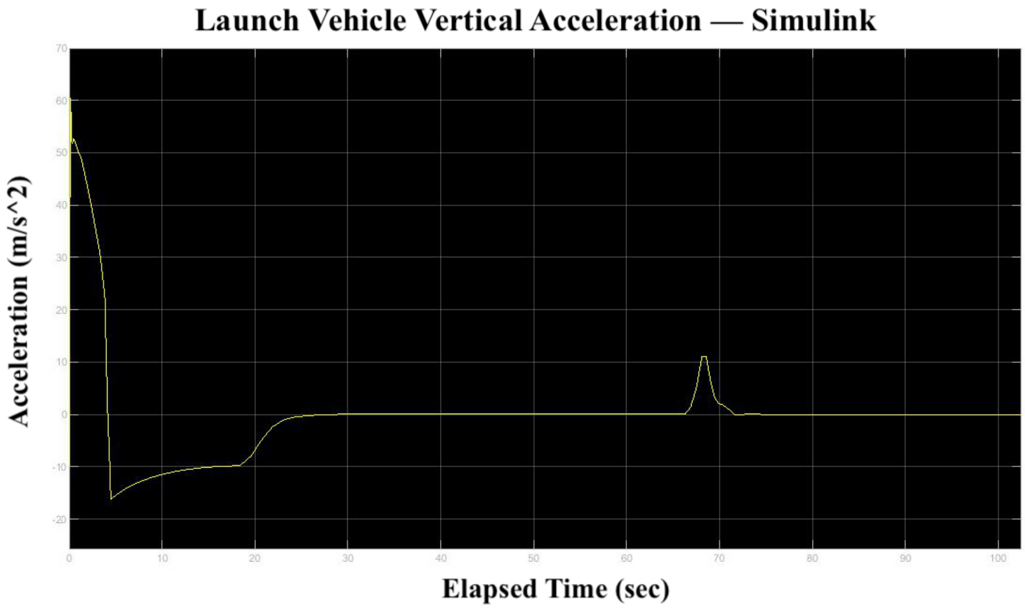


Figure 4.11- Launch Vehicle Vertical Acceleration During Flight — Simulink

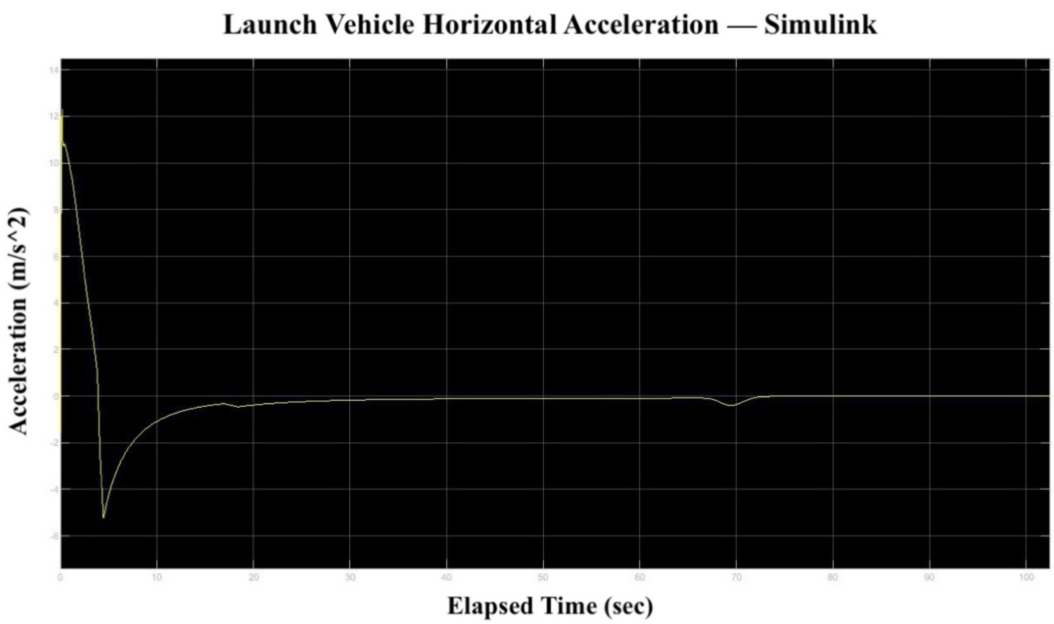


Figure 4.12- Launch Vehicle Horizontal Acceleration During Flight — Simulink

Taking a dynamics approach, the launch vehicle should always maintain a vertical acceleration equal to the local gravitational acceleration, g (approximately $9.81 m/s^2$ or $32.2 ft/s^2$ near Earth's surface), after burn and ignoring drag. Observing the accelerations experienced by the launch vehicle in the above graphs, the maximum acceleration experienced after the vehicle has finished ascending is approximately 2g during main parachute deployment; this quantity is about twice that of the terminal velocity acceleration forces experienced under drogue. Furthermore, due to Subteam Requirement S.P.1.6, the Lander must be deployed after the deployment of the main parachute; this means that the failure condition of R&D backdriving under main still satisfies all designated requirements. Since backdriving the stepper motor will not cause immediate harm to the R&D electronics, this scenario was deemed acceptable by the team.

On the other hand, the team remains wary of the acceleration forces experienced under drogue. Luckily, as just discussed, the main parachute accelerations are predicted to be twice that of drogue; this makes sense, as the drogue parachute approximately begins producing drag with only the horizontal drift velocity of the launch vehicle at apogee and then settles higher towards the drag of terminal velocity. Essentially, the drogue parachute begins with minimal force and converges upon the weight force of the launch vehicle. That said, if the R&D system is able to handle the loads associated with main parachute deployment, it follows that it should be able to handle the flight loads of drogue. The Payload Team will take this observation into account during descent testing.

While Simulink was not utilized to determine quantities that would be unable to be generated through a simple static problem, it provided a solid basis for investigation and confirmed the results of static analyses performed by the team. Additionally, it brought to light the knowledge that the acceleration experienced under drogue is actually slightly greater than 1g as suggested before; since the launch vehicle gains a significant drift velocity throughout its flight, a horizontal component of approximate magnitude $0.4 m/s^2$ is also experienced. While this quantity has been considered far too insignificant to cause issue with the proposed testing solution (~5% of g), it makes it clear that a full safety factor of 2.0 will not be achievable with the current configuration. A required safety factor of 1.5 is entirely possible, however.

4.2.2.1.3 Electronic Hardware

The R&D system electronics serve the purpose of retaining the Lander inside the rocket until the target altitude of 700 feet is achieved. At that point, a stepper motor drives a lead screw that releases the payload. To accomplish this, the electronics follow the basic design shown below.

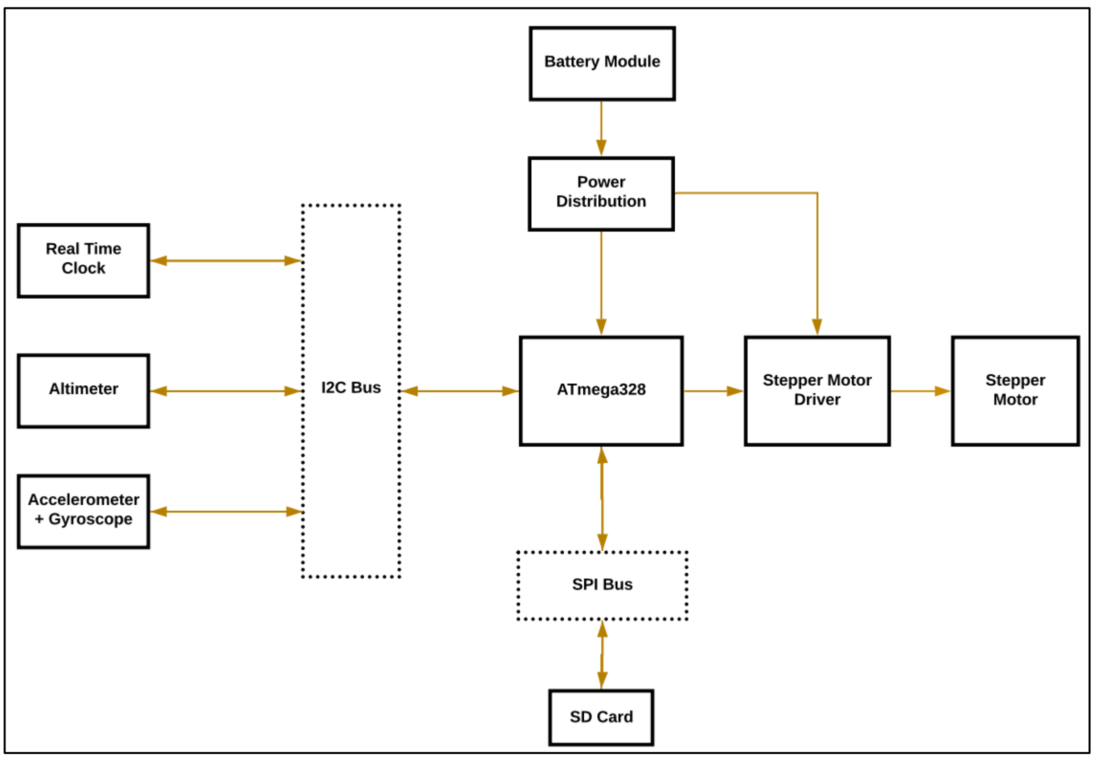


Figure 4.13: R&D Electrical System Overview

The R&D microcontroller handles all the data processing and control logic for the R&D system. It interfaces with the sensors and motor to enable it to control each part of the system. The microcontroller was chosen to be the ATmega328. This was due to its ease of use, low power consumption, and the fact that it has enough IO pins for the task.

In order to detect the wanted payload release point, the team has decided to implement two sensors, an altimeter and an accelerometer. The chosen altimeter is the BMP280 pressure sensor, which will act as the primary sensor. This was chosen due to its ease of interface and reliability. Additionally, the BMP280 was tested in the team's subscale launch, which performed as expected. The ATmega328 will read from the BMP280 to determine the altitude of the payload. When the target altitude is reached, the ATmega328 will drive the stepper motor. In case of a read failure or some other error, an accelerometer was also implemented. The

LSM6DS33 was chosen as the accelerometer. Like the BMP280, this was also tested in the subscale launch. The purpose of the accelerometer is to serve as a backup for signaling when to drive the stepper motor. The main parachute will deploy at an altitude of 900 feet, which is near the payload deployment range of 500 – 700 feet. The accelerometer will be able to detect the main parachute deployment and utilize acceleration measurements as well as the real time clock to determine the appropriate release point.

In addition to the sensors, a real time clock was also implemented. The PCF8563 was chosen because it has few complimentary components, communicates via I2C, and has a programmable interrupt. The real time clock was added for the purpose of accurate time keeping. It also serves to timestamp the data logs. The data collected by the R&D system will be logged on the SD card. This way, the data received by the sensors can be reviewed after the launch. It also serves as a method of verifying that the system works during testing, in that the team can review what the system experiences and make adjustments accordingly.

The motor driver has undergone a recent change. Originally, the team decided to use the DRV8833 motor driver. After testing it, the team discovered that there was an issue with thermals. The chip would overheat and cause it to go into thermal shutdown, which would turn it off until safe temperatures were reached. After putting a heat sink on the chip, it would still go into thermal shutdown in less than 30 seconds when keeping the motor in a stationary position. The team decided to switch to the A4988 motor driver after discovering this issue. This motor driver was chosen because team members have had experience with this driver in the past, and it was readily available for testing. After more testing, the A4988 did not run into the thermal issue that the DRV8833 did. A useful feature that will be utilized is the sleep function of the chip. When put in sleep mode, the motor driver draws almost no power, which will be useful for conserving power when on the launch pad. The motor driver will be in sleep mode until the accelerometer detects launch. When launch is detected, it will be moved out of sleep mode in order to utilize the holding torque of the motor.

The motor the team has chosen for the R&D system is the Nema 17 17HS15-1684S stepper motor. A stepper motor was chosen because of its holding torque. The payload will be held in place by a lead screw attached to the motor. The team will utilize the stepper motor's holding torque to prevent the lead screw from spinning while the rocket is in descent. The holding torque of the stepper motor will prevent the lead screw from spinning too soon, which would deploy the payload above the target release window.

Figure 4.14: R&D Battery Selection

The battery for the R&D system will be a 7.4 V Lipo battery. It has a capacity of 1500 mAh, which will be enough power to satisfy the need for 2 hours of launch ready state before the launch and have plenty of power left to drive the stepper motor during flight.

Figure 4.15: R&D Key Switch

The switch between the power and the rest of the R&D system is a key switch. This ensures the switch is not accidentally flipped before launch. Additionally, the switch is rated to handle 1 amp at 125 VAC, which is much more power than the R&D system will ever draw.

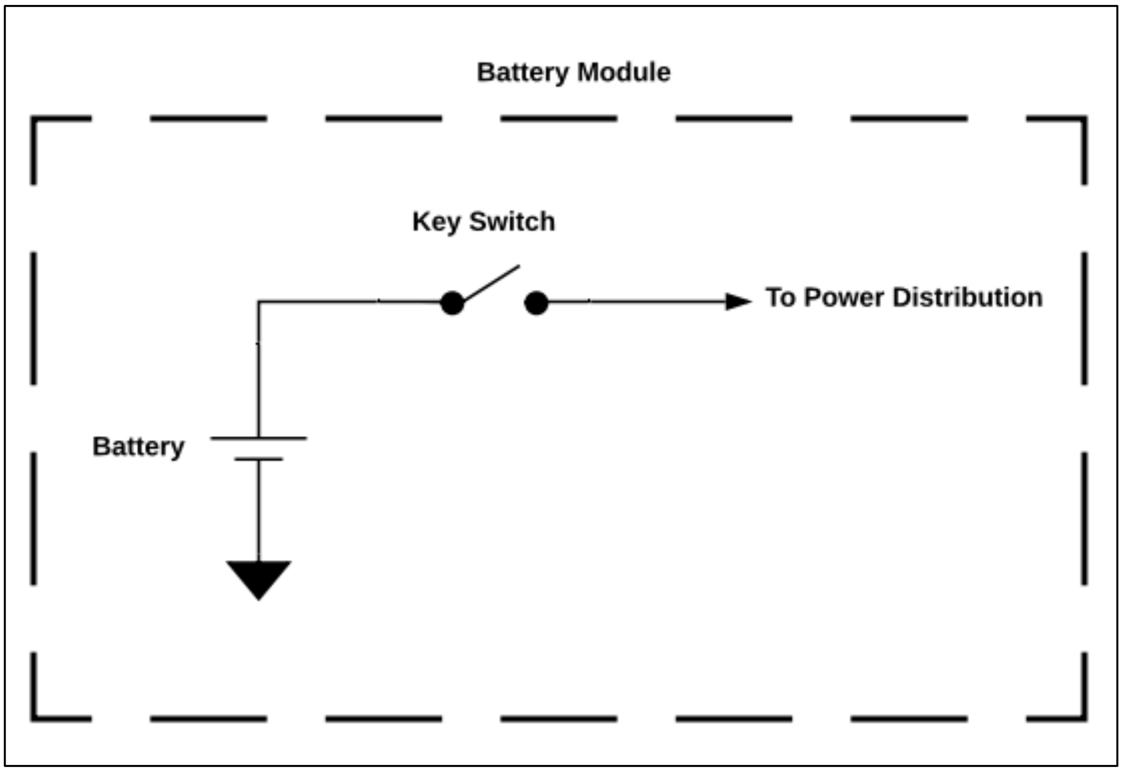


Figure 4.16: Battery Module Schematic

The R&D system will keep its battery module separate from the R&D PCB. The battery module will consist of both the battery and the key switch. The key switch is wired in series with the battery. This allows the team to control when the R&D system has power. This also contributes to safety. If the system has no power, there is no risk of shock to a team member.

The following pages contain the full R&D electrical schematic. Due to its size, it is broken into two pages, even though the full R&D electrical system is contained on one PCB.

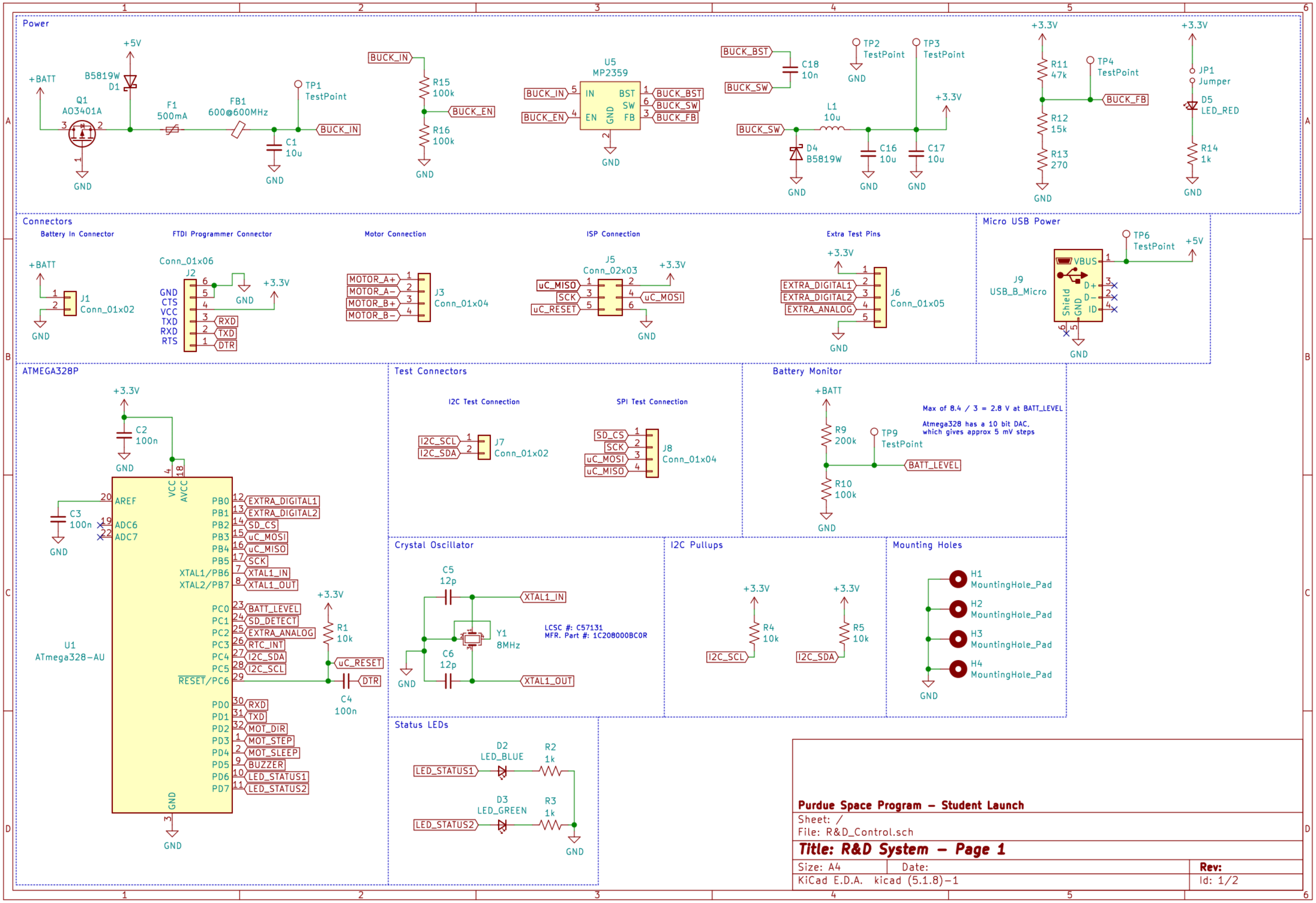


Figure 4.17 Electrical Schematic for Retention and Deployment System Page 1

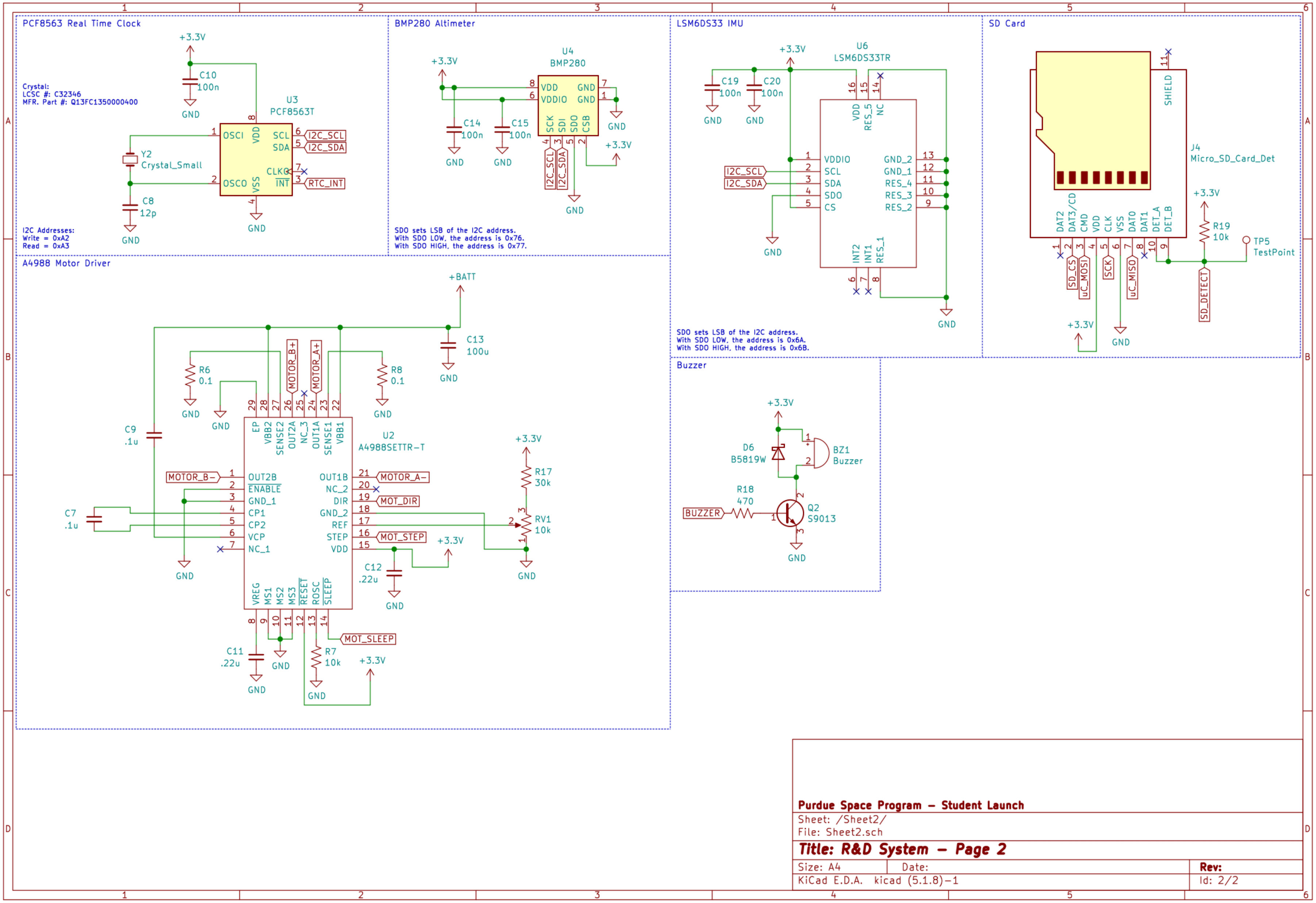


Figure 4.18 Electrical Schematic for Retention and Deployment System Page 2

4.2.2.1.4 Software Design

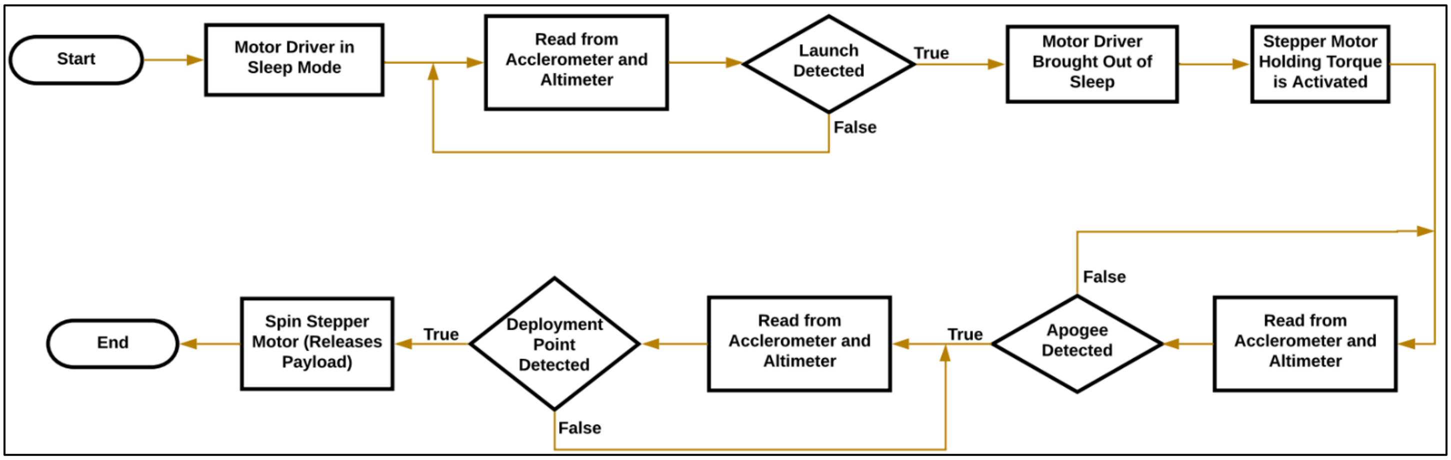


Figure 4.19: R&D Software Diagram

The goal of the R&D software is to wait until the appropriate time and then release the payload. The inner workings of this goal are slightly more complicated. The software can be broken into two distinct phases: before launch and during launch.

Before launch, the software keeps the R&D system in a state of readiness and power conservation. To do this, it puts the motor driver into sleep mode. All other components are kept out of sleep mode and in normal operation. This is so that they can detect the launch. If there is a rapid increase in magnitude of acceleration or a rapid increase in altitude, the system will be able to identify the occurrence of launch. This is important because the software can now pull the motor driver out of sleep mode and activate the holding torque of the stepper motor so that the payload does not prematurely deploy.

During launch, the software monitors for key points and maintains the holding torque of the stepper motor. The software will monitor the altimeter and wait until the appropriate altitude of 700 feet is reached during descent. Once the altitude is reached, the software will signal the stepper motor to spin, causing the payload to deploy. As a backup signal, the accelerometer will detect both the drogue and main parachute deployments. It will ignore the drogue parachute deployment, as that occurs too high to deploy. If, for whatever reason, the software fails to identify the altitude from the altimeter, the accelerometer will detect main parachute deployment, which occurs at 900 feet. This is near the payload deployment range of 500-700 feet. Therefore, the system can use the accelerometer and the real time clock to determine the approximate altitude of the system, basing off the main parachute deployment altitude of 900 feet. After the payload is deployed the R&D software has served its purpose and completed its task.

4.2.2.2 Descent and Landing Subsystem

During the time between PDR and CDR, the design of the parachute as well as its detachment method have undergone changes to better detail how the parachute will integrate with the rest of the Lander. In particular, the parachute was changed to a 2-foot Fruity Chutes parachute because it allowed the Lander to reach a target terminal velocity of 20 fps, and the Lander detachment method was given redundancy in its implementation for testing. While the initial nichrome-based solution appears promising in conceptual prototype testing, there are increasing questions about the solution's ability to complete its task relative to proposed alternatives. Since the safety measures included in any of the proposed final designs have been discussed at length and solidified, the team feels that this solution can be maintained and tested before the first Payload Demonstration Flight without much concern. If this particular design fails, the Lander could possibly be unable to complete its mission but will still have been tested to prove that it would not jeopardize the safety of the ground team.

4.2.2.2.1 Selected Parachute

The Payload Team decided to use a Fruity Chutes standard 24-inch parachute to slow the descent of the Lander to about 20 fps. This decision was made as it allows the Lander to travel faster than the rest of the rocket, preventing the Lander from colliding with the rocket. The parachute will be tied to the Lander via a 50 lbf rated nylon rope and two bowline knots—one on the parachute and one on an eyebolt of the Lander. The Payload Team decided that such a rope would be able to withstand the forces acting on it by performing a free-body analysis of the Lander as it drops from the Payload Bay. For analysis purposes, the team assumed that the

Lander drops for 1.5 seconds before the parachute fully opens, that the parachute has a coefficient of drag of 1.55 (according to Fruity Chutes), the density of air is 1.1015 kg/m^3 (at an altitude of 1100 ft MSL where the Lander will be dropped), that the parachute opens in negligible time upon leaving the deployment bag, and that the nylon may experience a maximum force of 50 lbf before yielding. The team performed this analysis by equating the tension in the nylon with the drag force on the parachute at the instant of opening, then the team used the drag equation to find that the expected tension would be 12.425 lbf with a nylon factor of safety of 4.02. This is to say the Payload Team expects no safety concerns to arise from the use of a 50 lbf rated nylon rope. The stated assumptions utilize parachute opening timings that are far more extreme than initial testing suggests.

The parachute will be lightly folded into a parachute deployment bag and stuffed into a 1-inch space at the top of the payload bay. This will prevent the parachute from prematurely opening, which is necessary to ensure the parachute will not tangle with the shock cord attached to the nose cone. The deployment bag will be tied to the interior of the payload bay with a 50 lbf rated nylon cord. This is to say the payload team expects no safety concerns to arise from the use of a 50 lbf rated nylon rope. The parachute will be lightly folded into a parachute deployment bag and stuffed into a 1-inch space at the top of the payload bay. This will prevent the parachute from prematurely opening, which is necessary to ensure the parachute will not tangle with the shock cord attached to the nose cone. The deployment bag will be tied to the interior of the payload bay with a 50 lbf rated nylon cord.

Figure 4.20: Chosen Parachute—Fruity Chutes 24 Inch

Figure 4.21: Parachute Wadded into Parachute Bag

4.2.2.2.2 Lander Detachment Method

The payload team decided to continue with the idea to use 36-gauge nichrome wire to sever the nylon rope upon landing. This system consists of a battery connected via copper wire to around 8 inches of nichrome. This was determined to be the amount of nichrome in order to maintain current in the wire at a stable 400 mA, which was deemed to be the minimum current required to sever the nylon. Further testing will be required to determine the exact quantity of nichrome which will work with the Lander in its grounded state. Using this setup results in a nylon cutting method that takes several minutes in order to sever the rope. The Payload Team intends to try different amounts of nichrome in the future to determine the optimal current for cutting the nylon. Due to concerns that the wire will take too much time or energy to cut, the Payload Team has decided to implement a backup solution if future testing suggests nichrome is not a valid solution. The Payload Team formulated a rack-and-pinion setup that mechanically separates the parachute by moving a bar the nylon rope will be tied to. To highlight the subtlety of this change, the structural design of the D&L the Lander is shown below.

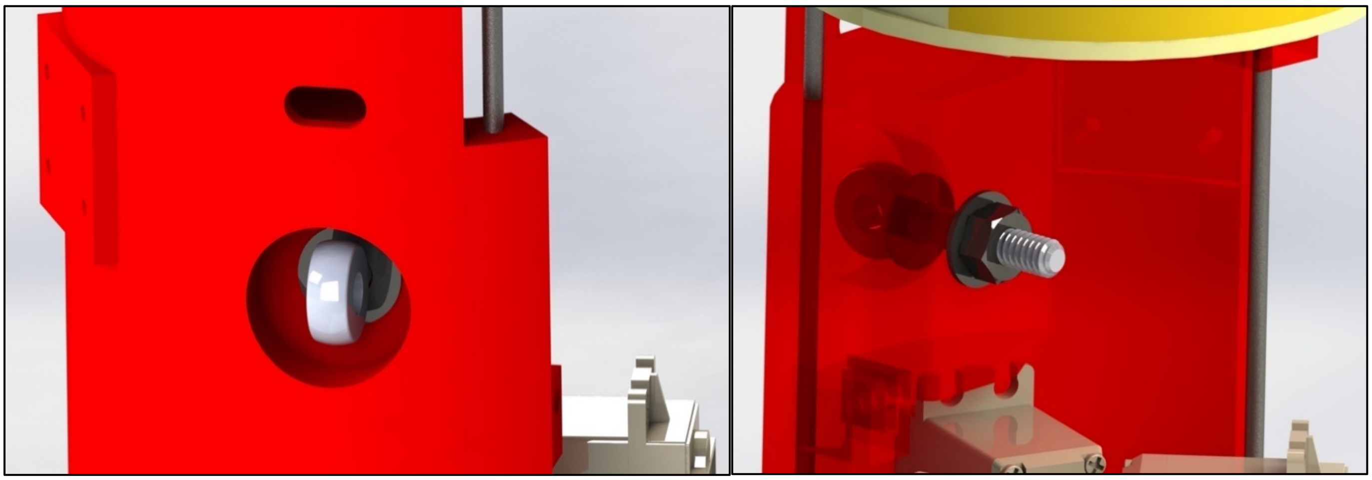


Figure 4.22 Descent Lander Plate (Colored Red, Outside Opaque [Left], Inside Translucent [Right])

In the above figure can be seen the outer portion of the Descent and Landing subsystem. Points of interest include the recessed 5/8" carbon steel eyebolt port and nichrome slot on the wall above. This design has to compensate for competing space needs and optimizes local material strength with access points. Notably, the load applied by the parachute during descent will be transferred into the eyebolt through the internal ¼-20 nut and the load distributing washer and into the Lander wall. In order to maximize the load carrying capacity and mitigate bending rupture, the eyebolt port is located within 30° axially from the nearest steel support threaded rod. These threaded rods form the skeleton of the entire Lander and will act to distribute loads across the other Lander wall plates. Additional testing will be required to ensure that this configuration will be able to satisfy Project Requirement G.2.4.1. Primary concerns in this case include the amount of bend experienced by the Lander walls and threaded rod under loads exceeding 1.5 times that of the Lander's previously mentioned maximum expected descent-slowing load of 12.425 lbf. Additional testing will be required to determine whether the designed structure can handle such loads.

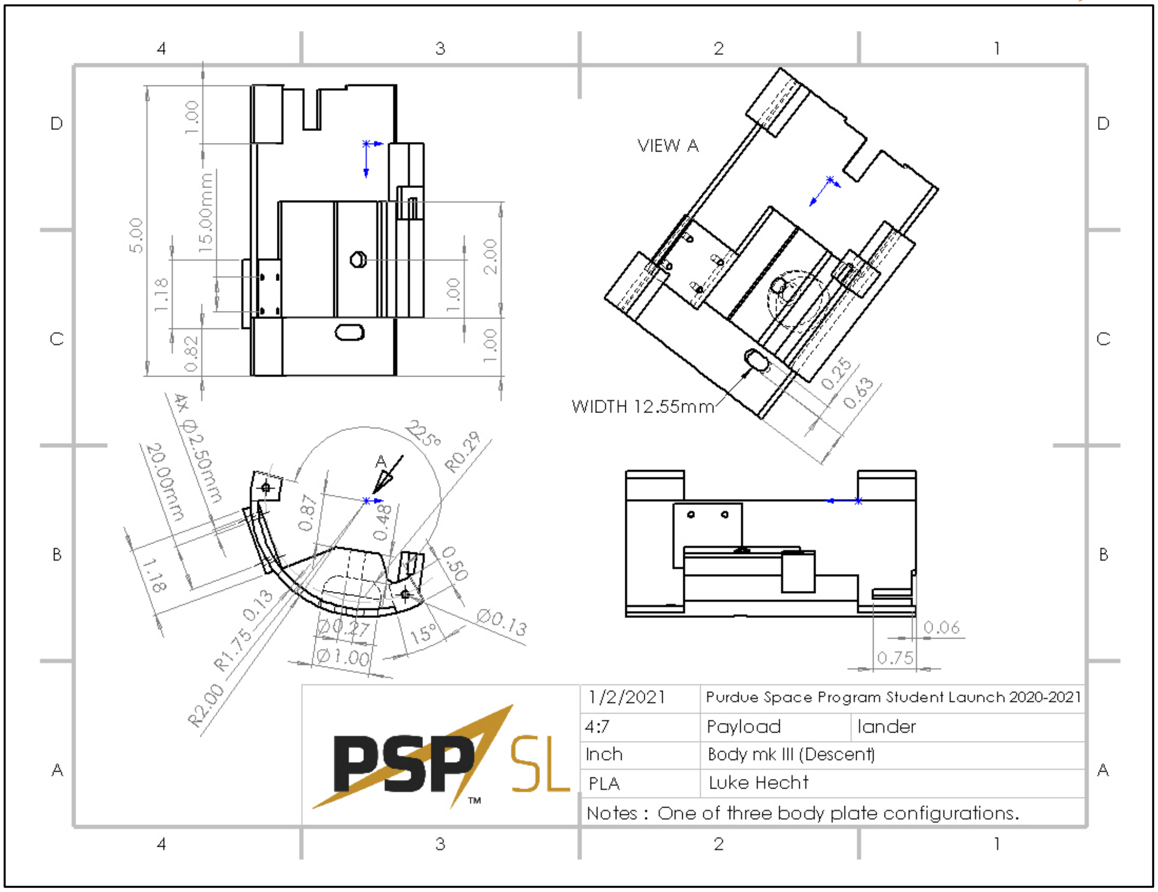


Figure 4.23 Descent Plate Drawing

In the event that Lander D&L nichrome testing does not yield positive results as planned, the team has prepared to be able to undertake a CAD redesign of this system. Since the safety of the ground team is of utmost priority, much consideration has gone into the design of the parachute attachment and release mechanisms. That said, the team has concluded that all designs must at least maintain the level of structural safety employed during descent. In this case, components of concern include the parachute, the nylon rope and its attachment points, and the integration of the nylon attachment point with the Lander body. At this point, the only item from that list that is subject to change is the nylon attachment point with the Lander. This attachment point entirely depends on the method of detachment chosen, nichrome or not. Therefore, even if the chosen parachute release method changes, only the parachute attachment point will need to be modified. This allows for consistent assurance of safety across methods. In preparation for this kind of radical post-test change, the team has already formed a concept for a servo-based sliding-plate nichrome attachment and release solution. This solution would remove the need for nichrome, but it would also greatly disrupt the available space within the lander and its weight distribution. This design does not have formal CAD prepared at this point.

4.2.2.2.3 Lander Detachment Electronics

The electronic control of the nichrome will be implemented on the Primary PCB, discussed later in Section 4.2.2.5.3. The current for the nichrome will come directly from the battery, through a solid-state relay (SSR). The relay will be controlled by the microcontroller in the LCS. The schematic below shows the connections to the SSR.

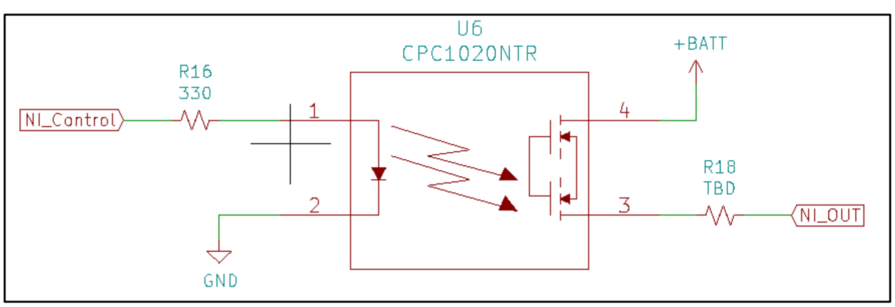


Figure 4.24 Schematic of connections to the SSR

This circuit is also shown in the context of the entire Lander Control System in the primary PCB schematic at the end of this section. The nichrome control pin, NI_Control, goes from the microcontroller through a current limiting resistor, R16, before going into the control side of the SSR. This resistor protects the microcontroller pin and the LED from being damaged by too much current. The relay is normally open, so any power failure in the microcontroller will result in the relay turning off.

When the LCS has determined it is safe to detach the parachute, the NI Control pin will be set, and the relay will turn on. A discussion of how the LCS will determine if it is safe to release the parachute is contained within the LCS Software Design Section below. Once the relay is turned on, current will flow through the nichrome. R18 on the schematic is a current limiting resistor for the nichrome. As the final length, and therefore resistance, of the nichrome will be determined in future tests, the value of this resistor has yet to be determined. It is likely that the nichrome will present enough of a load that R18 will not be necessary, in which case it would be replaced by a 0 Ohm resistor.

The NI_OUT tag on the schematic will go to a connector on the Primary PCB, and the connector will go to copper wire that goes outside of the lander and connects to the nichrome. This means that the nichrome will not be heating up the inside of the lander during operation. This also allows for less nichrome to be used, which will lower the load resistance and increase current.

4.2.2.3 Self-Orientation Subsystem

The Self Orientation Subsystem (SOS) acts to align the Lander with the local gravitational vector within the angular bounds designated by Requirement P.4.3.3 and constrained by Subteam Requirements S.P.1.5 and S.P.1.9 to ensure proper function. The system activates once the Lander is on the ground, and after the descent and landing subsystem has completed all action. The Lander will begin on its side and three servos will actuate simultaneously in a process described below, causing the Lander to orient vertically.

Since PDR, the team has continued the design of the “3-Pinwheel” SOS. The team has created all relevant components, assembled them in CAD, and has devised a solution for attaching all parts to all other necessary parts. All modifications to the lander, such as component geometry for PICS, were made with SOS in mind, e.g., attempted to keep CoM low and reduce total weight of the Lander.

Finalizations of the SOS included the modification of the legs, the addition of a PCB and battery to control and power the servos, other hardware, and an exact method to attach all necessary components. The design has largely remained the same as the initial proposed subsystem, with no radical changes in geometry.

4.2.2.3.1 Mechanical Design

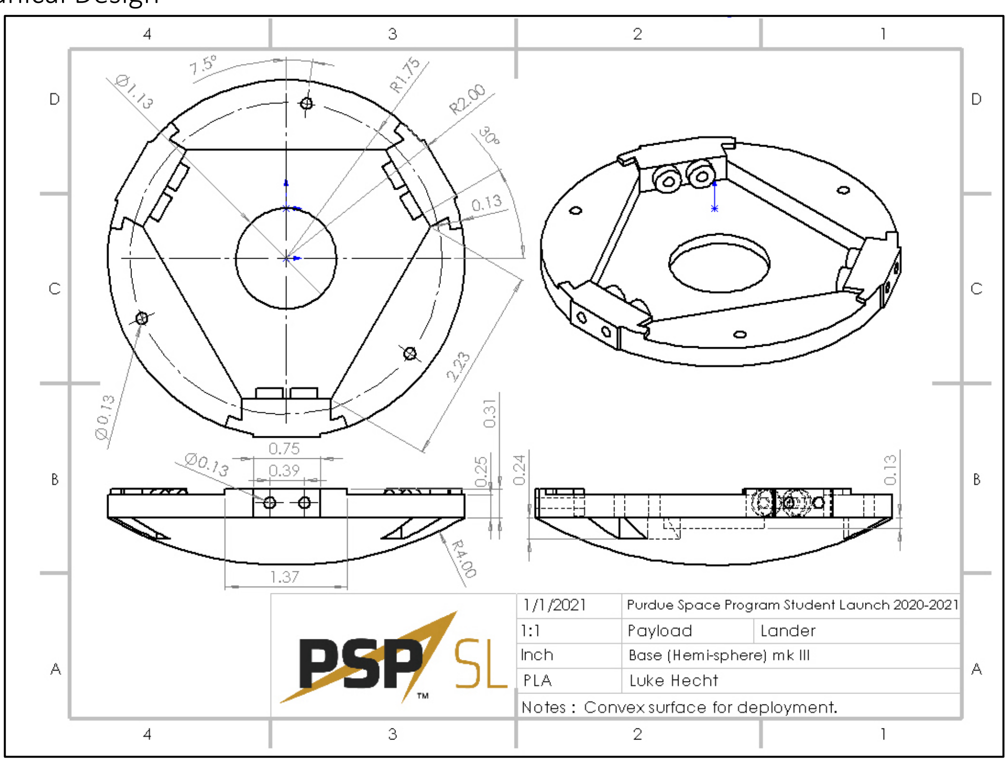


Figure 4.25 SOS Hemispherical Base Drawing

The SOS's main component is the base of the lander, which will be 3D printed. The base contains a convex bottom to assist with R&D as explained above. Further geometry includes mounts for the servos to be screwed into, and three holes for the threaded rod which is intended to hold the Lander together. The team will use three goBILDA 25-2 Servos mounted in a triple axially symmetric manner. These servos will each have an aluminum control horn to match the specific spline of the servo, which in turn will be screwed into 3D printed legs. These legs will begin in an upright position, nearly parallel to the lander's cylindrical form, and will actuate to become perpendicular to the Lander in a process described below.

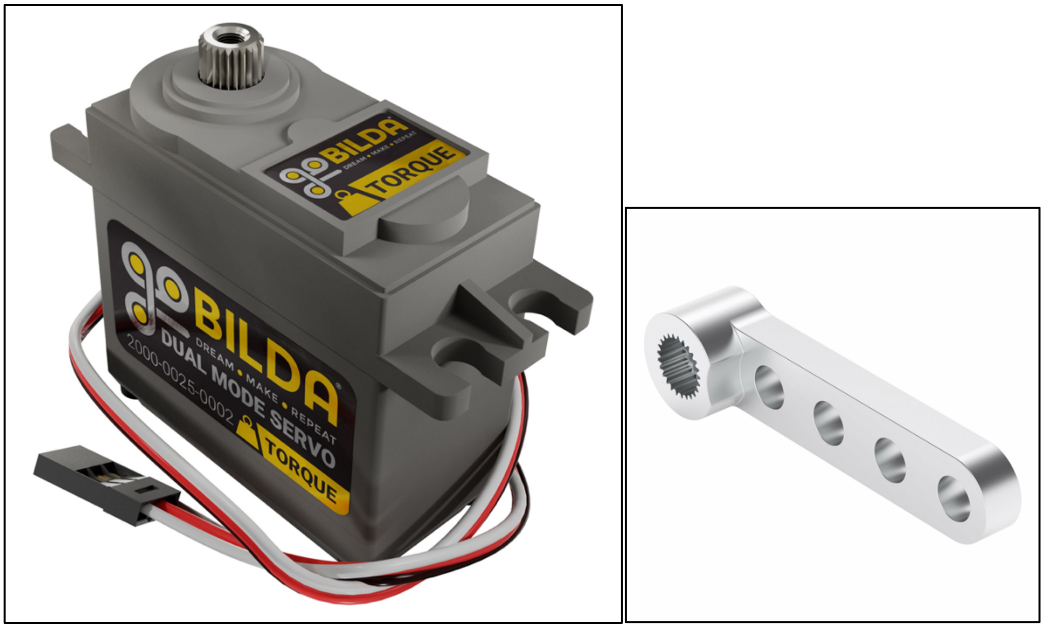


Figure 4.26 goBUILDA 25-2 Servo (Left) and associated 1900 Series Single Servo Arm (Right)

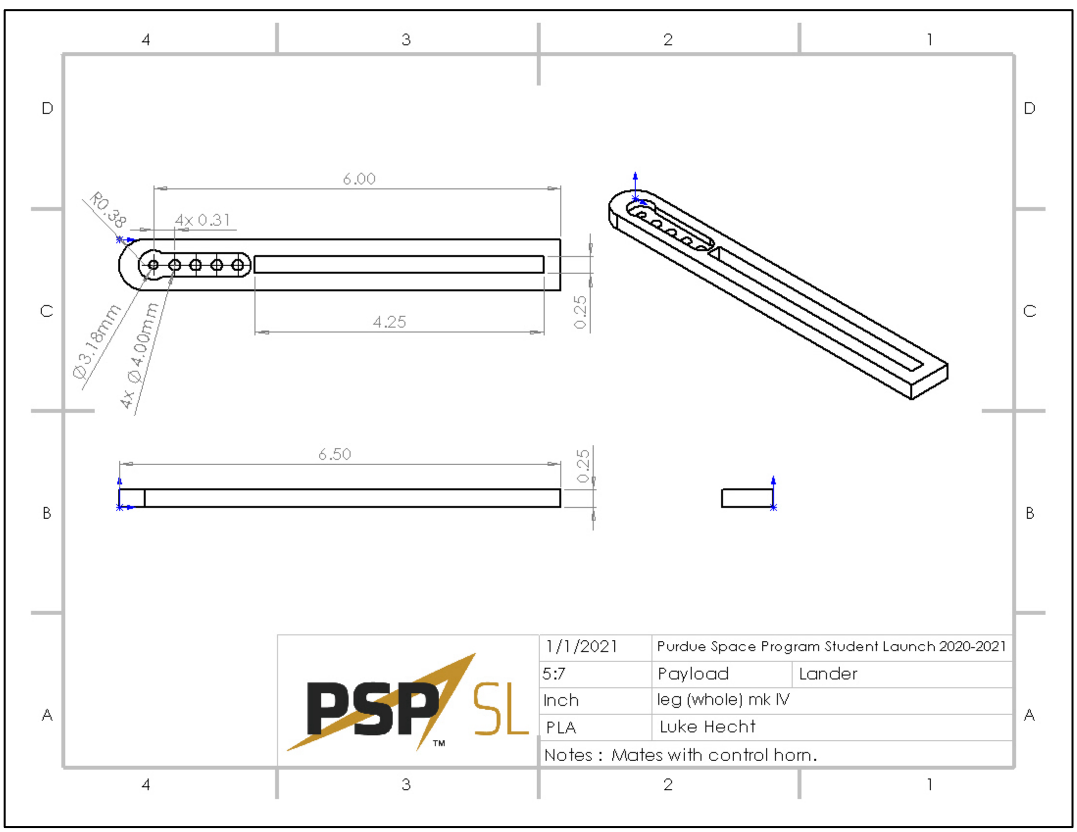


Figure 4.27: SOS Leg Drawing

The shell of the Lander has been redesigned into 3 similar pieces, each containing unique geometry for a specific task. Each piece has a gap for the servo to connect with the control horn through. The shell pieces fit together in a looped hinge-like configuration, with the threaded rods running through their length and keeping them fixed to the base, to each other, and to the cupola. All geometry within the lander, including that which is meant to fix the PCB in place, assist with Descent and Landing, or contributes to the PICS has all been designed with the SOS in mind. The team worked to optimize the Lander by minimizing total mass, center of mass distance, and length of the Lander while still ensuring structural stability and function of all systems.

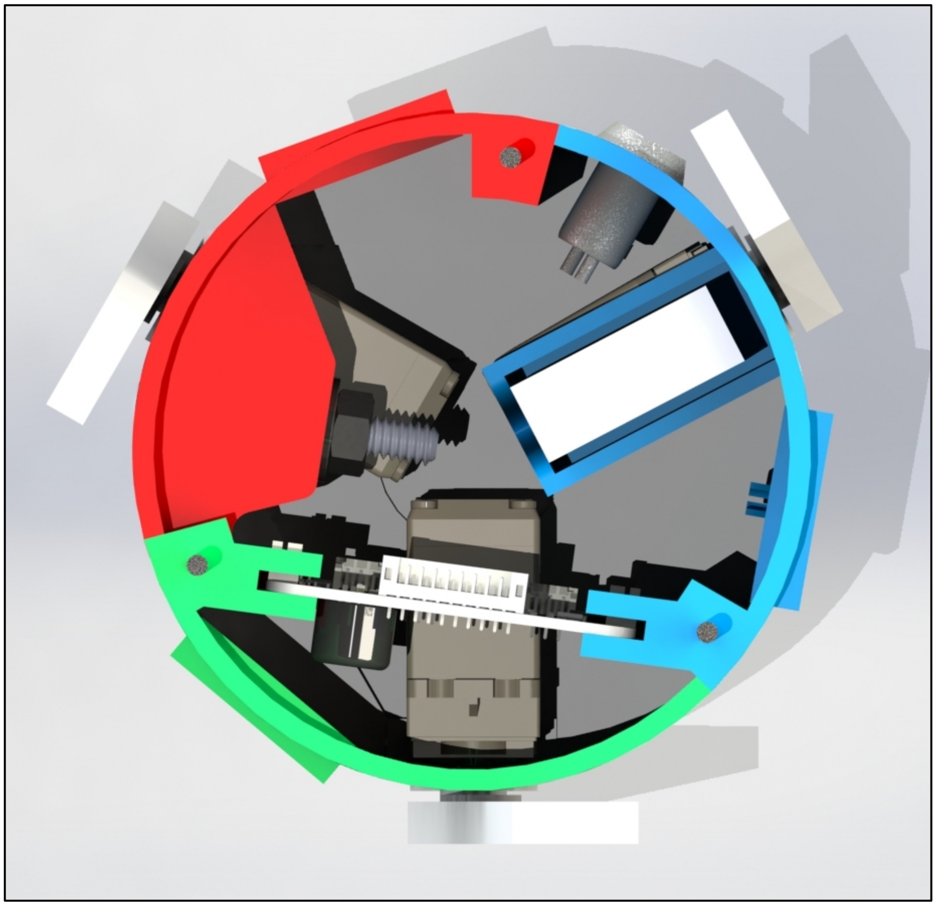


Figure 4.28 Lander Overhead View Showing SOS Shell Structure

The Lander is held together by M3 threaded rods running through the length of its central body. These threaded rods will have nuts on either end to fasten all structural components together. M3 threaded rods were chosen because the team believed they would be adequate in their ability to hold the Lander together, even in possible failure states. Since the majority of the Lander will be 3D printed, including this metal skeleton will provide a consistent level of rigidity without incurring much extra weight. The Lander will have a total length of 7.25” end to end, which is slightly smaller than initially anticipated. This change in length has no negative effects, as a shorter lander is easier to orient for numerous reasons, including a smaller moment of gravity, and better positioning of the center of mass nearer to the center of the base.

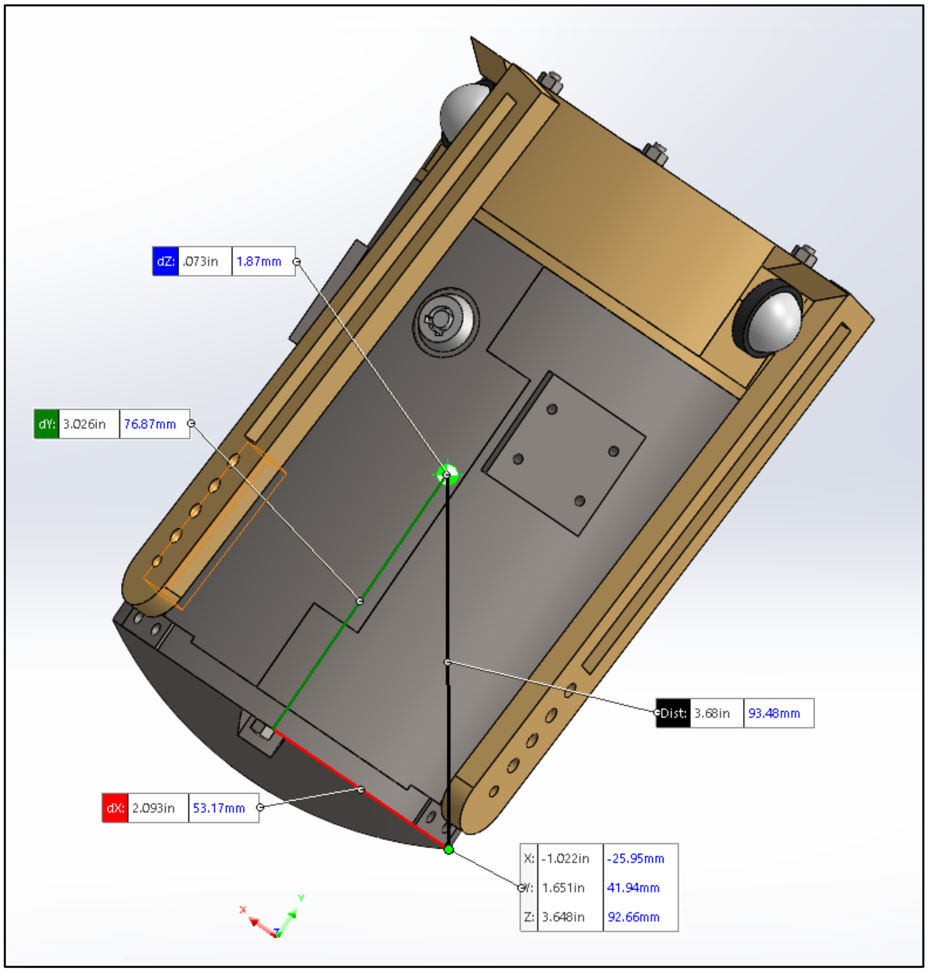


Figure 4.29 Lander CoM Preview

4.2.2.3.2 Software Design

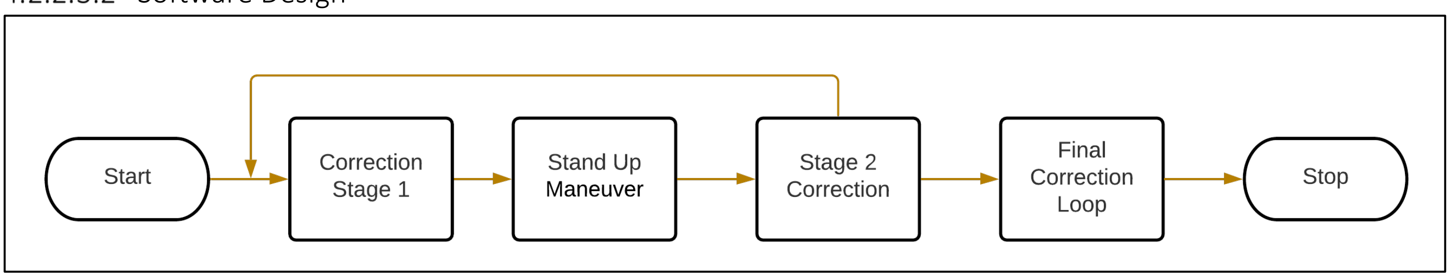


Figure 4.30: SOS Software Overview

The first step of orientation is Stage 1 correction. The purpose of Stage 1 correction is to determine if the Lander is in a position where it can successfully orientate before even attempting any maneuvers. Extensive testing will be conducted to determine in what scenarios the Lander can and cannot safely complete an orientation maneuver. One such example would be if the Lander were on slope that was too steep for the Lander to orientate without falling over—or similarly, any time that the base of the Lander lies at a greater elevation than the cupola. If the Lander attempted to orientate in this situation, it could tumble with the legs extended. This would risk damaging the legs and could jeopardize the success of the mission. In a situation like this, the Lander would perform a corrective maneuver, which would be a quick movement of a single leg in an attempt to get the Lander to roll to a better position. Since the leg would quickly return the starting the position, the Lander would now tumble with the legs retracted, which is much less likely to cause damage. If the Lander does not move, the Lander will attempt the same maneuver with different legs until it moves. Once the Lander moves, the new position is detected, and the detection process is repeated. Once the Lander passes Stage 1 Correction, it will move on to the Stand-up maneuver.

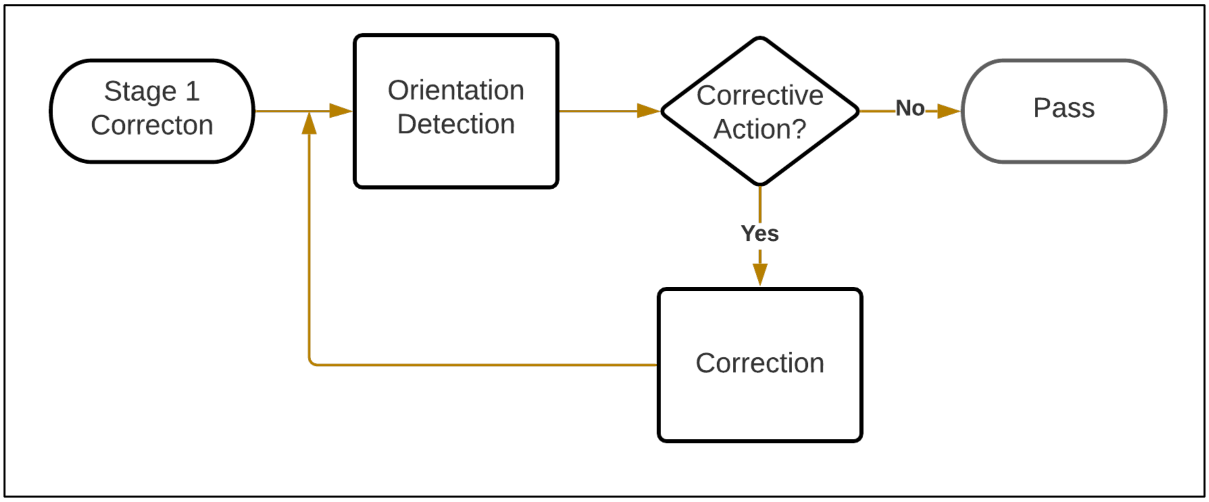


Figure 4.31: SOS Stage 1 Correction Process

The purpose of the stand-up maneuver is to get the Lander from laying on its side to an upright position. To accomplish this, all three orientation servos will be set to move concurrently to the same pre-determined "standing" location. This motion was proposed by a member of the team and confirmed through preliminary mechanical testing to perform correctly under most normal circumstances as ensured by Stage 1 Correction. Once upright, the Lander will move into Stage 2 Correction.

The purpose of Stage 2 Correction is similar to Stage 1. Now that the Lander is upright, the Lander will detect if the orientation target, within 5 degrees of vertical, is within the range of motion of the servos. Ideally, through testing, the team will be able to have Stage 1 Correction detect most of these situations. However, Stage 2 is still necessary to ensure that the Lander does not end up in an endless loop trying to perform the final correction maneuver as a result of a failed stand-up maneuver. If the Lander does not pass Stage 2, then the Lander will retract the legs, perform a corrective maneuver, and then return to Stage 1 Correction. If the Lander passes Stage 2, it will move into the Final Correction Maneuver.

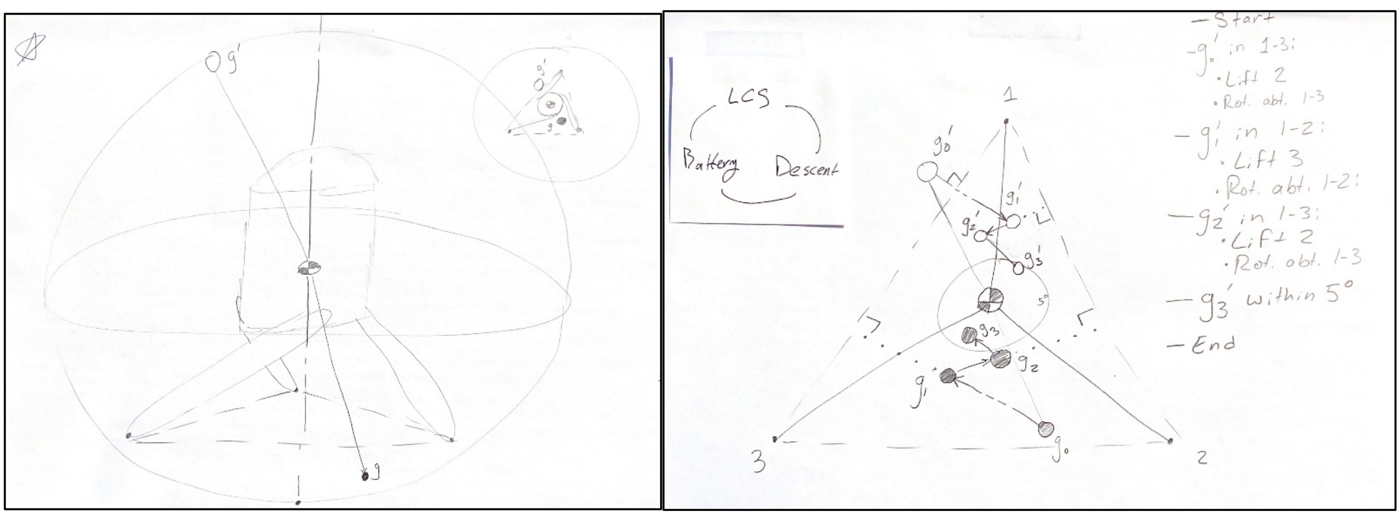


Figure 4.32 Preliminary SOS Final Correction Algorithm Sketches (Overview [Left], Polar Projection [Right])

This final maneuver is considered the most challenging operation by the team, seeing as there is a high level of precision required to achieve a 5° angular tolerance from the local gravitational vector—the vector being defined as vertical. While the team has not constructed any formal code prototype of this operation, the team has discussed a possible implementation. Above is a preliminary conceptual sketch of this operation. The procedure involves recognition of the local vertical gravitational vector by the Lander's IMU. This vector is referred to as g' in the diagrams. In this scheme, the LCS must first recognize the location of the surface contact points of its legs; using these contact points, the top-down polar projection of the Lander can be constructed with the CoM as its center and three reference areas bounded by projected unit vector lines to each contact point. With this projection prepared, the LCS will actuate the SOS servo opposite of the current region bounded by the two nearest unit vectors to g'. This will cause the Lander to lift in a single direction, moving g' towards the vertical. Checking after every step, if the LCS detects that it has moved into a new bounded region, it will change which servo it actuates. This also will require recalculation of the contact point unit vectors since legs move after each operation. In this manner, the SOS is likely to orient in one direction initially and then alternate between two servos as it converges upon the vertical. While this algorithm is likely to require intense calculation, the team recognizes that it is best to proceed with this operation as slowly as possible to avoid loss of balance.

4.2.2.3.3 Mechanical Testing

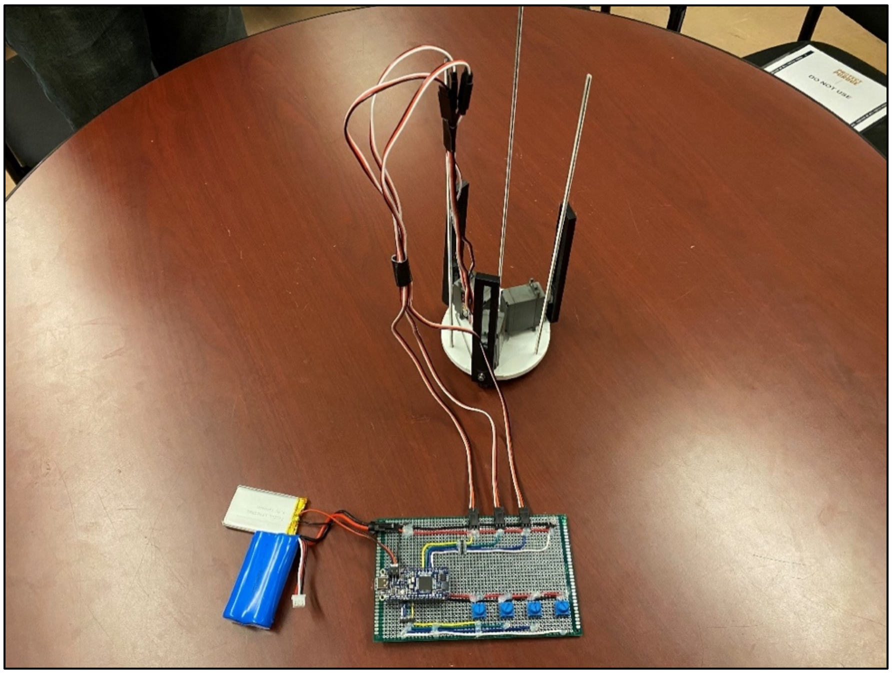


Figure 4.33 Initial SOS Prototype Testing Apparatus

In order to verify the SOS orientation concept proposed by the team, the team constructed a skeleton model of the final design and connected it to a customized servo test board. Together, the team was able to present this model with new and interesting terrain challenges in order to learn its behavior. In the end, the team has determined that this proposed design will function as intended, though further testing in a natural environment will be necessary to verify this statement.

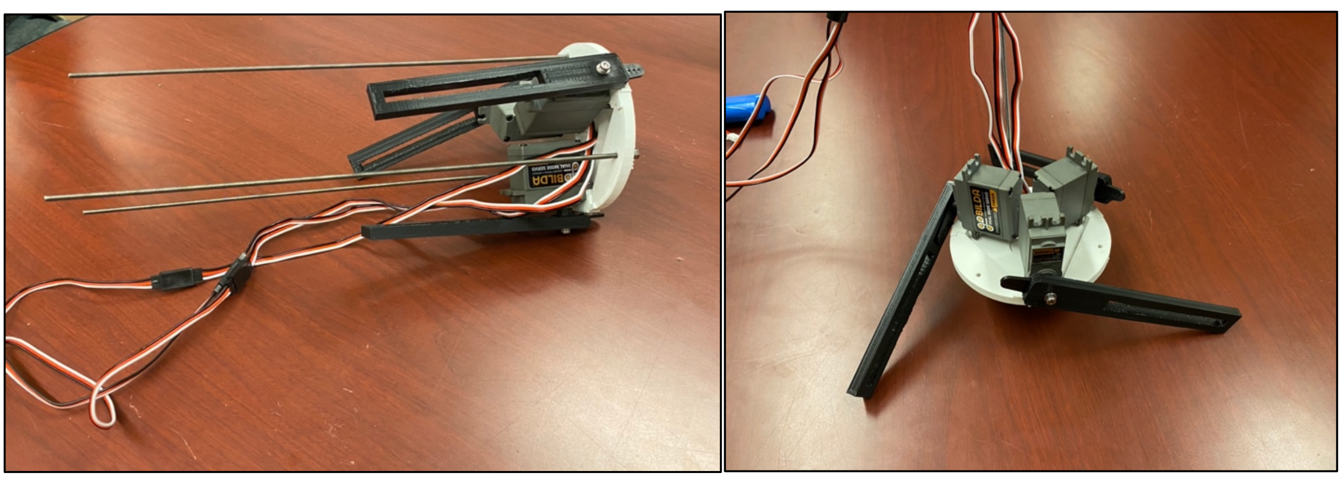


Figure 4.34 SOS Orienting on a Flat Surface

As expected, the SOS performed the best on a flat surface where all three legs have the opportunity to contact the ground. By simply turning all three control knobs equally, the team was able to cause the SOS to upright upon its base. In order to test this design further, the team introduced slanted surfaces in order to simulate farmland terrain—something that returning team members remember posing problems for rover designs in prior years. In this case, the SOS appeared to perform just about the same once it had descended into the trough. While slanted surfaces required a bit of shuffling in order to achieve a proper stance, the team felt that this such shuffling would be reasonably implementable in code.

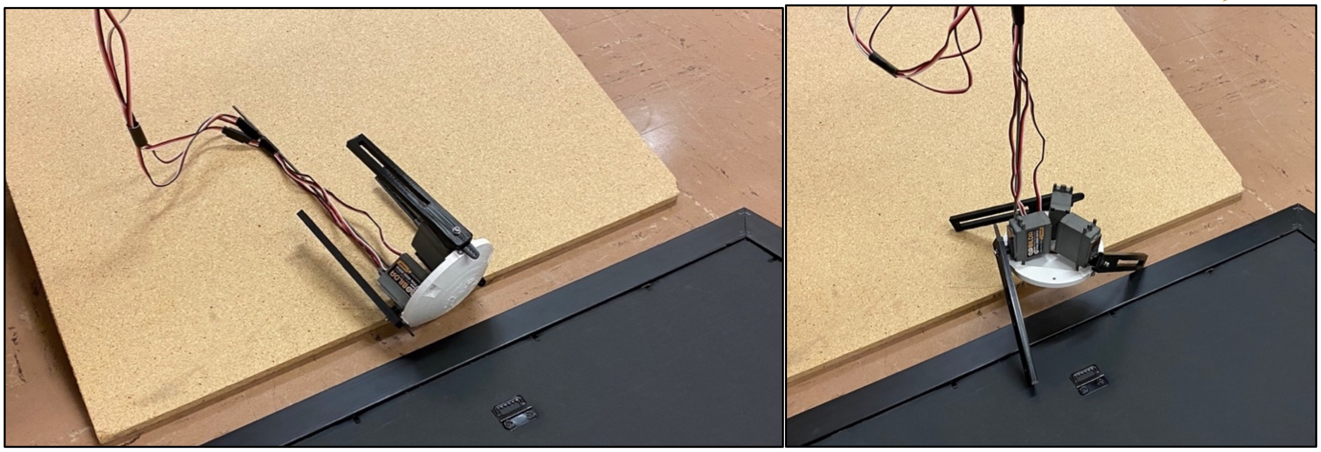


Figure 4.35 SOS Orienting on Roughly Simulated Farmland

4.2.2.4 Panoramic Image Capture Subsystem

The Panoramic Image Capture Subsystem (PICS) will be responsible for capturing the images that make up the panoramic photo. The entire subsystem consists of 3 cameras, a housing to mount and protect the cameras, and a GPIO extender. All the components of the PICS are connected to the Secondary PCB located at the top of the Lander. The GPS and radio transceiver used by the Lander Control Subsystem (LCS) are also mounted on the Secondary PCB because of its location at the top of the lander.

One of the challenges the team faced while designing the PICS is the height of the camera modules and their effect on the total height of the Lander. Mounting the camera modules directly to the Secondary PCB would have been ideal because it would hold the cameras in place and be easy to connect. However, this caused the PICS to add too much height to the Lander and caused the Lander to be outside the constraints for height. To solve this, the team decided to sink the camera modules into the Lander such that the camera lenses are just above the PICS PCB. This reduced the height of the PICS to fit within the Lander's size constraints.

4.2.2.4.1 Structural Design

The PICS apparatus will be located at the opposite end of the Lander to the SOS base—in a location named the "Cupola" after the viewing deck of the International Space Station. This cupola is positioned in such a way to provide an adequate viewing altitude for the PICS system. After orientation, this section will be located higher than any other system in the Lander. Additionally, the high location of the cupola allows for the centrally positioned GPS to operate with increased efficiency. This part requires placement far away from the electrically noisy main bay.

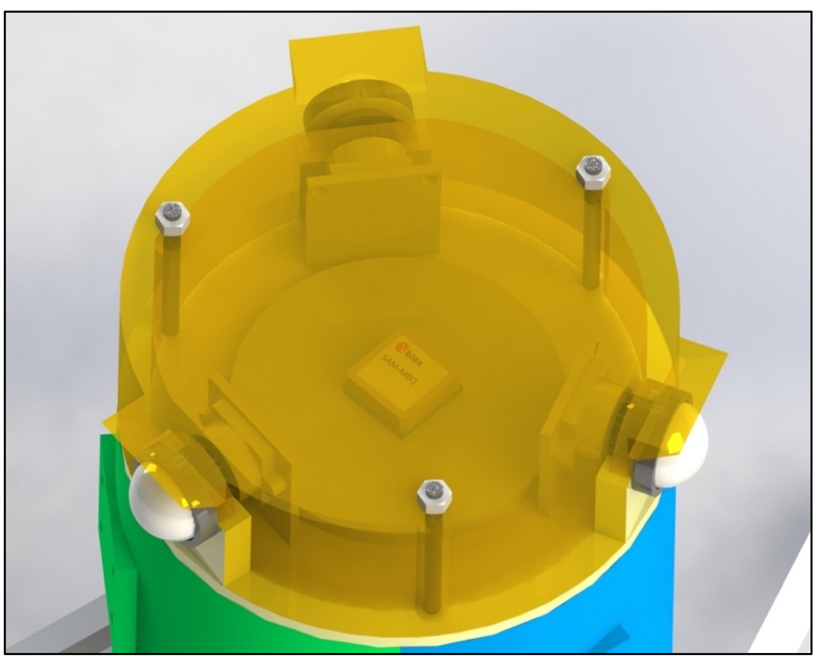


Figure 4.36 PICS Cupola (Translucent Orange), Attachment Plate (Yellow) and internal Secondary PCB

The Secondary PCB has been designed to be circular, matching the shape of the Lander. This PCB will be attached to a circular plate which rests on the main section of the Lander. This cupola-plate arrangement allows for the disassembly of the Lander without the need to disassemble the PICS apparatus. The interface between the Lower Cupola Plate and Upper Cupola allows for mechanical meshing with the PICS cameras for the purposes of support and protection. When the Lander is unbolted, the Upper Cupola may be removed with ease, allowing for modular access to this bay.

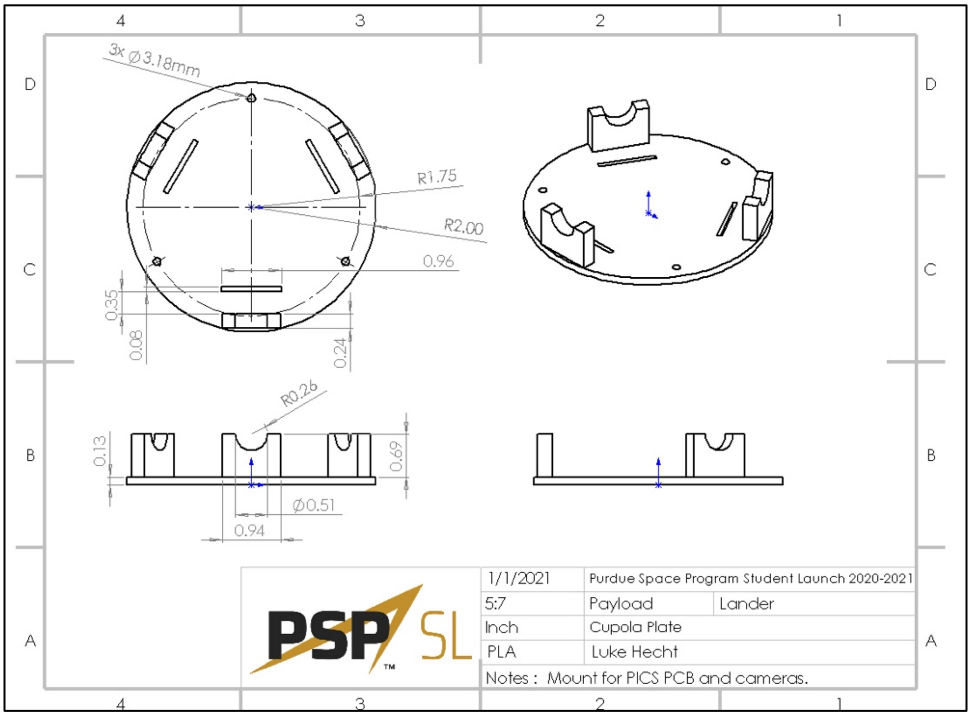


Figure 4.37 PICS Lower Cupola Plate Drawing

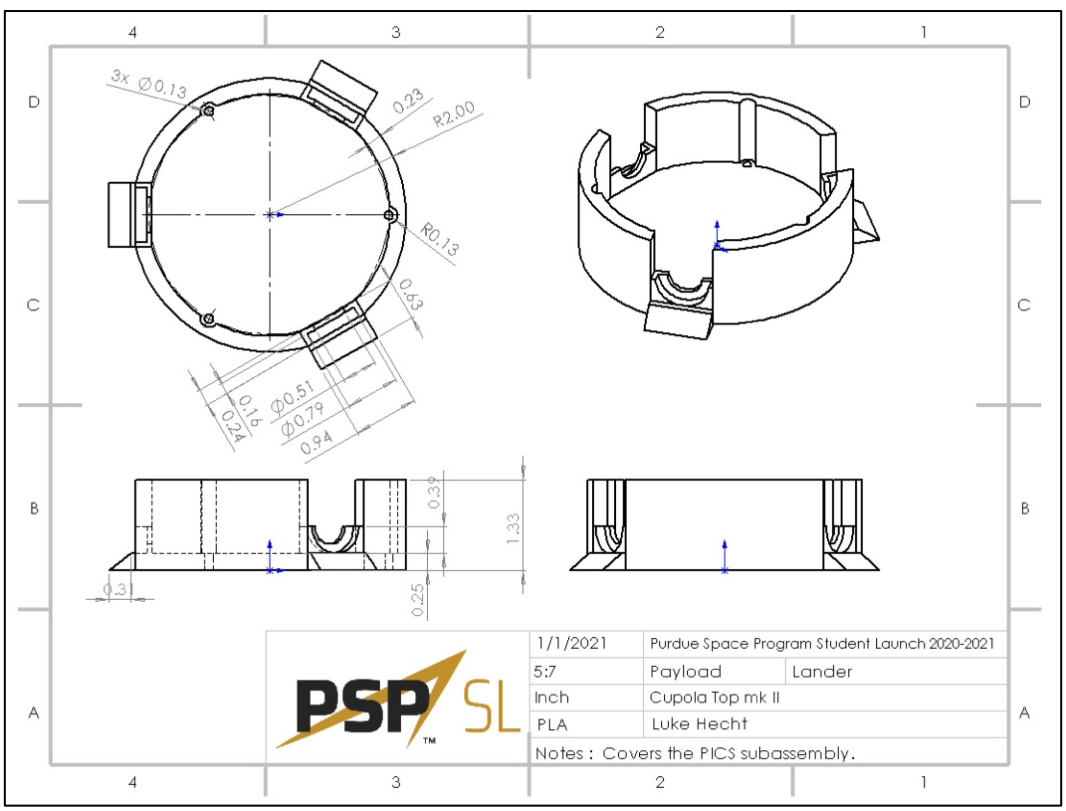


Figure 4.38 PICS Upper Cupola Drawing (Upside-down for Viewing)

4.2.2.4.2 Electronic Hardware

The major components on the PICS are its camera modules. The camera modules that the team has chosen are the Arducam 2MP Plus OV2640 SPI Camera.

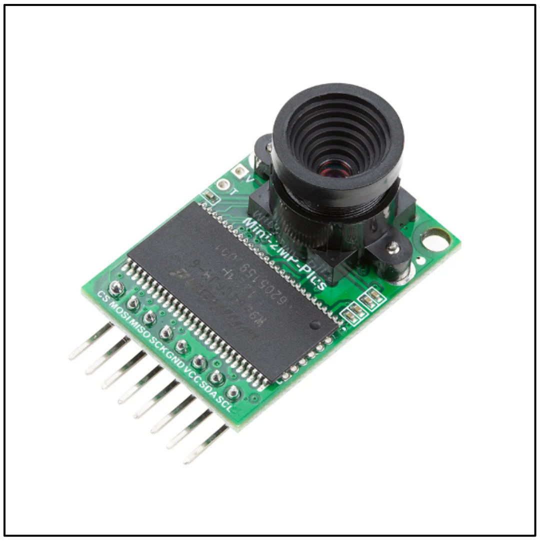


Figure 4.39: Arducam 2MP Plus OV2640 SPI Camera

The module includes an OV2640 image sensor to take the photos and an onboard microcontroller and digital storage unit that handle the configuration and capturing the photo from the image sensor. The camera sensor can be configured via I2C and the image data is transferred via SPI. Since the onboard microcontroller handles most of the interactions with the image sensor, it is easier to integrate the module with the Lander's control system. The tradeoff is that the module is larger than just a bare image sensor and will take up more space in the Lander.

Figure 4.40: Fisheye Lens

The camera lens that is included with the camera module does not have a high enough horizontal FOV to create a panoramic photo with three cameras. To replace the stock lens, the team selected a fisheye lens with a horizontal FOV of 185 degrees. This gives the PICS more than enough coverage to take three images that can be combined into a panoramic photo. Fisheye lenses have distortion at the edges of the photo, but since the combined FOV of all three cameras is greater than 360 degrees the distorted parts of the images will be cut out when the panoramic photo is created. One downside to this lens is that the lens sticks out farther than its casing. This creates the possibility of the lens being damaged upon the Lander's impact with the ground. To minimize this risk, the legs from the SOS will be angled to cover the lenses and protect against direct impact with the ground.

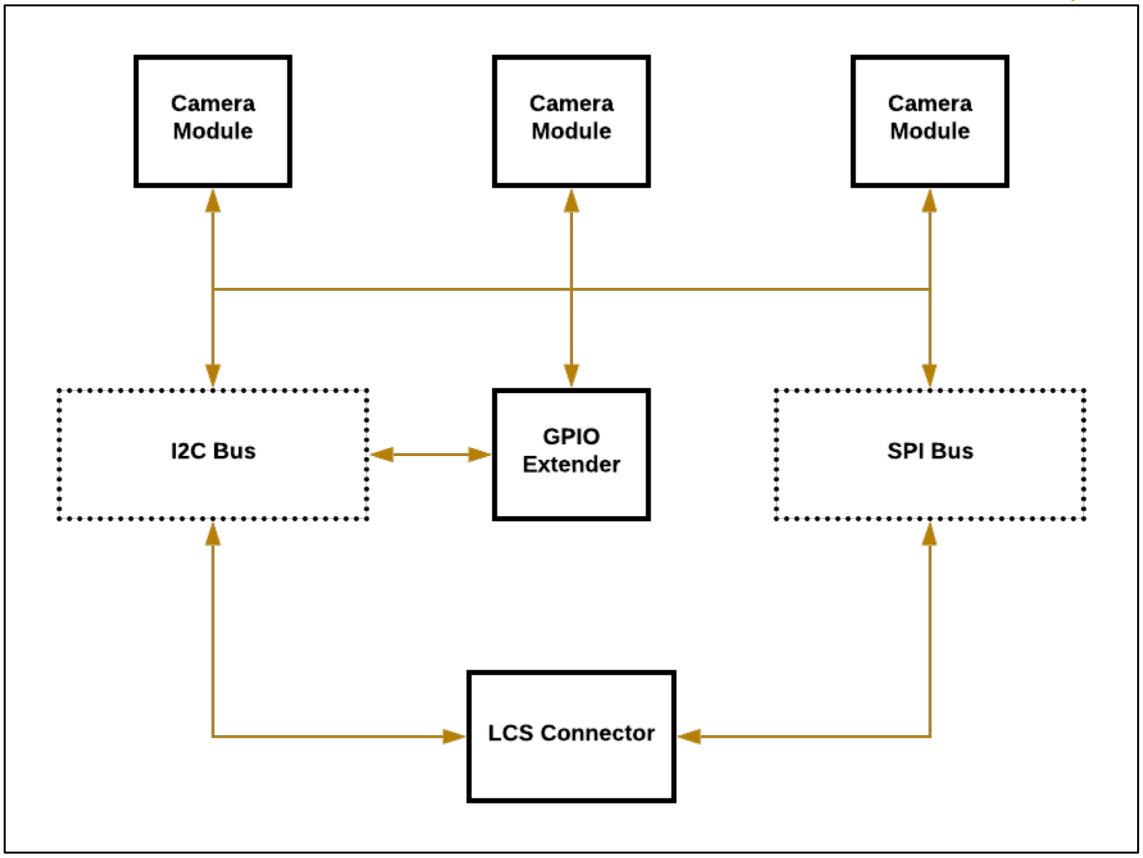


Figure 4.41: PICS Wiring Diagram

The diagram above shows the different devices of the PICS, how they are connected, and the communication protocols used by the devices. The camera modules are connected to the Secondary PCB via connectors on the bottom of it. All other electrical components related to the PICS are located on the Secondary PCB. The PICS is connected to the LCS on the primary PCB via a connector on the bottom of the Secondary PCB. This connector supplies the power, connects the devices to the SPI and I2C bus, and has 3 GPIO ports. The PICS uses the 3.3V regulated power from the primary PCB so there is no need for a power switch on PICS because when the LCS is switched off the PICS is also off. All three cameras are connected to both the SPI and I2C busses. Due to SPI requiring a chip select pin to be pulled low to select the device to communicate with, a GPIO extender is used to decrease the number of GPIO ports needed from the LCS. The GPIO extender is controlled via the I2C bus and each of its GPIO outputs is connected to a camera module. One of the GPIO ports is also connected to the radio transceiver because it also uses SPI. There is a red LED connected to one of the GPIO extenders GPIO outputs that will be used to indicate that the PICS is on and the GPIO extender is operational.

The following page contains the full schematic of the Secondary PCB.

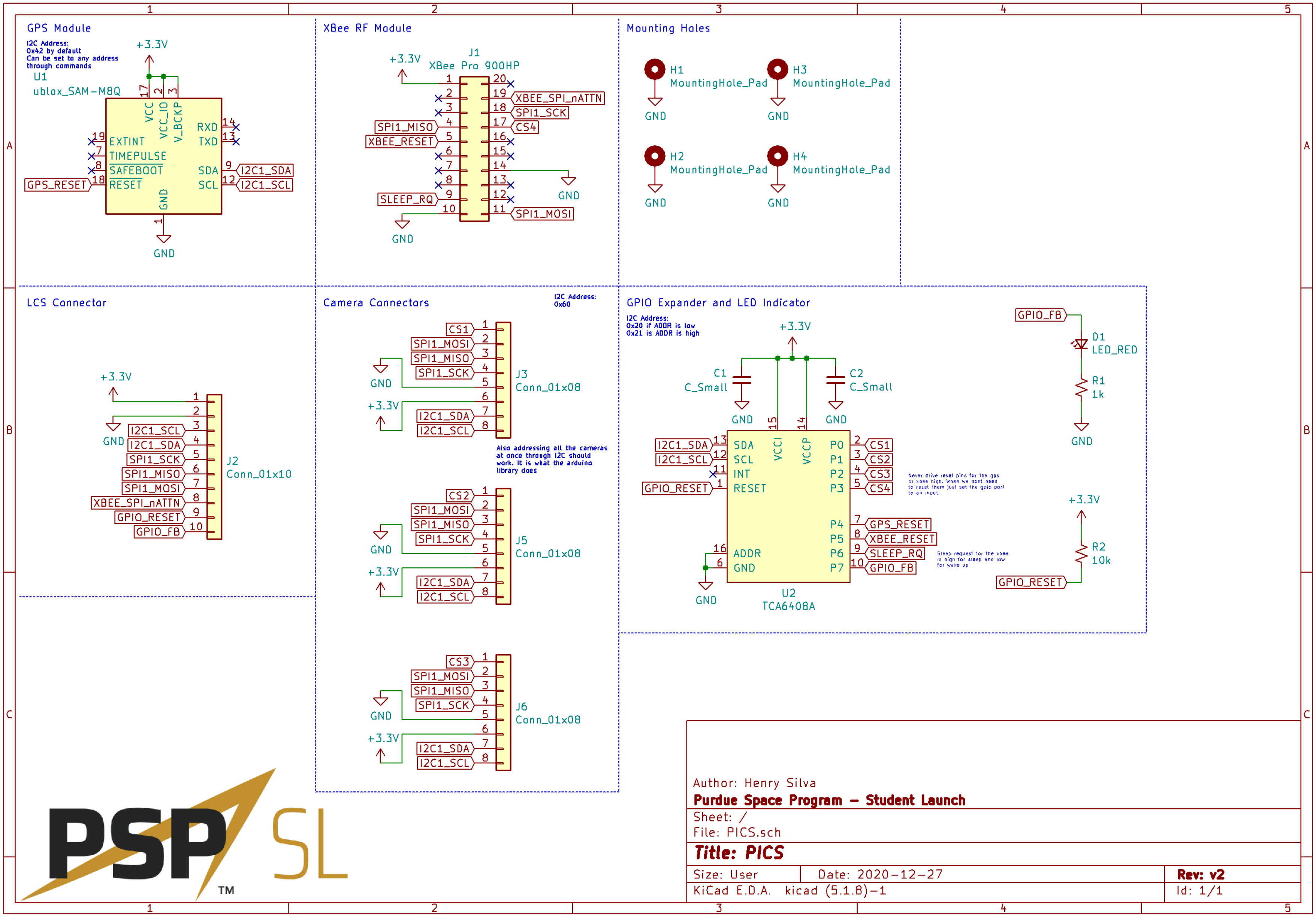


Figure 4.42: Schematic of secondary PCB

4.2.2.4.3 Software Design

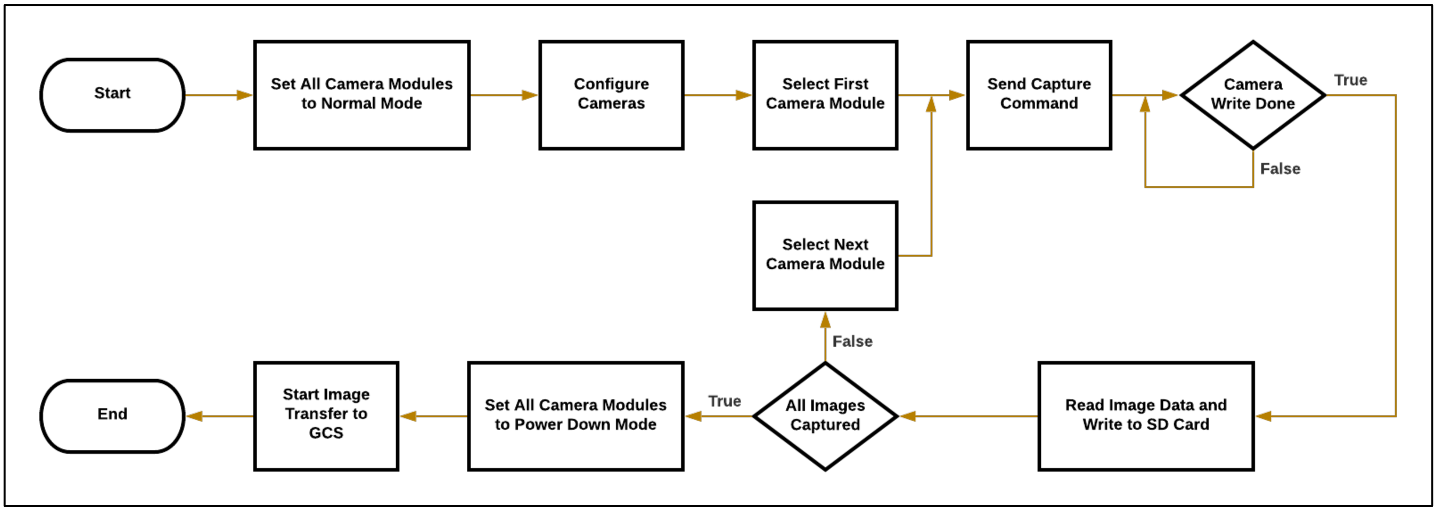


Figure 4.43: PICS Image Acquisition Process

Although the PICS is fully controlled by the LCS, the process for capturing and storing the images will be described here. This process will start once the SOS has completed self-orientation. At this point, all camera modules will have been set to power down mode during system initialization when the LCS is powered on. The first step is to set all the camera modules to normal mode. Setting the camera modules to normal mode turns on the image sensor and allows images to be captured. Once all camera modules are set to normal mode, all the image sensors are configured to output a JPEG compressed image with a resolution of 320x240. All three image sensors are configured simultaneously via I2C.

After successful image sensor configuration, the LCS will start to collect the image data from each camera module one at a time. To begin, the LCS will select the next module that has not taken a picture and send it a capture command via SPI. The LCS will then poll the camera module until the camera write done flag has been set. It should take the camera module no more than a second to receive the image data from the sensor and be ready to be read by the LCS. Once the camera writes done flag is set, the LCS will begin to read the image data and write it to a new file on the SD card. The LCS will read the image data one byte at a time, storing the read byte and then reading the next byte. The LCS will read and store the image data till the image is fully read. Reading one byte at a time and storing it will be slower than reading all bytes at once but will use less of the LCS microcontroller's RAM. Each image's data will be placed in a separate file. The process will be repeated until an image has been read from all three camera modules. With the three images needed to make up the panoramic photo in the SD card, the LCS will begin the process for transferring the images to the LCS and the PICS image acquisition process will be complete.

4.2.2.4.4 PICS Testing

In order to be fully confident in the PICS design, the team has done some testing to prove that the camera modules in our planned configuration will work. A test platform was made to position two camera modules similar to how they will be positioned in the Lander.

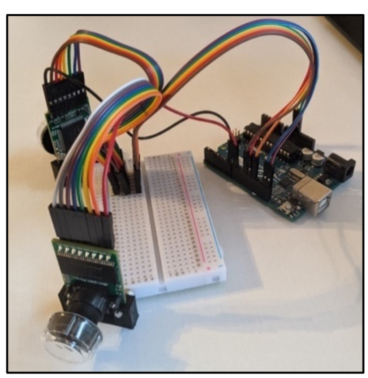


Figure 4.44: PICS Test Platform

The test platform shown in the above figure positions two camera modules 120 degrees apart and 2 inches from a center point. This is approximately how they will be positioned in the Lander minus the third camera module. The one difference in the test platform compared to how the cameras will be positioned in the Lander is that they are flipped upside down. An Arduino Uno was used to test the cameras because the company that makes the camera modules also provides a program that lets the user change the configuration options and take photos easily. Using the two camera modules, a test was conducted to see if they could produce an image spanning at least 240 degrees.

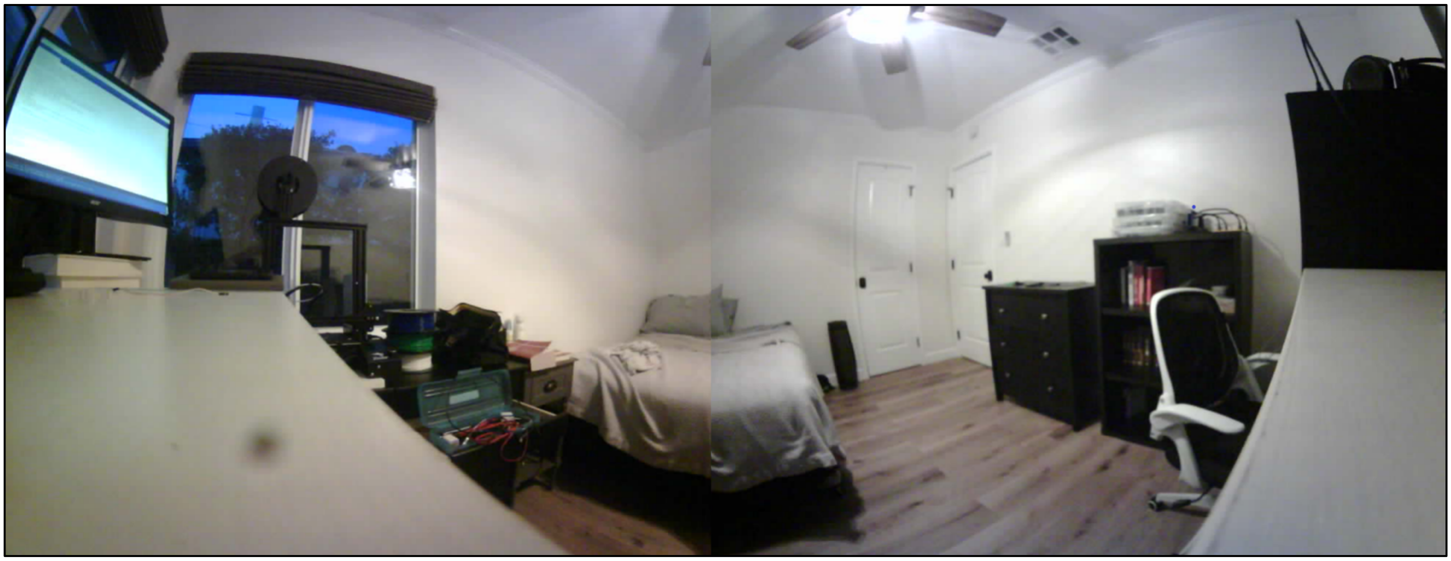


Figure 4.45: PICS Test Picture

To conduct the test, the platform was placed at the edge of a table towards a room with enough object to make combining the two photos easier. A picture was taken by each camera module and saved. The resolution of the images taken is higher than planned to better show the results of the test. The two images were then combined in photo editing software. Part of one of the images was cut to remove the overlap. As can be seen above, the two camera modules were able to successfully produce a partially panoramic image. The photos were slightly offset from one another which caused parts of the images to not line up at the transition point. The distortion due to the fisheye lenses is also noticeable towards the top and corners of the images. This is still a success because the Lander will most likely be taking a photo of a flat horizon. Ideally, no large objects will be sufficiently close to the Lander to make the distortion noticeable and the resolution will also not be as high so slight offsets in position will not be obvious.

4.2.2.5 Lander Control Subsystem

The Lander Control Subsystem (LCS) controls all the functions of the Lander and interfaces with both the Panoramic Image Capture Subsystem (PICS) and the Self Orientation Subsystem (SOS) to execute their primary functions. The LCS includes the Lander's microcontroller, altitude sensor, IMU, and battery. Using one microcontroller to control all the subsystems of the Lander reduces the complexity of the software and electrical design while still keeping the subsystems modular. All the Lander's subsystems are connected internally with wires and connectors. The LCS executes and manages all the Lander's functions so that they are executed at the correct time.

4.2.2.5.1 Structural Design

The LCS takes up the most space within the Lander's central bay. In the three-plate arrangement of the Lander, the LCS involves the usage of both the LCS plate and Battery Plate. Requiring communication with all electrical components of the Lander, this system forming the geometric heart of the Lander makes much sense. Given the two-plate arrangement of the LCS, deconstructing the Lander in its entirety allows for the modular deconstruction of the LCS, providing full access to its respective parts and future wirings.

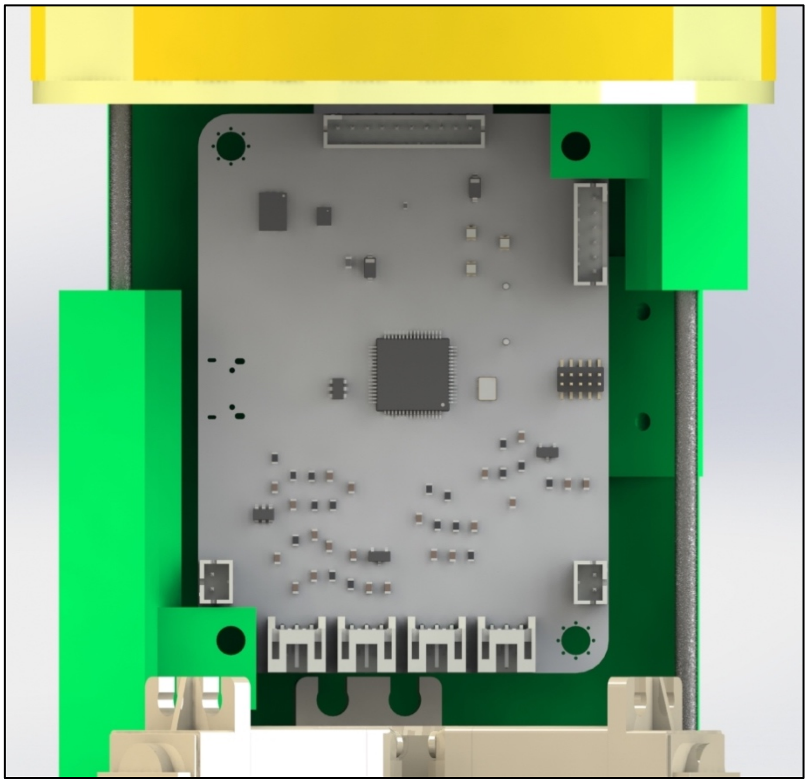


Figure 4.46: LCS Plate with PCB Render (Colored Green)

The LCS PCB is rectangular and flat, as common PCB's tend to be. However, this presented the challenge of attachment within a circular body. To solve this, the team prioritized the attachment of the LCS to be as close to the structural threaded rods as possible. While this provides high structural rigidity, the design required the compliance with the Descent and Battery configurations of the Lander's wall plate to attach correctly. For final construction, the LCS PCB will be required to be removed before the central plates can be separated, however the LCS PCB can remain attached to its plate for storage and protection purposes during transport and testing.

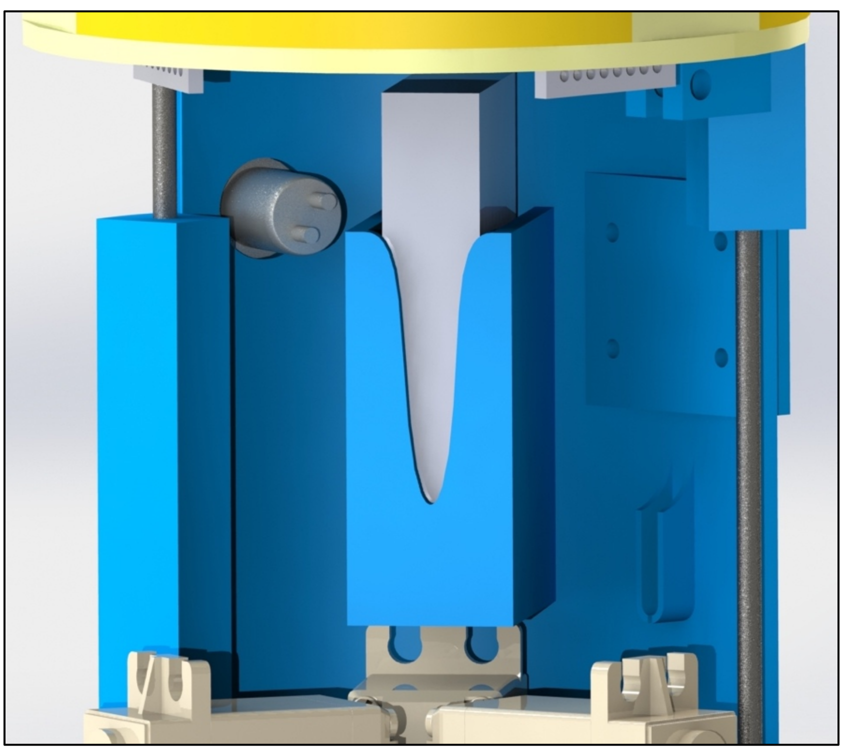


Figure 4.47: Lander Battery (Gray) with Key Switch (Left Side) and Reed Switch Holder (Right Side)

Adjacent to the LCS plate is the Battery plate. While simple, this plate stores the heaviest component of the Lander, the Battery. Additionally, to its sides are positioned the power key switch and sleep-state reed switch holder. In order to comply with the needs of the SOS, the battery holder was requested to remain as central to the Lander as possible. Placing the battery directly against the wall would provide more internal space but would greatly shift the Lander's CoM off its central axis.

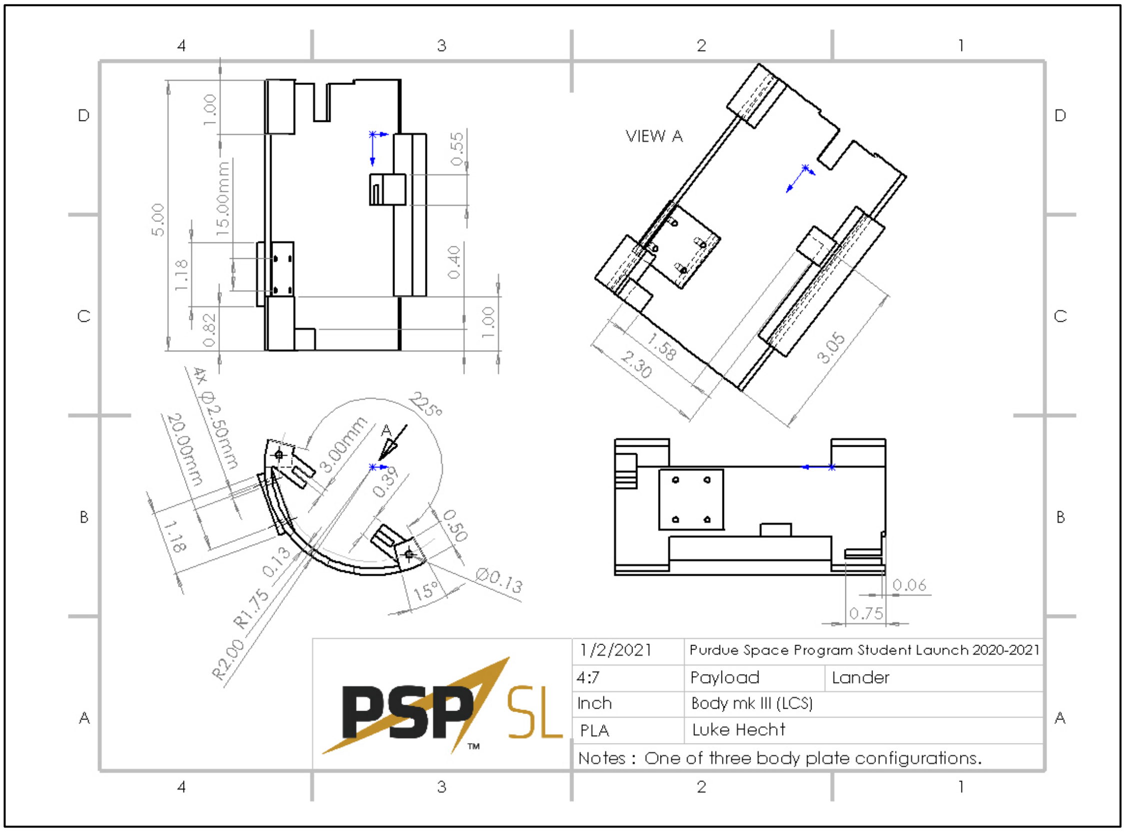


Figure 4.48 LCS Plate Drawing

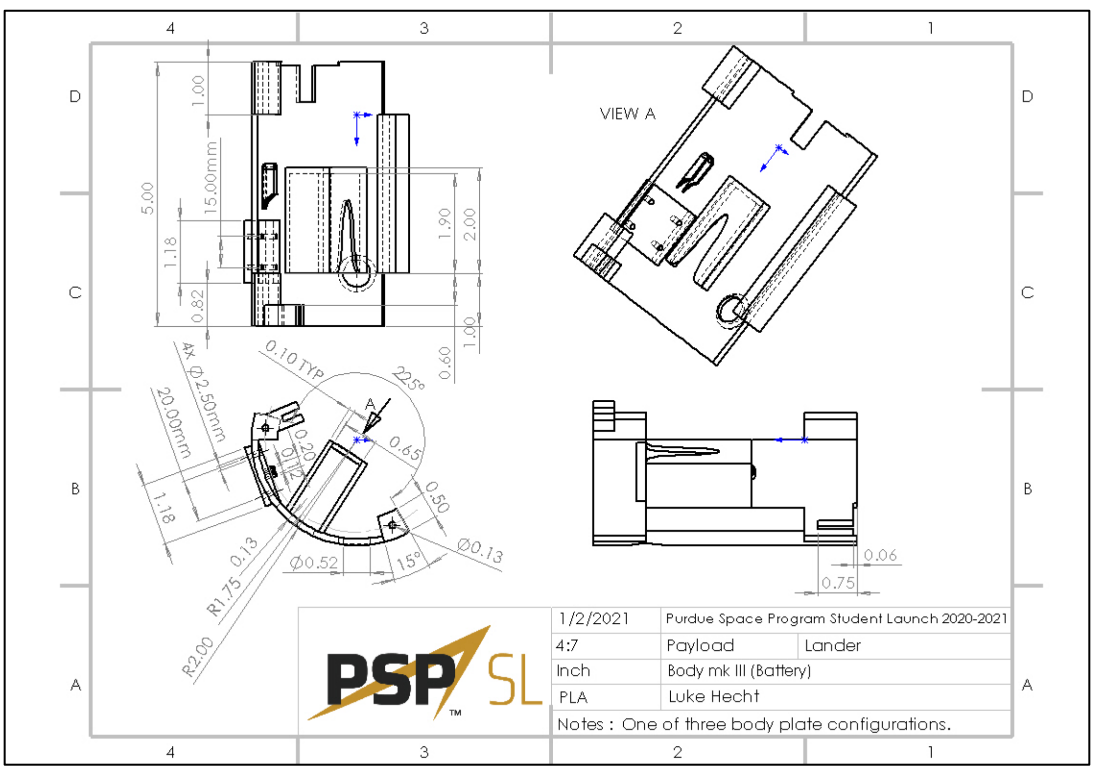


Figure 4.49 Battery Plate Drawing

4.2.2.5.2 Electronic Hardware

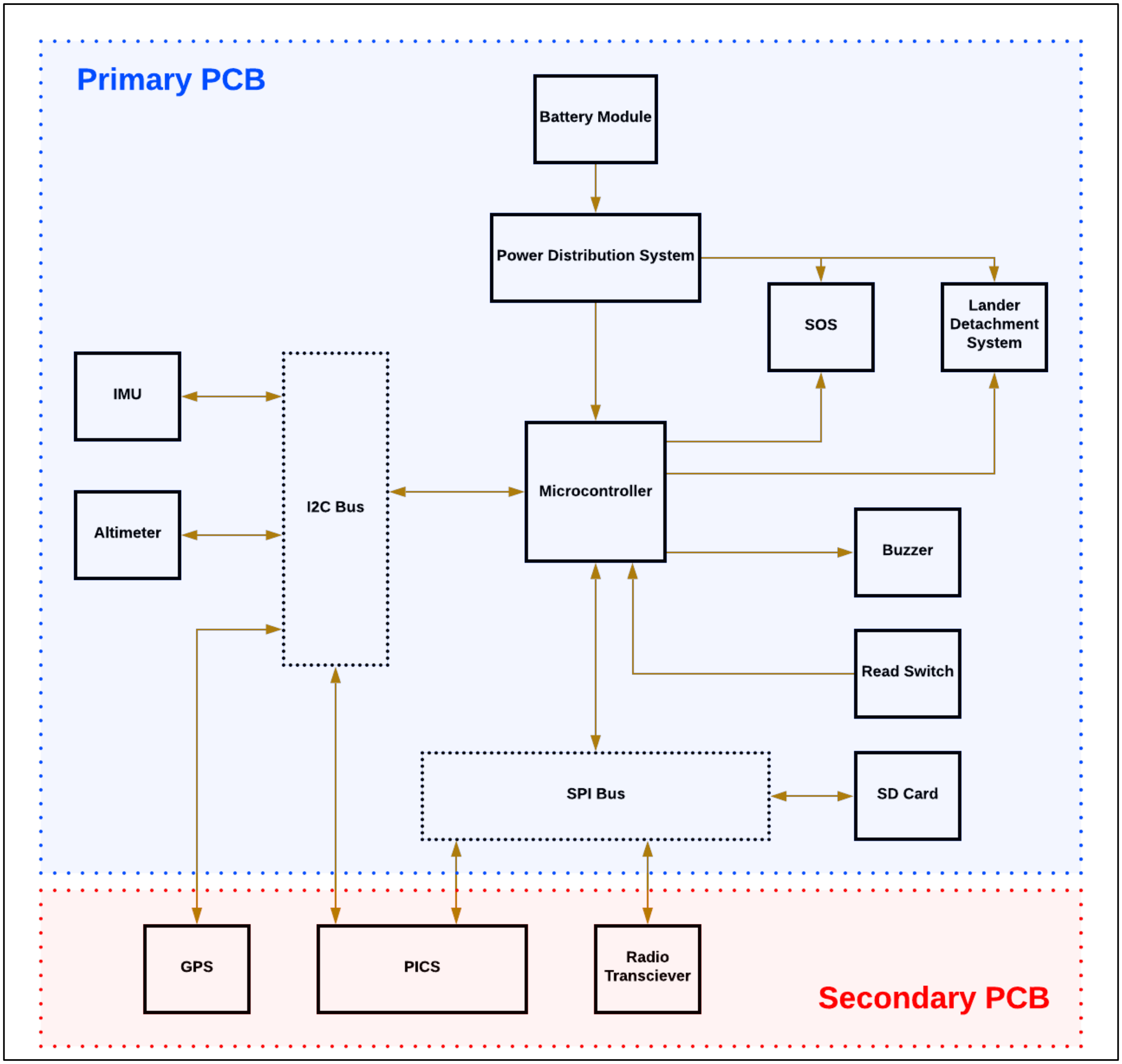


Figure 4.50: LCS Wiring Diagram

The diagram above shows the major components of the LCS and how they are connected. The diagram also shows the communication protocols used by the devices. The Lander contains two PCBs. The Primary PCB includes the main microcontroller, sensors, the power distribution system, and the SD card. The Secondary PCB includes the GPS, radio transceiver, and the whole PICS.

The microcontroller for the LCS is the STM32F405RG. We have selected it because of its high pin count and ability to interface with all the devices in the LCS, PICS, and SOS. It also features a sleep mode which reduces its current draw significantly while still being able to be woken up by a signal read from one of its GPIO pins. The STM32F405 has a max clock speed of 168MHz. This is more than fast enough to meet the processing requirements of the LCS software and will most likely be run on a lower speed to reduce current draw. Another advantage of using the STM32F405 is its internal USB to serial converter which makes implementing a USB port on the PCB for debugging much easier.

Figure 4.51: Battery and Key Switch

The battery module includes a battery and a key switch. The battery is a Venom Fly 30C 1300mAh 7.4V LiPo Battery. After determining the approximate power needed for the duration of the Lander's mission, the team decided that the smallest and lightest battery would be best due to the limited space and weight requirement of the Lander. A LiPo battery was chosen because they are commonly used in small electronic vehicles. The continuous discharge rating of the battery is 39A which is more than enough for the Lander. XT60 connectors will be used to connect the battery module to the Primary PCB.

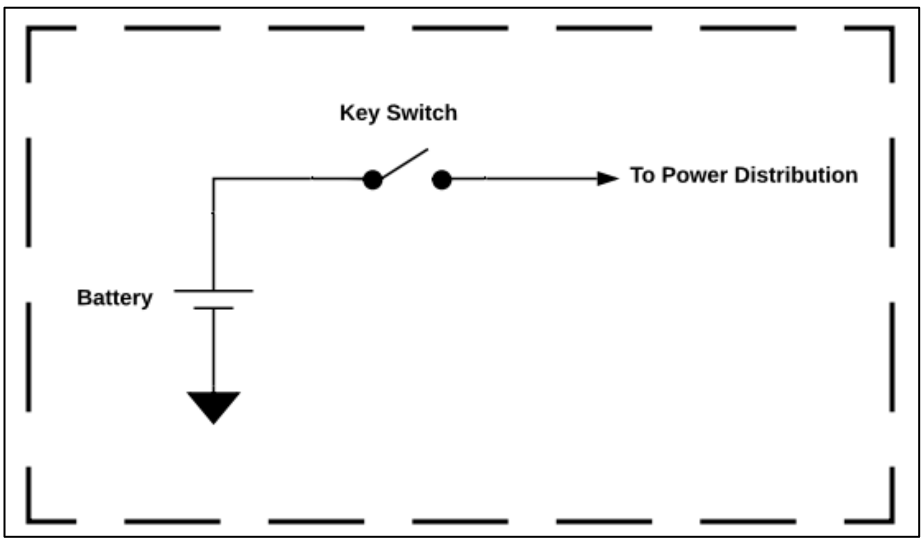


Figure 4.52: Battery Module

The keyswitch will be used to control the power into the Primary PCB. Using a keyswitch ensures that the Lander will not accidentally be turned on or off. As the diagram above indicates, the battery and key switch will be wired in series. The keyswitch is the same one used by the Retention and Deployment Subsystem.

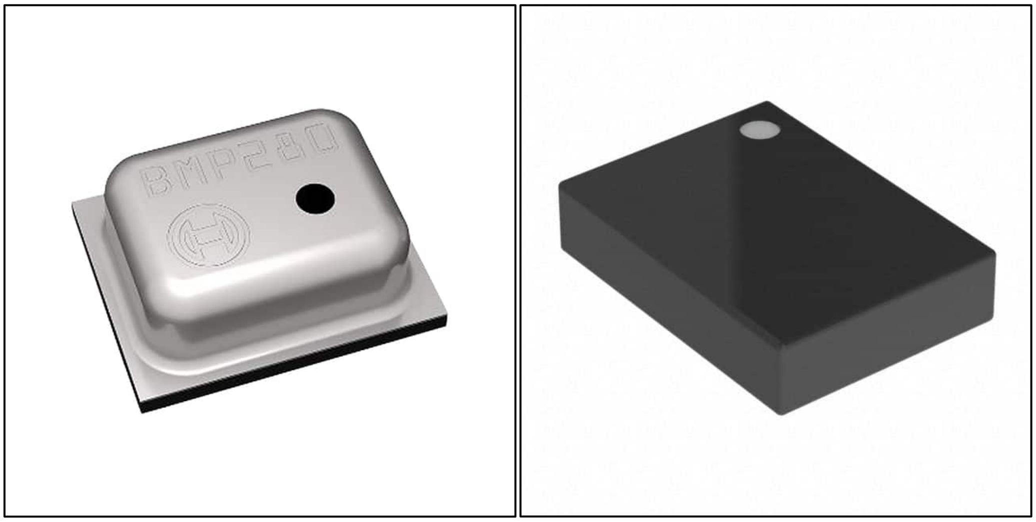


Figure 4.53: BMP280 and BNO085

The two main sensors that the LCS will use during the mission are its altimeter and IMU. The altimeter that we have selected is the BMP280. It was selected due to its ease of implementation and reliability. The IMU that we have selected is the BNO085. The BNO085 is one of the only low-cost absolute orientation sensors in a small package on the market. The sensor handles all the intense calculations necessary to determine the absolute orientation of the Lander during self-orientation. The sensors included in the BNO085 are an accelerometer, gyroscope, and magnetometer. The data from all three sensors can be accessed as well.

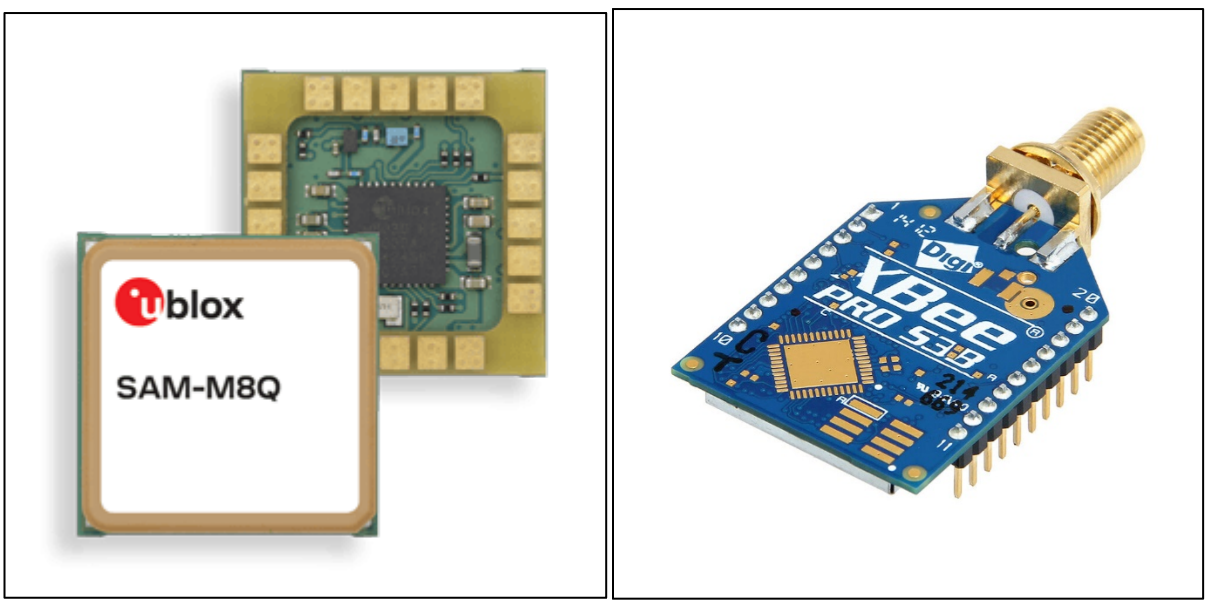


Figure 4.54: GPS and RF Modules

The GPS and radio transceiver that will be used in the LCS are the u-blox SAM-M8Q GPS Module and the Digi XBee-PRO 900HP RF Module. The u-blox SAM-M8Q module is a all-in-on GPS module that is easy to incorporate into design, even for those with little RF experience. The main microcontroller will communicate with the GPS over I2C using u-blox's UBX protocol. The GPS module will be on the Secondary PCB, which will be placed at the top of the Lander. The PCB will create a large ground plane below the GPS. This configuration will shield the GPS from other RF interference and give it a clear view of the sky.

The XBee PRO 900HP will be mounted on the bottom side of the Secondary PCB. It will be mounted towards the edge of the board so the antenna can wrap upwards. The confined space may cause problems with RF interference, particularly when the XBee is transmitting while the GPS is acquiring a signal. If problems arise during testing, the team may need to consider moving the Xbee. If necessary, the team will also modify the LCS software to not allow the Xbee to transmit while the GPS is acquiring a signal.

The Xbee PRO 900HP will use Digi's proprietary protocol that is already programmed onto the Xbee. This protocol includes features for interference immunity and error detection. The main microcontroller will communicate with the Xbee using SPI communication.

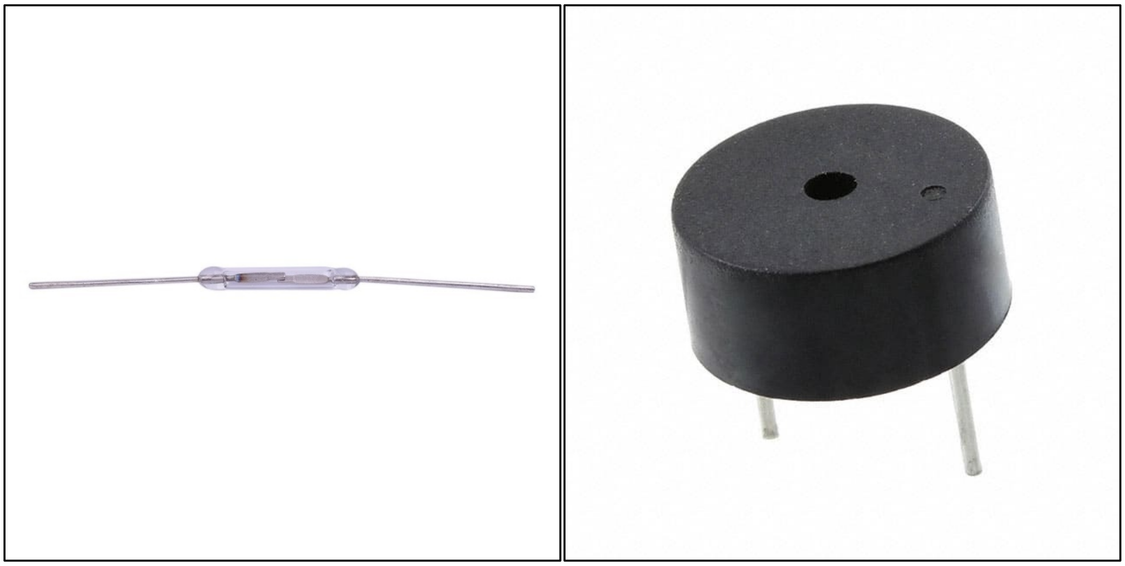


Figure 4.55: Reed Switch and Buzzer

The reed switch and buzzer are two simple, but important components of the LCS. The reed switch is normally open and will be used to detect when the Lander has left the Payload Bay. When the Lander is loaded into the Payload Bay, a small neodymium located on the inside wall of the Launch Vehicle will trigger the reed switch to close. When the Lander leaves the Payload Bay, the reed switch will open, and an interrupt attached to the microcontroller's GPIO pin will cause the microcontroller to exit sleep mode. This solution for detecting the Lander leaving the Payload Bay was deemed to be simplest compared to other solutions. The buzzer will be used to communicate errors or other messages to the team while the Lander is inside the Launch Vehicle. The Primary PCB has several LEDs to indicate power on and status, but they are not able to be viewed while the PCB is in the Lander. The buzzer will be important to determine if all sensors and the radio transceiver have initialized properly.

Figure 4.56: JST XH Connector

The team has chosen vertical XH Connectors by JST for all the connectors in the LCS. They feature a locking mechanism that will ensure that the connections do not come loose due to vibrations. The connectors have a current rating of 3A which is more than enough. Although these connectors are not as space efficient as other JST connectors, we decided on the XH series because it is still small enough while having a high current rating making it a versatile connector. The team decided that having a standard connector that the whole team used would be better than having several different connectors.

The following page contains the full schematic for the Primary PCB.

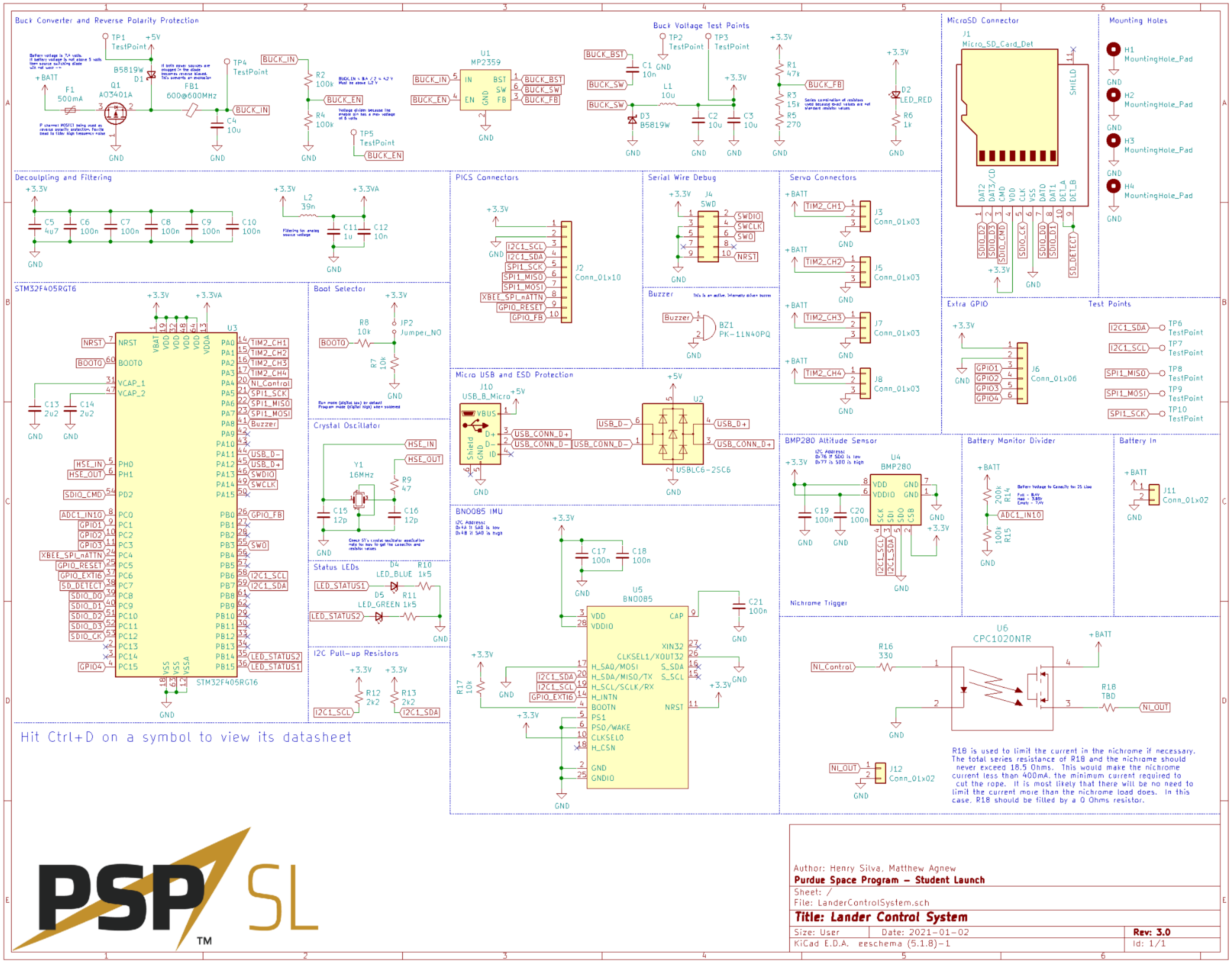


Figure 4.57: Primary PCB Schematic

4.2.2.5.3 Software Design

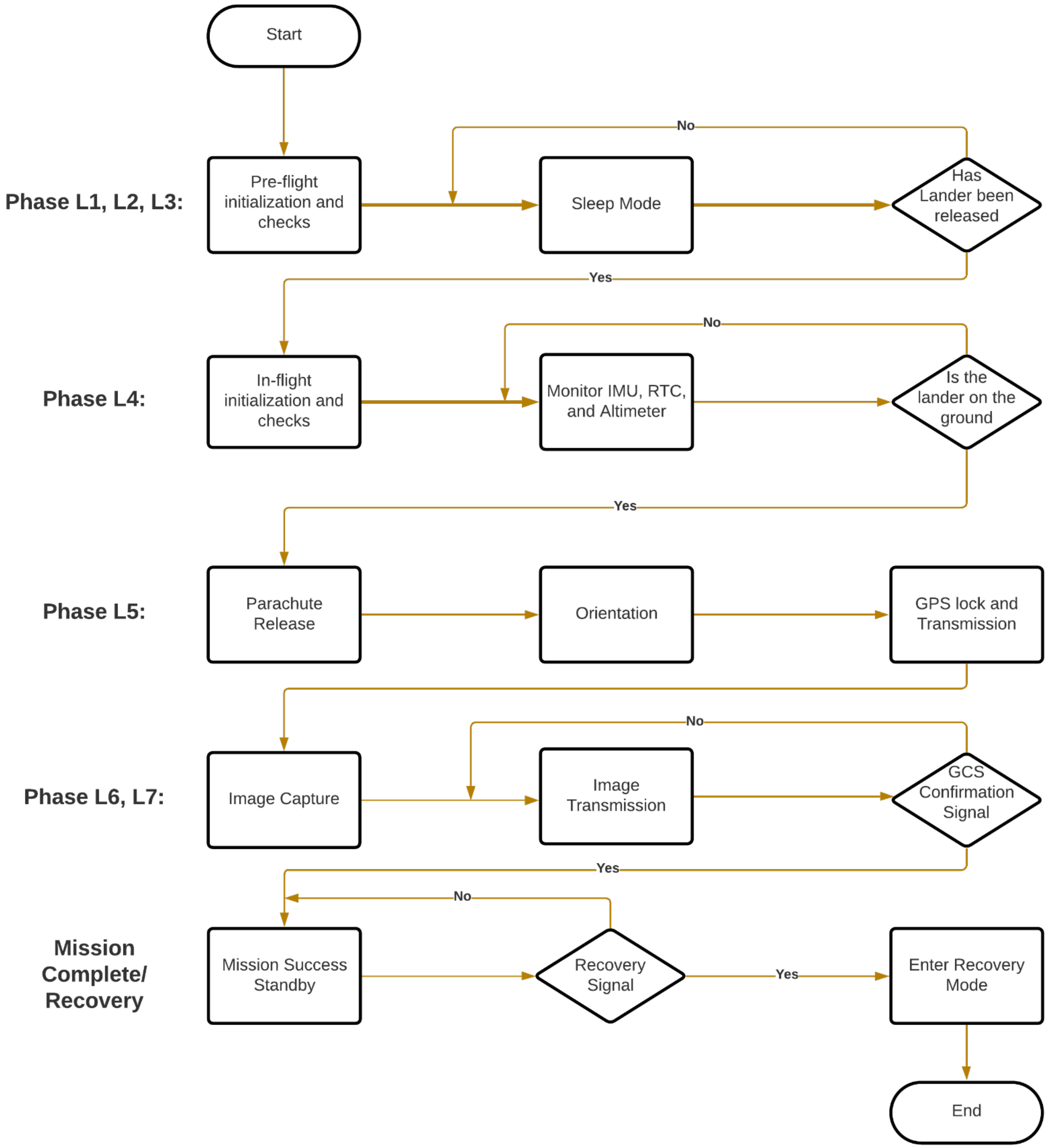


Figure 4.58: LCS Software Overview

After the Lander has been turned on, the LCS will begin the initialization phase. The initialization and system check phase will test every system in a simulated mission except for the parachute release system. After booting up, the microcontroller will poll all the peripheral devices to ensure all devices are working properly. The microcontroller will also instruct the Xbee to establish a connection with the GCS. If there are errors that the LCS cannot resolve then the LCS will use the buzzer to signal an error state, and if a connection has been established, error codes will be transmitted to the GCS. From there it will be up to the team to resolve errors and restart the device.

If there are no errors with any of the peripheral devices and the Xbee has established a connection with the GCS, the lander will wait for a signal from the GCS to begin orientation. The team will then send a command from the GCS to begin orientation. The orientation measurement will be proceeded by a beep from the buzzer to alert nearby team members. This first orientation will be a test to ensure the orientation system is functioning prior to launch. Once orientation is complete the LCS will instruct the GPS module to acquire a signal. Establishing a GPS signal prior to launch will ensure that there are no errors with the GPS system. In addition, the GPS will have already calculated data for satellites in view. This will allow for the GPS to get a faster position fix later in the mission.

As the GPS is acquiring position data the LCS will begin to go through a test of the PICS system.   The PICS software design is detailed an above section. After images have been acquired, they are transmitted to the GCS along with the GPS data. The LCS will then wait for a command to transition to the flight ready state.

Once the team is satisfied that the Lander is ready for launch, a command from the GCS will instruct the lander to prepare for launch. The lander will return the legs to the starting position and set all peripheral devices, except for the Xbee, into their respective sleep modes. The LCS will then wait until the reed switch detects the magnet in the payload bay. Once the magnet is detected, the LCS will send a confirmation message to the GCS and then the XBee will be but into sleep mode. The microcontroller will also go into sleep mode and will remain in this state until the reed switch detects that the lander had left the payload bay.

Once the Lander has left the payload bay the reed switch will open and trigger an interrupt on the microcontroller. The microcontroller will come out of sleep mode and begin the in-flight initialization. The microcontroller will begin start the real time clock (RTC) and instruct the Xbee and GPS modules to begin acquiring signals. The microcontroller will also begin to monitor altimeter and IMU data, as well as transmitting data back the GCS for the team to monitor. Reading data from the RTC, IMU, and altimeter will ensure that the LCS does not detach the parachute too early because of a glitch in any one sensor. The team will run simulation to determine the expected time it will take the Lander to reach the ground after release. The team will also conduct tests to determine what a landing will look like in the IMU data. The LCS will not trigger the parachute release until both the RTC, IMU, and altimeter show have shown values that would indicate a landing. If the Lander does not detect landing after a period significant period of time, then there must be an error somewhere in the system. In this failure mode, the LCS will not proceed until it receives confirmation from the GCS that it is safe. If the Xbee has a connection to the GCS, the LCS will capture images using the PICS system as well as all relevant data and transmit it all to the GCS. The system will then wait for a command from the GCS which will either instruct the Lander to abort the mission and enter recovery mode or to continue with the mission as planned.

If everything functions correctly, the LCS will detect a successful landing and trigger the parachute release mechanism. After the parachute has been released, the LCS will begin the orientation phase. Details of this phase were covered in the SOS software section. Since the GPS will most likely not be able to get a good position fix while the lander is on its side, the LCS will again poll the GPS after orientation. This data is transmitted back the GCS for the team to use in recovery. If the GPS does not immediately have position data, the LCS will begin the PICS process. After the GPS and image data are transferred to the GCS there will be an automatic confirmation reply. After the LCS receives confirmation of a successful transfer, the LCS will have completed its mission and will go into standby mode. The LCS will remain in standby mode until instructed by the GCS to transition to recovery mode. In recovery mode the Lander will return the legs to the starting position and will remain in this state until the Lander is turned off by the recovery team.

4.2.2.6 Ground Control Station

Figure 4.59 GCS Conceptual Render from Early 2020

The Ground Control Station (GCS) team computer was designed and constructed during the previous NASA-SL season for usage in the competition. With its outer Pelican 1550 case shell and carbon fiber panel inserts, its original purpose was to “enable real-time command and control of the UAV and R&D systems” involved in last year’s challenge. Its usage as the team’s central command center was impressive and groundbreaking, but nonetheless limited relative to the device’s full potential—thanks to world situations and lack of time. While the GCS was originally designed with a different competition in mind, it was also designed with a high level of versatility, and thanks to its central Raspberry Pi 4, much of that versatility could be leveraged in this year’s software design.

Therefore, this year, the team intends on reusing the physical structure of the GCS along with its electronic components, whether or not all components will have a defined use in the end. Seeing as the GCS is no longer necessary to provide real-time instruction to the Lander—only communication and data transmission—many of the switches present on the board will remain inactive. Despite this, the GCS’s built-in Raspberry Pi, keyboard, trackpad, display screens, and power distribution remain be fully reusable for the purposes of the Planetary Landing System.

For a full internal analysis and electronic breakdown of the GCS, please refer to PSP-SL's 2019–2020 documentation.

4.2.2.6.1 Upgrade Plans

While the GCS is planned to be reused, there remains a few action items that must be handled in order to prepare the device for further use. As previously discussed, the GCS’s Raspberry PI 4 computer will remain in use, and the GCS’s physical construction will be left as-is. However, the software loaded onto the GCS will be subject to rework. For starters, the GCS’s old GUI will be outdated for this year’s competition and will require a redesign from the ground up. The GCS must also be made compatible with Lander data communication protocols; while the XBee installed in the GCS is currently the same model to be used this year, last year’s designs involved a more simplistic two-way communication.

In order to prepare for use as the team's computer once again, the GCS must be reassembled and tested as though it were a new computer. While the team has the GCS in its possession, the device has yet to be powered on this season. Thankfully, the team is familiar with its base components and rests assured that faulty components can be replaced in time for integration testing.

Figure 4.60 Raspberry Pi 4 Computer

4.2.3 **PLS System Integration**

4.2.3.1 PICS, LCS, D&L, and SOS as the Lander

The Lander consists of four main subsystems, each with a purpose that is essential to the success of the PLS. The Lander was conceptualized as such a modular arrangement to allow for team members to work to their strengths with a semi-individualized modular assembly. The end result, while internally complex, is a sleek outer design with a modest goal: The Lander must land, orient, take a panoramic picture, and share that picture with the ground team. The final design of the Lander brings together many unique systems with their own required form factors into a single and harmonious assembly.

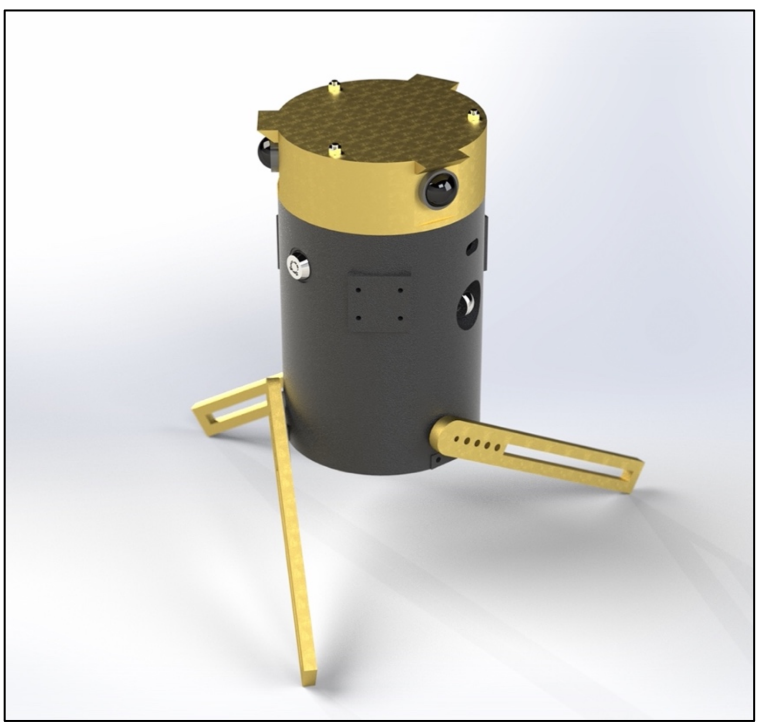


Figure 4.61 Lander Overall Render (Standing)

The overall design of the Lander provides the optimal amount of access space while still being retained by the R&D. This cylindrical form factor can be easily accepted by the equally cylindrical Payload Bay. By retracting the SOS's legs, the Lander is able to be retained by the R&D's three rail carriages and stored during flight. On the inside of its lower bay, the Lander is highly active with its central control PCB of the LCS and the SOS's three actuation servos positioned just below. The upper bay is home to the PICS camera array essential to the completion of the mission. The design of the Lander is capable of parachute descent as well, accepting the attachment of a nylon rope through its side eyebolt of the D&L. While appearing simple and elegant, the Lander contains a concert of intermingling components.

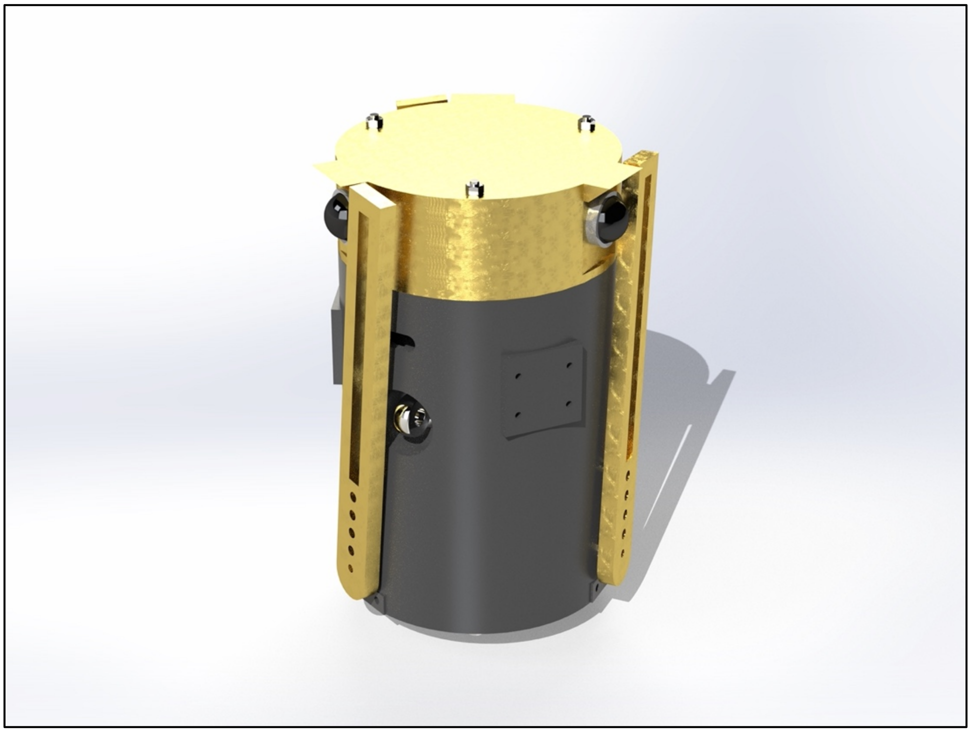


Figure 4.62 Lander Overall Render (Closed)

In order to produce this design, the team considered multiple ways of attaching the Lander's main subsystems. Even by PDR, the team had decided upon the cylindrical shape, but the question remained how so many different pieces of hardware and electronics could be fit in one place. Going off of last year's UAV design, the team considered employing a similar stacked plate structure; discussion ensued about the orientation of stacked plates, how many, and how much space would be available for attachment. Given that the Payload Bay has a great length, the team opted to orient its stacked plates to be parallel with the Payload Bay's axis. However, there remained the question of how many plates and how could select plates still be accessed from the outside?

Two observations came together to produce the final design. First, the R&D design was being built to be radially symmetric, meaning that the Lander's attachment points must be as well. Next, the team was reminded of Apple's 2013 Mac Pro design, a concept which revolved around a triangular arrangement of motherboards and ports, topped with a toroidal fan. The team had recognized that by designing the stacked plates to be radially symmetric, they could be modularly assembled and disassembled. Additionally, the same wall plate part could be used to produce slightly different shapes and access points for different systems. Through discussion, the team landed on utilizing three central plates for the structural and computing center of the Lander. Meanwhile, the entire Lander would be supported from underneath by the SOS and capped from above by the PICS, a system that would benefit from being far above and away from the other subsystems. Thus, with confirmation of the triple axially symmetric design with R&D, the final Lander design was born.

ITEM NO.	PART ID	QTY.
1	Base (Hemi-sphere) mk III	1
2	Body mk III (Descent)	1
3	Body mk III (LCS)	1
4	Body mk III (Battery)	1
5	leg (whole) mk IV	3
6	Threaded rod (stand-in)	3
7	servo GoBuilda 25-2	3
8	Cupola Plate	1
9	3798K35	1
10	95462A029	1
11	90107A029	2
12	KO131A102 Keylock Switch	1
13	LanderControlSystem	1
14	Battery Stand-In	1
15	Cupola Top mk II	1
16	OV2640_Cam	3
17	1900-0025-0104	3
18	90725A020	6
19	SecondaryPCB_Stand-in	1

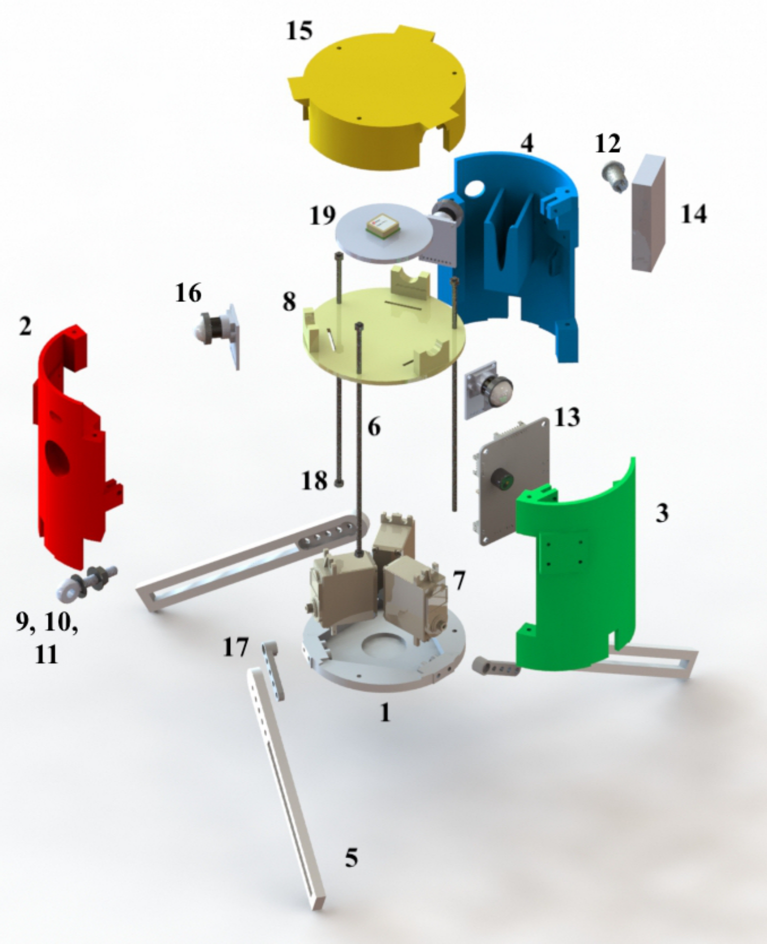


Figure 4.63 Pinwheel Lander Exploded View with BoM

Aside from the pre-purchased electronic components and some standard hardware, most of the Lander's unusual structure is entirely designed to be 3D printed. The team intends on capitalizing on Purdue University's 3D printing capabilities, utilizing Markforge Onyx Carbon Fiber filament in the construction of the Lander's main body. This material offers much greater durability and strength when compared to typical PLA. This quality is essential to the Lander, which unlike last year's UAV will be absolutely required fall onto and interact with unknown farmland terrain. This material is mostly black in color, an added bonus to the team, allowing for the University's colors to be presented. While it is likely that the Lander's legs will also be printed from this material for resilience purposes, the team is still considering the possibility of utilizing more standard gold silk PLA to once again present the University's colors. If initial testing proves that the Lander's legs have little expected damage, the switch to such material may be made. Alternatively, the team may paint these components manually. On a more crucial note, the Lander's Upper Cupola will be constructed from this standard gold silk PLA material. Not only will the cupola be the section of the Lander least likely to interact with the terrain, but the Lander's GPS cannot be confined by the radio-shielding conductive carbon fiber of the main Lander material, lest the team may lose its position. For the underlying skeleton of the Lander, the design will utilize size M3 threaded rods to hold the assembly together, as stated in the SOS section. By removing these threaded rods alone, all sections of the Lander become structurally independent, an essential quality for a modularity-inclined team.

4.2.3.2 Retention and Deployment with the Launch Vehicle

The successful operation of the R&D requires the cooperation with the Construction Team and their nosecone design. The Payload Team has worked with the Construction team to ensure that both systems can coexist in the upper airframe. For a depiction of the nosecone and its attachment plate, please refer to the Vehicle Criteria section of this report.

In order to accommodate the R&D electronics bay as well as the installed tracking device (see Tracking Devices) within the same coupler section, multiple holes will be drilled into the coupler-airframe interface band along with the standard holes for airframe attachment bolts. Specific to Payload Bay electronics, two ½" holes are required to be drilled in between two separate sets of these attachment holes. Additionally, one 0.2" static port hole will be drilled between two other attachment bolt holes in order to provide pressure reference for the R&D altimeter. These holes have been positioned in a manner to maximize the structural material between them, reducing the likelihood of material failure. Additionally, the separation of attachment points acts to reduce the PLS's effect on the launch vehicle's products of inertia.

4.2.3.3 Retention and Deployment with the Lander

The R&D and Lander are both compatible with PBCLinear MR12-0250-1 Sliding Rail, a custom request for the company in order to accommodate the Payload Bay. As previously stated, PBCLinear's provided model does not come with separable carriages, but be assured that the carriages would be fastened to the Lander during the entire mission. While these rails are reported to provide a smooth glide, testing and adjustment will certainly be required to ensure the effortless operation of the R&D deployment scheme. Within the Payload Bay, the rails are offset from the airframe walls, and may be adjusted with additional spacers or washers as the team sees fit for proper operation.

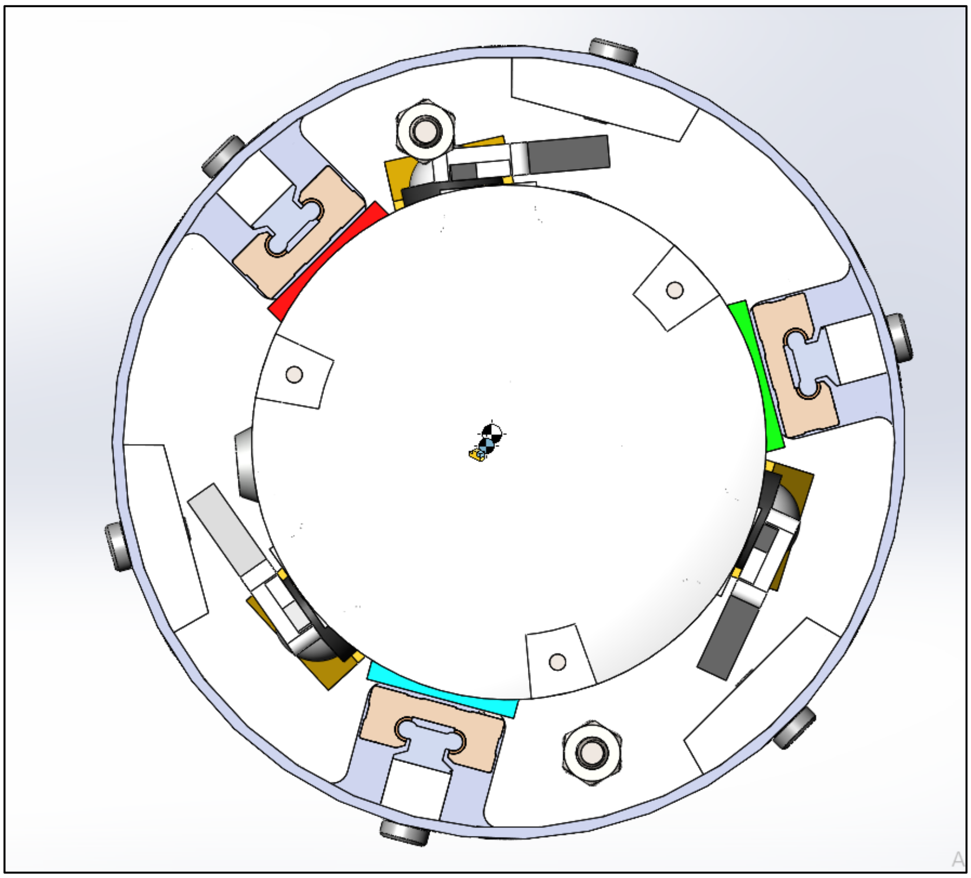


Figure 4.64 Lander within R&D Pizza Table Showing Alignment (Nosecone Removed)

Due to the immensely complex configuration of mutually exclusive linearly actuating components, much care was taken in the orientation of the Lander within the Payload Bay. In the end, the compounding of constraints has led to the final design to be rotationally locked. If each carriage and rail are not matched, then the system will not work exactly as intended. This is ensured by the existence of the Lander's internal reed switch, which requires a positioned magnet in order to turn on and off. This magnet will be placed adjacent to the Lander near to the reed switch. Since the reed switch is in a single spot on the Lander, lining up these parts is essential to Lander deployment function.

4.3 AeroBraking Control System

4.3.1 System Overview

The ABCS main design impetus is to assist in achieving the predicted apogee which has been calculated through OpenRocket at 4100 ft. Through a control system, the ABCS will determine the trajectory of the rocket and compute the necessary adjustments to a set of three aeroplates. The physical system in the lower rocket frame and is comprised of two main structures: the airbrake pad structure and the electronics bay. The electronics bay rests atop the airbrake system as shown below.

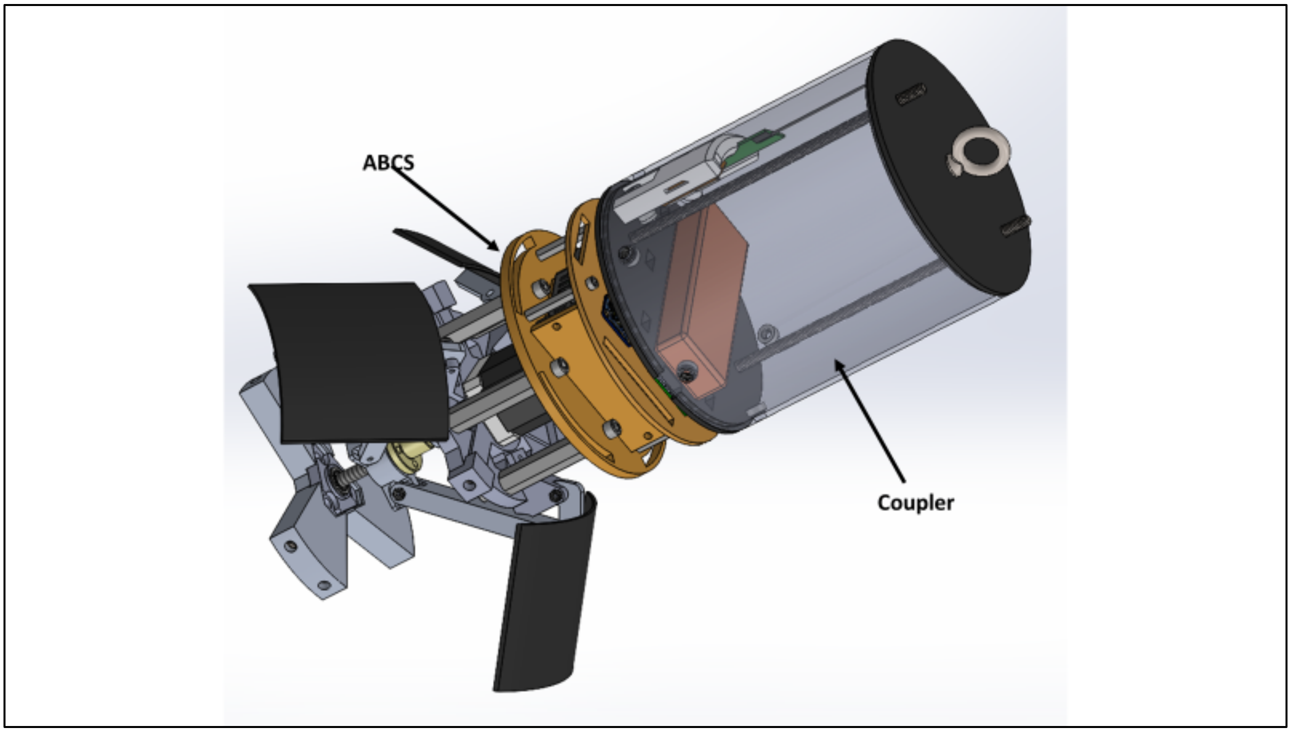


Figure 4.65 ABCS with Coupler

The upper end of the ABCS compartment is fastened to a coupler within which the battery that will power the system sits. On the aft side of the bay is the motor tube. The total mass estimate of the ABCS is less than 3 lb. This meets and undershoots team requirement S.P.0.3.

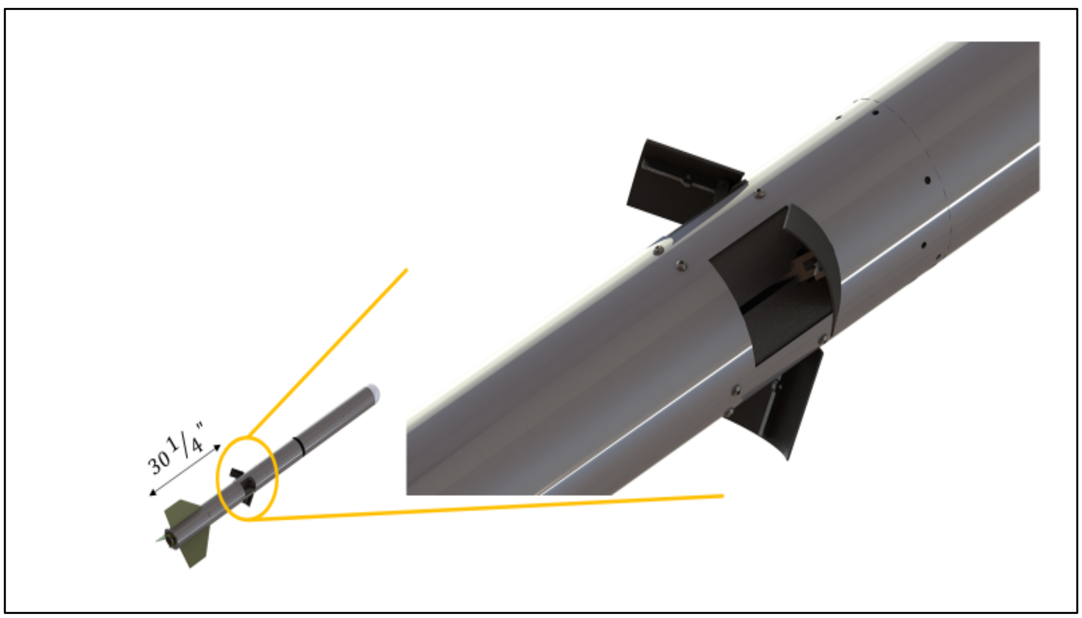


Figure 4.66 ABCS in Active state

The mechanical design has been refined for design and manufacturing success. Firstly, the electronics bay is the biggest addition to the physical design. Secondly, due to the limited manufacturing resources of the team, all pieces that were previously to be manufactured out of aluminum will now be 3D printed in Markforge 17-4 PH stainless-steel and Markforge Onyx Carbon Fiber. The metal printers were delayed due to lab access restrictions and a great demand.

4.3.2 **ABCS Detailed Design**

4.3.2.1 Mechanical Subsystem

The decision to move ahead with the umbrella design as concluded at PDR has kept the mechanisms virtually untouched. A stepper motor will rotate a lead screw that vertically translates amount. The Paddle Struts that are bolted to the lead screw mount will push and pull them in and out according to the input from the microcontroller.

The design has matured to include a well thought out electronics and sensor bay. This houses all the driver, MCU, and sensor units. The design includes further detail regarding integration of the E-bay and the rest of the rocket such as a safety switch and space for the battery. Many changes were made to the materials of certain components mostly from stock metal to 3D printed filament.

A first-time assembly of the ABCS hardware has proven that the design holds its own weight on wheels so to speak. With more testing planned, the structural and mechanical components will be fully verified.

4.3.2.1.1 Structural Design

Post-PDR, the ABCS design has changed to account for manufacturability. The new system retains all components and general functionality, as the system still uses the umbrella method with a coupler. The umbrella style of deployment ensures that major system failure modes do not impact the rest of the rocket, while the addition of the coupler ensures that the system does not compromise the vehicle's structural integrity. The coupler ensures that all essential components are kept enclosed by the coupler. As the rocket ascends, the interior of the system will experience minimal changes in pressure. The system still allows the team to increase or decrease the overall area of the aeroplates by lengthening them in design and reduces their retracted drag by enabling them to lie flush with the airframe when retracted. The system still uses linkages attached to a slide plate, which has its vertical position controlled by a lead screw and stepper motor.

The system's most load-bearing parts will be 3D-Printed from Markforge 17-4 PH stainless steel. All other essential components will be printed from Markforge Onyx Carbon Fiber material, excluding the electronics bay, which will be 3D printed out of PLA. Parts are now being 3D printed instead of being manufactured out of aluminum due to the team's time constraint. All components have been redefined by adding more support and thickness to account for the new Markforge Onyx Carbon Fiber and 17-4 PH stainless steel filament used during the 3D printing process. Finite element analysis was not conducted after the parts were redefined. If FEA were to be run on the new materials, it would be inaccurate due to material properties not being defined and the differences in printing techniques.

An electronics bay housing the entire system's electronics will be raised on six 2" standoffs above the Motor Plate. The electronics bay is 6" in diameter and will have two sections, the upper electronics bay and the lower, the entire bay being a length of 3”.

The team has constructed a MATLAB simulation that uses a numerically solved implicit equation of state to determine the motor torque vs. drag relationship and the motor position vs. aeroplate angle relationship. The results from this simulation have been used to size the motor at 75 mm x 75 mm x 621 mm and create a motor control code later discussed. The changes created to account for failure modes associated with using 3D printed parts do not affect the final mechanical system's length within the vehicle. The length is still 6.5", with an additional 3" of length for electronics.

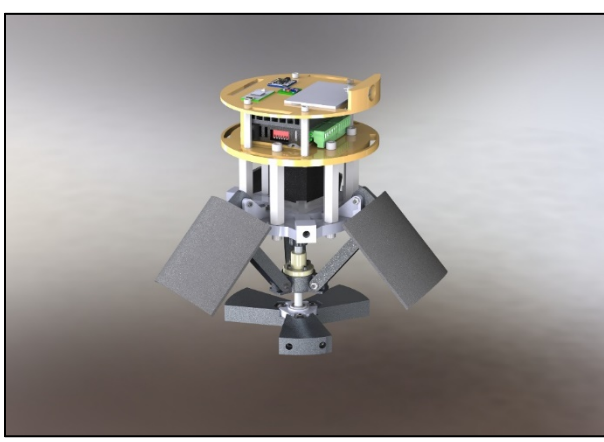


Figure 4.67 ABCS Mechanical System CAD Render

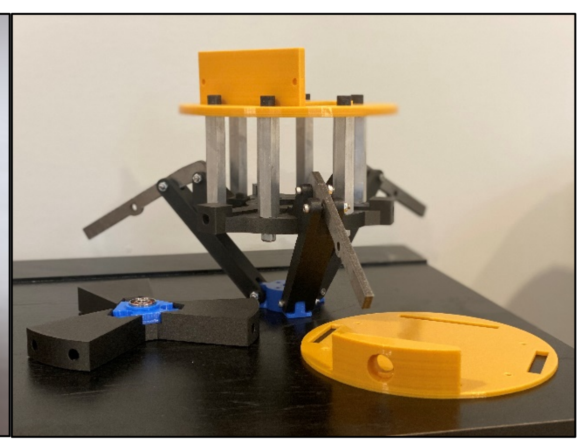


Figure 4.68 ABCS Physical Assembly

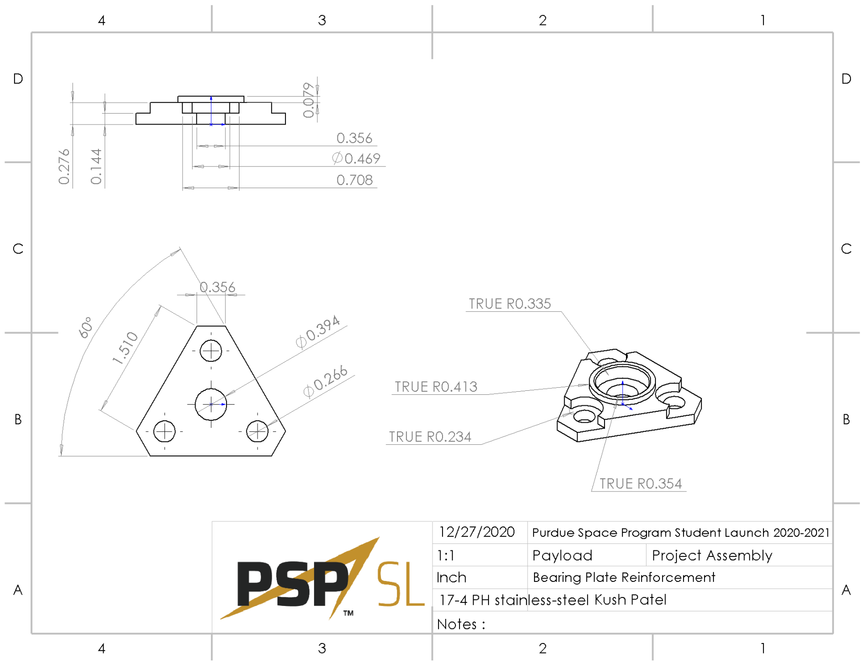


Figure 4.69 Bearing Plate Reinforcement Dimensional Drawing

The component shown above will be manufactured by 3D printing out of Markforge 17-4 PH stainless steel rather than Markforge Onyx Carbon Fiber. This is because the Bearing Plate, shown below, is a component that is seen to be load-bearing, and the team wanted to ensure that it can withstand all forces applied to it by the system. The location of the Reinforcement Plate is centered on the Bearing Plate. The Reinforcement Plate was designed so a more robust material (17-4 PH Stainless-Steel) can entail all extreme force before it is distributed throughout the Bearing Plate, hence its central location.

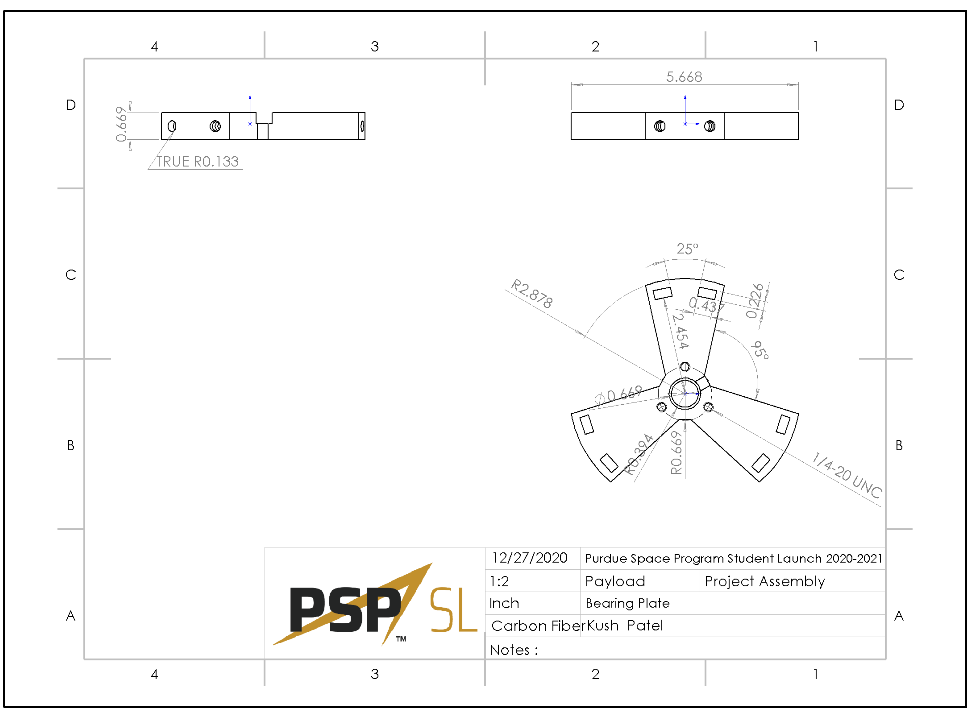


Figure 4.70 Bearing Plate Dimensional Drawing

The Bearing Plate shown above is the base of the system. Though it is load-bearing, it was still printed out of Markforge Onyx Carbon Fiber. The Reinforcement Plate mentioned above will account for the applied load on the Bearing Plate by the system. The end of the lead screw will be attached to this plate, along with the Bearing Reinforcement Plate, a bearing, and locknut. It is also one of the two points of attachment to the Airbrake Coupler and Airframe.

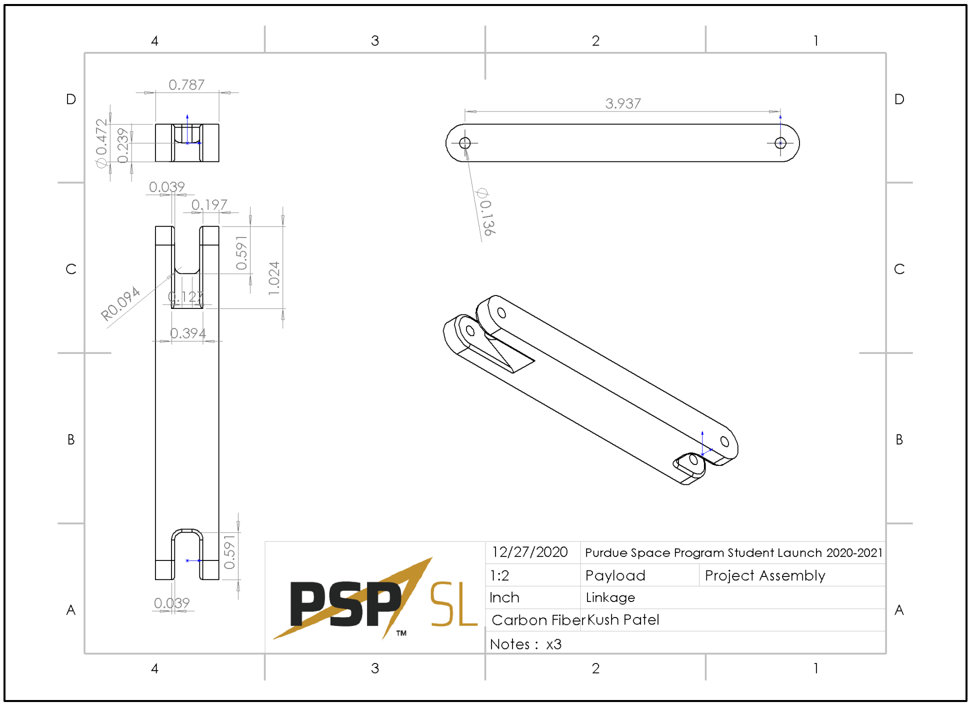


Figure 4.71 Linkage Dimensional Drawing

The linkages shown above attach to the Slide Plate, Motor Plate, and Paddle Struts and are used to help actuate the system. This system has three linkages because it will be actuating three aeroplates simultaneously. Three attachment points were created on the Slide Plate and Motor Plate for the linkages. The linkages are not load bearing and will be 3D printed out of Markforge Onyx Carbon Fiber.

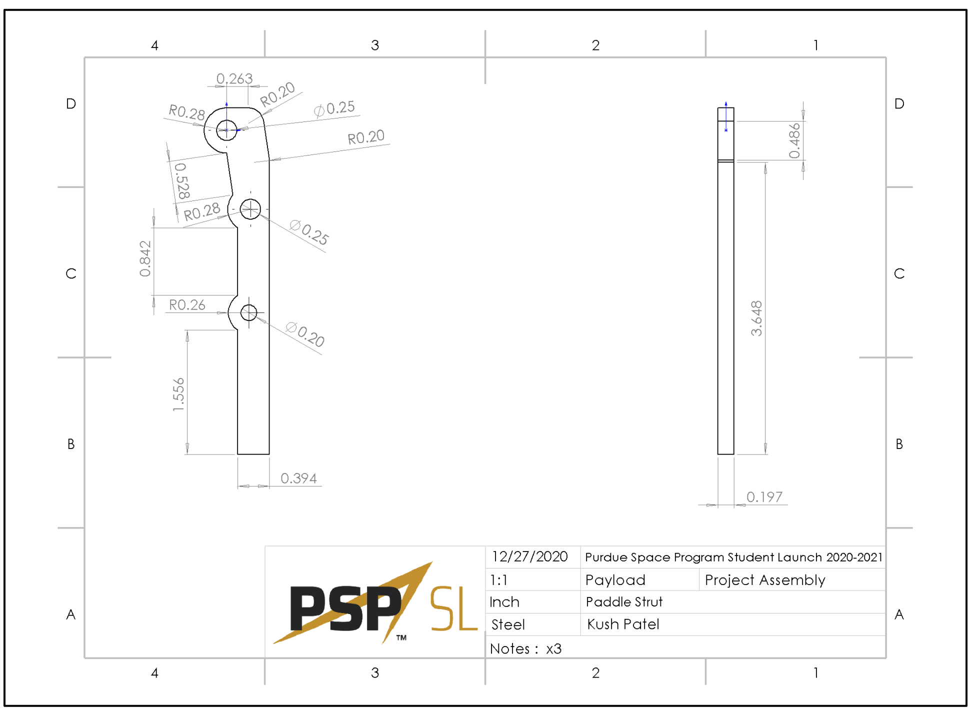


Figure 4.72 Paddle Strut Dimensional Drawing

The Paddle Struts shown above attach to the slide plate, motor plate, and linkages and are used to help actuate the system. The Paddle Struts are also the base for the aeroplates, which will be later epoxied on to the flat most surface shown to the right, above. The Paddle Struts are the most load bearing component of the system since they will encounter all the force pushed on the aeroplates first. Since they are the most load bearing, they will be laser cut out of steel. The team will not be 3D printing this component out of Markforge 17-4 PH stainless steel.

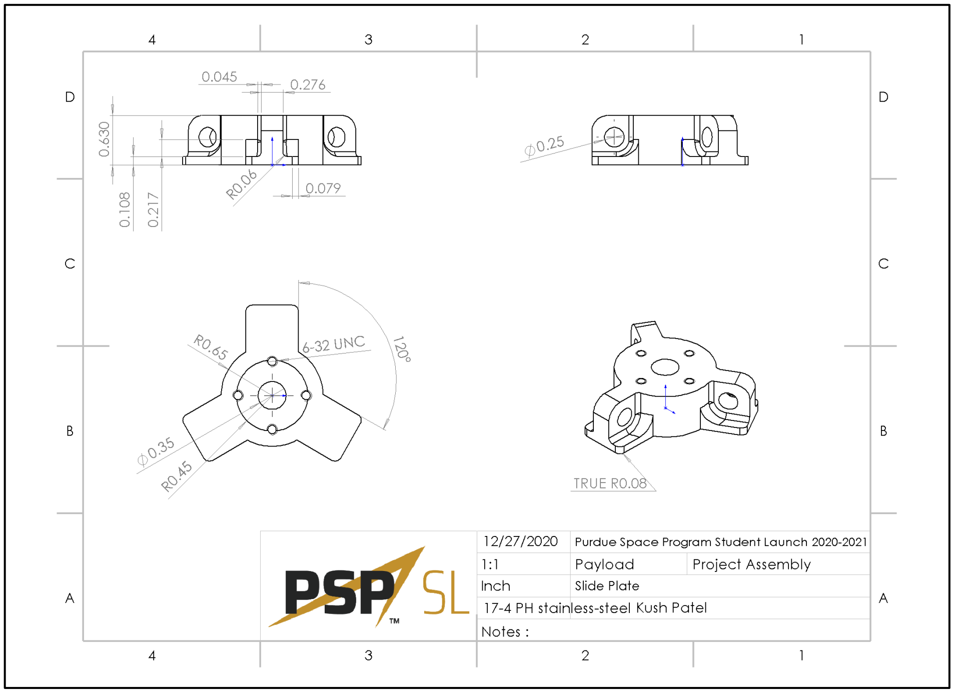


Figure 4.73 Slide Plate Dimensional Drawing

The slide plate is the base attachment point for the linkages. It is also what the system actuates on based on the stepper motor turning the lead screw. The team has considered this component as load bearing since it is the central component of actuation, encountering strain from the downwards force applied from the aeroplates while in air. The team has decided to 3D print the slide plate out of Markforge 17-4 PH stainless steel for that reason.

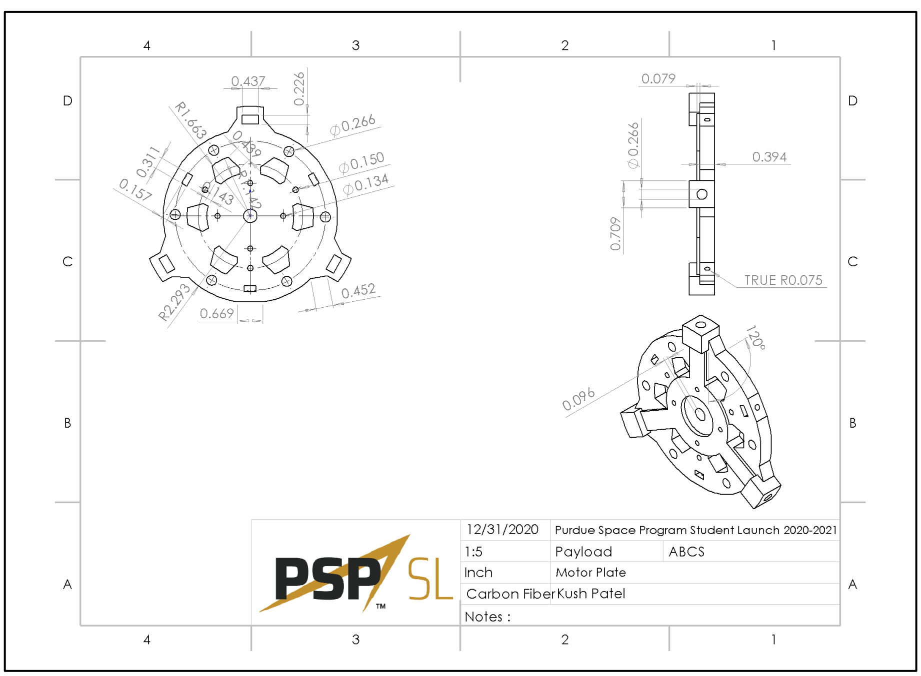


Figure 4.74 Motor Plate Dimensional Drawing

The motor plate shown above is the central component of the ABCS system. It is the attachment point of the paddle struts, electronics bay, and the motor. The motor plate will be 3D printed out of Markforge Onyx Carbon Fiber since the component is not load bearing.

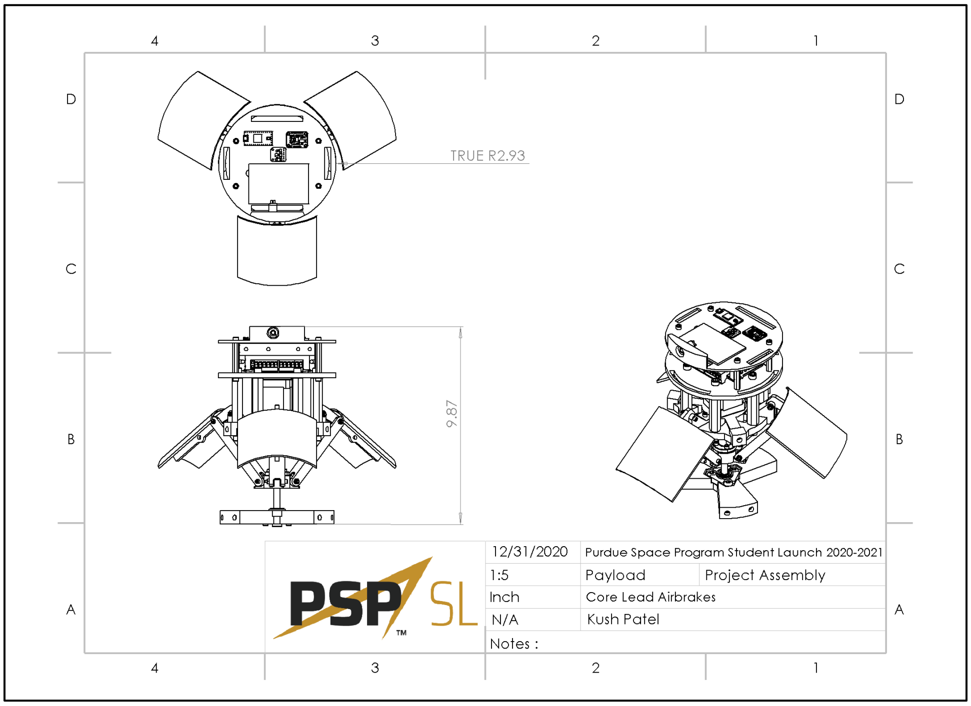


Figure 4.75 Mechanical System Dimensional Drawing

Above is the dimensional drawing for the full mechanical system, including the electronics bay. The total length is 9.87", and the system's maximum diameter is 6". Since PDR, most of the team's effort has been placed on refining the design within SolidWorks initial assembly, defining the control algorithm inputs, creating electrical schematics, and conducting heavy analysis of the aeroplates through *ANSYS* to derive comprehensible testing parameters such as the appropriate weight to mimic in a drag simulation test.

4.3.2.1.2 ABCS Aerodynamic Analysis

As shown in the render of the ABCS system, the drag-generating aeroplates outfitted on the ABCS were designed based on the drag experienced at the maximum possible speed. It was assumed that the maximum speed experienced by the aeroplates would be 154.4 m/s, which is the projected speed at the time of motor burnout. This maximum speed was determined based on the work done to specify the Mission Performance Predictions, as outlined previously in the report.

Firstly, hand calculations were completed for the drag force experienced by the aeroplates, which estimated a maximum drag force of 150 N. This result was computationally validated using ANSYS CFX via Steady State analysis for the max speed of 154.4 m/s. With the domain inlet speed set to the max speed, the drag force within CFX was calculated to 120 N. Convergence was confirmed for these simulations by implementing a user defined point which evaluated the velocity in the flow field downstream of the rocket. This was done to confirm if the simulation reached steady state when convergence was achieved. Although there is a 20% discrepancy between the hand calculations and CFX, it was established that a 1.20 safety factor would be applied to the CFX result due to the unknowns associated with confirming structural aptitude of the 3D printed ABCS structure. Therefore, after applying the safety factor of 1.20, the result of 144 N matched very closely with hand calculations and the rounded number of 150 N was used in executing additional analyses for the ABCS structure.

The validation of the CFD model was also useful in sizing the aeroplates themselves. The aeroplates used in the solid model used in validating the CFD work had width of 100 mm and length of 100 mm, for a max deployment angle. This was an entirely arbitrary size; therefore, the size of the aeroplates was optimized to minimize its area while maintaining its drag generating capabilities, assessed via the drag coefficient for a given geometry. Due to the dual-objective nature of this optimization problem—minimizing area while maximizing the drag coefficient—constrained multi-objective optimization was used to carry out the optimization work. The constraints included physical constraints limiting the width, length, and deployment angle of the system and one dynamical constraint which was to ensure that 120 N of drag force was generated, per plate. As for the drag coefficient, a bluff body relationship which relates the width and length was used as the model. The figure below is the Pareto Frontier, and feasible region in green, from the multi-objective optimization, and the red dot is the parameter point at which the CFD validation was completed. As shown in the figure below, the arbitrary parameters are actually very close to the optimal solution set; therefore, the design shown in the render of the ABCS system was validated and retained.

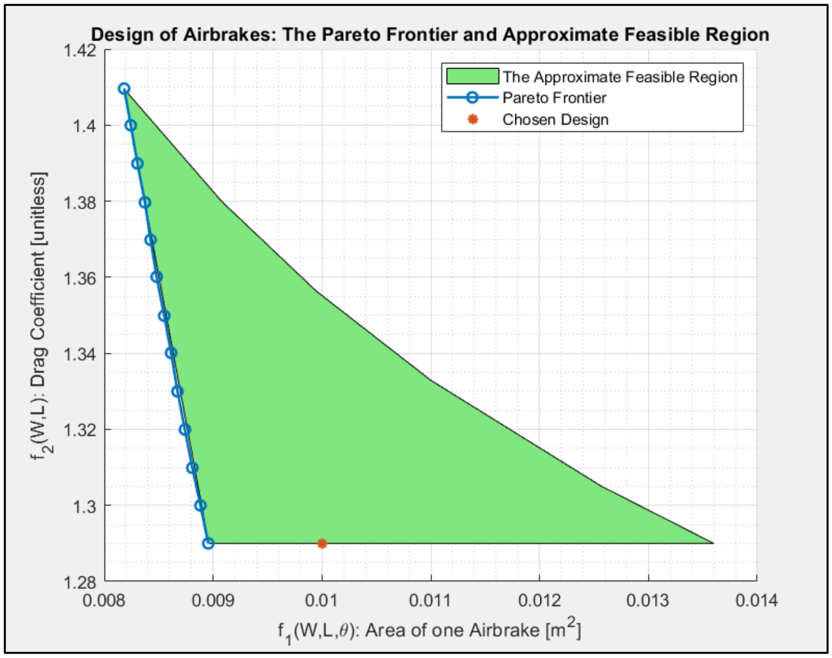


Figure 4.76 Pareto Frontier generated to capture the plate optimization solution

4.3.2.1.3 Selected Drive Motor Apparatus

The stepper chosen to perform the rotational motion required to actuate the ABCS mechanism was the Nema 17 17HS24-2104S Bipolar Stepper motor. This motor had enough torque to be able to provide sufficient speed and rotation to the main ABCS mechanism. This mechanism utilized the rotational force generated by the stepper motor to actuate the paddle struts to provide additional drag after burnout has been achieved.

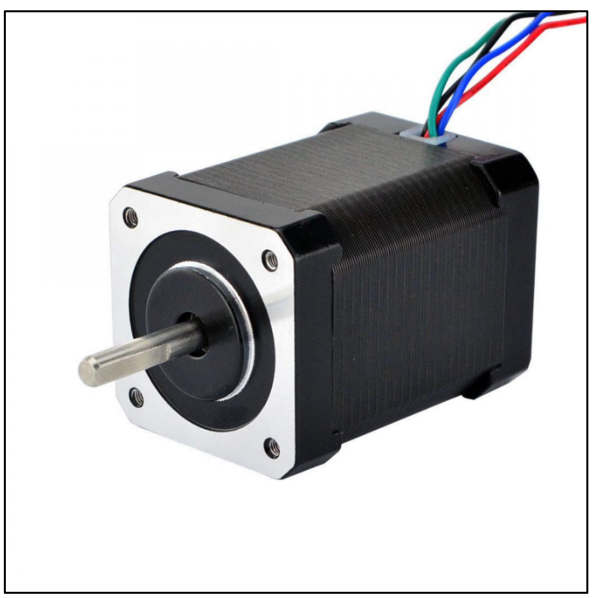


Figure 4.77: Nema 17 17HS24-2104S stepper motor

4.3.2.1.4 Assembly and Physical Testing

The entire assembly was built with the final components, and all parts integrated successfully with the system. The system has not been installed on the rocket. It will be installed after winter break. The actuation of the airbrake system showed promising results in tolerance and linkages clearing smoothly from the motor plate. Actuation also showed linkages clearing the bearing plate. Changes were made to the electronics bay; holes that were set as tapped were changed to clearance so the correct sized screw can fit when the electronics bay is 3D printed. Assembly of design has shown us that the system can work. Physical testing is next. During physical testing, the system will be vertically anchored to a beam. The paddle struts were designed to attach cables at 3 points where the team can apply 45 kg or more of tension to simulate the downwards force acted on all three of the aeroplates simultaneously. This test will show us the structural integrity at the largest possible force the team expects the system to encounter on the new Onyx Carbon Fiber and steel 3D printed parts. Physical testing will also show how well the motor can handle the system's applied torque when actuated against a force.

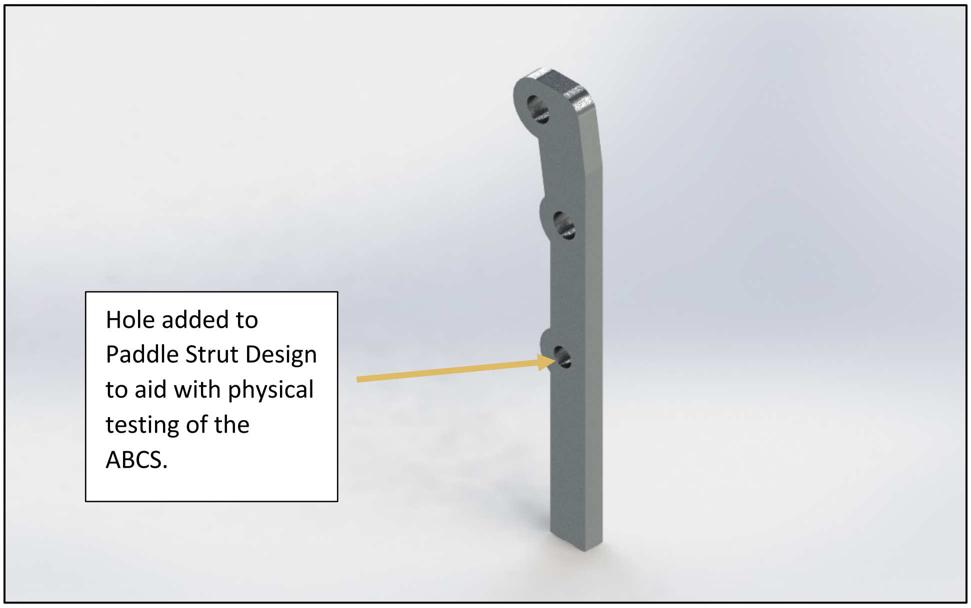


Figure 4.78: Paddle Strut Render

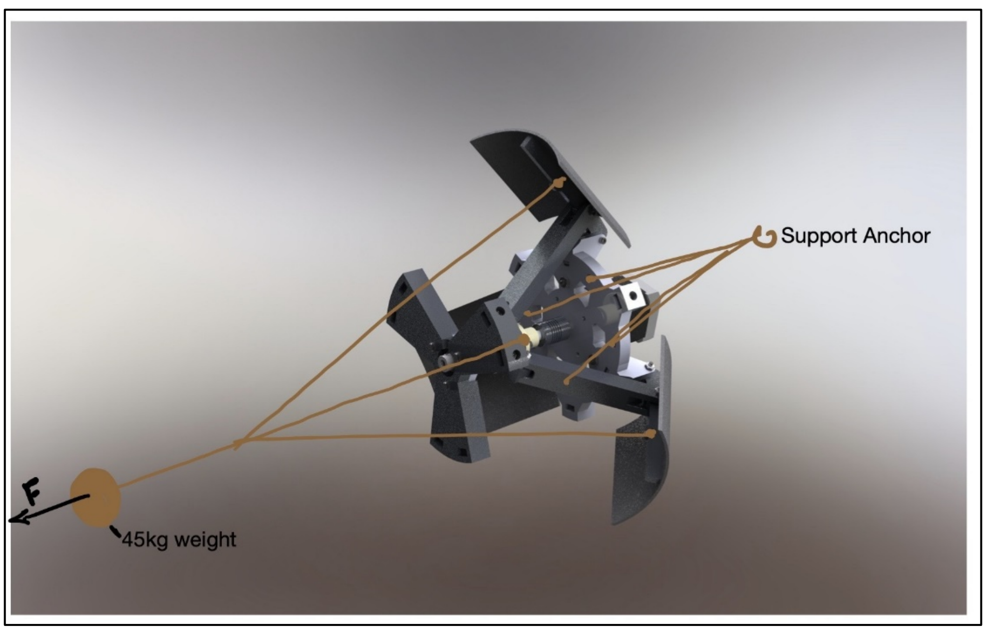


Figure 4.79 Physical Testing Mock-Up

4.3.2.2 Control System Electronic Hardware

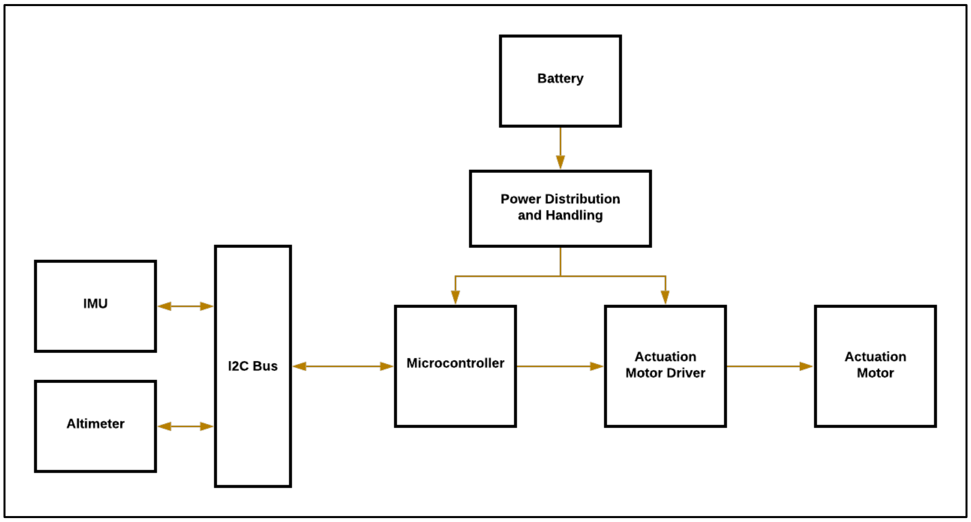


Figure 4.80 FBD for ABCS electronics

Leading up to PDR, the focus of the ABCS team with respect to the electronic hardware was to assess potential candidates for the flight hardware. After executing trade studies for the pressure sensor, IMU, and motor, the BMP280 pressure sensor, BNO085 IMU fusion sensor, and a NEMA 17 stepper motor were chosen. Further details for the hardware are given in the following sections. In preparation for CDR, the team focus has been on acquiring the hardware outlined below and preparing them for prototyping and verifying the electrical schematics. The schematics are also given below.

4.3.2.2.1 Sensor Array

The sensors onboard the ABCS are an altimeter and an IMU. The altimeter is a BMP280 barometric pressure and altitude sensor and the IMU is a BNO085 9-DOF orientation IMU. Three static pressure portholes around the airframe will provide adequate sample airflow to the sensor array. The sensors will be directly mounted to the PCB which will be affixed with insulated bonding or fasteners.

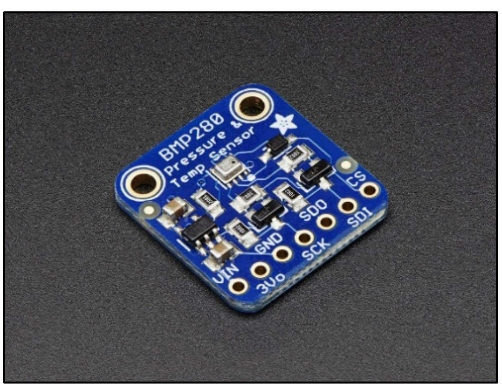


Figure 4.81 Adafruit BMP280

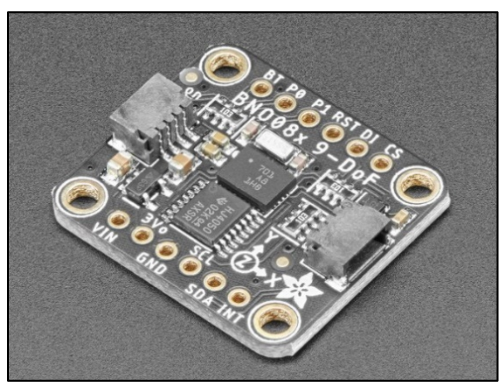


Figure 4.82 Adafruit BNO085

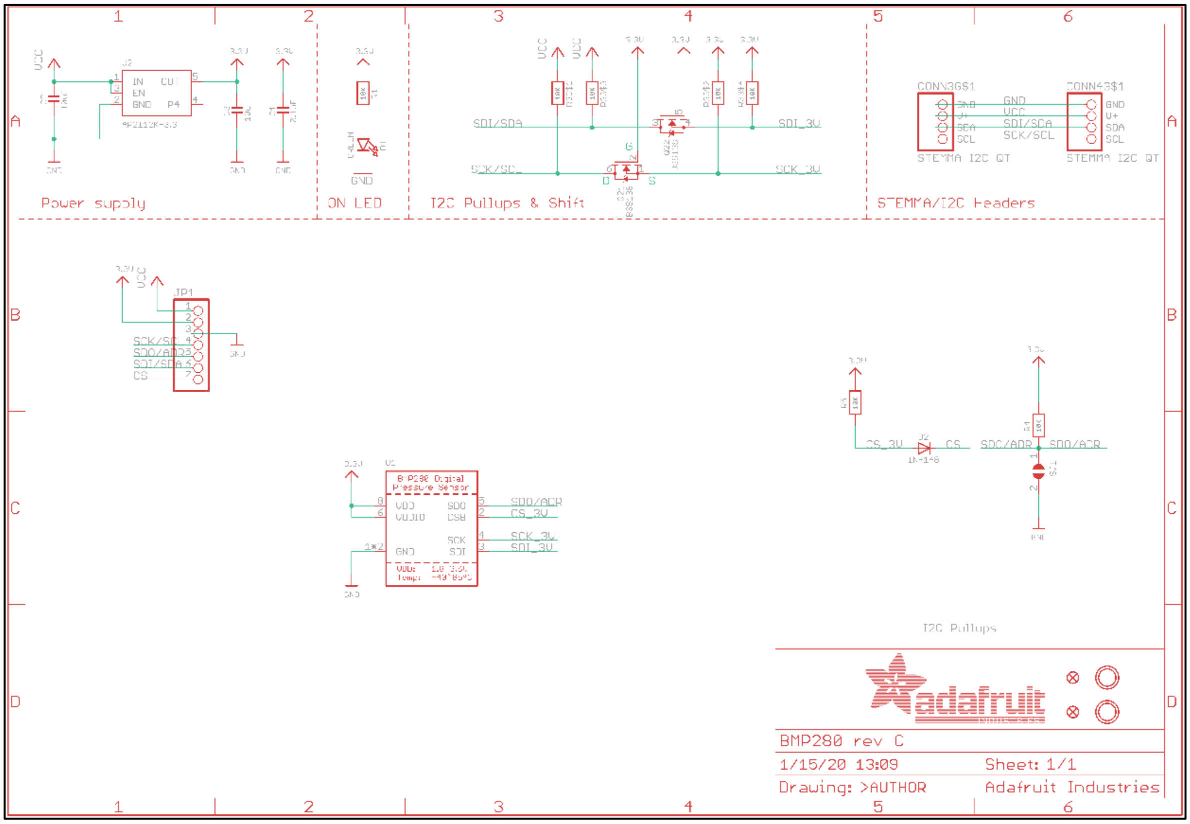


Figure 4.83 BMP280 Schematic

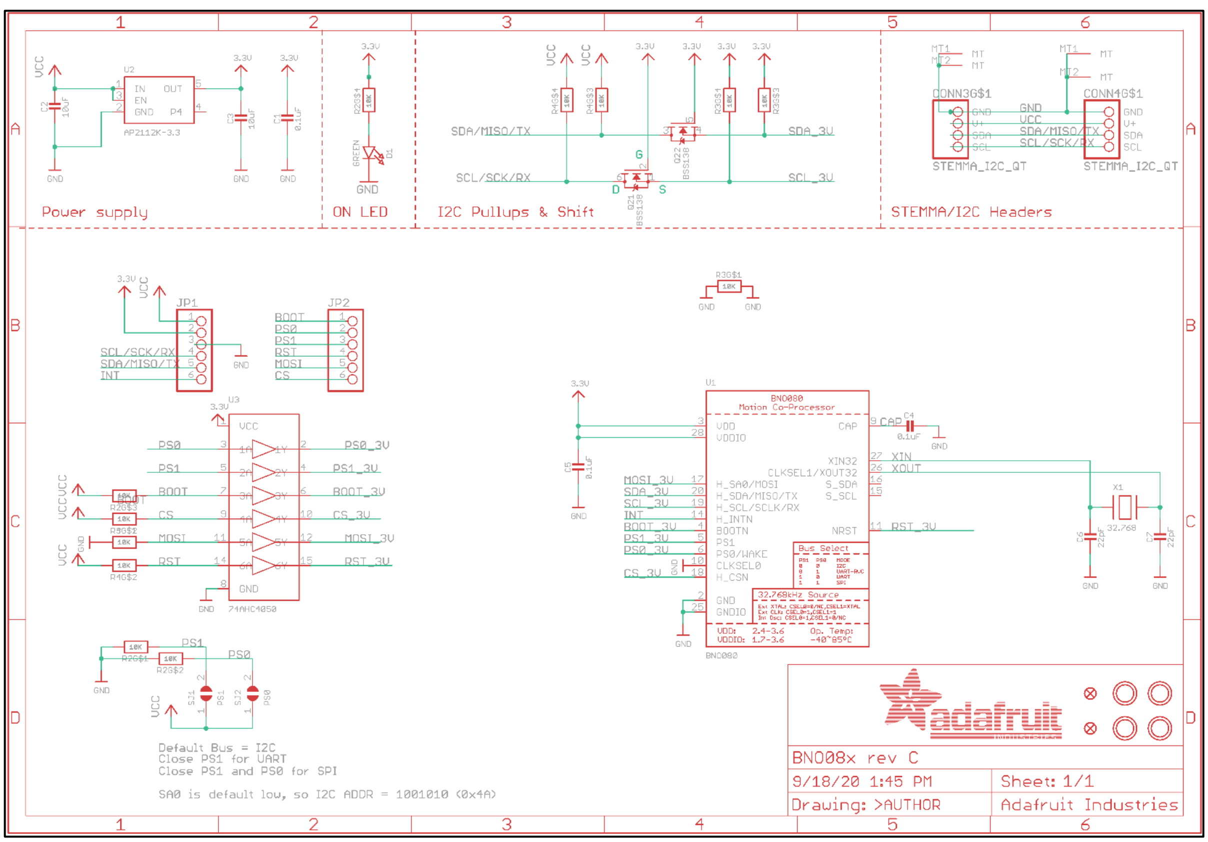


Figure 4.84 BNO085 Schematic

4.3.2.2.2 Motor Driver

The chosen stepper driver was a TB6600 stepper driver capable of delivering up to 4A of current to a wired stepper driver. This driver can be powered using 9V+, allowing the battery to be directly connected to it. The microcontroller will be wired through a 3.3V to 5V logic level converter to the stepper driver since the microcontroller is 3.3V logic and the driver runs on 5V logic. This driver will be able to step the stepper motor, change the direction of motion, and enable or disable the motor. These operations were desired properties of the final selected driver. The connection between the driver and the motor is a simple wiring of 4 wires from the motor to the driver based on predetermined motor configurations. The three connections from the microcontroller to the driver can be seen in the top left of Figure 4.84, labeled enable, direction, and pulse.

4.3.2.2.3 Battery Selection

The ABCS will be independently powered by a battery cell. The power source will be a 3S LiPo Battery Pack, 11.1V RC Lipo 3300mAh. In order to maximize space in the lower airframe, the battery will be secured onto the far side of the coupler plate as depicted below. Several Velcro strips will fasten the cell down to the plate. A test for the duration of the battery will be conducted to ensure that it may endure any prolonged wait on the launch pad.

Figure 4.85 3300mAh battery

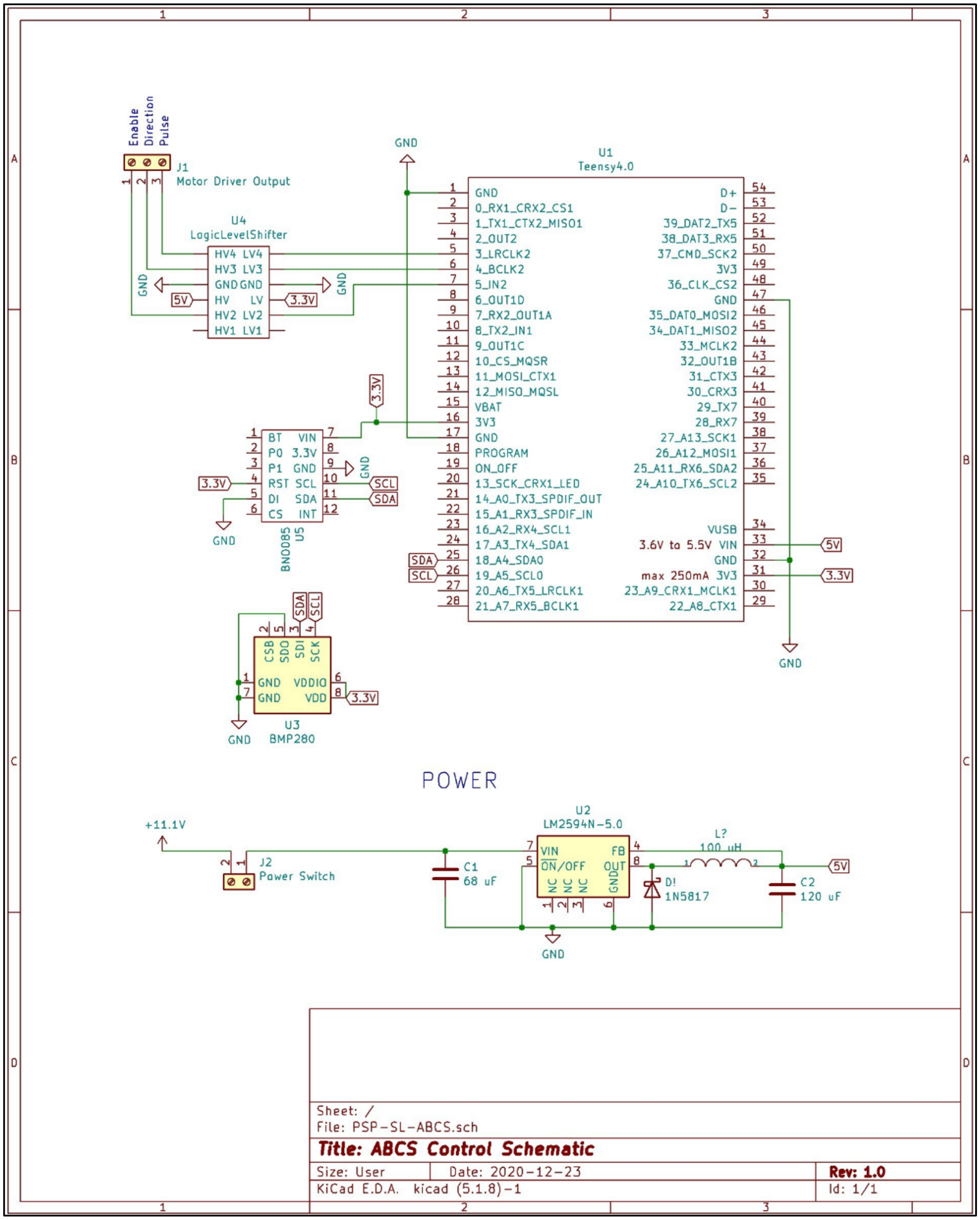


Figure 4.86: ABCS Electronic Schematic

4.3.2.3 Control System Design

The focus of the team prior to PDR was mainly to research relevant electronics that could be applicable to satisfy the goals of ABCS. To summarize, a Teensy 4.0 will serve as the system microcontroller, a BNO085 fusion sensor will be used to measure the launch vehicle states during various stages of coast, BMP280 will be used to measure altitude data, and a NEMA 17 stepper motor will be used to actuate the system. The reasoning behind the selection of these instruments was the fast-processing power of the microcontroller, the advanced signal processing capabilities of the BNO085, and the high precision control of the stepper motor. As outlined in the previous sections, integration of the electronics is nearly complete, and a functional prototype of the electronics is shown in the figure below. Following a functional prototype with respect to the schematics above, the development of flight hardware PCB and software will be the focus of the ABCS team, in preparation for FRR and as Project Voss moves towards its first flight test.

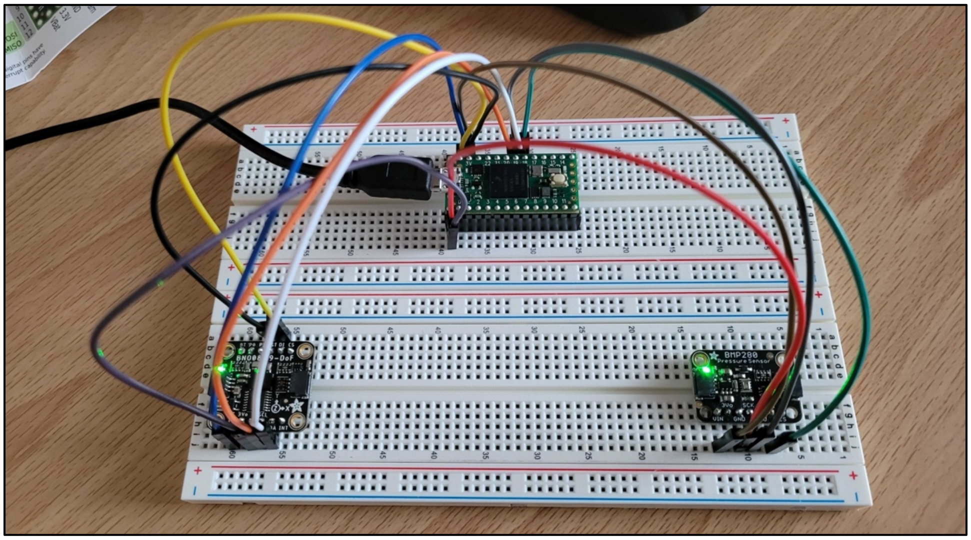


Figure 4.89: ABCS Electronics Prototype

4.3.2.3.1 Design Goals

The goals for the control system design for the ABCS system were to keep to data stream as simple as possible. It was clear from the beginning that the operating regime of the ABCS would very narrow, in terms of time spent in the coast phase by the rocket relative to the rest of the flight regime. However, the deceleration experienced by the rocket during the coast to apogee would quite significant, for example the change in rocket velocity between the time of burnout versus the velocity one second after burnout would be significant. Therefore, a powerful microcontroller like the Teensy 4.0 was required to handle the quickly changing rocket state expected with the actuation of the ABCS. The high processing power of the Teensy 4.0 also allows for the fulfilment of safety-related set of system objectives. Stability measuring characteristics about the primary axes of the rocket will be built into the software to ensure that the ABCS performs without compromising the stability of the vehicle.

4.3.2.3.2 System Input Stream

Once the switch is flipped on the ABCS, the entire system will be functional and running. The two sensors will be active from the beginning of the launch to the end. As the rocket completes burnout, the IMU will sense a decrease in the acceleration experienced by the rocket to detect initiation of the coast phase, triggering the control loop for the ABCS. For the sake of redundancy, a timer function will active the ABCS in case the burnout is not detected. Through three static portholes, the altimeter will take in and convert temperature and pressure readings to altitude data.

4.3.2.3.3 Control System Process

The ABCS will use a continuous control loop to convert sensor inputs into motor command outputs that will then be used to deliver the necessary drag force to achieve the desired apogee. Specifically, altitude and air pressure will be measured using the altimeter, and velocity and acceleration will be measured using the IMU. The following flowchart illustrates the decisions the software will be making during the launch vehicle's flight.

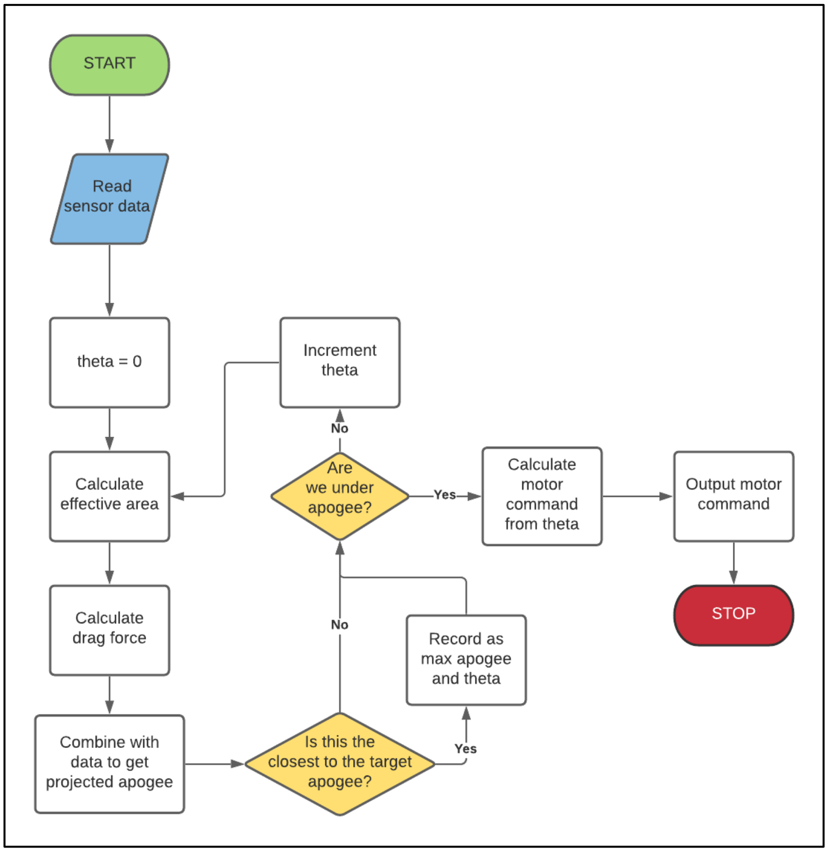


Figure 4.90: Flowchart of software

As shown in the flowchart, the logic can be divided into three parts. First, the control loop chooses a value for the angle of the aeroplates and uses this value to determine cross-sectional area of the launch vehicle along with the drag force this area would produce. Then, the sensor data (air pressure, altitude, velocity, and acceleration) will be used along with the hypothesized drag force to obtain an estimation of the apogee the chosen angle will lead to. This is repeated for as many angle choices as is necessary for the software to find the angle that will result in the least distance from our desired apogee. Finally, the angle is converted into a specific position on the ABCS lead screw, which allows the software to calculate the proper motor command and deliver said command to the stepper motor itself. The code this flowchart represents will run continuously during flight, ensuring that the aeroplates are always actuated at the proper angle to generate the needed drag force.

4.3.2.3.4 Redundancy and Safety

The ABCS is mechanical system not designed with physical redundancies in place. Instead, the system is inherently designed to ensure that the aeroplates fully retract if there are any anomalies encountered during flight. The decision to fully retract the aeroplates was made being cognizant of the fact that the rocket expect to fly nominally without ABCS functionality, due to the secondary nature of the ABCS. In the case of a mechanical failure, the system is designed so that the default state of the ABCS is with the aeroplates retracted. However, the material choices during the design ensured that a mechanical failure would be averted, the system's performance under static loading will be confirmed following final assembly.

To ensure that system responds appropriately within any other failure mode, the redundancies will be implemented within the flight software. The BNO085 IMU allows for the direct measurement of angular velocity and angular position about the three principal axes of the rocket. These quantities will be used to monitor any large deviations from expected values prior to the actuation of the ABCS. The thresholds against which the angular velocity and position will be compared will be developed using the Mission Performance Predictions. Also, as the IMU will be used to detect burnout through the transition from upwards acceleration to downwards acceleration, a redundant time-based trigger will also be implemented to take hold if the IMU fails to recognize motor cutoff; that is, if the IMU fails to detect burnout after a specified time, the ABCS will activate anyways assuming no other anomalies. The specific time will be based on the average time the motor takes to completely burn its fuel with some extra time allotted for safety. Additionally, the Mission Performance Predictions will be used to develop the performance thresholds for angular velocity and displacement assessments for ABCS actuation in the coast phase as well. The performance thresholds assessments will be written in the flight software so that the assessments take place after each instance of actuation to ensure that the rocket is not placed in adverse flight profile.

4.3.3 **ABCS System Integration**

4.3.3.1 Mechanical with the Launch Vehicle

The ABCS will be installed to the Airframe along with a coupler. There are nine attachment points offset from the three aeroplates. These attachment points are located on the Bearing and Motor plates. Each attachment point on the Airframe will be secured to each respective plate by a square nut, which will be embedded within the plates. The ABCS coupler is used to help enclose the interior components of the ABCS, ensuring the structural integrity of the system and the rocket by reducing the expected build-up of pressure if the system was not enclosed. The coupler is secured to the ABCS's nine attachment points to the Airframe. The coupler will be manufactured from G12 Fiber Glass and will be cut by using a Dremel. A cut tray will be 3D printed to help guide the team's cuts, ensuring everything attaches to the Airframe and actuates properly.

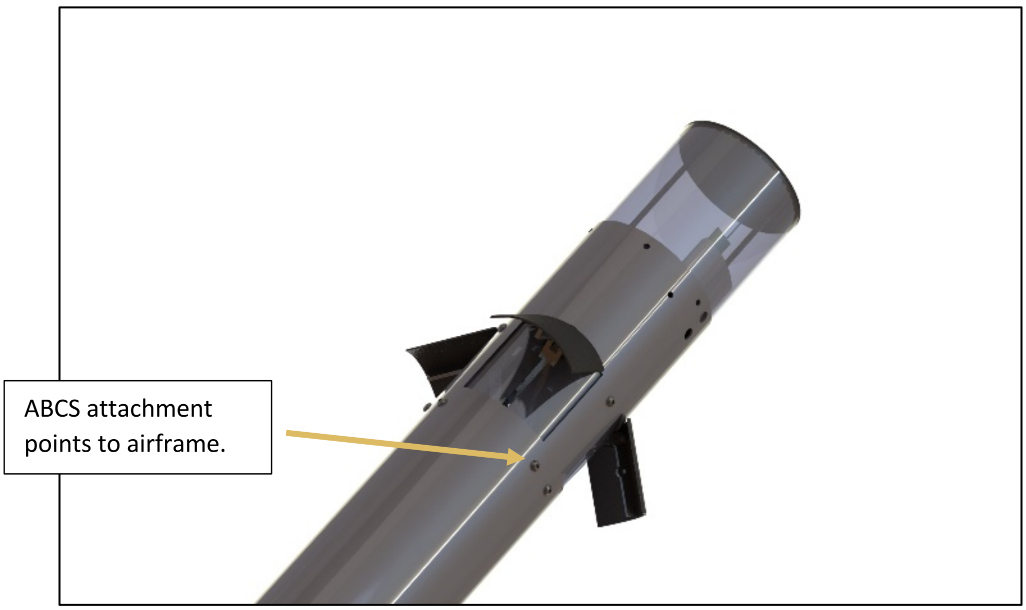


Figure 4.91 ABCS in Respect to Airframe

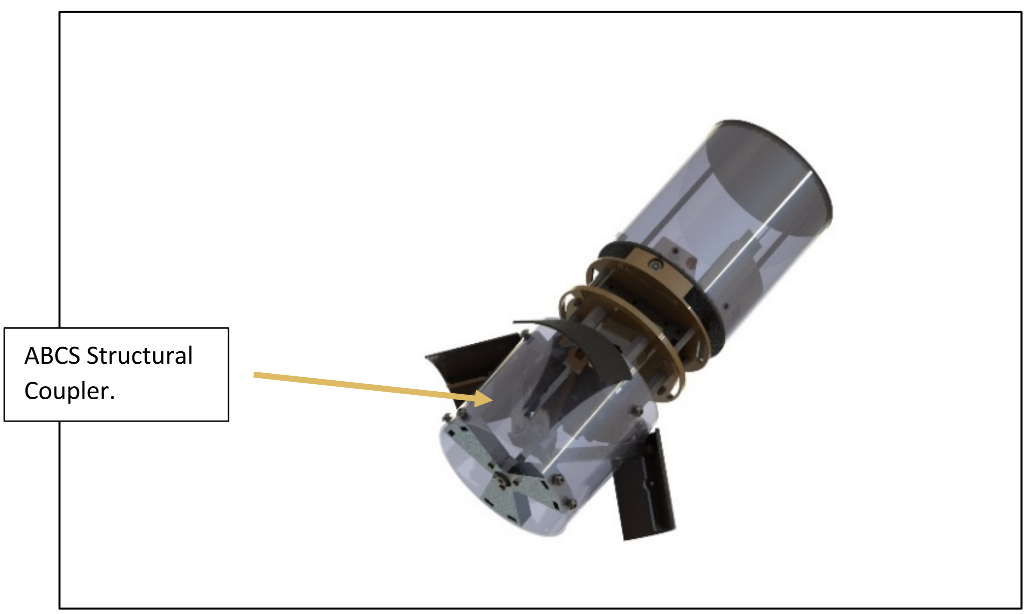


Figure 4.92 ABCS Coupler Attachment

4.3.3.2 Electrical with the Launch Vehicle

A few electrical components require attention to integration. Firstly, the safety switch for the ABCS will protrude through the couplers and airframe. Similar to other switches it will be bonded sufficiently to the mount surface inside the electronics bay. Another component requiring seamless integration is the battery. The wiring shall be carefully threaded through the upper coupler plate to the PCB. Furthermore, the altimeter will require static portholes that bring in the surrounding airflow through the structural coupler and airframe. Jigs that will surround the rocket will be 3D-printed in order to accurately drill three 6.5mm holes. Lastly, the upper coupler will host a GPS tracker unit in case the lower airframe gets lost in flight. The unit includes a switch, a battery, and a transmitter.

5 Safety

An integral part of a mission's success includes a thorough understanding of the hazards that the project will face. Threats to team personnel, the launch vehicle, the environment, and the project encompass the overall risk associated with the mission. Throughout the design, construction, testing, and launch of the vehicle, as well as successes and failures of previous flights, risks are identified and minimized. The team is aware that not every hazard can be foreseen nor mitigated, but acknowledgement of the known dangers to the project and a plan to minimize or eliminate these risks is the only way to maximize the safety of team, the safety of the project, and the success of the mission.

5.1 Operation Procedures

5.1.1 **Final Assembly and Launch Procedures/ Checklists**

In preparation for actual launch operations, the drafts for checklists have been created to maintain a clear flow of launch operations and ensure maximum personnel and vehicle safety. In addition, the creation of these checklists allows the team to create contingencies against worst case scenarios, namely misfires or unintended ballistic trajectories. While there is no way to truly lessen the danger associated with the latter, the creation of a contingency is the best method for reducing the chance of tragedy.

5.1.1.1 In Advance

<u>Programming the TeleMetrum Altimeter</u>

(Note: the LiPo battery can be charged by plugging it into the TeleMetrum and then plugging the TeleMetrum into a laptop with a micro-USB cable. The red light will turn green when the battery is fully charged. A switch does not need to be connected.)

1) Connect the LiPo battery and a switch to the TeleMetrum, then plug it into a laptop with AltOS installed (https://altusmetrum.org/AltOS/) using a micro-USB cable.
2) Choose Configure Altimeter and then turn on the TeleMetrum using the switch. It should appear as a device to select. Select the TeleMetrum device and continue to the settings window. The TeleMetrum should halt beeping altogether as a connection indicator.
3) Configure settings as desired. In this case, the main deploy altitude should be set to 900' and the apogee delay should be set to 0 seconds. Also ensure the following:
    a) Frequency: 434.550 MHz Channel 0
    b) Telemetry/RDF/APRS Enable: Enabled
    c) Telemetry Baud Rate: 9600 baud
    d) APRS Interval(s): 5
    e) Callsign: KD2IKO
    f) Maximum Flight Log Size (kB): 8192 (1 flight)
    g) Igniter Firing Mode: Dual Deploy
    h) Pad Orientation: Antenna Up
4) Choose Save to save the new settings to the TeleMetrum. If desired, the accelerometer can be calibrated by choosing Calibrate Accelerometer and the TeleMetrum rebooted by choosing Reboot.
5) Before any flight, choose Save Flight Data and delete all previous flights to ensure no actual flight logs are lost.

<u>Programming the StratoLoggerCF Altimeter</u>

1) Connect the DT4Ux cable to the USB-B → USB-A cable, then connect the DT4Ux cable to the data port of the StratoLoggerCF altimeter and the USB-B → USB-A cable to laptop with PerfectFlite DataCap installed (http://www.perfectflite.com/Download.html). Also connect a 9V battery and a switch to the StratoLoggerCF.
2) Open the DataCap software and turn on the switch. While the StratoLoggerCF is performing its initialization beeps, ensure the correct comm port (COM5) is selected by choosing Altimeter, then CommPort. Also choose Altimeter, then Setup to ensure the connection to the altimeter was successful. If it was, the serial number and current settings should appear, and the altimeter should halt its initialization beeps and begin beeping once every few seconds as a connection indicator.
3) Configure settings as desired. In this case, Preset 3 should be chosen, which sets the main deploy altitude to 700' and the apogee delay to 1 second. Also ensure the Siren Delay is set to 0 seconds. Choose Update Alt to save the settings to the StratoLoggerCF.
4) If desired, self-tests can be performed by choosing Altimeter, then Test.
5) There is no way to delete previous flights from the StratoLoggerCF. After 16 flights, the oldest will be automatically deleted.

<u>Setting Up the EggFinder Trackers</u>

1) Use a key to turn on the keylock switch of one EggFinder TX.
2) Plug the corresponding EggFinder RX into a laptop. The red LED should immediately come on, indicating the board has power. After one or two seconds, the green LED should then begin blinking, indicating that it is receiving data from the EggFinder TX.
3) On the Prolific serial driver webpage (https://prolificusa.com/product/pl2303hx-rev-d-usb-serial-uart-bridge-controller/), the correct driver to download is the "Windows Driver Installer Setup Program" at the bottom of the list. After downloading, extract the zip file and run the application "PL2303-WHQLDriver_Setup_v1230_20190815.exe", following the instructions to install the driver.
4) Download and run MapSphere (http://www.mapsphere.com/download/mapsphere). You will need to create a MapSphere account first before use.
5) Repeat steps 1 and 2 with the other EggFinder TX and RX on another laptop.

<u>Assembling the Avionics Bay</u>

- As specified in the CAD model, cut the two threaded rods to length using a Dremel and tap the altimeter mounting holes using an appropriately sized screw.
- Epoxy the switch band around the center of the avionics coupler and drill out all holes in the coupler/switch band assembly using the hole jig (static port holes, airframe interface screw holes, switch holes, and switch holder interface screw holes).
- On each key switch, solder a length of 22 AWG stranded wire onto each lead, twist them together, and add a wire sleeve at the top for support. Also solder each wire onto the metal contacts of the male JST connector and add a wire sleeve at the bottom for support.
- Insert the key switches into each switch holder and secure with the included nuts. Hot glue the switch holder interface hex nuts into place on the underside of the switch holder using appropriately sized screws as guides.
- Attach the switch holders with the key switches inside to the interior of the coupler using appropriately sized screws. Also glue the press-in nuts to the appropriate places on the interior face of the coupler with E6000.
- As specified in the CAD model, drill out all holes on each bulkhead (threaded rod holes, eyebolt holes, black powder canister holes, terminal block holes, and wire through-holes).
- On each bulkhead, attach an eyebolt using an appropriately sized lock nut and washer. Also, as specified in the CAD model, attach the black powder canisters using screws, washers, and hex nuts, and attach the terminal blocks using screws and hex nuts.
- Cut four more pieces of 22 AWG stranded wire to length and twist together into two sets. On each set, crimp the female metal JST contacts onto one end of each wire and slide them into the female JST connector. Add a wire sleeve at the interface for support. These will be the switch connection wires for each altimeter.
- Cut eight more pieces of 22 AWG stranded wire to length and twist together into four sets. These will be the drogue and main lighter connection wires for each altimeter.

<u>Preparing the Payload Electronics</u>

- Charge the Lander, R&D, and ABCS LiPo batteries.

- Remove and clear the memory storage drives (SD cards) of all data logging devices. This includes the Lander, R&D, and ABCS.
- Final software needs to be uploaded to the GCS and Lander.
- Reinsert all memory storage drives into their respective logging devices.
- Calibrate the inertial measurement units on the Lander, R&D, and ABCS.
- Calibrate the altimeters on the Lander, R&D, and ABCS.

5.1.1.2 Component Assembly

QUALITY WITNESS NOTE: Quality witnesses will be responsible for double checking a completed part with which they were not involved in its assembly or construction. Upon final assembly prior to launch, they will be responsible for answering if their step has been completed and witnessed.

Avionics Bay

- Black powder installation (PPE required: gloves and safety glasses):
    - Cut the fingertips off four fingers of a nitrile glove. Measure out 2g, 3g, 3g, and 4g quantities of FFFFG black powder using a gram scale and funnel one of each into a glove fingertip. Insert a lighter cut to size into the fingertip and seal shut using two small zip ties.
    - Insert the 2g and 3g black powder charges into the corresponding black powder canisters on the drogue bulkhead and the 3g and 4g black powder charges into the corresponding black powder canisters on the main bulkhead. Pack tightly with fireproof cellulose insulation and seal with masking tape. Screw the lighters into their corresponding terminal blocks.
- Screw each set of switch connection wires into the switch terminals of each altimeter. Also, screw a set of both drogue and main lighter connection wires into the corresponding terminals of each altimeter.
- Attach the TeleMetrum and StratoLoggerCF altimeters to their corresponding sets of built-in mounting posts on the altimeter sled using nylon altimeter mounting screws.
- Insert the (fully charged) 3.7V LiPo and 9V batteries into their corresponding compartments in the altimeter sled and connect the 3.7V LiPo battery to the TeleMetrum. Also, attach a 9V battery connector to the 9V battery and screw the connector into the battery terminals on the StratoLoggerCF.
- **QUALITY WITNESS - Avionics Bay Assembly:** Inspect avionics bay component for the following. If any are missing, damaged, or otherwise incorrect, halt launch procedures and direct attention to the avionics team lead, who will appropriately respond to the irregularity.
    - Inspect for presence of:
        - One washer on each of the 2 threaded rods on the drogue bulkhead side
        - 2 nuts on each of the 2 threaded rods on the drogue bulkhead side
        - One washer on each of the 2 threaded rods on the main bulkhead side
        - 2 nuts on each of the 2 threaded rods on the main bulkhead side
        - Altimeter sled
        - Battery guard
        - 4 nuts on each of the 2 threaded rods securing the avionics sleds
        - TeleMetrum altimeter
        - StratoLoggerCF altimeter
        - 4 wires connecting the TeleMetrum to the main and drogue terminals
        - 4 wires connecting the StratoLoggerCF to the main and drogue terminals
    - Pull test each wire connecting the altimeters to the ejection charges
    - Check that the batteries are connected to their respective altimeters
        - 3.7V LiPo battery connects to the TeleMetrum
        - 9V battery connects to the StratoLoggerCF
- On each threaded rod, screw on two hex nuts so that there is about 0.8” between the top of the second hex nut and the bottom of the threaded rod. Place a washer on each threaded rod.
- Slide the drogue bulkhead facing down onto the threaded rods.
- On each threaded rod, screw on two more hex nuts so that there is about 0.5” between the bottom of the first hex nut and the bulkhead.

- Slide on the altimeter sled with the battery compartment facing up so that it is touching the hex nuts just placed on the threaded rods. Feed the drogue lighter connection wires from each altimeter though the corresponding holes in the drogue bulkhead.
- Slide on the battery guard, then add two more hex nuts onto each threaded rod so that they are touching the battery guard.
- Slide the coupler over the components until it is touching the bulkhead.
- Connect the two sets of switch-to-altimeter JST connections together, and also screw the drogue lighter connection wires into the other ends of their corresponding terminal blocks on the exterior of the bulkhead. Make sure the switches are OFF.
- Feed the main lighter connection wires from each altimeter though the corresponding holes in the main bulkhead, then slide the bulkhead onto the threaded rods so that the coupler is sealed.
- Screw the main lighter connection wires into the other ends of their corresponding terminal blocks on the exterior of the bulkhead. Add a washer and two hex nuts to each threaded rod to secure everything together.

Thrust Structure (MFSS)

- Attach bottom closure plate to bottom centering plate via specified nuts/bolts
- Attach motor retainer to bottom closure plate with the 3 standoffs
- Connect Spar to top and bottom centering plate with required nuts/bolts
    - Repeat for remaining 2 spars
- Place fin inside spar, screw in all 5 screws
    - Repeat for remaining 2 fins
- Insert MFSS into aft of the launch vehicle
- Screw in 6 screws in top centering plate after motor installation (Note: motor installation occurs later in the launch procedures)
    - Repeat for bottom centering plate
- **QUALITY WITNESS – Thrust Structure Assembly:** Inspect MFSS assembly for the following. If any are missing, damaged, or otherwise incorrect, halt launch procedures and direct attention to the Construction Team Lead, who will appropriately respond to the irregularity.
    - Inspect for presence of:
        - Securement of all fins inside spars
        - MFSS secured into aft of launch vehicle

Payload Bay

- Lander:
    - Construct the associated subsystem plates: SOS, D&L, PICS, and LCS
        - The D&L attachment and release mechanisms must be installed.
        - The LCS board should be connected to essential components and the battery.
        - The battery and reed switch are loaded into their slots.
    - Assemble the Lander by inserting the skeletal threaded rods into the main attachment points.
        - The LCS board will be fastened AFTER the LCS, Battery, and D&L plates are assembled.
        - The SOS's servos should be attached and wired prior to attachment.
        - The PICS cupola should be constructed in its entirety before being fastened to the top of the Lander.
    - Prepare the parachute and attach to the D&L attachment mechanism.
- R&D:
    - Ensure that the R&D stepper is installed into the Payload Bay coupler.
    - Ensure that the R&D Pizza Table rail assembly has been installed into the Payload Section.
    - Assemble the Pizza Table and parachute bag apparatus. Set aside for vehicle integration with the Lander and nosecone.
    - Load R&D electronics into their holder. Set aside for vehicle integration with Payload Bay coupler.

ABCS

- Assemble the airbrakes mechanism, including central lead screw and aeroplates.
- Assemble the ABCS electronics bay.
    - Insert the battery into its coupler slot and connect to electronics bay.
- Integrate the mechanism and electronics bay.
- Integrate the ABCS and its couplers into the Lower Airframe.

QUALITY WITNESS – Payload Bay Assembly: Inspect Payload Bay assembly for the following. If any are missing, damaged, or otherwise incorrect, halt launch procedures and direct attention to the Payload Team Lead, who will appropriately respond to the irregularity.

- Inspect for presence of:
    - Proper installation of SOS system
    - Proper installation of D&L system
    - Proper installation of PICS system
    - Proper installation of LCS system
    - Proper installation of ABCS electronics bay
    - ABCS integration into Lower Airframe / Booster Section

5.1.1.3 Vehicle Assembly

- Quantity Witness Note: See the Quality Witness step for Vehicle Assembly below. The inspection for orange tape over all quick link fasteners and proper parachute folding technique must be done as those steps are occurring. The inspection for screws and shear pins in their proper locations must occur after the vehicle is fully assembled.
- Inspect all recovery system elements, including quick link fasteners and parachute shroud lines, for damage.
- In each shock cord, make three loops (one on each end and one 1/3 of the shock cord length from one end). For every 10' of shock cord, make one bundle of z-folds and tape it together with masking tape. Attach large quick links to every loop.
- Fold the main parachute on a tarp so that it is long and thin. Attach the drogue parachute and a Nomex blanket to the middle quick link of the 30' shock cord and the main parachute and another Nomex blanket to the middle quick link of the 60' shock cord. Flag each quick link with orange tape to signify it has been closed (both links).
- Attach the shorter end of the drogue shock cord to the eyebolt on the bulkhead of the drogue side of the avionics bay and the longer end to the eyebolt on the bulkhead of the booster section through the lower recovery section. Flag each quick link with orange tape to signify it has been closed. Reconnect the lower recovery section to the booster section using shear pins. A rubber mallet may be required.
- Insert the drogue parachute and shock cord into the lower recovery section and make sure they are adequately covered on the top with the Nomex blanket to protect them from ejection charge gases. Reconnect the lower recovery section to the avionics bay using screws.
- Attach the longer end of the main shock cord to the eyebolt on the bulkhead of the main side of the avionics bay and the shorter end to the eyebolt on the bulkhead of the payload section through the upper recovery section. Flag each quick link with orange tape to signify it has been closed. Reconnect the upper recovery section to the payload section using shear pins. A rubber mallet may be required.
- Insert the folded main parachute and shock cord into the upper recovery section and make sure they are adequately covered on the top with the Nomex blanket to protect them from ejection charge gases. The main parachute should be as loose as possible while still fitting in length into the upper recovery section. Reconnect the upper recovery section to the avionics bay using screws.
- Use a key to briefly turn on and off each keylock switch through the switch band and listen for initialization beeps from each altimeter to ensure all wiring is still intact.
- Measure the Center of Gravity of the vehicle and mark that location on the outside of the vehicle's body.
- **QUALITY WITNESS - Vehicle Assembly:** Inspect final vehicle assembly for the following. If any are missing, damaged, or otherwise incorrect, halt launch procedures and direct attention to Project Management, who will appropriately respond to the irregularity.
    - Inspect for presence of:
        - Proper packing of both drogue and main parachutes
        - Orange tape around the closed connections at these points:
            - Shock cord to main parachute
            - Shock cord to drogue parachute
            - Drogue shock cord to booster section
            - Drogue shock cord to drogue bulkhead
            - Main parachute shock cord to payload bay
            - Main parachute shock cord to main bulkhead

            - Shear pins at these points:
                - Upper recovery section to payload bay
                - Lower recovery section to the booster section
            - Screws at these points:
                - Upper recovery section to avionics bay
                - Lower recovery section to avionics bay

5.1.1.4 On Launch Site

- Briefing to team members by Safety or Systems Team Lead:
    - Timeline of events prior to, during, and after launch
    - Launch field etiquette
    - NAR minimum safe distance from launch vehicle
    - "Scatter" callout in case of ballistic trajectory
    - Identification of fire suppression and first aid equipment
    - Designate a "rapid response" person or persons to be the one(s) to perform duties such as administering first aid in the case of an emergency.
    - Designate spotters to keep track of the launch vehicle's descent and to point out its location as it falls.

Events marked with * may occur at the same time.

- *Selecting a launch area:
    - Student Mentor, Project Management, Construction Team Lead, and Safety Team Lead select a launch area that is free from wildlife intrusion and general obstructions.
    - Ensure a fire blanket has been placed under the pad if conditions at launch are dry enough to require it.
    - Safety Team Lead marks off NAR minimum safe distance for personnel and communicates it to the team.
- *Inspect all vehicle components for damage from travel.
    - If damage has occurred, Project Management must be notified to determine whether the launch may proceed.
- *Inspect motor, motor casing, and motor retainment system for damage.
    - If damage has occurred, Project Management must be notified to determine whether the launch may proceed.
- *Ensure two-way radios are functioning properly

<u>*Lander Initialization</u>

- The team will determine an appropriate area to initialize the Lander. The area should be relatively flat and free of any obstacles that may obstruct the Lander. Ideally this location will be close to the team's viewing area so the GCS will not need to be moved.
- The GCS will be powered on and initialized.
- The Lander will be placed on the ground on its side and will be turned on using the key switch.
- The Lander will automatically enter its initialization sequence.
    - The Lander will establish a connection with the GCS
        - If the Lander cannot establish a connection the buzzer will sound and the team will need to troubleshoot the problem.
    - Once the Lander has established a connection to the GCS the Lander will send codes for any errors that were detected during the initial checks. If there are no errors then the Lander will send a signal confirming it is ready for the next initialization phase.
        - If errors are received, the team will need to troubleshoot and resolve them before moving onto the next step
    - The member at the GCS will announce to nearby team members that the Lander is about to perform the orientation maneuver. This will be to ensure the safety of nearby team members and the safety of the Lander. This announcement will also alert non-payload team members so they can witness the glory of what the payload team has created.
    - While the Lander is orientating, team members will closely inspect the Lander for any defects or unexpected behavior. If something is noticed, project management will be notified and will decide if the issue needs to be corrected.

- Once orientated, the lander will acquire a position fix, and run the PICS system. The Lander will send the GPS data as well as the images to the GCS. The Lander will also report any errors detected. Team members will review this data and resolve any errors before continuing.
- Once satisfied the Lander is ready for launch, the team will send a command from the GCS to instruct the Lander to enter its Launch-Ready configuration. The team member entering the command must also notify nearby members that the Lander will be moving.
  - The Lander will return the legs to the starting position
  - The LCS will place all peripheral devices, except the Xbee, into their sleep modes.
- The Lander will now be ready to be loaded into the Payload Bay.

*ABCS Initialization

- The ABCS should already be installed into the Lower Airframe.
- The ABCS can be turned on utilizing its key switch.
- The ABCS will enter its initialization sequence, providing audible and/or visible feedback of flight-ready status.
- The ABCS is now armed, but will remain inactive until altitude and acceleration criteria are met for activation.
  - The Lower Airframe should be handled carefully; dropping the ABCS at this point could potentially cause malfunction and activation of the drag plates.

*Vehicle Integration

- R&D:
  - When the nosecone is prepared, it should be installed onto the Pizza Table attachment plate.
  - The R&D electronics and holder will be installed into the Payload Bay coupler before it is installed into the launch vehicle.
  - The R&D electronics holder is fastened into the Payload bay coupler by a single airframe bolt. The key switch should align with its outer hole.
  - AT THIS POINT, R&D-VEHICLE INTEGRATION SHOULD PAUSE UNTIL RESUMED.
  - The Payload Bay rear bulkhead can now be attached to the forward bulkhead and installed, paying special attention to avoid rotation of the forward bulkhead.
- Lander:
  - THE FIRST HALF OF INTEGRATION OF THE R&D MUST BE COMPLETE TO CONTINUE.
  - The Lander's parachute should be packed into the Pizza Table's deployment bag and held on top of the Lander cupola.
  - The Lander and Pizza Table/Nosecone can be loaded together into the Payload Bay.
    - The Lander should fit within the Pizza Table's confines.
    - The Lander must be installed into the only correct rail orientation. Failure to do so will not trigger the Lander's reed switch via magnet.
  - When the Lander detects the magnet in the payload bay it will send a message to the GCS confirming that the payload has been loaded then both the Xbee and main microcontroller will be put to sleep. If the payload is inserted into the payload bay and confirmation is not received by the GCS, then team members will inspect the Lander and the payload bay for the cause of this error. If necessary, the Lander will be removed from the bay and the problem corrected. The GCS must receive confirmation that the Lander is loaded before the team can move on to next steps. If the Lander is erroneously pulled out of sleep mode or detects an error after the Xbee is put to sleep, the buzzer will sound to notify team members.
  - AT THIS POINT, LANDER-VEHICLE INTEGRATION SHOULD PAUSE UNTIL RESUMED.
  - Utilizing on-board connections on the R&D PCB, the Pizza Table should be installed onto the R&D electronics servo lead screw, avoiding over-tightening of the Pizza Table.
  - R&D-VEHICLE INTEGRATION MAY RESUME

R&D Initialization

- THE FIRST HALF OF INTEGRATION OF THE LANDER MUST BE COMPLETE TO CONTINUE.
- The R&D can be turned on utilizing its key switch.
- The R&D will enter its initialization sequence, providing audible and/or visible feedback of flight-ready status.
- The R&D is now armed, but will remain inactive until altitude and acceleration criteria are met for activation.
  - The Payload Bay should be handled carefully; dropping the R&D at this point could potentially cause malfunction and activation of the Pizza Table mechanism.

- LANDER-VEHICLE INTEGRATION MAY RESUME

Motor Installation

- Prep and install motor (PPE required: gloves and safety glasses). Note: The Student Mentor is the ONLY person allowed to install the motor and ignition system.
    - Grease motor tube forward and aft closure threads.
    - Bolt on forward closure (with eye bolt attached).
    - Place one grain in motor tube.
    - Insert RUBBER washer.
    - Repeat last two steps for all motor grains.
    - Apply lubricant as necessary to O-rings.
    - Bolt the aft closure / nozzle onto the motor tube.

Note: Igniter and nozzle cap will be added once the launch vehicle is on the launch pad. Under no circumstances are they to be inserted prior to being on launch pad.

- Install motor into lower airframe.
- Safety Team Lead will call out team members responsible for all Quality Witness steps to ensure that the launch vehicle is ready for flight. These are:
    - Avionics Bay Assembly
    - Thrust Structure Assembly
    - Payload Bay Assembly
    - Final Vehicle Assembly
- If any Quality Witness step cannot be verified, halt launch proceedings and investigate any source of uncertainty. Launch operations may continue ONLY when all Quality Witness steps can be accounted for.
- If all Quality Witness steps are accounted for, the vehicle is ready for installation onto the launch pad.

Installing the Vehicle on the Launch Rail

- Check that the weather conditions remain favorable for launch
- Move launch vehicle to launch rail
    - NOTE: Only launch essential personnel and those carrying the launch vehicle are allowed to accompany the launch vehicle to the launch pad
    - Ensure launch rail is at least the minimum safe distance from spectators based upon the NAR minimum distance table
    - Ensure the launch controller is disarmed prior to installing the launch vehicle onto the pad
    - Ensure the launch pad is stable and is an adequate size for the launch vehicle being used
- Tilt launch rail and slide launch vehicle onto rail along rail buttons
- Ensure the launch vehicle slides smoothly along the launch rail.
    - If this is not the case, halt launch proceedings to lubricate the launch rail and check the rail buttons for proper alignment.

Altimeter Continuity

- Use a key to turn on both keylock switches for the altimeters through the switch band.
- The StratoLoggerCF should emit the following sets of beeps. One lower-pitched beep precedes each set. Ensure **bolded** events occur.
    - 3 beeps (indicating **Preset 3** was set).
    - 7 beeps, then 10 beeps, then 10 beeps (indicating a main deploy altitude of **700'** was set).
    - One very long beep (indicating an apogee delay of **1 second** was set).
    - Beeps corresponding to the apogee recorded in the previous flight in feet.
    - Beeps corresponding to the battery voltage in volts (ones place, then tenths place). Count to ensure the voltage is at least **above 8.0V**.
    - **3 continuity beeps** every 0.8s.
        - If only 2 continuity beeps – Indicates continuity on only main lighter
        - If only 1 continuity beep – Indicates continuity on only drogue lighter
        - If 0 continuity beeps – Indicates continuity on neither drogue nor main lighters
- The TeleMetrum should emit the following sets of beeps. Here *dits, dahs,* and other specific beeps are specified. Ensure **bolded** events occur.

  - Beeps corresponding to the battery voltage in volts (ones place, then tenths place). Count to ensure the voltage is at least **above 3.3V**.
  - *Dit*, *dah*, *dah*, *dit* (indicating the TeleMetrum is in **Pad Mode** and waiting for launch).
    - If only *dit*, *dit* – Indicates the TeleMetrum is in Idle Mode; ensure it is in the correct orientation (pointing up)
  - **3 continuity *dits*** every 5s.
    - If only 2 continuity *dits* – Indicates continuity on only main lighter
    - If only 1 continuity *dit* – Indicates continuity on only drogue lighter
    - If *brap* – Indicates continuity on neither drogue nor main lighters
    - If *warble* – Indicates storage is full; need to delete extraneous flights
- Use a key to turn on both keylock switches for the trackers.
- Ensure all static port holes are clear of debris.

<u>Installing Ignitor</u>

- Ensure ONLY the Student Mentor installs the ignitor.
- Ensure ignitor clips are clean and undamaged.
- Ground ignitor clips to ensure excess static charge has been dissipated.
- Install ignitor into the motor.
- Return to the viewing area.
- Ensure the ignition system has continuity.

<u>Setting Up the TeleDongle at the Launch Viewing Area</u>

- Assemble the TeleMetrum antenna. The longest prongs go at the bottom and the shortest go at the top.
- Plug the antenna into the TeleDongle, then plug the TeleDongle into a laptop with AltOS installed.
- Open AltOS and choose Monitor Flight. The TeleDongle should appear as a device to select. Select the TeleDongle device and continue to the telemetry window.
- Set the frequency to 434.550 MHz Channel 0 and baud rate to 9600 baud. Live telemetry from the TeleMetrum should now be appearing on the screen.
- Ensure all lights are **green**.
  - Battery, apogee igniter, and main igniter voltages are all **above 3.3V**.
  - On-board Data Logging is **Ready to record**.
  - **At least 4** GPS satellites are in solution. This may take a few minutes.
  - GPS Ready is **Ready**.
- Also ensure Site Map tab is filled with launch area.

<u>Setting Up the EggFinder Trackers at the Launch Viewing Area</u>

- Follow step 2 in the “Setting Up the EggFinder Trackers” section above.
- In MapSphere, choose GPS, then Configure. Choose the COM port the GPS is connected to, then OK.
- In the GPS Status tab to the lower right, GPS satellites should begin coming into view. **At least 3** GPS satellites must be in solution. This may take a few minutes.
- In the main map, **current location should now be shown** as an orange triangle and be tracking in real time.
- Repeat the above steps with the other EggFinder tracker. Another laptop must be used, although it may be the same one that the TeleDongle is plugged into.
- **The vehicle is now ready for launch.**

5.1.1.5 Countdown to Launch

- Ensure the launch and the flight are not angled towards any spectators or buildings.
- Check cloud ceiling and winds and make sure the skies around the launch area are clear.
- Ensure there are no obstructions or hazards in the launch area.
- Designate 2 rapid response persons to administer first aid and call for help, respectively.
- Designate 2 spotters to track launch vehicle’s flight path.
  - Spotters must point to the launch vehicle at all times.
- Remind spectators of the appropriate reaction to a ballistic trajectory and “scatter” call
  - If a “scatter” is called, all personnel must turn away from the launch vehicle and run for at least 10 seconds.

- Shortly before the countdown, give a loud announcement that the launch vehicle will be launched; if applicable to the situation, use a PA system.
- When launching, give a loud countdown of "5, 4, 3, 2, 1, LAUNCH!"
- Spotters are to follow the path of the launch vehicle and call any deviation or unusual behavior in the vehicle's flight (unsteady flight, sudden course deviation, etc.).
- Ensure deployment of drogue parachute is evident at most 4 seconds after apogee.
- If no sign of drogue deployment is apparent, see Troubleshooting below.
- Call a loud "Heads up!" (If needed, sound an air horn) in the case of any launch vehicles approaching the prep area or spectators; all who see the incoming launch vehicle should point at it as it descends.
- Make sure whoever is responsible for recovery is kept fully aware of the status of the launch vehicle (failed to launch, nominal in-flight, midair failure, returning for recovery, etc.).
- Communicate launch progress effectively to NASA officials, if needed.

5.1.1.6 After Vehicle Has Landed

- Assign teams to approach the main vehicle and payload. Note: Ensure these teams are only as large as needed to prevent unnecessary personnel from coming into contact with the vehicle and payload systems.
    - Give each team and the group of team members not approaching the vehicle or lander a two-way radio for communication.

<u>At the Landed Vehicle</u>

- Ensure the main parachute is secured and not dragging the vehicle along the ground.
- Take numerous photos at various angles of the landed vehicle.
- The StratoLoggerCF should emit the following sets of beeps. One lower-pitched beep precedes each set.
    - Beeps corresponding to the apogee in feet.
    - Beeps corresponding to the maximum velocity in miles per hour.
- The TeleMetrum should emit the following sets of beeps.
    - Beeps corresponding to the apogee in meters.
- Once these three values are recorded, use a key to turn off both keylock switches for the altimeters through the switch band. Note: Failing to record altimeter data before turning them off might result in disqualification from competition.
- Gather the sections of the vehicle and carry back to the launch viewing area.
- Inspect all vehicle components for damage. If damage is present, report it to proper team lead(s).

<u>Payload Ground Mission</u>

- Each launch will have a team member designated as the payload recovery lead (PRL). This team member will oversee locating the Lander and ensuring that all recovery tasks are performed.
- If necessary, the PRL will use GPS data to locate the Lander. If another team member reaches the Lander, they will wait so that the PRL will be the first to approach the Lander.
- If the wind is blowing the parachute away from the Lander, then the PRL will put another team member in charge of recovering the parachute.
- As the PRL approaches the Lander they will look for any visual damage to the Lander, paying close attention to damage that may present a danger to other team members such as sharp edges or battery damage.
- If the PRL determines there is an immediate safety threat with the Lander, the PRL will notify the safety lead via radio or send another team member. If a danger is present the PRL must stay near the Lander to warn other team members.
- If there is not an immediate safety threat but a danger is observed, then the PRL will clearly identify the dangers to nearby team members. The PRL will then make a decision if extra PPE, such as gloves or safety glasses will be needed to recover the Lander. If extra PPE is required, the PRL will instruct team members to acquire the necessary PPE.
    - Note: at this point the Lander is still in the upright position
- The next step is to verify that the nichrome is at a safe temperature. The PRL will take the temperature of the nichrome using a non-contact thermometer.
    - If the nichrome is a safe temperature then the recovery can continue on its normal path.
    - If the nichrome is still hot then the PRL will warn nearby team members. The PRL will then take great care to avoid the nichrome as they use the key switch to disable the Lander. The PRL will then wait for the nichrome to cool before continuing with recovery.

- Now that recovery is safe to continue the PRL will ensure the following tasks are completed:
    - Pictures of Landing area and Lander are taken.
    - Any damage and irregularities are documented for future analysis.
    - If the parachute detached, pictures of the parachute and the approximate distance from the Lander.
- The PRL will now use a radio, or send a team member, to request that the Lander is put into its recovery state. A team member will use the GCS to send the recovery command to the Lander. The Lander will then slowly return the legs to the start position causing it to fall back on its side. The PRL will ensure that team members are clear of the Lander for this operation.
- Once in recovery mode, the Lander can be disabled and returned to the team's prep area. If the Lander was disabled in the upright configuration due to a nichrome error, then it will need to be recovered in the upright position.
- Team members will partially disassemble the Lander documenting any damage or irregularities found inside the Lander. The batter will be disconnected and stored safe place. The SD card will also be removed and stored for later analysis.

<u>Downloading GPS Data from the EggFinder Trackers at the Launch Viewing Area</u>

- In MapSphere on one laptop, choose GPS, then Save GPS-log to save the GPS log as a raw GPS data file.
- Repeat the above step for the other EggFinder tracker on the other laptop.
- Use a key to turn off both keylock switches for the trackers.

<u>Post-Flight Tasks</u>

- Notify NASA officials to verify the results of the launch, if necessary.
- Disarm the launch controller.
- Place cap on launch rods, if necessary.
- Take down the launch pad, if necessary.
- Perform sweep of launch field to ensure no materials are unintentionally left behind.

5.1.1.7 In Days After Launch

<u>Downloading Flight Data from the TeleMetrum Altimeter</u>

1) Follow step 1 in the "Programming the TeleMetrum Altimeter" section above.
2) Choose Save Flight Data and then turn on the TeleMetrum using the switch. It should appear as a device to select. Select the TeleMetrum device and continue to the next window. The TeleMetrum should halt beeping altogether as a connection indicator.
3) Select the newest flight and save the raw TeleMetrum data file to the laptop.
4) To display a plot, statistics, and a map from the flight, choose Graph Data, then select the raw TeleMetrum data file just saved. The plot can be configured by choosing Configure Graph and selecting different options.
5) To convert the raw TeleMetrum data file to a CSV file, choose Export Data, then select the raw TeleMetrum data file just saved. Choose Save to save the CSV file to the laptop. Coordinate location information can also be saved by changing the Export File Type to Google Earth Data (.KML) and analyzed in the same way as the raw GPS data files from the EggFinder trackers two sections below.
6) The raw TeleDongle data file can also be saved and analyzed in a similar way when the TeleDongle is plugged into the laptop.

<u>Downloading Flight Data from the StratoLoggerCF Altimeter</u>

1) Follow steps 1 and 2 in the "Programming the StratoLoggerCF Altimeter" section above.
2) Choose Data, then Acquire. Select the newest flight and choose Start.
3) A plot and statistics from the flight should be displayed. Different plots can be displayed by selecting different options under Displayed.
4) To retrieve the numerical data, choose Data, then Inspect.
5) Choose Select All, then Copy. The data can then be pasted into an Excel document to be plotted and analyzed.
6) The raw StratoLoggerCF data file can also be saved by choosing File, then Save As. It may be opened later without a StratoLoggerCF interfaced to the laptop by choosing File, then Open, then the raw data file.

<u>Converting Raw GPS Data Files from the EggFinder Trackers</u>

1) On one laptop, select Choose File in GPSVisualizer (https://www.gpsvisualizer.com/) and choose the raw GPS data file that had been downloaded on launch day.
2) Choose JPEG map as the output format, select Map It, and download the image on the next page to save the map.

3) Go back to the first page, choose plain text table as the output format, select Convert It, and download the text file on the next page to save the coordinate location information. The coordinates can then be pasted into an Excel document and converted to a CSV file for further analysis.
4) Repeat the above steps for the other EggFinder tracker on the other laptop.

Payload Mission Aftermath

- The Lander will be disassembled. At this point, the subsystems can be evaluated to see if they sustained any damage further than what was observed on launch day.
- The SD card will be read to acquire the initial and final orientation as well as any data logged that we deem useful for analysis.
- Any batteries used on launch day will be charged.

5.1.1.8 Troubleshooting

In the case of a misfire:

- Wait a minimum of one minute before approaching launch pad.
- Disarm launch controller and avionics.
- Remove failed igniter and motor if needed.
- Determine if another attempt at launch is feasible.

In the case of unintended ballistic trajectory:

- If the launch vehicle is in freefall for longer than four seconds without any indication of parachute ejection (smoke from ejection charge, parachute deploying), those tasked with observing the trajectory will loudly announce "Scatter."
- All spectators of the launch are to immediately turn away from the direction of the launch vehicle and run for a minimum of 10 seconds.

In case of missing section of launch vehicle during descent:

- If any sections of the launch vehicle are present, inspect for signs indicating point of separation.
    - If failure mode can be determined, keep in mind any dangers that may be associated with the missing sections of the launch vehicle.
- Taking into account last known launch trajectory and wind, on a map or map-analogue identify the most likely location of missing part.
- Assemble team at the edge of the nearest road or other linear landmark.
    - Spread the team out with between 30 and 50 feet between adjacent team members.
    - Instruct team members to keep their gaze between 40 and 50 feet in front of them, scanning the ground in 180-degree arcs, walking in a straight line.
        - If applicable, follow ruts in the dirt from plowing devices or planting
- Once the far end of the search area has been reached, move the search party such that the last person in the line now stands where the first person was before the move
    - Move back in the direction of the initial linear landmark, and repeat search

5.2 Hazard Analysis Methods

The seriousness of a risk is evaluated by two criteria: the likelihood of an event to occur and the severity of the event should it happen or fail to be prevented. The breakdown of the methods used in the team's risk analysis and the assessment of personnel, vehicle failure mode, environmental, and project risks are discussed in the following sections:

5.2.1 Likelihood of Event

Category	Value	Gauge
Remote	1	Extremely unlikely to occur
Unlikely	2	Unlikely to occur
Possible	3	Average odds to occur
Likely	4	Above-average likelihood to occur
Very Likely	5	Very likely to occur/has occurred previously

Table 5.1: Event Likelihood Scale

5.2.2 Severity of Event

Category	Value	Health and Personal Safety	Equipment	Environment	Flight Readiness
Negligible	A	Negligible injury. No first aid required. No recovery time needed.	Minimal and negligible damage to equipment or facility. No required correction.	Negligible damage. No repair or recovery needed.	No flight readiness disruption.
Minor	B	Minor injury. Requires band-aid or less to treat. 5-10 minutes of recovery time required.	Minor damage. Consumable equipment element requires repair.	Minor environmental impact. Damage is focused on a small area. Little to no repair or recovery needed. Outside assistance not required.	Flight proceeds with caution.
Moderate	C	Moderate injury. Gauze or wrapping required. Recovery time up to one day.	Reversible equipment failure. Non-consumable element requires repair. Outside assistance not required.	Reversible environmental damage. Personal injuries unlikely. Outside assistance recommended. Able to be contained within team.	Flight delayed until effects are reversed.
Major	D	Serious injury. Hospital visit required. No permanent loss of function to any body part.	Total machine failure. Outside assistance required to repair.	Serious but reversible environmental damage. Outside assistance required. Personal injuries possible.	Flight on hold until system is removed.
Disastrous	F	Life-threatening or debilitating injury. Immediate hospital visit required. Permanent deformation or loss of bodily function.	Irreversible failure. Total machine loss. New equipment required.	Serious irreversible environmental damage. Personal injuries likely. Immediate outside assistance required. Area must be vacated. Needs to be reported to a relevant environmental agency.	Flight scrubbed or completely destroyed.

Table 5.2: Event Likelihood Scale

5.2.3 Risk Analysis

By cross examining the likelihood of an event with the impact it would have if it occurred, a total risk can be determined, and is detailed in the table below. The color code displayed is as follows:

- Green: Minimal risk
- Yellow: Low risk
- Orange: Medium risk

- Light red: High risk
- Dark red: Very high risk

		Severity				
		Negligible (A)	Minor (B)	Moderate (C)	Major (D)	Disastrous (F)
Likelihood	Remote (1)	A1	B1	C1	D1	F1
	Unlikely (2)	A2	B2	C2	D2	F2
	Possible (3)	A3	B3	C3	D3	F3
	Likely (4)	A4	B4	C4	D4	F4
	Very Likely (5)	A5	B5	C5	D5	F5

Table 5.3: Total Risk Scale

Prior to a plan for risk mitigation, many of the events listed in the following sections fall outside of the acceptable tolerance of Medium risk. Listed alongside these events are the team's risk mitigation plans, as well as verification metrics to ensure team compliance. Post-mitigation risk is also listed, ensuring all project risks are acceptable after mitigation.

5.2.4 Personnel Hazard Analysis

Hazard	Likelihood (Cause)	Severity (Effect)	Risk	Mitigation	Verification	Post Mitigation Risk
Burns from Motor	2 (Improper proximity to launch pad, touching engine too soon after landing)	C (Mild to moderate burns)	C2, Low	Maintain minimum safe launch distance from vehicle according to NAR standards. Wait an appropriate amount of time after launch to retrieve vehicle.	The Safety Team lead will ensure the minimum safe distance region is marked and communicated to team members at the launch.[1]	C1, Low
Contact with Airborne Chemical Debris	3 (Airborne particulate debris generated from construction or testing operations making direct contact with the body)	B (Minor burns, abrasions)	B3, Low	Install / use proper guard on machinery to prevent contact with debris if possible. Always wear appropriate PPE such as gloves, lab coats and breath masks.	Safety Team or relevant Team Lead will verify that each participating member is wearing appropriate PPE during construction and testing operations.	B1, Minimal
Direct Contact with Hazardous Chemicals	3 (Improper use or storage of chemicals leading to unintended contact with the body)	C (Moderate burns, abrasions)	C3, Medium	Minimize the need for hazardous chemicals (for example, see footnote [2]). Always wear appropriate PPE, such as gloves or lab coats, when working with chemicals.	Safety Team or relevant Team Lead will verify that each participating member is wearing appropriate PPE during construction and testing operations.	C1, Low
Dust or Chemical Inhalation	3 (Breathing in airborne particulate debris from construction or testing operations)	C (Short to long-term respiratory damage)	C3, Medium	Work in a well-ventilated area if possible. Always wear appropriate PPE for materials being worked with, such as a respirator.	Safety Team or relevant Team Lead will verify that each participating member is wearing appropriate PPE during construction and testing operations. Team members will not be allowed to work with hazardous materials without proper PPE.	C1, Low
Dehydration	3 (Failure to drink adequate amounts of water)	D (Exhaustion and possible hospitalization)	D3, Medium	Ensure all members have access to water at all team activities, including launch, testing, and construction operations.	Team members will be instructed to bring water to team activities, and team leads will ensure all members are properly hydrated by encouraging	D1, Medium

					members to drink water when not working.	
Heatstroke	3 (Extended exposure to high temperatures during team operations)	D (Exhaustion and possible hospitalization)	D3, Medium	Wear clothing appropriate to the weather, and ensure all members have access to water at team operations.	Safety Team or relevant Team Lead will ensure that team members are appropriately dressed and have enough water. If this is not the case, the team member will be sent to remedy their situation in whatever way the relevant Team Lead deems fit.	D1, Medium
Hypothermia	3 (Extended exposure to cold temperatures during team operations)	D (Sickness and possible hospitalization)	D3, Medium	Wear clothing appropriate to the weather, and ensure all members have access to a warm area to rest at launch, such as a heated car or inside of a building.	Safety Team or relevant Team Lead will ensure that team members are appropriately dressed for team operations. If this is not the case, the team member will be sent to remedy their situation in whatever way the relevant Team Lead deems fit.	D1, Medium
Electrocution	2 (Unintended contact with electrical systems that are faulty or improperly used or stored)	D (Potentially dangerous levels of electricity being passed through a team member, potential hospitalization)	D2, Medium	Give labels to all high voltage equipment warning of their danger and ground oneself when working with high-voltage equipment.	Members working with high voltage equipment must guarantee no open electrical components by inspection. Team Leads must allow only one member to work on electrical components at a time with proper PPE and student supervising.	D1, Medium
Entanglement with Construction Machines	3 (Unintended contact of loose hair, clothing, or jewelry with machines utilizing spinning or binding parts)	F (Severe injury, death)	F3, High	Secure loose hair, clothing and remove jewelry before operation machinery. Always wear appropriate PPE for the machine being worked with.	All use of construction machines will be done under the supervision of a person / people also trained on that specific machine. This is fulfilled by student supervisors at the locations where construction machines are found.	F1, Medium
Epoxy Contact	3 (Bodily contact with resin spill through improper use or storage of resin)	C (Mild skin irritation, possible allergic reaction, redness and rashes on skin)	C3, Medium	Minimize the need for hazardous chemicals (for example, see footnote [2]). Always wear appropriate PPE, such as gloves or lab coats when working with resin.	Team Leads must ensure all members working with hazardous chemicals are wearing proper PPE and are working in a safe environment.	C1, Low
Eye Irritation	3 (Airborne particulate debris entering unprotected eye, dry air / low humidity)	B (Temporary eye irritation)	B3, Low	Install / use proper guard on machinery to prevent contact with debris if possible. Always wear appropriate PPE such as face shields and safety glasses.	Team Leads must ensure all engineering controls available have been implemented and that proper PPE is always worn during team operations.	B1, Minimal
Hearing Damage	4 (Close proximity to loud noises)	D (Long term hearing loss)	D4, High	Seek alternative machines / methods to fabricate the desired part if possible. Always wear appropriate PPE such as earplugs	Team Leads must ensure all engineering controls available have been implemented and that proper PPE is always worn during team operations.	D1, Medium

				when using power tools and larger machines.		
Kinetic Damage to Personnel	2 (Forceful detonation of combustible or explosive materials near team members due to reckless actions or improper storage of materials)	D (Possible severe kinetic damage to personnel)	D2, Medium	Eliminate need for excitable materials if possible. Ensure team members are aware of excitable materials in the workspace and how to properly store and use them.	Team Leads must brief team members on the dangers of the current workplace prior to its use.	D1, Medium
Launch Pad Fire	2 (Completion of fire triangle on launch pad)	C (Moderate burns)	C2, Low	Prevent excess heat from occurring on the launch pad, as the fuel (vehicle motor) and oxygen (air) cannot be removed from the system. Have fire suppression systems nearby and use a protective ground tarp.	The Safety Team Lead is responsible for maintaining proper fire suppression equipment and for bringing it to all launch activities.[3]	C1, Medium
Injury from Falling Vehicle	3 (Vehicle striking team members due to a recovery system failure (ballistic trajectory) or a lack of awareness of vehicle descent under parachutes)	F (Severe injury, death)	F3, High	Keep all eyes on the launch vehicle during flight. Call “heads up” if vehicle is approaching team members under parachutes. Call “scatter” if vehicle is under ballistic descent	Team will be briefed on launch day procedures before the launch occurs by the Safety or Systems Team Lead., emphasizing the importance of keeping eyes on the launch vehicle during flight.[3]	F1, Medium
Injury from Falling Components	3 (Failure to keep all components securely attached to the launch vehicle, result of improper staging constraints, part failure, or excessive vibration during flight)	F (Severe injury, death)	F3, High	Keep eyes on the launch vehicle during flight. Call “heads up” if unintended components separate from the vehicle during flight.	Team will be briefed on launch day procedures before the launch occurs by the Safety or Systems Team Lead., emphasizing the importance of keeping eyes on the launch vehicle during flight.[3]	F1, Medium
Injury from Navigating Terrain	2 (Tripping over uneven ground, contact with poisonous plants, falling into fast-moving water)	F (Broken bones, infections, drowning)	F2, High	Do not attempt to recover the launch vehicle from dangerous areas. Seek professional aid to recover vehicle if it cannot be done by team members.	The Safety Team Lead will set boundaries to not cross at the launch location before the launch occurs and communicate that to the rest of the team.[1]	F1, Medium
Injury from Projectiles Launched by Vehicle Jet blast	2 (Debris striking team members because of a failure to properly clear launchpad, or failure to stand an appropriate distance from the launch vehicle during launch)	C (Moderate injury to personnel)	C2, Low	Clean the launchpad before use. Ensure all members are wearing proper PPE for launch. Ensure all team members are an appropriate distance from the launch vehicle when launching.	The Construction and Safety Team Leads will verify that the launchpad is clean and clear of debris before launch occurs. [1]	C1, Low
Physical Contact with Hot Sources	3 (Contact with launch vehicle parts which were	C (Moderate to severe burns)	C3, Medium	Turn off all construction tools when not in use. Team members must be	Team Leads must brief team members on the	C1, Low

	recently machined, improper use of soldering iron or other construction equipment)			aware of potential hot surfaces created during machining. Always wear appropriate PPE.	dangers of the materials prior to their use.	
Physical Contact with Falling Construction Tools or Materials	3 (Materials which were not returned to a safe location after use striking a team member)	D (Bruising, cuts, lacerations, possible severe physical injury)	D3, High	Brief personnel on proper clean-up procedures for working with tools and materials. Wear appropriate clothing and shoes for machine work.	Team Leads and / or relevant supervisors must ensure team members are aware of proper procedures for cleaning up the current workplace.	D1, Medium
Premature Ignition	2 (Short circuit, improper installation of motor and / or ignitors)	C (Mild burns)	C2, Low	Prepare energetic devices only immediately prior to flight. Ensure ignitor leads are shorted before attachment to the motor.	The Safety and Systems Lead must ensure that the proper personnel prepare the ignition system for flight.[4]	C1, Low
Downed Power Lines	2 (Launch vehicle becomes entangled in power lines, knocking them within range of personnel)	F (Fatal electrocution)	F2, High	Attempt to launch away from power lines. If vehicle entanglement occurs, call the power company and stand clear until proper personnel arrive.	Any team member must alert all team members of the hazard if spotted. The Safety Team Lead must ensure all members are stood clear of the area until certified personal clean up the area and verify it is safe.	F1, Medium
Power Tool Cuts, Lacerations, and Injuries	3 (Carelessness or improper use of power tools, power tool malfunction or failure)	D (Possible hospitalization from damages)	D3, Medium	Ensure loose hair and clothing is tied back and jewelry is removed before operating power tools.	Team Leads must brief team members on the dangers of the current workplace prior to its use.	D1, Medium
Tripping Hazards	3 (Improper storage of materials and equipment, unsecured cables overhead and along the ground)	C (Bruising, abrasions, possible severe harm if tripping into construction equipment)	C3, Medium	Brief personnel on proper clean-up procedures for working with tools and materials. Wear appropriate clothing and shoes for machine work. Tape loose cords or wires to the ground if they must cross a path which is used by personnel.	Team Leads must brief team members on the dangers of the current workplace prior to its use.	C1, Low
Unintended Black Powder Ignition	3 (Accidental black powder exposure to flame or enough electric charge near black powder)	F (Possible severe hearing damage or other personal injury)	F3, High	Label containers storing black powder, ensure black powder is only handled by those with relevant safety training.	Project Management and the Team Leads must verify that the handling of black powder is only done and supervised by team members qualified to handle it.	F1, Medium
Workplace Fire	2 (Unplanned ignition of flammable substance, overheated workplace, improper use or supervision of heating elements, or improper wiring)	F (Severe burns, loss of workspace, irreversible damage to project)	F2, High	Have fire suppression systems nearby, prohibit open flames, and store energetic devices in Type 4 magazines as stated in the CFR, Title 27.	Team Leads must brief team members on the dangers of the current workplace prior to its use and ensure all materials are being properly stored. The Safety Team Lead must ensure that fire suppression systems are available and acknowledged by the team	F1, Medium

					members when the team is in the workplace.	

Table 5.4: Personnel Hazard Analysis

For verification of certain mitigation plans, see the following footnotes:

1. *5.1.1.4 On Launch Site, "Selecting a Launch Area"*
2. *3.1.6.2.2.1 MFSS FEA Static Studies Summary*
3. *5.1.1.4 On Launch Site, "Briefing to team members by Safety or Systems Team Lead"*
4. *5.1.1.4 On Launch Site, "Installing Ignitor"*

5.2.5 Vehicle Failure Modes and Effects Analysis

Hazard	Likelihood (Cause)	Severity (Effect)	Risk	Mitigation	Verification	Post Mitigation Risk
Airframe Failure	1 (Buckling or shearing of the airframe from poor construction or use of improper materials, faulty stress modeling)	F (Partial or total destruction of vehicle, ballistic trajectory)	F1, Medium	Use appropriate materials according to industry standards and previous flight experiences of the airframe, bulkheads, fasteners, and shear pins. Make use of reliable building techniques, confirm analyses with test launches.	Construction Team will ensure proper materials are being used. If the airframe does not perform well in test launches, perform another test launch with a new airframe design before confirming flight readiness.	F1, Medium
Failure to Ignite Motor	2 (Lack of ignitor continuity)	A (Recycle launch pad)	A2, Minimal	Check for continuity prior to attempted launch.	Ignitor continuity shall be checked prior to launch. [5]	A1, Minimal
Motor Detonation	1 (Motor defect, assembly error)	F (Partial or total destruction of vehicle)	F1, Medium	Inspect motor prior to assembly and closely follow assembly instructions.	Motor inspection shall be performed by Student Mentor prior to launch.[6]	F1, Medium
Instability	1 (Stability margin of less than 1.00)	F (Potentially dangerous flight path, loss of vehicle)	F1, Medium	Measure physical center of gravity and compare to calculated center of pressure.	Have measured physical center of gravity documented and marked prior to arriving at the launch site.[7]	F1, Medium
Motor Expulsion	2 (Improper retention methods)	F (Risk of recovery failure, low apogee, falling debris)	F2, High	Use positive retention method to secure motor.[9]	Motor retainment inspection shall be performed by Student Mentor prior to launch.[6]	F1, Medium
Premature Ejection	2 (Altimeter programming, poor venting)	F (Zippering, potential loss of vehicle components)	F2, High	Check altimeter settings prior to flight and use appropriate vent holes. Test altimeter in similar conditions to those to be experienced at launch.	Altimeter settings shall be checked prior to launch.[8]	F1, Medium
Loss or Damage of Fins	2 (Poor construction or improper materials used)	F (Partial or total destruction of vehicle)	F2, High	Ensure proper analysis has been completed on the thrust structure.	Fin slots in the thrust structure must be designed to properly contain the fins.[9]	F1, Medium
Damaged / Destroyed Nose Cone	2 (Poor construction or improper materials used, damage from previous flights, poor storage, or transportation)	F (Partial or total destruction of vehicle)	F2, High	Use materials and building techniques appropriate to high-power rocketry.	Check for nose cone damage prior to flight and ensure that nose cone is secured to vehicle for test flights.[10]	F1, Medium
Ejection Charge Failure	3 (Not enough power from improper charge	F (Ballistic trajectory,	F3, High	Perform ground test to ensure ejection charges	Ground test of ejection charges will occur after the CDR deadline.	F1, Medium

	sizing, electrical failure)	destruction of vehicle)		sufficiently separate vehicle sections.		
Altimeter Failure	3 (Loss of connection or improper programming)	F (Ballistic trajectory, destruction of vehicle)	F3, High	Perform altimeter settings and continuity check prior to launch.	See footnote [11].	F1, Medium
Payload Failure	3 (Electrical failure, program errors, dead battery)	D (Disqualified, objectives not met)	D3, Medium	Test payload prior to flight, check batteries and connections. Assemble payload with care to prevent errors.	Full payload testing will occur after the CDR deadline.	D1, Medium
Heat Damaged Recovery System	2 (Insufficient protection from ejection charges)	F (Parachute damage, excessive landing velocity, potentially ballistic trajectory)	F2, High	Use appropriate protection methods, such as Nomex blankets.	Check that proper recovery system protection methods are installed before launch.[12]	F1, Medium
Broken Fastener	1 (Excessive force from launch or descent)	F (Ballistic trajectory)	F1, Medium	Ensure proper fasteners are purchased and used based on the seller's reputation and the products past use in flight.	Inspect fasteners before launch to ensure they are not damaged.[7]	F1, Medium
Motor and Fin Support Structure Failure	2 (Excessive force from motor, poor construction)	F (Partial or total destruction of vehicle, ballistic trajectory)	F2, High	Design thrust structure according to analysis of material strength and performance under stress, make use of reliable building techniques, confirm analyses with test launches.	Thrust structure design will be subject to testing and analysis before launch.[9]	F1, Medium
Battery Overcharge / Leakage/ Ignition	3 (Unsupervised/undocumented charge, battery puncture)	F (Destruction of battery, potential ballistic trajectory of vehicle)	F3, High	Ensure batteries are documented and supervised if charging. Properly house and place batteries in launch vehicle.	Reminders will be set by testing personnel to track battery charging tests.	F1, Medium
Premature Black Powder Ignition	2 (Accidental exposure to flame or sufficient electric charge)	F (Partial destruction of vehicle, premature stage separation)	F2, High	Ensure design has sufficient distance/ protection from outside, and motor, charges, and batteries.	Ensure by design and testing that black powder wells secure from other systems. Ground test of ejection charges will occur after the CDR deadline.	F1, Medium
Destruction of Bulkheads	2 (Poor construction or improper bulkheads chosen which cannot withstand launch forces)	F (Partial or total destruction of vehicle, ballistic trajectory)	F2, High	Use appropriate materials according to analysis of materials and previous flight data, make use of reliable building techniques, confirm analyses with test launches.	Bulkheads will be visually inspected for damage prior to launch.[9] [10]	F1, Medium
Motor Angled Incorrectly	2 (Poor construction, damage from previous flights,	D (Lower launch vehicle stability, launch vehicle does not follow	D2, Medium	Ensure proper measurements and alignments are made during construction, ensure there is no rush to	Inspect motor and motor retainer prior to launch to ensure proper installation.[6]	D1, Low

	poor storage or transportation)	desired flight path)		attach the motor tube. Implement checklists to ensure proper constraint and alignment of the motor within the thrust structure		
Premature Stage Separation	3 (Premature ejection, shear pin or fastener failure)	F (Possible recovery failure and damage to or loss of vehicle, ballistic trajectory)	F3, High	Check altimeter settings prior to flight, use appropriate vent holes, choose shear pins and fasteners suitable for flight.	Redundant altimeter will be used by design.[8] Inspect shear pins and fasteners for proper installation.[12]	F1, Medium
Forgotten or Lost Components	3 (Carelessness with launch vehicle components, failure to take note of inventory before attempting to launch)	D (Launch vehicle does not launch at the desired launch time)	D3, Medium	Ensure all launch vehicle components are accounted for prior to departure to launch field. Bring backup parts to launch field as necessary.	Team Leads are responsible for assigning the transportation of their section of the vehicle to the launch field.	D1, Medium
Launch Vehicle Disconnects from Launch Rail	2 (High wind speeds, failure to properly use the rail buttons, faulty rail buttons)	F (Partial or total destruction of vehicle, ballistic trajectory which endangers personnel, onlookers, and property on the ground)	F2, High	Use physical analysis to ensure the rail buttons are properly aligned and working as planned, double check the rail buttons are properly attaching the launch vehicle to the launch pad before launch, test rail buttons with subscale flights.	Rail buttons will be inspected prior to launch for cracks, misalignment, or other inaccuracies.[10]	F1, Medium
Flight Path Interference	2 (Wildlife in the air, unforeseen obstacles such as a loose balloon)	F (Minor to severe change in the vehicle's flightpath, possible ballistic trajectory)	F2, High	Ensure there are clear skies above before launching, ensure an FAA waiver has been obtained for the designated launch area. Hold launch until flight path is clear.	Visually inspect the surrounding launch area to make sure no incoming wildlife or loose objects appear.[13]	F1, Medium
High Launch Rail Friction	3 (Faulty installation of rail buttons, faulty setup of launch rail, faulty installation of launch vehicle on launch rail, failure to properly lubricate launch rail as needed, weather conditions cause excess friction)	B (Launch vehicle does not follow the designated flight path well, lower maximum height, failure to leave pad)	B3, Low	Set up the rail using instructions which come with the product, use lubrication on the rail as needed according to weather and rail type, ensure the launch vehicle is properly installed on the launch rail.	Launch rails will be tested by tactile inspection to insure proper lubrication.[14]	B1, Minimal
Failure to Ignite Propellant	2 (Faulty motor preparation, poor quality of propellant, faulty igniter, faulty	F (Launch vehicle does not immediately launch and is a considerable	F2, High	Purchase motor and ignitors only from reliable sources, Team Mentor must install motor and ignitors, determine if the	Team Mentor is the only one allowed to install motor and ignitors.[5] [15]	F1, Medium

	igniter power source, damage to motor)	hazard until it is confirmed that it will not launch, changes to igniters or launch vehicle required)		igniters chosen work well during subscale testing.		
Propellant Fails to Burn for Desired Duration	2 (Faulty motor preparation, poor quality of propellant, damage to motor)	C (Launch vehicle does not follow the designated flight path well, lower maximum height, if drastic change in maximum height the ejection charges for recovery may not deploy)	C2, Low	Purchase motor and ignitors only from reliable sources, check the motor for damage prior to launching, Team Mentor must install motor and ignitors.	Team Mentor is the only one allowed to install motor and ignitors.[5 15] Inspect motor prior to launch to ensure proper installation.[6]	C1, Low
Propellant Explosion	1 (Faulty motor preparation, poor quality of propellant, damage to motor)	F (Ballistic trajectory, catastrophic destruction of vehicle, possible harm to bystanders)	F1, Medium	Purchase motor and ignitors only from reliable sources, check the motor for damage prior to launching, Team Mentor must install motor and ignitors.	Team Mentor is the only one allowed to install motor and ignitors.[5 15] Inspect motor prior to launch to ensure proper installation.[6]	F1, Medium
Payload Computer Failure	3 (Electrical failure, program error, poor setup of wiring causes a connection to come undone, forgotten connection, battery failure)	F (Disqualified, objectives not met, loss of electronic control, improper payload deployment)	F3, High	Test payload prior to flight, check batteries and connections before flight.	Ground test payload in flight like conditions, inspect software before use, monitor payload during VDF. Payload testing will be done after the CDR deadline.	F1, Medium
Power Loss to Avionics Bay and/or Payload	3 (Faulty wiring, battery failure, poor setup of wiring causes a connection to come undone, forgotten connection)	F (Disqualified, objectives not met, failure to correctly trigger ejection charges)	F3, High	Test the reliability of the wiring and batteries through subscale flights, check batteries and connections before flight.	Perform continuity checks for altimeters prior to launch, visible wires will be inspected for nicks or damage prior to launch.[11]	F1, Medium
Arming System Failure	3 (Faulty arming system, faulty wiring, battery failure, poor setup of wiring causes a connection to come undone, forgotten connection)	F (Disqualified, objectives not met, failure to correctly trigger ejection charges)	F3, High	Ensure the avionics bay is successfully communicating with the team prior to flight, test arming system through test launches.	Ensure communication between avionics bay and the team is established and reliable right before launch.[16]	F1, Medium
Stages Fail to Separate	3 (Faulty ejection charge, excessive strength is used to hold stages together, altimeter failure)	F (Launch vehicle does not follow desired flight path, possible ballistic trajectory, lower	F3, High	Examine ejection charges for damage before launch, ensure proper functionality of the altimeters, ejection charges, and interstage	Ejection charge testing will be performed to ensure charges can separate stages, and dual altimeters will be employed to enable redundancy.	F1, Medium

		maximum height, damage to the launch vehicle)		joints, have a secondary ejection charge for each stage separation.		
Main Parachute Fails to Deploy	2 (Poor design of where parachute is in launch vehicle, poor sealing of parachute chamber, poor loading of parachute, faulty parachute or ejection charge, altimeter failure)	F (Main parachute does not slow down the launch vehicle, recovery failure, ballistic trajectory)	F2, High	Any team member who seals or packs the parachute chamber must be supervised by at least one other team member, examine parachute and ejection charges for damage before launch, have a secondary ejection charge in case of emergency which is larger than the first.	Ejection charge testing will be done to ensure charge effectively deploys parachute.	F1, Medium
Drogue Parachute Fails to Deploy	2 (Poor design of where parachute is in launch vehicle, poor sealing of parachute chamber, poor loading of parachute, faulty parachute or ejection charge, altimeter failure)	F (Drogue parachute does not slow down the launch vehicle, recovery failure, ballistic trajectory)	F2, High	Any team member who seals or packs the parachute chamber must be supervised by at least one other team member, examine parachute and ejection charges for damage before launch, have a secondary ejection charge in case of emergency which is larger than the first.	Ejection charge testing will be done to ensure charge effectively deploys parachute.	F1, Medium
Parachute Shroud Lines Break	1 (Poor shroud line materials, improper ejection of recovery system, damage from previous flights or transportation)	F (Possible recovery failure, ballistic trajectory)	F1, Medium	Only buy parachutes from reliable sources, remove threats to parachute integrity from the parachute housing, check the recovery system for damage before launch.	Examination of the shroud lines and parachutes must occur before packing into the main vehicle.[7]	F1, Medium
Shock Cord Breaks or Disconnects	1 (Faulty shock cord, damage to shock cord, poor connection to the launch vehicle)	F (Parachute disconnect from the launch vehicle, recovery failure, ballistic trajectory)	F1, Medium	Any team member who connects the shock cord to the launch vehicle must be supervised by at least one other team member, check the shock cord for damage before and after flight, only buy shock cords from reliable sources, analyze the shock cord with test flights.	Orange tape must be placed over the fasteners connecting different vehicle components together.[12]	F1, Medium
Tangled Parachute or Shock Cord	2 (Faulty or damaged shock cord or parachute, poor packing of shock cord and/or parachutes, poor sizing of parachutes or shock cord,	F (Shock cord or parachutes may not fully extend or inflate, possible ballistic trajectory, possible failed recovery)	F2, High	Only buy parachutes and shock cords from reliable sources, any team member who seals or packs the parachute chamber must be supervised by at least one other team member, examine parachutes and shock cord for damage before launch, check	Ensure parachute packing is observed by at least one other team member with knowledge of the recovery system.[12]	F1, Medium

	unstable or ballistic flight)			performance of parachutes and shock cord in test flights, appropriately follow recommended sizing for shock cord and parachutes.		
Parachute Comes Loose from Launch Vehicle	2 (Failure of recovery system mount on the launch vehicle body, poor shroud line materials, improper ejection of recovery system, damage from previous flights or transportation)	F (Recovery failure, ballistic trajectory)	F2, High	Only buy parachutes from reliable sources, check the recovery system for damage before launch, double check that the recovery system is properly mounted before launch.	Ensure parachute packing is observed by at least one other team member with knowledge of the recovery system.[12]	F1, Medium
Parachute or Shock Cord Catch Fire	2 (Not enough space given between ejection charge and parachute, poor insulation of parachute, poor parachute packing, faulty or poorly chosen ejection charge)	F (Shock cord or parachutes do not fully achieve their goal, possible ballistic trajectory, possible failed recovery, damage to internal launch vehicle components)	F2, High	Any team member who packs the parachute or ejection charges must be supervised by at least one other team member, use recommended sizing methods for ejection charges, confirm proper placement and packing methods of ejection charges and parachutes with test flights.	Ensure parachute packing is observed by at least one other team member with knowledge of the recovery system.[12]	F1, Medium
ABCS Failure to Deploy	3 (Software error, mechanical failure)	B (Improper final vehicle altitude)	B3, Low	Design software and mechanics according to expected flight conditions.	Monitor ABCS performance during VDF.	B2, Low
Erratic Vehicle Path from ABCS Failure	3 (Software error, mechanical failure)	F (Partial or complete destruction of vehicle, possible ballistic trajectory)	F3, High	Design software and mechanism according to expected flight conditions. Ensure system enters low drag state during all failure modes.	Perform ground testing of various failure modes to ensure ABCS disengages if erratic movement is detected. Monitor ABCS performance during VDF.	F1, Medium
Airframe Zippering	2 (Excessive deployment deceleration)	F (Partial or complete destruction of vehicle)	F2, High	Properly time ejection charges and use an appropriately long tether.	Ensure design of vehicle and thrust structure are sound, test and observe vehicle at full scale launch prior to Huntsville.	F1, Medium
GPS Lock Failure	2 (Interference or dead battery)	F (Loss of vehicle)	F2, High	Ensure proper GPS lock and battery charge before flight.	Ensure GPS signal is established before flight.[16]	F1, Medium
Insufficient Landing Speed	3 (Improper load, higher coefficient of drag for the parachutes than needed, higher surface area of the parachutes than needed)	B (Unexpected changes in flightpath and landing area, increased potential for drift)	B3, Low	Use subscale flights to determine if the subscale parachutes were accurately sized, use recommended and proven-to-work parachute sizing techniques for full scale vehicle.	Avionics Lead must ensure the proper parachute is purchased and used.	B1, Minimal

Excessive Landing Speed	3 (Parachute damage or entanglement, improper load, improperly sized parachute)	F (Partial or total destruction of vehicle)	F3, High	Properly size, pack, and protect parachute.	Avionics Lead must ensure the proper parachute is purchased and used. Parachute packing must be observed by at least one other team member with knowledge of the recovery system.[12]	F1, Medium

Table 5.5: Vehicle Failure Modes and Effects Analysis

For verification of certain mitigation plans, see the following footnotes:

5. *5.1.1.4 On Launch Site, "Installing Ignitor"*
6. *5.1.1.4 On Launch Site, "Inspect motor, motor casing, and motor retainment system for damage"*
7. *5.1.1.3 Vehicle Assembly*
8. *5.1.1.1 In Advance, "Programming the TeleMetrum Altimeter" and "Programming the StratoLoggerCF Altimeter"*
9. *3.1.6.2.2.1 MFSS FEA Static Studies Summary*
10. *5.1.1.4 On Launch Site, "Inspect all vehicle components for damage from travel"*
11. *5.1.1.4 On Launch Site, "Altimeter Continuity"*
12. *5.1.1.3 Vehicle Assembly, "QUALITY WITNESS - Vehicle Assembly"*
13. *5.1.1.4 On Launch Site, "Selecting a Launch Area"*
14. *5.1.1.4 On Launch Site, "Installing the Vehicle on the Launch Rail"*
15. *5.1.1.4 On Launch Site, bullet point that starts with, "Prep and install motor"*
16. *5.1.1.4 On Launch Site, "Setting Up the EggFinder Trackers at the Launch Viewing Area"*

5.2.6 Environmental Hazard Analysis

A note regarding the Environmental Hazard Analysis: The following two tables are split by whether the launch vehicle or the environment is harmed by the other. This is a change made from PDR because the new format appears more streamlined and is easier to read.

Hazards to Environment						
Hazard	**Likelihood (Cause)**	**Severity (Effect)**	**Risk**	**Mitigation**	**Verification**	**Post Mitigation Risk**
Pollution from Exhaust	5 (Combustion of APCP motors)	A (Small amounts of greenhouse gasses emitted)	A5, Medium	Use only launch vehicle motors approved for use by the National Association of Rocketry, Canadian Association of Rocketry, or Tripoli Rocketry Association.	Launch vehicle motors in consideration will be purchased and installed by the team's Student Mentor to ensure compliance.	A5, Medium
Pollution from Vehicle Itself	3 (Loss of components from vehicle in surroundings)	C (Materials degrade extremely slowly, possible harm to wildlife or water contamination)	C3, Medium	Ensure all parts are attached to each component of the vehicle and that the components of the vehicle are properly attached. Scavenge for fallen parts after launch is completed.	Quality Witness checks shall be performed as part of the assembly of each vehicle component and the integration of the components into the larger vehicle.[17]	C1, Low
Pollution from Team Members	2 (Failed disposal of litter, improper cleanup procedures, members walk through important plant life, farming fields, sod, etc.)	D (Litter may degrade extremely slowly, wildlife may consume harmful litter, destruction of crops)	D2, Medium	Brief team members on proper cleanup procedures, foster a mindset of leaving no trace at launch sites, only the minimum number of required team members should retrieve the launch vehicle.	Follow societal standards and leave cite cleaner than was found, make sure disposable equipment is kept track of and guaranteed to remain at designated locations. This will occur in the Safety briefing upon team arrival to the launch field.[18]	D1, Medium
Vehicle collisions with Man-made	2 (Failure to properly predict	F (Damage to public property or	F2, High	Do not launch under adverse conditions which	Simulate results for vehicle trajectory.[19]	F1, Medium

Structures or with Humans	trajectory, failure to choose an appropriate launch area)	private property not owned by the team, damage to team equipment, serious damage to team personnel or passerby)		may affect the course of the launch vehicle, run simulations which analyze the launch vehicle's trajectory mathematically and physically, choose a launch area which is not close to civilization, follow launch procedures closely.	Project Management and the Avionics Team Lead must ensure that the actual launch is ran in a similar way to that which was simulated. Safety Team Lead will monitor weather conditions prior to launch.[20]	
Battery Leakage	3 (Absence of or damage to battery casing causing puncture to battery)	C (Possible toxic acid leak, heavy metal contamination, degradation and harm to plant and animal life)	C3, Medium	Batteries will be individually enclosed in plastic casing, parachutes will be selected to reduce landing kinetic energy below levels that will damage the casing.	Examine the battery casing for damages prior to launch.[21] It is assumed that if the kinetic energy of the vehicle landing is less than the maximum outlined in the Handbook[22], the battery casing will not be damaged to the point of battery puncture. Fulfillment of the kinetic energy upon landing requirement is calculated.[23]	C1, Low
Fire to Surroundings	3 (Exhaust caused by launch vehicle engine)	F (Possible spread of wildfire, damage to wildlife or landscape)	F3, High	Ground will be cleared per NAR standard, fire extinguishers will be on hand.	Safety Team Lead will ensure compliance with NAR safety standard on minimum clear area.[24] Safety Team Lead is responsible for bringing fire suppression equipment to the launch field.[25]	F1, Medium
Kinetic Damage to Buildings	2 (Launch vehicle veers off trajectory causing landing in occupied area)	D (Repairable destruction to building)	D2, Medium	Choose launch site that is remote enough to make this risk negligible.	Safety Team Lead must ensure minimum distance from building exceeds minimum building distance as established by NAR safety standard.[24]	D1, Medium
Kinetic Damage to Terrain	4 (Launch vehicle has excessive landing speed)	A (Creation of small ground divots, mild inconvenience to wildlife and flora)	A4, Medium	Parachute selection must ensure that vehicle does not land with an excess of kinetic energy.	Avionics Team has verified proper vehicle landing kinetic energy.[23]	A1, Minimal

Table 5.6: Hazards to Environment

For verification of certain mitigation plans, see the following footnotes:

17. *5.1.1.3 Vehicle Assembly, "QUALITY WITNESS - Vehicle Assembly"*
18. *5.1.1.4 On Launch Site, "Briefing to team members by Safety or Systems Team Lead"*
19. *3.4.1.3 Simulink*
20. *5.1.1.5 Countdown to Launch, as well as various other points in launch procedures*
21. *5.1.1.4 On Launch Site, "Inspect all vehicle components for damage from travel"*
22. *2021 NASA Student Launch Handbook and RFP, Proposal / Statement of Work, Section 3.3*
23. *3.4.1.3 Simulink, Table 2*
24. *5.1.1.4 On Launch Site, "Selecting a Launch Area"*
25. *5.1.1.4 On Launch Site, "Briefing to team members by Safety or Systems Team Lead"*

Hazards from Environment						
Hazard	**Likelihood (Cause)**	**Severity (Effect)**	**Risk**	**Mitigation**	**Verification**	**Post Mitigati on Risk**

Landscape	3 (Vehicle contact with trees, brush, water, power lines, wildlife)	F (Inability to recover launch vehicle, damage or destruction of vehicle or vehicle components)	F3, High	Angle launch vehicle into wind as necessary to reduce drift. Choose a launch area that is flat and clear of large flora.	Safety Team Lead, Project Management and the Student Mentor shall pick a launch area that is free from obstructions.[27]	F1, Medium
Humidity	2 (Climate, poor forecast, improper storage of launch vehicle when not in use)	C (Rust on metallic components, failure of electronics components)	C2, Low	Use as little ferrous metal as possible in vehicle design, store vehicle indoors when not in use.	Team Leads should be aware of this hazard throughout the design process of the vehicle and its components.	C1, Low
Winds	3 (Poor forecast)	D (Inability to launch, excessive drift)	D3, Medium	Angle into wind as necessary, abort launch if wind exceeds 20 mph (NAR High Power Rocketry Code, point 9)	Safety Team Lead will monitor weather conditions prior to launch.[26]	D1, Medium
High Temperature	3 (Poor forecast)	C (Heat related injury or damage to vehicle components)	C3, Medium	Ensure team is wearing appropriate clothing for extended periods of time in hot environments. Keep launch vehicle in shaded area until before launch.	Safety Team Lead or Project Management must notify the team of weather on day of launch or manufacturing to wear proper clothing, Safety Team Lead must ensure mitigation is strictly followed due to weather conditions.	C1, Low
Low Temperatures	3 (Poor forecast)	C (Cold-related personnel injuries, frost on ground, ice on vehicle, clogging of vehicle ventilation, change in launch vehicle rigidity and mass, higher drag force on launch vehicle)	C3, Medium	Ensure team is wearing appropriate clothing for extended periods of time in cold environments. Keep the launch vehicle at room temperature or bundled in materials which hold in heat, if ice appears anywhere on the launch vehicle, do not launch and return it to a warm location.	Safety Team Lead or Project Management must notify the team of weather on day of launch or manufacturing to wear proper clothing, Safety Team Lead must ensure mitigation is strictly followed due to weather conditions.	C1, Low
Wildlife Contact with Launch Vehicle	1 (Failure to accurately predict trajectory, unexpected appearance of wildlife, poor choice of launch area)	D (Damage to vehicle components, damage to wildlife, unexpected trajectory close to the ground)	D1, Medium	Launch in an open area with high visibility, be aware of the surroundings when choosing a launch area and launching.	Safety Team Lead, Project Management and the Student Mentor shall pick a launch area that is free from obstructions and places where wildlife could be found.[27]	D1, Medium
Wildlife Contact with Launch Pad	1 (Failure to monitor the launch pad, poor choice of launch area)	D (Possible inability to launch the launch vehicle, unpredictable launch behavior or trajectory)	D1, Medium	Launch in an open area with high visibility, be aware of the surroundings when choosing a launch area and launching, if animals tamper with the launchpad, do not launch.	Safety Team Lead, Project Management and the Student Mentor shall pick a launch area that is free from obstructions and places where wildlife could be found.[27]	D1, Medium

Table 5.7: Hazards from Environment

For verification of certain mitigation plans, see the following footnotes:

26. *5.1.1.5 Countdown to Launch, as well as various other points in launch procedures*
27. *5.1.1.4 On Launch Site, "Selecting a Launch Area"*

5.2.7 Project Risks Analysis

Hazard	Likelihood (Cause)	Severity (Effect)	Risk	Mitigation	Verification	Post Mitigation Risk
Insufficient Funding	4 (Lack of revenue, parties unwilling / unable to contribute due to COVID-19 complications)	F (Inability to purchase parts and construct the vehicle)	F4, Very High	Create and execute a detailed funding plan properly, minimize excessive spending by having multiple members check the necessity of purchases.	Each subteam must verify purchases with Project Management and the Business Team Lead to ensure the team is still within their given budget. Program's Treasurer and Business Lead will work to create a uniform document for member purchases.	F1, Medium
Failure to Receive Parts	2 (Shipping delays, out of stock orders)	F (Cannot construct and fly vehicle)	F2, High	Order parts while in stock well in advance of needed date. Seek other vendors if product is out of stock.	Team Leads must ensure parts are ordered at least a month before the parts are scheduled to be used by referencing the GANTT chart.[28] A month was chosen to give ample time for shipping, especially during holidays and in the pandemic.	F1, Medium
Damage to or Loss of Parts	2 (Failure during testing, improper part care during construction, transportation, or launch)	F (Cannot construct or fly vehicle without spare parts)	F2, High	Order extra parts in case some need to be replaced. Take care in handling parts for transportation, launch, and construction operations.	Team Leads must confirm a minimum number of parts needed so the team can obtain duplicates for needed items. They must also assign responsibility for more important and expensive parts as they see fit.	F1, Medium
Rushed Work	3 (Rapidly approaching deadlines, unreasonable schedule expectations)	D (Threats of failure during testing or the final launch due to a lower quality of construction and less attention paid to test data)	D3, Medium	Set deadlines which both keep the project moving at a reasonable pace and leave room for unforeseen circumstances.	Team Leads shall verify that projects are being completed before the deadline arrives as denoted by the GANTT chart.[28]	D1, Medium
Major Testing Failure	2 (Improper construction of the launch vehicle, insufficient data used before creating the launch vehicle's design)	F (Damage to vehicle parts, possible disqualification from the project due to a lack of flight data, an increase in budget for buying new materials, delay in project completion)	F2, High	Ensure parts used fall within specifications of required use. Take care to perform tests correctly.	Team Leads / team members who write testing procedures must ensure that proper parts and methods are being used in tests.	F1, Medium
Unavailable Test Launch Area	2 (Failure to locate a proper area to launch vehicle, failure to receive an FAA waiver for any launch)	F (Disqualification from the project due to a lack of flight data)	F2, High	Secure a reliable test launch area and FAA waiver well in advance of the dates on which test launch data is required.	Project Management must work with the Student Mentor of the team to ensure the launch site is able to be used. The Safety Team Lead must ensure that the Student Mentor of the team has submitted the	F1, Medium

					proper FAA waiver and that it is approved.	
Loss or Unavailability of Work Area	4 (Construction, building hazards, loss of lab privileges, COVID restrictions imposed by Purdue University)	D (Temporary inability to construct vehicle)	D4, High	Follow work area regulations and have secondary spaces available.	Inform members of proper work area etiquette to prevent loss of lab privileges. Project Management and Team Leads must regularly confirm that the team has access to secondary locations if the need arises.	D1, Medium
Failure in Construction Equipment	1 (Improper long-term maintenance of construction equipment, improper use or storage of equipment)	C (Possible long-term delay in construction)	C1, Low	Ensure proper maintenance and use of construction equipment and have backup equipment which can be used in case of an equipment breakdown.	Team members involved in the construction of the vehicle must inspect equipment before and after use to confirm it is functioning properly, consulting the supervisors of the area in which work is being done if any doubts arise.	C1, Low
Insufficient Transportation	3 (Insufficient funding or space available to bring all project members to launch sites or workplace)	C (Loss of labor force, team members lose knowledge of what is happening with the project, low attendance to the final launch)	C3, Medium	Organize and budget for transportation early, keep track of dates on which large amounts of transportation are needed.	Project Management and Team Leads must organize transportation at least two weeks prior to major activities to make sure either enough drivers are secured, or buses are rented.	C1, Low
Inactivity / Low Availability of Personnel	3 (Members are unable or unwilling to work due to an increase in classwork, COVID-19 restrictions, or other mandatory activities)	F (Low attendance, loss of team members, labor shortages, inability to construct vehicle)	F3, High	Ensure all team members have an important role in the design process, shift extra personnel resources to needed areas.	Team Leads and Project Management will ensure the GANTT chart [28] is followed and all members are engaged with the design process.	F1, Medium
Damage by Non-Team Members	2 (Accidental damage caused by other workspace users)	F (Extensive repairs necessary, delay in construction)	F2, Medium	Separate all pf the team's components from other areas of the workspace as necessary.	All team members must ensure only team members can have access to vehicle components by using the storage mechanisms allotted to the team, including cabinets and shelves.	F1, Medium
Vehicle damage During Transit	2 (Mishandling during transportation)	F (Inability to fly launch vehicle)	F2, High	Protect all launch vehicle components during transit.	Personnel transporting vehicle components must ensure the vehicle is secured with padding and bracing.	F1, Medium
Weather Delays	3 (Poor weather conditions during tests or launches such as high wind speeds, ice and frost, or storms)	F (Possible disqualification from the project due to a lack of flight data)	F3, High	Have multiple dates available on which test launches can be conducted in case of adverse weather conditions.	Project Management must set adequate launch and backup launch dates on the GANTT chart.[28]	F1, Medium

Table 5.8: Project Risks Analysis

For verification of certain mitigation plans, see the following footnote:

28. *6.6 Project GANTT Chart*

6 Project Plan

6.1 Avionics Testing

6.1.1 Altimeter Ejection Vacuum Test - VT.A.5.3

Test Objective: Fulfill requirement S.A.5.1: Both altimeters need to be able to consistently ignite both ejection charges at the appropriate times.

Testing Variable: The testing variable is the number of times the lighters are ignited by the TeleMetrum and StratoLoggerCF altimeters.

Success Criteria: Both altimeters must ignite the drogue parachute lighters at apogee (or 1s after apogee) and the main parachute lighters at the correct altitude during descent.

- For the TeleMetrum altimeter, the magnitude of the difference between the apogee altitude and the altitude the drogue lighter ignites at must be less than 500' for all three trials.
- For the TeleMetrum altimeter, the altitude the main lighter ignites at must be between 900 ± 50' for all three trials.
- For the StratoLoggerCF altimeter, the drogue delay (the time between apogee and ignition of the drogue lighter) must be between 0.75 and 1.75s (as it is programmed to be 1s) for all three trials.
- For the StratoLoggerCF altimeter, the altitude the main lighter ignites at must be between 700 ± 50' for all three trials.

Why it is Necessary: Both altimeters must be able to ignite both ejection charges at the correct times in flight in order to ensure the successful recovery of the vehicle and validate the choices of altimeters.

Methodology:

(Note: If desired, both altimeters can be tested at the same time for a total of only three rather than six trials)

1) One large hole will be drilled into a sheet of plexiglass. A wine stopper will be placed into this hole and a small ring of plumber's putty will be placed around it in order to prevent air from escaping.
2) A smaller hole will be drilled to the side of the larger one (this will act as a pressure release hole to simulate descent).
3) To test one altimeter, a lighter will be connected to each the drogue and main outputs, and a battery and switch also connected. This system (along with the AltimeterOne turned on and set to Real Time mode) will be placed in the glass bowl, with the switch and the lighters hanging over the rim of the bowl to allow easy access to turn the altimeter on and off as well as to allow the lighters to ignite in a non-constrained environment. If the TeleMetrum is being tested, it will be placed pointing up.
4) A larger ring of plumbers' putty will be placed around the rim of the bowl, over the lighters and switch wires. The prepared sheet of plexiglass will then be placed over the bowl and pressed down until there is a uniform seal around the entire perimeter. Extra plumbers' putty will be placed around the exposed wires as needed.
5) A small piece of plumbers' putty will be used to seal the pressure release hole, then the altimeter will be switched on and allowed to complete its initialization routine. It is important that these steps are completed in this order because if the chamber is sealed after the altimeter is switched on, it might detect the small drop in pressure and start the launch.
6) The wine bottle air remover pump will then be used to remove air through the stopper. Once the process of removing air is halted at the expected apogee altitude (the digital display of the AltimeterOne will indicate when this occurs), the drogue lighter is expected to ignite (or one second after apogee for the StratoLoggerCF altimeter).
7) Finally, the small piece of plumbers' putty will be very slightly lifted away from the plexiglass to slowly allow air back inside it, causing the altitude to decrease according to the AltimeterOne. The main lighter will be expected to ignite at pressures corresponding to an altitude of 900' (or 700' for the StratoLoggerCF altimeter).
8) The flight data will be downloaded onto a laptop for analysis.
9) The procedure will be repeated two more times for a total of three trials, then three more times with the other altimeter.

Impact of Results: If both altimeters pass this test, no action will be required to correct the performance of lighter ignition, and it can be expected that the altimeters will eject the parachutes with no issues during launch. If one or both altimeters fail this test, a complete retest will need to be conducted on the altimeter(s) that failed in order to determine and correct the issue, and new altimeters may be considered.

Results and Conclusions: This test will be conducted in late January 2021.

6.1.2 Black Powder Ejection Test - VT.A.2.1, VT.A.3

Test Objective: Fulfill requirements S.A.2.1: Parachutes will be completely protected with a Nomex blanket on the side of the ejection charges, and S.A.3: The black powder canisters will create appropriate separation between the airframe sections.

Testing Variable: The testing variable is the amount of separation on the ground between the correct airframe sections both the drogue and main side black powder canisters result in.

Success Criteria: Both black powder canisters must separate the correct airframe sections the appropriate amount on the ground, not damage any vehicle components, and fully eject the parachutes.

- Black powder canister on the upper recovery section side of the avionics bay: ignition must result in at least 6’ of separation between the upper recovery section and the payload section for at least one amount of black powder equal to or greater than 3g.
- Black powder canister on the lower recovery section side of the avionics bay: ignition must result in at least 6’ of separation between the lower recovery section and the booster section for at least one amount of black powder equal to or greater than 2g.

Why it is Necessary: Both black powder canisters must separate the correct airframe sections the appropriate amount on the ground, not damage any vehicle components, and fully eject the parachutes in order to ensure the successful recovery of the vehicle and validate the choices of all of these components.

Methodology:

1) The black powder canister on the upper recovery section side of the avionics bay will be filled with 3g of black powder. Specifically, the black powder will be measured out using a gram scale and poured into the cut tip of a finger of a disposable latex glove, which will then be zip-tied shut with the end of a lighter also placed in there. This will then be placed into the black powder canister, which will be packed with dog barf and covered with masking tape to prevent anything from falling out.
2) The other end of the lighter will be connected to the terminal block on the avionics bay, and a 10’ extension wire will also be connected to the other end of the terminal block.
3) The main parachute and a Nomex blanket will be attached off-center to the 60’ shock cord via a loop and quick link. The longer end will be attached to the eyebolt on the bulkhead of the upper recovery section side of the avionics bay, and the shorter end will be attached to the eyebolt on the bulkhead of the payload section through the upper recovery section. The main parachute and Nomex blanket will be packed in flight configuration in the upper recovery section, which will then be reconnected to the avionics bay using screws. The upper recovery section will also be reconnected to the payload section using shear pins.
4) The extension wire will have been threaded through one of the switch holes so it can be accessed from the outside of the vehicle. A remote detonator will be connected to the extension wire.
5) The person conducting the test will stand 40’ away from the system and will set off the remote detonator. The ejection charges will then be expected to ignite and result in the separation of the two sections connected by shear pins. If they do indeed separate, the distance between them will be measured in feet using a tape measure.
6) If the above success criteria were not met, the procedure will be repeated using increasing amounts of black powder (in 1g increments) until 6’ of separation is achieved. This last amount of black powder will then be recorded as the ideal amount of black powder.
7) The procedure will also be repeated for the black powder canister on the lower recovery section side of the avionics bay (with the drogue parachute inserted and attached on the other side to the booster section with the 30’ shock cord), with 2g of black powder.

Impact of Results: If both black powder canisters pass this test, no action will be required to correct the performance of airframe separation, and it can be expected that the black powder canisters will successfully separate the correct airframe sections and eject the parachutes with no issues during launch. If one or both black powder canisters fail this test, the following responses will be taken: if the upper recovery section and payload section separation is less than 6', black powder will be added in 1g increments from the initial 3g until 6' of separation is achieved. If the lower recovery section and booster section separation is less than 6', black powder will be added in 1g increments from the initial 2g until 6' of separation is achieved. These retests will be conducted until 6' of separation of the two airframe sections is achieved by both the drogue and main side black powder canisters. Then, these ideal amounts of black powder will be used in the new vehicle design.

Results and Conclusions: This test will be conducted in early February 2021.

6.1.3 Altimeter Continuity and Battery Drain Test - VT.A.5.1, VT.A.5.2, VT.A.6.1, VT.A.6.3

Test Objective: Fulfill requirements S.A.5.1: Altimeters will continue to function across all likely flight temperatures, S.A.5.2: Both altimeters will achieve and maintain continuity consistently throughout flight, S.A.6.1: Altimeter batteries will supply usable voltage and current for 1 hour longer than the given pad time of 2 hours, and S.A.6.3: Altimeter batteries will not fail to function at any flight temperature and will function properly at a variety of temperature extremes.

Testing Variables: The testing variables are the number of continuity beeps emitted by both the TeleMetrum and StratoLoggerCF altimeters and the voltages of both the 3.7V LiPo and 9V batteries.

Success Criteria: Both altimeters must maintain continuity and receive adequate power from their respective batteries for 3 hours powered on in both temperature extremes, and the voltages of both batteries must remain the same after 18 hours powered off in warm weather.

- Warm-weather test: Must be above 75°F.
- Cold-weather test: Must be below 35°F.
- Every continuity measurement of both the StratoLoggerCF and the TeleMetrum altimeters must be 3 beeps (full dual-deployment continuity).
- In the powered-on test, the voltage of the 9V battery must not drop below 8V.
- In the powered-on test, the voltage of the 3.7V LiPo battery must not drop below 3.3V.
- In the powered-off test, the voltage of each battery must remain exactly the same after 18 hours.

Why it is Necessary: Both altimeters must maintain continuity and receive adequate power from their respective batteries for 3 hours powered on in both temperature extremes, and the voltages of both batteries must remain the same after 18 hours powered off in warm weather in order to ensure the successful recovery of the vehicle and validate the choices of all of these components.

Methodology:

Powered-On Test (Warm and Cold Weather)

1) A note was made of the current temperature.
2) One new 9V battery was connected to the StratoLoggerCF altimeter using a 9V battery connector, and a switch was also connected. A lighter was connected to each of the drogue and main outputs as well.
3) The altimeter was powered on using the switch and allowed to complete its initialization routine. Then, the system was left for 3 hours.
4) Every 0.5 hours (including at 0 hours and 3 hours), the voltage of the battery was recorded using a multimeter, and the number of continuity beeps that were being emitted was also recorded.
5) The procedure was repeated with the 3.7V LiPo battery and the TeleMetrum altimeter. However, since a multimeter cannot be used to measure the voltage of a 3.7V LiPo battery, the voltage was measured by briefly flipping the switch off and then on again, restarting the TeleMetrum. The number of initialization beeps (which represent the current voltage level detected by the TeleMetrum) was then recorded as the voltage measured for that interval of time.
6) The entire test was conducted in both the early fall and in the winter in order to verify that full continuity and adequate voltage supplied to the altimeters can consistently be achieved in both warm and cold weather.

Powered-Off Test (Warm Weather Only)

1) In warm weather only with the same setup as in the powered-on test, but with the altimeters powered off, a voltage reading of each battery was taken before and after 18 hours of everything being wired together in flight configuration.

Impact of Results: If both altimeters and batteries pass this test, no action will be required to correct the continuity and power delivery performance, and it can be expected that the altimeters will eject the parachutes with no issues during launch. If one or both altimeters or batteries fail this test, a complete retest will need to be conducted on the altimeters or batteries that failed in order to determine and correct the issue, and new altimeters or batteries may be considered.

Results and Conclusions: This test was conducted on both November 10, 2020 and December 15, 2020 to test the components in both warm and cold weather conditions. Both altimeters consistently demonstrated continuity over the entire 3-hour period in both temperature extremes with no issues. Therefore, it can be said that **both altimeter systems pass the continuity test for warm and cold weather**. Both batteries also remained well above the safety margins for voltage in both temperature extremes when powered on. When powered off and in warm weather, neither battery voltage measured as changing in the entire 18-hour period, but when powered on, the voltage measured decreased slightly over the 3-hour period. However, since this decrease is low and the batteries were able to remain above their respective thresholds, it can be said that **both altimeter systems pass the battery drain test as well**. No design changes need to be made.

Picture:

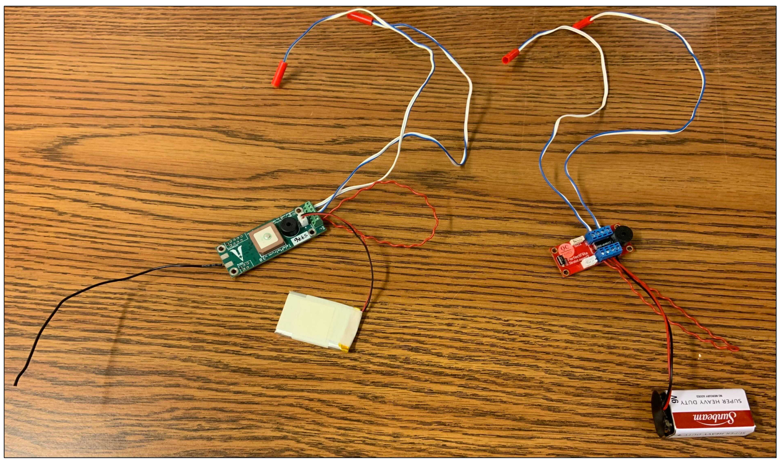


Figure 6.1: Altimeter Continuity and Battery Drain Test Setup

6.1.4 Parachute Drop Test - VT.A.2

Test Objective: Fulfill requirement S.A.2: Parachutes will open consistently within an appropriate distance range or time frame to allow for full deployment after ejection.

Testing Variables: The testing variables are the elapsed time between the weight being dropped and the drogue parachute fully opening and the final estimate for the total drop distance required for the main parachute to open fully, including shock cord extension.

Success Criteria: Both parachutes must fully deploy within their respective maximum parameter.

- The elapsed time between the weight being dropped and the drogue parachute fully opening must be below 1s for each trial.
- The final estimate for the total drop distance required for the main parachute to open fully, including shock cord extension, must be below 150’.

Why it is Necessary: Both parachutes must fully deploy within their respective maximum parameter in order to ensure the successful recovery of the vehicle and validate the choices of parachutes.

Methodology:

1) The 60’ shock cord was marked with blue tape in 5’ increments and draped over the top edge of the parking garage to serve as a vertical distance marker.
2) The drogue parachute was attached to the center of the 30’ shock cord via a loop and quick link, and the ends of the shock cord were tied a few times around the 50lbm weight (simulating the weight of the launch vehicle) and secured with another quick link.

3) The drogue parachute was packed in the old upper airframe with the Nomex blanket wrapped around it (in flight configuration), which was then held over the top edge of the parking garage. With a running timer on one smartphone in view of another smartphone also video recording the drop, the weight was tossed over the top edge of the parking garage.
4) This procedure was repeated three times for a total of three drops of the drogue parachute.
5) This procedure was also repeated three times for a total of three drops of both the subscale and main parachutes. However, these were not timed.
6) When later analyzing the video recordings, the elapsed time between the weight being dropped and the drogue parachute fully opening was recorded for each trial. For the subscale parachute, the distance between the parachute leaving the airframe and fully opening was recorded for each trial. For the main parachute, the distance between the parachute leaving the airframe and hitting the ground was recorded for each trial, as well as the approximate percentage the parachute was open to just before hitting the ground.
7) The distance to open values (these not including the extension of the shock cord) of the drogue and subscale parachutes were plotted against parachute surface areas (hemispherical model). Linear and exponential models were then created from this data.
8) The percentage opened values of the main parachute were plotted against precise drop distance, and an exponential model was then created from this data. This model was used the estimate the total drop distance required for the main parachute to open fully (100%).
9) The surface area of the main parachute was input into both the linear and exponential distance to open models created from the drogue and subscale data to output two more estimates of the total drop distance required for the main parachute to open fully. The one that is closer to the estimate from the main parachute percentage data was then averaged with that estimate. Finally, 30’ (the extension of the 60’ shock cord that will be used with the main parachute in flight, when doubled up) was added to that number to produce the final estimate.

Impact of Results: If both parachutes pass this test, no action will be required to correct the deployment performance, and it can be expected that the parachutes will deploy with no issues during launch. If one or both parachutes fail this test, a complete retest will need to be conducted on the parachute(s) that failed in order to determine and correct the issue, and new parachutes or packing methods may be considered.

Results and Conclusions: This test was conducted on November 14, 2020. After plotting the data for trials involving the full-scale main parachute, an extrapolated estimation of 82’ is required for the parachute to fully open. The data from the full-scale drogue and subscale parachutes were then used to create linear and exponential models that show the relationship between distance dropped once fully open versus parachute surface area. The linear model showed a closer approximation to the extrapolated result with an estimation of 100’ for the full-scale main parachute to open completely. This approximation was averaged with the extrapolated data point, and an estimation of 91’ is required for the full-scale main parachute to completely open. Including the extension of the 60’ shock cord that will be used with the parachute (30’ when doubled-up), a final estimation of 121’ is required for the full-scale main parachute to completely open. This value is less than the 150’ maximum as per the project requirements, meaning **the full-scale main parachute passes this test**. All three trials for the full-scale drogue parachute saw it open fully within the 1s maximum time period set by the project requirements, including the extension of the 30’ shock cord that will be used with the parachute (15’ when doubled-up). The conclusion can therefore be made that **the drogue parachute also passes this test**. Seeing as both parachutes passed the testing requirements, there is no need to retest and project requirement S.A.2 has been verified. Also, no design changes need to be made.

Picture:

Figure 6.2: One Drogue Parachute Trial in Parachute Drop Test

6.2 Payload Testing

6.2.1 Weight Testing — VT.P.0.1, VT.P.0.2, VT.P.0.3, VT.P.0.4

Test Objective: Fulfill requirements S.P.0: The overall mass of the payload systems shall not exceed 16lbm, S.P.0.1: The overall mass of the lander subsystem shall not exceed 3lbm, S.P.0.2: The overall mass of the retention and deployment subsystem shall not exceed 5lbm, and S.P.0.3: The overall mass of the ABCS shall not exceed 8lbm.

Testing Variables: The testing variables are the mass of each system of the Payload in pound-mass, determined through measuring pound-force.

Success Criteria: Mass of the Payload systems including Lander subsystem, Retention and Deployment subsystem, and ABCS properly fulfill set weight requirements.

Why it is Necessary: A smaller total weight is necessary to obtain the required minimum thrust-to-weight ratio of 5:1, as defined by S.V.9. Additionally, properly coordinated weight is essential to achieving the desired apogee.

Methodology:

1. Assemble individual components in flight configuration.
2. Measure the mass of each system with a scale.
    a. Measure the combined mass of the Payload.
    b. Measure the individual mass of the Lander Subsystem.
    c. Measure the individual mass of the Retention and Deployment Subsystem, subtracting the mass of the vehicle airframe.
    d. Measure the individual mass of the ABCS, its couplers, and attachment fasteners, subtracting the mass of the lower airframe (including all motor components).

Impact of Results: Should any system exceed weight requirements, further work would be required to optimize the materials used in construction of components, whether that be design changes or other relevant compromises. Should each system meet weight requirements, no further modifications would be necessary.

Results and Conclusions: This test has not been conducted and will be done early 2021.

6.2.2 ABCS Physical Testing — VT.P.2.1

Test Objective: Fulfill requirement: G.2.4.1 TD: Verify that the ABCS can function correctly through physical testing at a factor of safety of at least 1.5.

Success Criteria: The ABCS can withstand the maximum simulated drag load of 120N applied on each Aeroplate with a factor of safety at or above 1.5, meaning 180N each.

Why it is Necessary: The test ensures that the ABCS will be operational for the flight of the vehicle.

Methodology:

1. The system will be anchored vertically to a beam.
2. Attach cables to each of the paddle struts and then apply 55 kg of tension in order to simulate the downwards force on the entire system simultaneously. This will simulate the flight load with a safety factor of 1.5. If the ABCS performs under this load without breakage, then the ABCS has passed the test.

Impact of Results: If the system performs as expected when the maximum anticipated force is applied, then no further action needs to be taken and the ABCS can be fully integrated into the rocket. If there is a problem encountered during testing, modifications will be made to the system and retested until the desired outcome is achieved. This test has the added benefit of showing how well the motor can handle applied torque from the system while encountering a force.

Results and Conclusions: This test has not yet been conducted but will be done early 2021.

6.2.3 ABCS Battery Testing — VT.P.2.2

Test Objective: Fulfill requirement S.P.2.7: The battery powering the ABCS must be able to withstand idle operation for a minimum of 2 hours.

Success criteria: The ABCS would be able to receive adequate power from the battery for 2 hours powered on.

Why it is necessary: This test will ensure that the ABCS will be able to function when necessary during the flight of the vehicle.

Methodology:

1. The motor assembly system will be connected to a multimeter and the battery. As the motor is deployed, the current will be measured which will then be used to calculate how long the battery can support active motor usage.
2. To test the idle operation of the motor assembly it will be connected to the battery and multimeter and left turned on in the idle position. The voltage will be recorded every 0.25 hours until at least the 2 hour mark is reached. The voltage drop will then be analyzed in order to determine if it adequately meets the voltage needs of the system.

Impact of Results: If the test goes as expected, the battery and motor combination is adequate and is ready to be fully integrated into the rocket. If the battery will not be able to withstand the idle operation for the full 2 hours, then modifications will be made and further tests will be made until this is able to be accomplished.

Results and Conclusions: This test has not yet been conducted but will be done early 2021.

6.2.4 ABCS IMU Testing — VT.P.2.3

Test Objective: Fulfill requirement S.P.2.8: The data output from the inertial sensor must be useful for the MCU. If the output from the MCU is raw data, additional electronics must be designed to convert raw data into a usable format for the MCU.

Success criteria: The data outputted from the inertial sensor is accurate.

Why it is necessary: This data would be used in order to monitor the behavior of the rocket during flight so that any necessary adjustments to the ABCS can be made.

Methodology:

1. To test the IMU the sensor will first be turned to 45, 90, and 180 degrees in order to compare the output data to reality.
2. A second test will verify the IMU's acceleration accuracy by tilting the sensor which should have an outputted acceleration equal to gravity.

Impact of Results: If the test goes as expected, the IMU will be fully integrated into the system's software. If the test does not perform as expected, troubleshooting will be done to determine the source of the issue whether it is hardware or software based and testing will continue until the IMU is accurate.

Results and Conclusions: This test has not yet been conducted but will be done early 2021.

6.2.5 ABCS Activation Testing — VT.P.2.4

Test Objective: Fulfill requirement S.P.2.6: The ABCS must be able to fully activate and deactivate control surfaces in under [1s] seconds.

Success Criteria: The ABCS can operate within the allotted time of 1 second.

Why it is necessary: This will ensure that the system is able to operate as quickly as necessary during flight.

Methodology:

1. An analysis will be done in order to determine how long it should take for the aeroplates to be fully deployed.
2. In order to fully test the activation/deactivation time, the plates will be loaded with 45 kg of tension evenly split between the three plates. The system will then be timed as it activates and deactivates in order to verify that it is within the allotted time of 1 second.

Impact of Results: If the test goes as expected no changes to the ABCS are necessary. If the activation and deactivation are not each able to be completed in the allotted time, modifications to the software and/or hardware will be made, and testing will continue until the requirement is met.

Results and Conclusions: This test has not yet been conducted but will be done early 2021.

6.2.6 PLS R&D Deployment Testing — VT.P.1.1

Test Objective: Determine whether requirements: S.P.1.4, S.P.1.18, S.P.1.19, and S.P.1.21 will be fulfilled or not.
Success criteria: The Lander analogue stays nearly immobile during in the payload bay while it is suspended and while it is suspended and shaking/swaying. The team can switch the R&D electronics from pre-flight to flight ready without disassembling the payload bay during the test. Finally, the Lander Analogue deploys under 5 seconds during both the static test and the swaying test.

Why it is necessary: The test ensures the payload bay and R&D system protect and deploy the Lander as designed.

Methodology:

1. Fully assemble the payload bay and insert Lander Analogue into the bay. Visually inspect fit of Lander Analogue.
2. Close payload bay and attempt to switch R&D electronics from pre-flight status to flight ready status.
3. Suspend the payload bay from the test stand.
4. Induce both vibration and swaying of the payload bay to simulate flight conditions and inspect Lander Analogue for movement or damage.
5. Signal the R&D electronics to deploy the Lander Analogue and time the process from signal sent to full deployment.
6. Perform the previous step under various swaying condition, both different intensities and different direction of sway.

Impact of Results: If the payload bay assembly perform as expected no action is necessary. If significant movement or any damage of the Lander Analogue is observed during the test modification of the payload bay will be necessary to better secure the Lander. If the deployment time is observed to be over 5 seconds during any of the tests the team will determine whether it is feasible to start the deployment process earlier in the flight so that Lander ejection occur in the correct altitude range. Additionally, the times recorded will be used to determine how much earlier to begin the deployment. If the Lander Analogue fails to deploy in any of the test's significant modifications to the ejection method might be necessary. This is also true for if the R&D electronics prove impossible to switch from pre-flight to flight ready during the test.

Results and Conclusions: This test has not yet been conducted but will be done early 2021.

6.2.7 PLS R&D Retention Testing — VT.P.1.2

Test Objective: Ensure that the R&D complies with Subteam Requirement S.P.1.18 and Project Requirement G.2.4.1TD. These requirements demand that the Lander remains protected within the R&D under flight loads until the point of deployment with at least a 1.5 failure point factor of safety.

Success Criteria: The R&D is able to retain the Lander under a Payload Section deceleration force equivalent of at least 2g.

Why it is necessary: While the software design of R&D must activate within the required altitude bounds, there remains the possibility of the failure of R&D to retain the Lander before the desired time and location. Therefore, the R&D must be tested to ensure that it will operate correctly under worst-case flight loads. As discussed in the CDR mechanical testing theory section, while the deployment of the main parachute generates the most deceleration of about 2g, the team would be satisfied by the forced back-drive of the R&D system under such conditions, seeing as the R&D will be designed to release when this point of flight is detected. Meanwhile, the team is concerned about the deceleration caused during drogue descent; this quantity is much less—marginally greater than 1g. Therefore, to ensure a middle ground for safety design, the team has decided to implement a test plan to exceed the 1.5 FoS requirement with respect to the drogue descent stage. Ultimately, the team decided on elevating this virtual FoS to 2, planning to expose the R&D to typical deceleration experienced during main parachute deployment.

Methodology:

1. The execution of this test will be in the same style as a typical tensile test. However, for sake of testing, a particular testing rig must be produced.
2. The Lander, R&D Pizza Table, nosecone, and nosecone attachment plate should be weighed together.
1. The Payload Bay must be assembled to its completed state. For the purposes of this testing rig, the vehicle's nosecone will be removed.
2. The Lander should be loaded into the R&D and secured.
3. The Payload Bay will be suspended by its eyebolt.
4. A weight equal to the Lander, R&D Pizza Table, and nosecone attachment plate PLUS twice the nosecone should be secured to the nosecone attachment plate. This will statically simulate 2g of deceleration.
5. The team should wait until oscillation of the Payload Bay has ceased or until the Pizza Table is back-driven out from the R&D. If the latter occurs, the R&D has failed the test.
6. Record the displacement distance of the nosecone attachment plate—if any—for future reference.
7. Remove the applied load.
8. Repeat steps 3 through 8 two more times to ensure short-term cyclic loading integrity.
9. The team should inspect the R&D for internal damage. If the R&D visibly back-drives at all during this test, the team should reconsider the safety of the design.

Impact of Results: The results of this test will inform whether the R&D retention design has enough strength and resistance to complete its descent task under the worst possible situations. If the design is shown to fail under these loading conditions, then the R&D must be redesigned to handle more load. If the R&D is shown to handle this level of loading without detectable damage, then the R&D will be ready for flight.

Results and Conclusions: This test is incomplete.

6.2.8 PLS RF Transceiver Testing — VT.P.1.3

Test Objective: To verify the RF design can fulfill requirement S.P.1.15 which states that the Lander must be able to transfer image data to the ground control station within 1 mile.

Success Criteria: The Lander will be able to transfer image date to the GCS at a range of one mile.

Why is it necessary: The primary purpose of the Planetary Lander System is to capture a panoramic image and transmit the image back the team. Failure to transmit the image from the Lander to the GCS would result in an overall mission failure for the PLS. This test will ensure that the selected RF solution is capable transmitting image data from any location within the landing zone.

Methodology:

To fully verify the functionality of the RF solution would require the Lander to be completely assembled to ensure no other components are causing interference. To ensure that problems are identified as soon as possible the team will conduct a preliminary test and a final test. The preliminary test will not require the Lander to be fully assembled and can be conducted much earlier.

The preliminary test will take place in more ideal conditions and will not require the full Lander assembly. This test will identify any issues with the transceiver at the component level and verify the transceiver can transmit at the desired range in ideal conditions. The final test will be to test the integration of the transceiver with the entire Lander assembly and verify that the transceiver will function at a system level. Since this test includes the entire Lander assembly, it will verify that the Lander electronics and outer materials are interfering with the transceiver operation.

The preliminary test procedures are as follows:

1. The RF transceiver will be connected to a remote system that is cable of storing and transmitting an image. This system will simulate the Lander and can be a test assembly of the Lander Control System or another system capable of completing the test.
2. The “local” RF transceiver will be connected to a device capable of recording the RSSI and packet loss from the transceiver. This device will simulate the GCS and will remain stationary during the duration of the test. This device can be the GCS, but it is not required for this test.
3. The simulated GCS and Lander devices will be powered on and a connection will be established. The Lander device will transmit an image that will be displayed on the GCS. Once this initial check has been completed the Lander device will be powered down. The GCS device will remain powered on.
4. Utilizing a GPS enabled phone, or similar device, the Lander device will be walked .25 miles away from the GCS device.
5. The Lander device will be powered on and the time it takes to establish a connection will be measured.
6. Once a connection is established, the Lander device will transmit a sample image to the GCS device. At the GSC device, team members will verify the integrity of the image and record the RSSI and packet loss reported by the transceiver.
7. The Lander device will be powered down.
8. Steps 4-7 will be repeated at increments of .25 mile until the Lander device has been tested at 1 mile away from the GCS device

The final test procedures are as follows:

1. The RF transceiver will be installed on the fully assembled Lander. The Lander will have modified code specifically created for this test, as well as a preloaded sample image on the SD card. The lander must be in its final state for this test.
2. The other RF transceiver will be installed on the GCS. The GCS will not need to be in its final state for this test, however, it must have software capable of receiving and displaying an image. The GCS must also be capable of displaying the RSSI and packet loss data from the transceiver. The GCS will also be placed on the table or stand so that it is in the same or a similar configuration to the on it will be in on launch day.
3. The GCS and Lander will be powered on. The Lander will transmit an image to the GCS. Once the team verifies that the Lander and GCS software are functioning as expected, the Lander will be powered off. The GCS will remain on.
4. Utilizing a GPS enabled phone, or similar device, the Lander will be walked .25 miles away from the GCS. Once there, the Lander will be placed on its side as if it had just landed.
5. The Lander will be powered on and the time it takes to establish a connection will be measured.
6. Once a connection is established, the Lander will transmit the sample image to the GCS. At the GCS device, team members will verify the integrity of the image and record the RSSI and packet loss reported by the transceiver.
7. The Lander will be powered down.
8. Steps 4-7 will be repeated at increments of .25 mile until the Lander has been tested at 1 mile from the GCS.

Impact of results: Successful completion of this test verifies that the selected transceiver can carry out the required image transfer at the maximum range it may be required too. If the selected Transceiver fails the preliminary test, then the team will need to search for a component level replacement. This means that the team will seek to replace the transceiver, antenna, or both. If the transceiver passes the preliminary test but fails the final test then the transceiver is not functioning at the system level. This most likely means that interference from other systems on the Lander is causing a failure. In this situation the team will need to take steps to mitigate interference by making design modification on other systems on the Lander. If it is not practical to make these modifications, then the team will have to replace will need to upgrade the transceiver or antenna.

Results and conclusions: Neither the preliminary nor the final test have been conducted. The preliminary test will take place in January of 2021 and the final test will be conducted once the Lander has been fully assembled and final modifications have been made.

6.2.9 PLS Battery Drain Testing — VT.P.1.4, VT.P.1.5, VT.P.1.6

Test Objective: To verify that the Planetary Lander System is compliant with S.P.1.11 and S.P.1.12. These requirements dictate that all PLS subsystems must have sufficient battery life to sustain their pre-flight state for 18 hours, and their launch-ready state for an additional 2 hours.

Success Criteria: The Planetary Lander System will contain enough battery to successfully perform it mission after staying in a pre-flight state.

Why it is necessary: If the battery were to drain too much during the pre-flight and launch-ready phases, then the battery may not contain enough charge for the system to carry out its tasks, resulting in a failed mission. To avoid this, the team will test the batteries to identify any faults before the design is finalized.

Methodology:

The Lander, R&D, and GCS can be tested separately or together using the following methodology.

Pre-flight drain test:

1. The battery is connected to its charger until it is fully charged.
2. The battery is disconnected from its charger and the starting voltage is measured using a multimeter.
3. The battery is then connected to the electronics it will be connected to on launch day. The electronics will remain off.
4. After 18 hours the battery will be disconnected from the electronics and the voltage will be measured again.

Launch-Ready test:

1. The battery is connected to its charger until it is fully charged.
2. The battery will be disconnected from the charger and connected to its respective control system.
3. The control system will be turned on and put into its launch-ready configuration.
4. The system will be left on for at 2 hours for R&D and 3 hours for the GCS and Lander.
5. The voltage of the battery will be measured every 0.5 hours (including 0 hours and 3 hours)

After the tests have been completed the test data will be analyzed to determine if the battery still contains sufficient charge and voltage to complete the mission.

Impact of Results: If the selected batteries cannot sustain the PLS subsystems for the required time then the team will need to make modifications and conduct more tests. These modifications could include changes to the electronics or software to reduce power consumption or changing to a bigger battery.

Results and Conclusions: This test is incomplete

6.2.10 PLS Panoramic Image Capture Test — VT.P.1.7

Test Objective: To verify that the Planetary Lander System, and more specifically the Panoramic Image Capture Subsystem (PICS), is compliant with requirements S.P.1.10, S.P.1.13, and S.P.1.14. These requirements ensure that the Lander can both capture and send the images necessary to create a panoramic photo in expected terrain conditions.

Success Criteria: The Planetary Lander System will successfully capture and transfer image data to the Ground Control Station (GCS) while also storing a local version of the image data on the LCS's SD card. The GCS will receive and process the image data into a panoramic image which will display on its monitor.

Why it is necessary: The capture and generation of a panoramic photo that displays the surroundings of the Lander is an essential part of its mission. If the team does not ensure that all systems involved with the photo's successful display on the GCS are fully functional then it is almost ensured that the systems will not work when they are expected to. This test will reveal any bugs or errors in the systems and narrow down the causes which will speed up the development process.

Methodology:

Test software will be loaded onto the LCS prior to testing to allow for manual control of the Lander's subsystems through a connection from the Lander's radio and the GCS. The GCS will also have an application created specifically for manual control of the Lander for testing purposes.

1. The Lander, completely assembled and powered on, will have all its legs in a standing position and placed upright in terrain simulating the expected conditions of the launch field.
2. The GCS will be in proximity of the Lander and turned on. The Lander will begin initialization and attempt to establish a connection with the GCS. Once connection has been established, the Lander will communicate if initialization of all system was successful.
3. If initialization is successful, the GCS be powered off and moved at least a quarter mile away from the Lander. The Lander will continue to try to establish a connection of the GCS.
4. The GCS will then be powered on and wait to connect to the Lander.
5. Once connection with the GCS has been established, a command will be sent to the Lander from the GCS to start its image capture protocol.
6. The Lander will send a notification to the GCS that all images have been captured and stored on the SD card. If the Lander is not able to capture and store the photos, an error message will be sent to the GCS.
7. If the image capture was successful, the GCS will send a command to the Lander to start the image transfer process.
8. Once the images are transferred, each photo will be inspected prior to image processing to confirm that the images were captured successfully and unobstructed by the terrain.
9. The GCS will then process the images into one panoramic photo. After processing is complete, the panoramic photo should automatically be shown on the GCS's display.
10. The panoramic photo is inspected to conclude if the image processing was successful.
11. The Lander is turned off and disassembled to allow access to the SD card.
12. The SD card is read to confirm that the images were saved successfully.

Impact of Results: The test is constructed in such a way that allows any failures that occur to be associated with a specific system within the Lander or GCS. If the entire test is successfully conducted, then the team can be sure that all systems and software for the Lander and GCS are functional and integrated correctly.

Results and Conclusions: This test is incomplete

6.2.11 PLS SOS Orientation Testing — VT.P.1.8

Test Objective: To verify that the Planetary Lander System is compliant with P.4.3.3 which states that the Lander will self-level within 5 degrees of vertical.

Success Criteria: The Lander can self-level within 5 degrees of vertical.

Why it is necessary: The ability to self-level is critical to the Lander's mission. If the Lander is not able to self-level, the Lander's mission will be considered a failure. Conducting this test will allow the team to determine if changes must be made to the Self Orientation Subsystem, whether they be mechanical or software.

Methodology:

Test code will be uploaded to the Lander for this test. The GCS or another RF capable device will send commands to the Lander to control it remotely. The Lander will have to be in its final state and fully assembled for this test.

1. Place the Lander on its side in terrain simulating the area of the launch field. The GCS or RF capable device should be on and ready to connect to the Lander.
2. Turn on the Lander through its key switch and wait for it to connect to the GCS.
3. Measure the starting orientation of the Lander using a measuring tool.
4. Once connected, a command will be sent to the Lander to begin self-leveling.
5. The Lander will run through the stages of self-leveling and will send progress notifications to the GCS.
6. Once the Lander has completed self-orientation, measure the orientation of the Lander using a levelling device.

7. Verify that the external measurement and the orientation measurement sent by the Lander are within a reasonable margin of measure.

Impact of Results: This test will ensure total functionality of the Self Orientation System before any demonstration flights. Successful completion of this test will verify that the SOS system is capable of self-leveling. If the Lander fails the test, the team will need to modify software and/or the mechanical systems until the Lander meets the requirement. If successful, P.4.3.3 will still not be completely verified until a separate wind-release test—discussed below in the wind-release test procedure—has been completed. If this test has been completed but a failure to complete the wind-release test results in design changes, then this test will need to be completed again with the new changes.

Results and Conclusions: This test is incomplete.

6.2.12 PLS D&L Wind-Release Testing — VT.P.1.9

Test Objective: Ensure that the Lander can satisfy Subteam Requirement S.P.1.9, stating that the Lander is able to withstand 10 mph wind while grounded.

Success Criteria: The Lander is able to remove its connection with its parachute in order to sustain 10 mph winds. This must be assured to occur despite orientation.

Why it is necessary: In order to orient properly, the Lander must not be affected by any external loads, whether that be by the parachute or by ambient winds. The purpose of this test is to verify the efficacy of the D&L detachment mechanism in separating from the parachute. Without its parachute, the Lander should be able to sustain such winds while grounded without falling over.

Methodology:

1. This test involves the operation of the D&L under various conditions, proving that the D&L will be able to ensure the stability of the Lander in its upcoming orientation phase. This test is multi-part but should cover every foreseeable D&L worst-case scenario.
1. Assemble the Lander with its parachute attached.
2. Enable the D&L system remotely to avoid any heat hazard caused by the descent solution.
3. Bring the Lander outside, clear of personnel or buildings. For D&L mechanisms utilizing nichrome, seek a suitable non-flammable ground surface such as gravel.
4. Orient the Lander in one of these possible configurations:
5. Lander on its side, attachment cable sticking up.
6. Lander on its side, attachment cable sticking to the Lander's side.
7. Lander on its side, attachment cable pressed into the ground, but led to the side of the Lander. This is to ensure operation while crushed.
8. Enable the D&L release mechanism while clear of the Lander.
9. Wait for 5 minutes or until the D&L has visibly operated correctly. If the D&L does not operate or the Lander catches fire, disable the D&L immediately and utilize a fire extinguisher.
10. Observe whether the D&L has successfully disconnected the parachute attachment cable. Retrieve parachute if necessary.
11. Repeat testing process for all three main orientations.

Impact of Results: The successful operation of the D&L will determine its ability to be used during the vehicle flight. If the D&L is unable to operate reasonably well within 5 minutes, then the D&L design must be modified to work correctly. Additionally, if the design is deemed too dangerous for operation, consideration of other designs must be made.

Results and Conclusions: This test is incomplete.

6.2.13 PLS D&L Structural Testing — VT.P.1.10

Test Objective: Verify the D&L's compliance with Project Requirement G.2.4.1TD, requiring that the PLS withstand all expected flight loads with a 1.5 failure point factor of safety.

Success Criteria: All elements of the D&L will be able to withstand loading of at least 19lbf without discernable damage for at least 3 cycles.

Why it is necessary: In order to ensure that the Lander may safely descend under its parachute, all points of connection with the parachute to the Lander must be capable of sustaining worst-case descent loads. Considering that the D&L will attempt to sever the point of attachment with the parachute once grounded, it must be proven that this will not occur before the D&L/LCS system decides. Through static analysis, the drag experienced by the parachute is assumed to be equal to the tension in the attachment cord; calculation available in this report puts this load at 12.425 lbf. With a FoS of 1.5, this results in the success value of 19 lbf.

Methodology:

1. Testing the structural integrity of the D&L requires that the entire Lander assembly be physically present, ensuring that both the D&L itself and Lander body is cable of withstanding such loads. Therefore, this test will involve the static loading test of the entire Lander and D&L.
1. The Lander must be assembled to its full physical state, including its parachute but without the R&D deployment bag.
2. The Lander will be suspended by the parachute's metal attachment ring. All drag loads will be assumed to have been transferred to this ring during descent.
3. Beneath the Lander will be suspended an at least 19 lbf weight, producing an attachment cord tension of at least 19 lbf. This test will most likely round to 20 lbf to account for available weights. The weight should be wrapped around the Lander's central body to distribute the load internally.
4. The team will wait until suspension oscillation has ceased before removing the applied weight. If the Lander shows any visible or audible sign of yielding, the test has been failed.
5. The team will repeat steps 3 and 4 two more times to ensure short-term cyclic loading integrity.
6. The team should disassemble the Lander to inspect for internal damage. By the discretion of the testers, the team will report whether the D&L is ready for flight.

Impact of Results: The results of this test will inform whether the D&L structural design has enough strength and resilience to complete its descent task under the worst possible situations. If the design is shown to fail under these loading conditions, then the D&L must be redesigned to handle more load. If the Lander is shown to handle this level of loading without detectable damage, then the D&L will be ready for flight, whether or not the disconnection mechanism is utilized.

Results and Conclusions: This test is incomplete.

6.3 Derived Requirements & Verification Plans

Requirement ID	Requirement Summary	Verification Type(s)	Verification Plan ID(s)	Verification Plan / Prerequisite Requirement Summary	Status
N.1.1	All work will be completed by the team specifically for this year's competition. A mentor will assist with handling of potentially explosive or flammable devices.	D	N/A	PSP-SL members shall demonstrate the new work they have completed through milestone documentation and presentations.	In Progress
N.1.2	The team will provide and maintain a project plan describing all aspects of the project.	D	N/A	The team will submit an up-to-date project plan with all milestones.	In Progress
N.1.3	For security reasons Foreign National team members will be identified by PDR.	D	N/A	The team will submit a list of FN team members with PDR.	Complete
N.1.4.1-3	The team will create launch week team member roster by CDR consisting of students engaged throughout the year and a single adult mentor.	D	N/A	The team will submit a list of team members and the project mentor with CDR.	Complete
N.1.5	The team will engage more than 200 participants in STEM activities.	D	N/A	The team will submit relevant outreach activity forms within two weeks of a given activity.	In Progress
N.1.5.1[TD]	The team will host virtual software tutorials for its members and the greater college community	D	N/A	The team will submit relevant outreach activity forms within two weeks of a given activity.	Incomplete
N.1.6	The team will establish a social media presence.	D	N/A	The team will submit a list of active team social media accounts.	Complete
N.1.7-10	All deliverables will be properly formatted and emailed to the USLI team by the specified deadlines.	D	N/A	The team will submit properly formatted deliverables on time at all milestones.	In Progress
N.1.7.1	All subteams will complete milestone editing 3 days prior to the official NASA deadline. After this time only the Project Management team will have access to the documentation to perform final edits.	D	N/A	The team will submit properly formatted deliverables on time at all milestones.	In Progress
N.1.11	The team will use proper teleconferencing equipment for all calls with the USLI team.	D	N/A	The team leads will perform professional video calls for all milestone meetings.	In Progress
V.1.12	The launch vehicle will use USLI standard launch rails and pad configurations.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
N.1.13	The team will identify an experienced mentor.	D	N/A	The team will submit information about the team mentor in CDR.	Complete
N.1.14	The team will track, and report hours spent working on all milestones.	D	N/A	The team will submit member timesheets with all reports.	In Progress

N.1.14.1 TD	The team will set up a software tool to allow members to submit their working hours.	D	N/A	The team will submit member timesheets with all reports.	In Progress
V.2.1	The vehicle's apogee shall be between 3,500 and 5,500 feet.	P		The team will conduct analyses and tests to verify this requirement with the ABCS active and deactivated. This verification will also include the vehicle demonstration flight.	In Progress
B.2.1.2 TD	The launch vehicle will actively control its apogee using an AeroBraking Control System (ABCS). Using the ABCS, the vehicle will reach within 15 ft of the PDR target apogee.	P		The ABCS team will conduct a multifaceted verification of the ABCS system to ensure its ability to function as intended.	Incomplete
N.2.2	The team will declare a target altitude at PDR.	D	N/A	The team will submit its target altitude in PDR.	Complete
A.2.3	The launch vehicle shall contain a commercially available barometric altimeter for recording apogee.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
G.2.4	The vehicle shall be designed to be recoverable and reusable.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
G.2.4.1 TD	The vehicle shall withstand all expected flight loads with a minimum safety factor of 1.5.	P (A, T)	VT.P.2.1 VT.P.1.2 VT.P.1.10	All subteams will independently verify the strength of their subsystems through analysis and testing.	In Progress
V.2.5	The launch vehicle shall have a maximum of 4 independent sections.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
V.2.5.1-2	Couplers at inflight separation points shall be at least 1 cal in length. Nose cone couplers shall be at least ½ cal in length.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
V.2.6	The launch vehicle shall be able to launch within 2 hours of flight authorization.	D	D.M.2.1	VDF will demonstrate the team's ability to prepare the launch vehicle for flight.	Incomplete
V.2.6.1 TD	The launch vehicle will be assembled in a quality conducive assembly area separate from the launch site. A quality conducive assembly area has (but is not limited to) the following attributes: climate control not necessitating thermal protective clothing, bright overhead lighting, and access to tools and components.	D	D.M.2.1	VDF will demonstrate the team's ability to prepare the launch vehicle for flight.	Incomplete
V.2.7	The launch vehicle and payload shall be able to remain in the flight ready configuration for at least 2 hours.	P		All subteams will perform battery drain testing on their subsystems.	Complete
V.2.7.1 TD	The launch vehicle and payload shall be able to remain in the pre-flight state for at least 18 hours.	P		All subteams will perform battery drain testing on their subsystems.	Complete

V.2.7.2 TD	The transition between the pre-flight state and flight ready state will not require the disassembly of the launch vehicle.	I	I.M.3.1	The systems manager will inspect all checklists to ensure procedural compliance.	In Progress
V.2.8-9	The vehicle shall be capable of being launched via a 12 VDC firing system as provided by the launch services provider	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
V.2.10	The launch vehicle shall use an APCP motor.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
V.2.10.1-2	The final motor choice shall be declared by CDR. Any changes after CDR must be approved by the RSO.	D	N/A	The team will submit its motor selection in the CDR milestone report.	Complete
V.2.11	The launch vehicle will be limited to a single stage.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
V.2.12	The total impulse of the launch vehicle shall not exceed 5120 Ns (L-Class).	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
V.2.13.0-3	Pressure vessels on the vehicle must be approved by the RSO and maintain safe standards.	N/A	N/A	No pressure vessels will be included in the vehicles design.	Complete
V.2.14	The vehicle shall have a minimum stability margin of 2.0 cal at rail exit.	P	D.M.2.1,	The avionics and construction subteams will independently verify the launch stability of the vehicle. Compliance will also be demonstrated during VDF.	In Progress
B.2.14.1 TD	The ABCS shall not reduce the stability margin below 2.0 cal at any point, under any failure mode.	P	D.M.2.1,	The team will perform FMEA and other analyses on the ABCS system to ensure compliance. Compliance will also be demonstrated during VDF.	In Progress
V.2.15	The vehicle shall not have any structural protuberance forward of the burnout CoM. Excepting aerodynamically insignificant camera housings.	I/A	I.M.1.1	The team will inspect the PDR design for forward structural protuberances. If any are present, the team will perform CFD analysis to ensure aerodynamic insignificance.	Complete
V.2.16	At rail exit the vehicle shall have a minimum velocity of 52fps.	P	D.M.2.1,	The team will perform launch analysis to ensure proper rail exit velocity.	Complete
V.2.17	A subscale rocket will be successfully flown by CDR.	D	N/A	The team will submit subscale altimeter data with CDR.	Complete
V.2.17.1	The subscale rocket shall resemble and perform similarly to the full-scale rocket but will not be the full-scale rocket.	I	I.M.1.2	The team will inspect the subscale vehicle design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
V.2.17.2	The subscale rocket shall contain an altimeter to record apogee.	I	I.M.1.2	The team will inspect the subscale vehicle design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete

V.2.17.3	The subscale rocket will be newly constructed for the 2021 competition	I	I.M.1.2	The team will inspect the subscale vehicle design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
N.2.17.4	Proof of the subscale flight shall be included in CDR	D	N/A	The team will submit subscale altimeter data with CDR.	Complete
N.2.18	The team shall complete the following demonstration flights.	P	N.2.18.1-2	The team will verify all prerequisite requirements.	In Progress
N.2.18.1	The team will fly the launch day vehicle in its final configuration in order to validate its flight capabilities. This Vehicle Demonstration Flight (VDF) has the following success criteria.	P	V.2.18.1.1-9	The team will verify all prerequisite requirements.	Incomplete
G.2.18.1.1	The vehicle and recovery system will have functioned as designed.	P		All subteams will complete post-launch system assessments.	Incomplete
N.2.18.1.2	The full-scale rocket must be newly designed and constructed for the 2021 competition.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
P.2.18.1.3	The payload does not have to be flown during VDF.	N/A	N/A	The team overrides this requirement with N.2.18.2.2.	Complete
P.2.18.1.3.1-2	If the payload is not flown, a mass simulator will be used to simulate the payload mass and will be located in approximately the same location as the payload CoM.	P		If included, the effect of the payload mass simulator will be quantified by the payload team.	Incomplete
P.2.18.1.4	If the payload effects the external surface of the rocket or manages the total energy of the vehicle, those systems will be active during VDF.	I	I.M.3.1.1	Before VDF, the systems manager will ensure all protrusions and energy management systems are present.	Incomplete
V.2.18.1.5	During VDF, the vehicle shall use the declared launch day motor.	I	I.M.3.1.1	Before VDF, the systems manager will ensure the declared launch day motor is installed in the vehicle.	Incomplete
V.2.18.1.6	The vehicle shall have the launch day ballast configuration for the VDF.	I	I.M.3.1.1	Before VDF, the systems manager will inspect the vehicle for proper ballasting	Incomplete
N.2.18.1.7	The team will not modify the vehicle after VDF without permission from the RSO.	D	N/A	The vehicle present at LRR will be identical to the vehicle discussed in FRR.	Incomplete
N.2.18.1.8	Altimeter data will be provided in the FRR report to prove a successful flight	D	N/A	The team will submit VDF altimeter data in FRR.	Incomplete
N.2.18.1.9	VDF must be completed by the FRR submission deadline. If a re-flight is required, an extension may be granted.	D	N/A	The team will submit VDF altimeter data in FRR	Incomplete
N.2.18.2	The team will fly the launch day payload aboard the launch day rocket in a successful Payload Demonstration Flight. This PDF will be considered	P	P.2.18.2.1-3	The team will complete all prerequisite requirements for PDF.	Incomplete

	successful if the vehicle experiences stable ascent and the following requirements are met.				
P.2.18.2.1	The payload will be fully retained until the intended point of deployment, and all R&D mechanisms will function as intended and suffer no damage	I	I.M.3.2	All subteams will complete post-launch system assessments.	Incomplete
N.2.18.2.2 TD	VDF will contain the final payload system, unless waiting until the completion of the payload would bar the team from satisfying requirement N.2.19.1	I	I.M.3.1	The systems manager will inspect the vehicle for proper installation of the payload system before flight.	Incomplete
G.2.18.2.3 TD	Test launches will only be attempted if all subsystem designs are frozen and thorough assembly protocols have been created.	I	I.M.3.2	The systems manager and project management team will conduct a survey of subteam leads and confirm their confidence in the vehicles ability to have a safe and successful flight.	Incomplete
N.2.19	An FRR Addendum will be required for teams completing PDF or VDF re-flight after the FRR report deadline.	D	N/A	The team will submit FRR Addendum if required.	Incomplete
N.2.19.1	The FRR Addendum must be submitted for all teams whose circumstances require its submission.	D	N/A	The team will submit FRR Addendum if required.	Incomplete
N.2.19.2	If the PDF fails, the team will not be permitted to fly at the competition launch.	N/A	N/A	N/A	Complete
N.2.19.3	If the PDF partially fails, the team may petition the RSO for permission to fly the payload at launch week.	N/A	N/A	N/A	Complete
N.2.20	All separable components will have the team's name and Launch Day contact information clearly visible.	I	I.M.3.1	The systems manager will inspect the launch vehicle and payload for proper labeling before all flights.	Incomplete
N.2.22.0-10	The vehicle will not use any of the following prohibited design features or modes: • Forward Firing Motors • Motors that expel titanium sponge • Hybrid Motors • Motor Clusters • Friction Fit Motors • Exceed Mach 1 at any point • Ballast exceeding 10% of the unballasted weight • Transmitters with individual power greater than 250 mW • Transmitters which create excessive interference	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete

	• Excessive / dense metal. Lightweight metal will be permitted for structural purposes				
V.3.1TD	The vehicle will contain an In-Flight Video Recording (IFVR) system to record flight video for downloading after recovery.	P		The IFVR team will verify the functionality of the IFVR system through a variety of DIAT methods.	Incomplete
V.3.1.1TD	The IFVR will have at least two sensors, aligned aft and radially, and may have a sensor aligned forward.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
V.3.1.2TD	The IFVR will be considered a vehicle element, not a payload experiment.	D	N/A	The construction team will be solely responsible for the IFVR and IFVR documentation will be included in the vehicle construction section of all reports.	Complete
A.3.1	Vehicle recovery process will abide by the requirements A.3.1.1 – A.3.1.13	P	A.3.1.1 – A.3.1.13	The team will complete all prerequisite recovery requirements.	In Progress
A.3.1.1	The main parachute will be deployed no lower than 500 feet.	P		The team will ensure the proper deployment of the main parachute through a variety of verification and design methods	In Progress
A.3.1.2	The apogee event will contain a delay of no more than 2 seconds.	P		The team will ensure the proper deployment of the drogue parachute through a variety of verification and design methods	In Progress
A.3.1.3	The motor will not be ejected at any point.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
A.3.1.4 TD	The recovery process will be designed to minimize shock to the vehicle.	P		The team will ensure the minimization of shock through various subteam requirements.	Complete
A.3.2	The team will perform a ground ejection test for all electronically initiated recovery events.	T	N/A	The team will submit ejection test results with FRR.	Incomplete
A.3.3	Each independent section of the launch vehicle will have a maximum kinetic energy of 75ft-lbf (101J)	P		The team will ensure acceptable landing energy through verification up to and including post launch examination of flight telemetry.	Incomplete
A.3.4	The recovery system will contain redundant, commercially available altimeters.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
A.3.5	Each altimeter will be equipped with a commercially available, dedicated power supply.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete

A.3.6	Each altimeter will be armed (placed into the flight-ready state) by a dedicated mechanical arming switch	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
A.3.7	The A&R system shall not be capable of disarmament due to flight sources.	P		The team will design and test the avionics bay to ensure that disarmament due to flight sources is impossible.	In Progress
A.3.8	A&R electrical circuits will be completely independent of payload electrical circuits.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
A.3.9	Removable shear pins will be used for both parachute compartments.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
A.3.10	The recovery area will be limited to a 2,500 ft. radius from the launch pad.	P		The team will verify the acceptability of the recovery area through a variety of methods.	In Progress
A.3.11	The descent time of the launch vehicle (apogee to touch down) must be less than 90 seconds.	P		The team will verify the acceptability of the vehicle descent time through a variety of methods.	In Progress
A.3.12	The launch vehicle will have a tracking device which transmits its position to a ground station.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
A.3.12.1	Any untethered component of the launch vehicle will contain a tracking device.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
A.3.12.2	All electronic tracking devices will be fully functional during launch day	I	I.M.3.1	Before launch, the systems manager will inspect all tracking devices' downlinks.	Incomplete
A.3.12.3 [TD]	Any tethered component of the launch vehicle will contain a tracking device.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
A.3.13	The recovery system will not be adversely affected by other electronics devices during flight.	P		The team will verify the electronic resilience of the avionics system through a variety of methods.	In Progress
A.3.13.1	Recovery system altimeters will be located in a compartment separated from other RF/EM emitting devices.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
A.3.13.2-4	Recovery system electronics will be shielded from other RF/EM emitting devices.	P		The team will verify the electronic resilience of the avionics system through a variety of methods.	In Progress
P.4.2	The payload will consist of a planetary lander capable of ejection during descent which will self-right during or after landing. After leveling the system will take a 360-degree panoramic photo of the landing site and transmit the photo to the team.	P	P.4.3, D.M.2.2	The team will complete all prerequisite requirements and demonstrate success in the payload demonstration flight.	In Progress

P.4.3	The landing system will adhere to requirements P.4.3.1.-P.4.3.4.4	P	P.4.3.1.-P.4.3.4.4	The team will complete all prerequisite requirements.	In Progress
P.4.3.1	The landing system will be completely jettisoned from the launch vehicle between 500 & 1000 ft AGL. The landing system must land within the external borders of the launch field. The landing system will not be tethered to the launch vehicle.	P		Once specific subteam requirements defined by the payload team have been verified, this requirement will be verified.	Complete
P.4.3.2	The vehicle will land in an upright orientation or will be capable of self-orienting autonomously.	P		Once specific subteam requirements defined by the payload team have been verified, this requirement will be verified.	InProgress
P.4.3.3	The landing system will self-level within 5 degrees of vertical.	P (A, T)	VT.P.1.8	Once specific subteam requirements defined by the payload team have been verified, this requirement will be verified.	In Progress
P.4.3.3.1	The lander must autonomously self-level.	P		Once specific subteam requirements defined by the payload team have been verified, this requirement will be verified.	In Progress
P.4.3.3.2	The landing system must record pre- and post-leveling orientation data. This data will be provided in PLAR	P		Once specific subteam requirements defined by the payload team have been verified, this requirement will be verified.	Incomplete
P.4.3.3.2.1	PDF orientation data will be provided in FRR	D	N/A	The team will submit PDF orientation data in FRR.	Incomplete
P.4.3.4	After self-leveling the lander will produce a 360-degree panoramic image of the landing site and transmit it to the team.	P		Once specific subteam requirements defined by the payload team have been verified, this requirement will be verified.	Incomplete
P.4.3.4.1	Image receiving hardware will be located within the team's assigned preparation or viewing area.	D	I.M.3.1	The team will display image receiving hardware to the NASA RSO before launch.	Incomplete
P.4.3.4.2	Only transmitters on board the vehicle during launch will be permitted to operate outside of the preparation or viewing areas.	D	I.M.3.1	The team will display image receiving hardware to the NASA RSO before launch.	Incomplete
P.4.3.4.3	After landing, the payload may use transmitters with a power greater than 250 mW.	N/A	N/A	N/A	Complete
P.4.3.4.4	The team will provide the 360-degree panoramic image in PLAR.	D	N/A	The team will submit the final panoramic image in PLAR.	Incomplete
P.4.4	The payload will adhere to requirements P.4.4.1-6	P	P.4.4.1-6	The team will complete all prerequisite requirements.	In Progress
P.4.4.1	Black Powder and/or similar energetics will only be used for in-flight recovery systems.	I	I.M.1.1	The team will inspect the vehicle and payload design at PDR to ensure compliance with NASA requirements. Relevant design aspects will be frozen after the submission of PDR.	Complete
P.4.4.2	Teams will abide by all FAA and NAR rules and regulations.	I	I.M.3.1	The systems manager will inspect the vehicle and payload before launch to confirm FAA and NAR compliance.	Complete
P.4.4.4	UAS payloads will be tethered to the vehicle and will not be released until RSO permission has been granted.	D	N/A	The team will inform the RSO of the relative location of the payload throughout flight.	Complete

P.4.4.5	UAS payloads will abide with all FAA regulations.	I	I.M.3.1	The systems manager will inspect the vehicle and payload before launch to confirm FAA and NAR compliance.	Complete
P.4.4.6	Any UAS weighing more than .55lbs will be registered with the FAA and be marked with its registration number.	I	I.M.3.1	The systems manager will inspect the vehicle and payload before launch to confirm FAA and NAR compliance.	Complete
P.4.5 TD	The payload team will be responsible for the design, manufacture, and operation of the ABCS	D	N/A	Project management will monitor the proper division of labor across the team.	In Progress
H.5.1	The team will use a launch and safety checklist which will be included in FRR and used in LRR and for all launch day operations	D	N/A	The team will submit all checklists with FRR.	Incomplete
M.5.1.1 TD	The team will utilize checklists for all pre-flight operations including but not limited to: A&R assembly, Payload assembly, Motor installation, and Vehicle integration.	D	N/A	The systems manager will create and review all checklists before use.	In Progress
M.5.1.2TD	The team will not launch a vehicle until the Systems Manager is satisfied with the status of all pre-flight checklists.	D	I.M.3.1	The systems manager will receive confirmation of checklist completion from all subteam leads.	In Progress
H.5.2	The team will identify a student safety officer who is responsible for all sub requirements of requirement M.5.3.	D	N/A	The team will submit information regarding its selected safety officer in the Proposal	Complete
H.5.3	Safety officer responsibilities are defined in H.5.3.1 -H.5.5	P	H.5.3.1-H.5.5	The team will complete all prerequisite requirements	In Progress
H.5.3.1	The safety officer will monitor team activities will an emphasis on safety during operations H.5.3.1.1-9 and H.5.3.2-4.	D	S.1.1	The safety officer will affirm their responsibility for the safety of the team.	In Progress
H.5.3.1.1-9	Safety officer will oversee all of the following operations: • Vehicle and Payload design • Vehicle and Payload construction • Vehicle and Payload Assembly • Vehicle and Payload ground testing • Subscale launch tests • Full-scale launch tests • Launch Day • Recovery Activities • STEM Engagement Activities	D	S.1.1	The safety officer will affirm their responsibility for the safety of the team.	In Progress
H.5.3.2	Ensuring the implementation of safety procedures for construction, assembly, launch and recovery.	D	S.1.1	The safety officer will affirm their responsibility for the safety of the team.	In Progress

H.5.3.3-4	Maintain and lead the development of team hazard analyses, failure mode analyses, and MSDS/chemical inventory data.	D	N/A	The team will submit hazard analyses and FMEAs in all relevant milestone reports.	In Progress
H.5.4	The team will follow all guidance from the local rocketry clubs RSO and will be in constant communication to ensure safety.	D	S.2.1	All team members will sign pledges affirming their intention to follow all local, state and federal regulations regarding the project.	Incomplete
H.5.5	The team will abide by all rules set by the FAA	D	I.M.3.1	The systems manager will inspect the vehicle and payload before launch to confirm FAA and NAR compliance.	Complete
N.6.1	At the NASA Launch Complex, the team must satisfy requirements N.6.1.1-4	P	N.6.1.1-4	The team will complete all prerequisite requirements.	In Progress
N.6.1.1	Teams must pass LRR during launch week.	D	N/A	The team will pass LRR during launch week.	Incomplete
N.6.1.2	The team mentor must be present for vehicle preparation and launch.	D	S.3.1	The team will not proceed with launch procedures without the team Mentor.	In Progress
N.6.1.3	The scoring altimeter must be presented to the NASA scoring official upon recovery.	D	N/A	The NASA RSO will receive the scoring altimeter after flight.	Incomplete
N.6.1.4	Teams may only launch once.	D	N/A	The team will only attempt a single flight.	In Progress
N.6.2.1	At Commercial Spaceport Launch Sites (local launch fields), the team must satisfy requirements N.6.2.1-8.	P	N.6.2.1-8	The team will complete all prerequisite requirements.	In Progress
N.6.2.1	The launch must occur at a NAR or TRA insured launch.	D	I.M.3.1	The systems manager will inspect the vehicle and payload before launch to confirm FAA and NAR compliance.	Incomplete
N.6.2.2	The launch site RSO will inspect the rocket and payload and determine its flight-readiness.	D	I.M.3.1	The team will not launch until receiving RSO approval.	Incomplete
N.6.2.3	The team mentor must be present for vehicle preparation and launch.	D	S.3.1	The team will not proceed with launch procedures without the team Mentor.	Incomplete
N.6.2.4	The team mentor and Launch Control Officer (LCO) will report any anomalies during ascent or recovery on the Launch Certification and Observations Report (LCOR).	D	N/A	The team will submit LCOR after flight	Incomplete
N.6.2.5	The scoring altimeter will be presented to the team's mentor and the RSO.	D	I.M.3.1	The RSO will receive the altimeter after flight.	Incomplete
N.6.2.6	The mentor, RSO, and LCO will complete all applicable sections in the LCOR.	D	N/A	The team will submit LCOR after flight.	Incomplete
N.6.2.7	The RSO and LCO shall not be affiliated with the team, team members, or academic institution.	D	N/A	The RSO and LCO will affirm their status on the LCOR.	Incomplete
N.6.2.8	Teams may only launch once.	D	N/A	The team will only attempt a single flight.	In Progress

6.3.1 Vehicle Subteam Requirements

Requirement ID	Requirement Summary	Satisfies Project Requirement:	Verification Type (s)	Verification Plan ID(s)	Verification Plan Summary	Status
S.V.1	The nose cone material will not interrupt RF signals.	V.2.17.2	D	VD.V.1	Attempt to communicate with components using RF signals through the nosecone.	Incomplete
S.V.2	The vehicle will reach an altitude of 4100' feet within a margin of error of ε = 200ft with no effect from airbrakes.	V.2.1	A	VA.V.2	The team will analyze predicted altitude through OpenRocket and RAS Aero simulations. Data from VDF will be reviewed after flight.	In Progress
S.V.3	The vehicle and its individual components will withstand at least 2 times the expected stresses without experiencing plastic deformation or destruction at any point during flight.	G.2.4.1	A	VA.V.3	The team will perform FEA simulations on all load bearing components and perform nondestructive testing on safety critical components.	Complete
S.V.4	The vehicle will remain in proper orientation for the duration of the flight.	V.2.14	A/T	VAT.V.4	OpenRocket or RasAero simulations will model flight path deviation, and the flight path will be monitored during VDF.	Incomplete
S.V.5	Critical flight components will always remain on the interior of the launch vehicle during flight.	V.2.15	A	VA.V.5	Team will use SolidWorks simulations to construct vehicle with vital components secured in the interior which will then be confirmed by inspection during construction.	Incomplete
S.V.6	The vehicle body and structure will be manufacturable with facilities accessible to the team or purchased as constructed.		I	VI.V.6	All design and purchases will be finalized with confirmation of proper facilities to manufacture said design.	Complete
S.V.7	The vehicle will be constructed within the specified mass, diameter, and height limits with the utmost precision as laid out in the proposal sheet to ensure it does not break apart under extreme stress.	V.2.5	I	VI.V.7	Once construction on the rocket is complete it will be compared to all design specifications and measured to ensure compliance	In Progress

S.V.8	The vehicle will have its Coefficient of Drag (Cd) determined.		A/D	VAD.V.8	Drop test of a 3D printed scaled airframe model with an accelerometer attached to it and a velocity sensor to measure terminal velocity OR the team will inspect results of the Subscale flight, and apply scaling	Complete
S.V.9	The launch vehicle will have a minimum thrust-to-weight ratio of 5:1.	V.2.16	I	VI.V.9	OpenRocket simulations will confirm the launch vehicle's thrust-to-weight ratio to be at least 5:1.	Complete
S.V.10	IFVR will be able to capture video for three hours continuously.	V.3.1	T	VT.V.9	The fully equipped IFVR system can be run for three continuous hours to verify memory and power capacity	In Progress
S.V.11	IFVR will capture forward, aft, and outward facing view for entire launch	V.3.1.1	D	VD.V.11	All three cameras will be tested on a mockup of the rocket to verify full video coverage of all viewpoints.	In Progress
S.V.12	Each subcomponent of the IFVR (one camera, computer, power source and memory card) will not weight more than 0.5lbm	V.3.1	T	VT.V.12	All fully assembled subcomponents of the IFVR will be weighed independently.	In Progress
S.V.13	All protrusions of the IFVR will contain housings that render the protrusions aerodynamically insignificant.	V.3.1	A	VA.V.13	Proper and extensive calculations will be conducted to endure that the housings are aerodynamically insignificant.	In Progress
S.V.14	The IFVR initiation system will be easily accessible and not require deconstruction.	V.3.1	D	VD.V.14	This design requirement can be demonstrated during construction and verified by successful initiation	In Progress
S.V.15	The IFVR will contain a notification system as to when the cameras are recording.	V.3.1	T	VT.V.15	The IFVR subsystems will be tested with their corresponding notification systems to ensure that footage is being captured.	In Progress
S.V.16	The IFVR systems will be secured so that the footage provided is clear and steady.	V.3.1	D	VD.V.16	The IFVR system will be tested in flight conditions and secured and shaking will be analyzed.	In Progress

6.3.2 Recovery Subteam Requirements

Requirement ID	Requirement Summary	Verification		Verification Plan Summary	Status
		Type(s)	Plan ID(s)		
S.A.1	Shock cord will be adequately long for each parachute.	D	VD.A.1	The subscale vehicle launch will demonstrate that the shock cord length is appropriate. For the main parachute this will be 60' and for the drogue parachute this will be 30'.	Complete
S.A.1.1	Parachutes will be tied to the shock cord off center to prevent two airframe sections from knocking together after separation.	I, D	VID.A.1.1	The team will inspect the shock cord and parachutes to ensure they are tied together off center. Subscale vehicle launch will demonstrate that the two sections separate without colliding.	In Progress

S.A.1.2	Shock cord will be z-folded with tape in appropriate increments to prevent tangling while being stored in the vehicle and to reduce shock during deployment.	I	VI.A.1.2	The team will inspect the shock cord stored within the vehicle and ensure it is folded to prevent tangling and reduce shock.	Incomplete
S.A.2	Parachutes will open consistently within an appropriate distance range or time frame to allow for full deployment after ejection.	T	VT.A.2 Parachute Drop Test	The parachute drop test will verify the drogue and main parachutes successfully deploy at the correct points of flight. For the drogue parachute, this is opening no more than 1s after being released, and for the main parachute, this is opening no more than 150’ after being released.	Complete
S.A.2.1	Parachutes will be completely protected with a Nomex blanket on the side of the ejection charges.	T	VT.A.2.1 Black Powder Ejection Test	The black powder ejection test will verify the parachutes are completely protected from the ejection charges.	Incomplete
S.A.2.2	Parachutes will be packed loosely to slide out easily during ejection.	I	VI.A.2.2	The team will inspect parachute packing before each flight.	Incomplete
S.A.2.3	The main parachute will utilize a slide ring to reduce shock loading during deployment.	I	VI.A.2.3	The team will inspect the preparation of the parachute prior to flight and verify the use of a slide ring before launch.	Discontinued
S.A.3	The black powder canisters will create appropriate separation between the airframe sections.	T	VT.A.3 Black Powder Ejection Test	The black powder ejection test will verify the black powder canisters are able to create 6’ of separation on the ground.	Incomplete
S.A.4	All avionics coupler components will be secured throughout the duration of flight. No components will be freely suspended within the compartment.	I	VI.A.4	Team will inspect avionics coupler and ensure components are secured.	Incomplete
S.A.4.1	Avionics coupler components will be organized. Wires and cords will be grouped together to prevent entanglement and damage.	I	VI.A.4.1	Team will inspect avionics coupler and ensure all components are organized and grouped together to prevent entanglement.	Incomplete
S.A.4.2	Avionics coupler components must be able to withstand all shock loads.	D	VD.A.4.2	The subscale launch will demonstrate that all avionics coupler components remain in place throughout the duration of flight.	Complete
S.A.5	Altimeters will record accurate readings and perform according functions throughout the duration of flight.	I	VI.A.5	The team will inspect that all altimeter related requirements have been fulfilled before PDR submission. Major design considerations will be frozen after this point.	Complete
S.A.5.1	Altimeters will continue to function across all likely flight temperatures.	T	VT.A.5.1 Altimeter Continuity and Battery Drain Test	The altimeter continuity and battery drain test will verify the altimeters can achieve continuity and provide readings in a temperature range from 35°F to 75°F. This range represents the likely temperature extremes for flight scenarios.	Complete

S.A.5.2	Both altimeters will achieve and maintain continuity consistently throughout flight.	T	VT.A.5.2 Altimeter Continuity and Battery Drain Test	This test will verify that the two altimeters are able to establish and maintain continuity in flight. This will be signaled by continuity beeps in sets of 3.	Complete
S.A.5.3	Altimeters will consistently ignite ejection charges at specific times throughout flight. The primary altimeter will ignite drogue and main charges before the redundant altimeter.	T	VT.A.5.3 Altimeter Ejection Vacuum Test	The altimeter ejection vacuum test will simulate the ascent and descent of the vehicle and verify each altimeter ignites at the correct time. For the primary altimeter, this means lighting the drogue charge at apogee and the main charge at an altitude of 900'. For the redundant altimeter, this means lighting the drogue charge 1s after apogee and the main charge at an altitude of 700'.	Incomplete
S.A.6	Altimeter batteries will function properly and ensure successful altimeter function for the duration of flight.	I	VI.A.6	The team will inspect that all battery related requirements have been fulfilled and ensure the coupler design meets flight expectations before PDR. Major design considerations will be frozen from this point forward.	Complete
S.A.6.1	Altimeter batteries will supply usable voltage for 1 hour longer than the given pad time of 2 hours.	T	VT.A.6.1 Altimeter Continuity and Battery Drain Test	The altimeter continuity and battery drain test will verify the altimeter batteries ability to power the altimeters for a 3-hour duration. Voltage readings will be taken every 30 minutes to ensure the altimeters would continuously function.	Complete
S.A.6.2	The avionics coupler will include battery shielding or casing to prevent battery damage in case of ballistic impact. This casing must not be compromised by any other coupler components.	I, D	VID.A.6.2	Team will inspect avionics coupler and ensure batteries are correctly located within casings. Subscale vehicle launch will demonstrate the integrity of the casings.	In Progress
S.A.6.3	Altimeter batteries will not fail to function at any likely launch temperature. They will function properly at a variety of temperature extremes.	T	VT.A.6.3 Altimeter Continuity and Battery Drain Test	The altimeter continuity and battery drain test will verify the altimeter batteries work at both 35°F and 75°F temperature extremes. These bounds represent the extremes for likely launch temperatures.	Complete
S.A.7	Key switches will prevent disarmament of the altimeter and ejection systems throughout flight. Only key switches will be able to engage or disengage these systems.	I, D	VID.A.7	The team will inspect the avionics coupler upon launch preparation and ensure the system is engaged. Subscale vehicle launch will demonstrate that no flight forces disengage the system.	In Progress

6.3.3 Payload Subteam Requirements

Requirement ID	Requirement Summary	Satisfies Project Requirements (If Applicable):	Verification		Verification Plan Summary	Status
			Type(s)	Plan ID(s)		

S.P.0	The overall mass of the payload systems shall not exceed 16 lbm.	N/A	T	VT.P.0.1	Measure the combined mass of the Payload experiment.	Incomplete
S.P.0.1	The overall mass of the lander subsystem shall not exceed 3 lbm.	N/A	T	VT.P.0.2	Measure the individual mass of the Lander subsystem.	Incomplete
S.P.0.2	The overall mass of the retention and deployment subsystem shall not exceed 5 lbm.	N/A	T	VT.P.0.3	Measure the individual mass of the R&D.	Incomplete
S.P.0.3	The overall mass of the ABCS shall not exceed 8 lbm.	N/A	T	VT.P.0.4	Measure the individual mass of the ABCS.	Incomplete
S.P.1.1	When the lander lands, the lander will remain in an operational state.	G.2.4.1 TD	A, D		Test function after landing and analyze possible failure modes.	Incomplete
S.P.1.2	Once deployed and free in the air, the lander should maintain a 6" clearance from all elements of the main vehicle.	G.2.18.1.1 P.2.18.1.4 P.4.3.1	D		During VDF, demonstrate that no vehicle components collide during descent and stay reasonably outside of this range.	Incomplete
S.P.1.3	The landing distance between any element of the main vehicle and lander should be greater than 10'.	G.2.18.1.1	D		Measure to ensure that the final landing distance is greater than the allotted distance.	Incomplete
S.P.1.4	Upon activation, the lander will fully deploy from the vehicle in under 5 s.	P.4.2 P.2.18.2.1 P.4.3.1	D, T	VT.P.1.1	Static test the functionality of the R&D system to deploy in under the allotted time frame.	Incomplete
S.P.1.5	The lander should be able to orient in terrain with a surface irregularity of 5"	P.4.2 P.4.3.2 P.4.3.3	A, D		Design to reliably function during the projected worst case geometric situation.	In Progress

	maximum crest height relative to trough and with maximum 6′ distance between local crests.					
S.P.1.6	The lander will be deployed under main parachute descent between an altitude of 700′ and 500′ AGL.	P.4.2 P.4.3.1	A, D		Verify through on-board altimeter data post-landing.	Incomplete
S.P.1.7	The lander must establish signal connectivity with the ground station capable of transmitting the image upon landing.	P.4.2 P.4.3.4	D		Static test of the transmission capability of the sender and receiver.	Incomplete
S.P.1.8	When taking a panoramic photograph, no component of the lander will obstruct the view of the cameras.	P.4.2 P.4.3.4	D		Test PICS to ensure a high quality image can be produced.	In Progress
S.P.1.9	Lander must be able to withstand 10 mph wind while grounded without being moved.	P.4.2 P.4.3.2 P.4.3.3	D, T	VT.P.1.9	Test the ability of the Lander to remain upright in an in-situ wind test.	Incomplete
S.P.1.10	The final panoramic photo produced and transmitted must be of a high enough quality to inspect the lander's surrounding area and horizon.	P.4.2 P.4.3.4	I, T	VT.P.1.7	Team verifies the result of image processing after the mission has completed.	Incomplete
S.P.1.11	Both the lander and its associated subsystems must have sufficient battery life to be in a launch-ready state for at least 2 hours.	V.2.7	A, T		All batteries must be drain tested to ensure proper functionality.	In Progress
S.P.1.12	The lander and its associated subsystems must be able to sustain a pre-flight state for a minimum of 18 hours.	V.2.7.1 TD	A, T		All batteries must be drain tested to ensure proper functionality.	In Progress

S.P.1.13	After the panoramic photo has been produced, the GCS must display the panoramic photo.	P.4.2 P.4.3.4	T	VT.P.1.7	Team verifies the result of image processing after the mission has completed.	Incomplete
S.P.1.14	Camera must be able to take a photo above the maximum dirt level within 10' radius of the landing site by 6".	P.4.2 P.4.3.4	T	VT.P.1.7	Place the lander in a similar environment as it will be expected to perform in and see it the photos it takes is clear of the trough.	Incomplete
S.P.1.15	Lander must be able to transfer image data to ground control station within 1 mi.	P.4.2 P.4.3.4	T		Move the lander 1 mile from the GCS and test whether it can transmit the image. Almost test if it gets blocked by the dirt (of the trough).	Incomplete
S.P.1.16	Lander must be able to land within 1 mi of ground control station.	P.4.3.1 P.4.3.4	A, D		Measure after VDF to ensure that the final landing distance is less than the allotted distance.	Incomplete
S.P.1.17	Lander must have some way to transmit its landing location.	N/A	I		The Lander will contain an operational GPS transmitter throughout the mission.	In Progress
S.P.1.18	Payload must be securely contained during flight until deployment.	P.2.18.2.1	D, T	VT.P.1.1 VT.P.1.2	The retention subsystem successfully retains the Lander during VDF.	Incomplete
S.P.1.19	The retention method must protect the lander from all flight loads such that it remains operational.	G.2.4.1 TD	A, D, T	VT.P.1.1	The retention subsystem successfully protects the Lander under simulated flight loads.	Incomplete
S.P.1.20	Lander system will have team name and launch day contact information clearly visible on the lander itself.	N.2.20	I		The team verifies the presence of both items on the body.	Incomplete
S.P.1.21	Payload system must be able to transition from pre-flight to flight ready without taking apart the rocket, through usage of the GCS.	V.2.7.2 TD	D, T	VT.P.1.1	The team will demonstrate that the system will be capable of modulating between these states.	Incomplete

S.P.1.22	Lander subsystem must be able to transition to a state of autonomous orientation, through usage of the GCS.	P.4.3.3.1	D		The team will demonstrate that the system will be capable of modulating between these states.	Incomplete
S.P.2.1	The ABCS shall never put the vehicle fins into a stall condition under any failure mode.	B.2.14.1 TD	A,D		The ABCS will be verified utilizing aerodynamic simulation methods. The ABCS will be shown not to induce a stall condition during VDF.	Incomplete
S.P.2.2	The ABCS shall never reduce the stability margin of the vehicle below 2.1 cal under any failure mode.	B.2.14.1 TD	A, D		The team will verify the stability modification of the vehicle through OpenRocket.	In Progress
S.P.2.3	When used, the ABCS will bring the vehicle altitude to within 100 ft of the target apogee.	B.2.1.2 TD	D		The vehicle and ABCS altimeters will produce final apogee data for review.	Incomplete
S.P.2.4	If the ABCS suffers a mechanical failure, the vehicle will not deploy the ABCS.	B.2.14.1 TD G.2.18.1.1	D		The ABCS will be shown to only operate in a completed, non-broken state.	Incomplete
S.P.2.5	The ABCS will only operate after the vehicle burn has completed.	B.2.1.2 TD B.2.14.1 TD	T		The ABCS will be tested to respond to the simulated flight loads associated with a successful burn.	Incomplete
S.P.2.6	The ABCS must be able to fully activate and deactivate control surfaces in under 1 s seconds.	B.2.14.1 TD	T	VT.P.2.4	The ABCS Mechanical Subsystem will be tested to ensure its complete operation time is under the allotted time.	Incomplete
S.P.2.7	The battery powering the ABCS must be able to withstand idle operation for a minimum of 2 hours.	V.2.7	T	VT.P.2.2	Battery drain tests will be conducted on the ABCS.	Incomplete

S.P.2.8	The data output from the inertial sensor must be useful for the MCU. If the output from the MCU is raw data, additional electronics must be designed to convert raw data into a usable format for the MCU.	N/A	D, T	VT.P.2.3	The inertial sensor can successfully communicate useful data to the MCU.	Incomplete

6.4 Budgeting

6.4.1 Line-Item Budget

Item	Quantity	Unit Cost	Extended Cost	Subteam	Manufacturer
JST SM-2 Connector Cables	1	$6.98	$6.98	Lander	Amazon
10k Potentiometer kit	1	$6.49	$6.49	Lander	Amazon
Arducam M12 - 140 deg	2	$16.99	$33.98	Lander	Amazon
25 Tooth Aluminum Control Horn	4	$4.99	$19.96	Lander	goBilda
Nichrome 80-250' - 36 Gauge Resistance Wire	1	$6.49	$6.49	Lander	Amazon
SEACHOICE Braided Utility Line 1/8" x 100' 40151 White	1	$7.44	$7.44	Lander	Amazon
MR12-0250-1 Sliding Rail (Custom Request)	3	$51.22	$153.66	Lander	PBCLinear
Female Threaded Hex Standoff Aluminum, 1/4" Hex, 1-1/2" Long, 8-32 Thread	4	$0.85	$3.40	Airbrakes	McMaster
Fasteners (packs of 50)	1	$9.08	$9.08	Airbrakes	McMaster
Teensy 4.0 Board	1	$29.10	$29.10	Airbrakes	Amazon
BMP280 Altimeter	1	$10.65	$21.30	Airbrakes	Adafruit
BNO085 IMU Fusion Sensor	1	$22.09	$32.74	Airbrakes	Adafruit
PCB Prototyping Kit	1	$9.99	$9.99	Airbrakes	Amazon
11.1V Lipo Battery	1	$42.99	$42.99	Airbrakes	Amazon
25’ Ethernet Chord	1	$11.99	$11.99	All	Amazon
Aluminum Extruder Feeder	1	$11.99	$11.99	All	Amazon
BNO055 IMU Fusion Sensor	1		$-	Airbrakes	
200A Watt Meter	1	$13.59	$14.54	All	Amazon
Gold MH Build Series PLA Filament - 1.75mm (1kg)	1	$21.39	$21.39	All	MatterHackers
Black MH Build Series PLA Filament - 1.75mm (1kg)	1	$21.39	$21.39	All	MatterHackers
Zinc-Plated Alloy Steel Socket Head Screw 1/4"-20 Thread Size, 1" Long, 25 pack	1	$6.37	$6.37	Airbrakes	McMaster
Zinc-Plated Alloy Steel Socket Head Screw 8-32 Thread Size, 1/2" Long 50 pack	1	$5.97	$5.97	Airbrakes	McMaster
Black-Oxide Alloy Steel Socket Head Screw 1/4"-20 Thread Size, 3/4" Long, 50 pack	1	$7.86	$7.86	Airbrakes	McMaster
Female Threaded Hex Standoff Aluminum, 1/2" Hex, 2-1/2" Long, 1/4"-20 Thread	6	$4.97	$29.82	Airbrakes	McMaster
TB6600 4A 9-42V Stepper Motor Driver	1	$11.52	$12.33	Airbrakes	Amazon
Crimping Tool Kit	1	$40.99	$40.99	All	Amazon
Coupler, Airframe, and Bulkheads	1	140	$178.87	Construction	MadCow Rocketry
Motor tube, retainer, and rail buttons	1	28.9	$43.83	Construction	MadCow Rocketry

Centering plates and Switch Band	1	16	$28.99	Construction	MadCow Rocketry
2 Masterhacker 1kg, 1.75mm PLA	1	39.98	$42.78	Payload-Airbrakes	Masterhackers
10x Adhesive Mount 1/4" 20 Nut	3	$7.26	$21.78	Construction	McMaster
Velco Cinch Straps	1	$12.97	$12.97	Payload	Amazon
Metric Tap Set	1	$12.37	$12.37	Payload	Amazon
M2 Screw Set	1	$9.99	$9.99	Payload	Amazon
Motor Tube	1	$9.00	$23.93	Construction	MadCow
Motor Retainer	1	$12.95	$12.95	Construction	MadCow
1010 Rail button	1	$6.95	$6.95	Construction	MadCow
G10 Fiber Glass Sheet 3/32"	1	$14.00	$26.85	Construction	Wildman
3" G12 Switch Band	1	$4.00	$16.99	Construction	MadCow
3"x38mm Centering plate	2	$6.00	$12.00	Construction	MadCow
3"x9" Coupler	1	$22.00	$22.00	Construction	MadCow
3" Stacked Bulkhead	2	$10.00	$20.00	Avionics	MadCow
1/4"-20 Eyebolt	1	$3.21	$3.21	Avionics	McMaster
H148R Reload	1	$34.99	$34.99	Construction	Wildman
3"x5' Fiberglass Tube	1	$100.00	$138.87	Construction	MadCow
48in. Rocketman High Performance CD 2.2 Parachute	1	$115.00	$115.00	Avionics and Recovery	Rocketman Parachutes
USB Data Transfer Kit	1	$24.95	$34.25	Avionics and Recovery	PerfectFlite
144in. Rocketman High Performance CD 2.2 Parachute	1	$385.00	$385.00	Avionics and Recovery	Rocketman Parachutes
StratoLoggerCF Altimeter	1	$54.95	$64.25	Avionics and Recovery	PerfectFlite
3" x 24" PVC	1	$5.60	$5.60	Systems	Home Depot
15 5/16-18 Locknut	2	$3.74	$7.48	Systems	Home Depot
3/8 x 250 ft Rope	1	$27.19	$27.19	Systems	Home Depot

3" PVC Flange	1	$16.32	$16.32	Systems	McMaster - Carr
3" PVC 90 deg	1	$11.21	$11.21	Systems	McMaster - Carr
1" PVC Flange	1	$7.01	$7.01	Systems	McMaster - Carr
45mm Bearing	1	$20.99	$20.99	Systems	McMaster - Carr
50 5/16 Spring Washer	1	$13.38	$13.38	Systems	McMaster - Carr
5/16-18 1' Allthread	7	$3.57	$24.99	Systems	McMaster - Carr
10 5/16-18 3/8" Hex Bolt	1	$6.82	$6.82	Systems	McMaster - Carr
1.25 Square Steel Tube	1	$7.99	$7.99	Systems	McMaster - Carr
10 5/16-18 2.75" Hex Bolt	1	$4.11	$4.11	Systems	McMaster - Carr
1"x1"x1' Aluminum Stock	1	$4.92	$4.92	Systems	McMaster - Carr
10 1/2-13 2.5" Hex Bolt	1	$7.49	$7.49	Systems	McMaster - Carr
50 1/2-13 Locknut	1	$10.07	$10.07	Systems	McMaster - Carr
3"x5' PVC	1	$7.92	$7.92	Systems	McMaster - Carr
.25" Thrust Bearing	2	$4.40	$8.80	Systems	Servocity
.25" Set Screw Hub	1	$4.99	$4.99	Systems	Servocity
.25" Set Screw Collar	1	$1.69	$1.69	Systems	Servocity
.25" x 2" D Shaft	1	$1.69	$1.69	Systems	Servocity
2 in Plastic Wheel	1	$3.99	$3.99	Systems	Servocity
GoBilda Servo	1	$27.99	$27.99	Systems	Servocity
CUI AMT102 Encoder	1	$23.63	$23.63	Systems	Digikey

Table 6.1 Line Item Budget

6.4.2 Funding Plan

The team needs a total budget of $12,300 for the competition. More specifically, the team needs $7,750 for the technical aspects of the project, but only has $1,483 of this goal remaining to be raised. This year, the team has made several requests from the Department Heads of the Engineering Departments at Purdue. Due to the current limitations that the university is under and the economic hardships currently being experienced, there has not been as much fundraising within the university. The team will continue to apply for grants and will be beginning a crowdfunding campaign to complete the needed fundraising. In the past, the team has run a majorly successful crowdfunding campaign that raised more than the remaining 55% of the budget, so this is a very viable option, should the funding from the department heads not be feasible.

6.5 Educational Outreach

6.5.1 Purdue Space Day

With Purdue's unprecedented on-campus fall 2020 semester, many safety measures were put in place to ensure the health of students and faculty. These measures greatly limited the team's ability to volunteer to the greater Lafayette community. As a result, the team focused on Purdue Space Day. This annual event hosted by the School of Aeronautics and Astronautics was hosted virtually this year and was broadcast live to over 4,000 participants across the US and the world. The team assisted with planning and running this event in conjunction with an interdisciplinary team of volunteers equally passionate about space exploration.

6.5.2 Spring 2021 Plans

As both Purdue University and the city of West Lafayette begin to refine their student and citizen health mandates, the Student Launch team is currently searching for ways to bolster our impact on the community. Some proposed ideas for events include more virtual seminars and interactive livestreams, socially distanced demonstrations at nearby K-12 schools, and on-campus activities.

6.6 Project GANTT Chart

The team is pursuing an accelerated timeline for the project due to Purdue University's extended winter break. The team was able to complete a large portion of the manufacturing prior to leaving for break, which should allow for the vehicle to be completed in time. The detailed work breakdown can be seen in the following Gantt Chart.

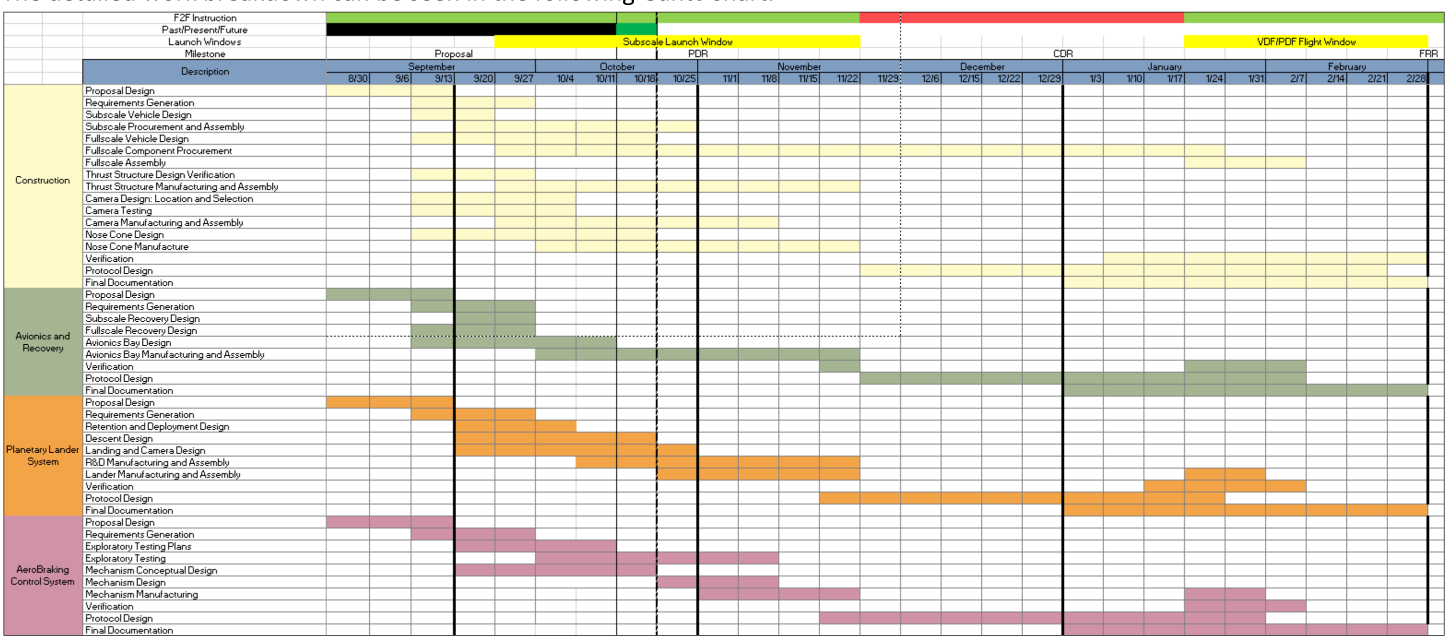


Figure 6.3: Project Gantt Chart Until FRR (03/08/2021)