Project Casper

Critical Design Review

(CDR)

Purdue University 2020

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West Lafayette, IN 47906
Purdue Space Program

Table of Contents

List of Figures

List of Tables

Acronyms and Abbreviations

Acronym / Abbreviation	Definition	Acronym / Abbreviation	Definition
PSP	Purdue Space Program	LED	Light-Emitting Diode
SL	Student Launch	LiDAR	Light Detection and Ranging
ABS	Acrylonitrile Butadiene Styrene	LRR	Launch Readiness Review
AGL	Above Ground Level	MAVLink	Micro Air Vehicle Link
AP/APCP	Ammonium Perchlorate (Composite Propellant)	MCC	Mission Control Computer
BJT	Bipolar Junction Transistor	MCS	Mission Control Software
BOM	Bill of Materials	MCU	Mission Control Unit
CAD	Computer-Aided Design	MMS	Mission Management System
CDR	Critical Design Review	MPH	Miles Per Hour
CF	Carbon Fiber	NAR	National Association of Rocketry
CFD	Computational Fluid Dynamics	NASA	National Aeronautics and Space Administration
CFRP	Carbon Fiber Reinforced Polymer	N/A	Not Applicable
CG	Center of Gravity	OT	Operational Testing
CM	Center of Mass	PDB	Power Distribution Board
CP	Center of Pressure	PDR	Preliminary Design Review

CPA	Control Panel Assembly	PDU	Power Distribution Unit
CTI	Cesaroni Technology Incorporated	PETG	Polyethylene Terephthalate Glycol
DC	Direct Current	PIC	Pilot-In-Command
DHA	Display Head Assembly	PLA	Polylactic Acid
DIU	Digital Imaging Unit	PPE	Personal Protective Equipment
DT	Developmental Testing	RC	Radio Controlled
DTED	Digital Terrain Elevation Data	RF	Radio Frequency
ESC	Electronic Speed Controller	RGB	Red/Green/Blue
FAA	Federal Aviation Administration	RSO	Range Safety Officer
FCC	Flight Control Computer	RTB	Return To Base
FCS	Flight Control System	R&D	Retention and Deployment
FEA	Finite Element Analysis	R&VP	Requirements and Verification Plan
FMEA	Failure Modes and Effects Analysis	SITL	Software In The Loop
FOS	Factor of Safety	SLI	Student Launch Initiative
FRP	Fiberglass-Reinforced Polymers	SME	Subject Matter Expert
FRR	Flight Readiness Review	SOW	Statement Of Work
GCS	Ground Control System	SRM	Solid Rocket Motor
GPIO	General Purpose Input and Output	STEM	Science, Technology, Engineering, and Mathematics
GPS	Global Positioning System	SS	Stainless Steel
GUI	Graphical User Interface	TAP	Technical Advisory Panel
HGL	Handheld GPS Locator	TRA	Tripoli Rocket Association
HTPB	Hydroxyl-Terminated Polybutadiene	UAV	Unmanned Aerial Vehicle
IMPS	Ice Mining and Procurement System	UAS	Unmanned Aerial System
IMU	Inertial Measurement	USB	Universal Serial Bus
IRI	Indiana Rocketry, Incorporated	VSD	Vehicle Status Delay
I_{SP}	Specific Impulse	V&V	Verification and Validation
I/O	Input/Output	WBS	Work Breakdown Structure
LCD	Liquid Crystal Display		

1. CDR Report Summary

The information in the following sections summarizes information about the 2020 PSP-SL team, its mentor, and the launch vehicle it will be using in this year's NASA (National Aeronautics and Space Administration) Student Launch competition.

1.1. Team Summary

Team Name	PSP-SL (Purdue Space Program - Student Launch)
Mailing Address	2604 Bristlecone Dr., West Lafayette, IN 47906
2020 Team Mentor	Victor Barlow
2020 Mentor Contact Information	vmbarlow@purdue.edu | (765) 414-2848 (Cell)
2020 Mentor TRA / NAR Certifications	NAR 88988, TRA 6839 TAP, Level 3 Certified

Table 1.1: PSP-SL Team Summary

1.2. PSP-SL 2020 Executive Board

Position	Name	Email
Project Manager	Luke Perrin	lperrin@purdue.edu
Assistant Project Manager	Michael Repella	mrepella@purdue.edu
Safety Team Lead	Noah Stover	nstover@purdue.edu
Payload Co-Team Lead	Josh Binion	binionj@purdue.edu
Payload Co-Team Lead	Hicham Belhseine	hbelhsei@purdue.edu
Avionics & Recovery Team Lead	Katelin Zichittella	kzichitt@purdue.edu
Business Team Lead	Natalie Keefer	nkeefer@purdue.edu
Social & Outreach Team Lead	Skyler Harlow	sharlow@purdue.edu
Construction Team Lead	Lauren Smith	smit3204@purdue.edu
Construction Team Mentor	Zach Carroll	carrollz@purdue.edu

Table 1.2: PSP-SL Executive Board

1.3. Launch Vehicle Summary

The following information provides a summarized version of the launch vehicle to be constructed by the PSP-SL team for this year's competition.

1.3.1. Launch Vehicle Size and Mass

The 2020 PSP-SL launch vehicle is a 6” inner diameter based launch vehicle. The launch vehicle is 125” long (tip to tail), including a 36” long nose cone (5:1 Von Karman), a 48” upper airframe, a 14” avionics bay (with a 2” switch band), 38” (40”) lower airframe, and a 1” long retainer. The predicted mass of the launch vehicle is 53.4lbm. The launch vehicle consists of the nose cone, two body tubes, three trapezoidal fins and a switch band, all of which are composed of G12 filament-wound composite fiberglass. The internal components contain a UAV (Unmanned Aerial Vehicle) payload system, two parachutes, a recovery system, the avionics bay, and a camera payload.

PSP-SL Launch Vehicle Dimensions	
Overall Launch Vehicle Diameter	6.17”
Overall Launch Vehicle Length	125”
Number of Sections (Outer Dia.)	3
Gross Lift Off Weight	53.4 lbm
Rail Size	144” x 15-15

Table 1.3: General Launch Vehicle Dimensions

1.3.2. Final Motor Choice

The 2020 PSP-SL team will be using a CTI 4 grain L1115-0 solid rocket motor. More discussion and analysis can be found in Section 5.3.

PSP-SL Final Motor Choice			
Manufacturer	Cesaroni Technology Inc. (CTI)		
Model	L1115-0		
Fuel	Ammonium Perchlorate, NH4ClO4 (AP 200)		
Oxidizer	Atomized Aluminum, Al		
Thrust Profile	Regressive		
Overall Weight	4404g	**Propellant Weight**	2394g

Table 1.4: Final Motor Choice

1.3.3. Recovery System

After thorough analysis and comparison, the 2020 PSP-SL team has decided to use the following altimeters and recovery hardware. These will be discussed more in Section 4.

Avionics & Recovery	
Main Parachute	Skyangle Cert-3 XXL (120”)
Main Deployment	800’ AGL (Above Ground Level) (700’ AGL redundant)
Drogue Parachute	Fruity Chutes Classic Elliptical (24”)
Drogue Deployment	Apogee (+1s redundant)
Primary Altimeter	Altus Metrum Telemetrum
Redundant Altimeter	Missile Works RRC3+ Sport

Table 1.5: Avionics and Recovery Overview

1.4. Payload Summary

Figure 1.2: Complete Payload System

“The Friendly Ghost” - Autonomous Unmanned Aerial System (UAS)

The chosen payload is comprised of an unmanned aerial vehicle (UAV) and a portable ground control station. The UAV is capable of fully autonomous or remotely piloted flight and will be equipped with a lunar ice mining and storage system. The ground control station (GCS) will control and monitor the UAV and launch vehicle telemetry. The UAV will be mechanically stored and retained in the launch vehicle during flight and recovery, and upon landing will autonomously move to a recovery zone to extract and store a lunar ice sample.

2. Changes Made Since Preliminary Design Review

2.1. Changes Made To Vehicle Criteria

Construction / Vehicle

Since the Preliminary Design Review (PDR), the team has made a few minor adjustments to the launch vehicle. One addition from PDR to CDR is two holes in the avionics bay for the camera payload that the team plans on implementing. On top of this, the team has reassessed the size of the ejection charges. This change is primarily due to poor assumptions and analysis of pressurization in both the upper and lower airframe.

Avionics and Recovery

The designs of the altimeter sled, battery compartment, battery guard, and Telemetrum antenna holder have completely changed since PDR. The reasons for these changes include the fleshing out of the design of the camera system in the avionics bay as well as concerns that the overall weight of the 3D printed parts was too high. With the original separated configuration, both the altimeter sled and the battery compartment would not be able to fit in the avionics bay on one side of the camera sled. However, placing them on either side would necessitate crossing battery wires over the camera sled. Also, many cutouts had to be made in these two separate, large parts in order to get them to be an acceptable weight, and eventually it became clear that this design was no longer viable. While the battery guard will remain approximately the same, the avionics sled and the battery compartment will now be combined into a single part. On one side will be the Telemetrum and the 9V battery (for the RRC3+ Sport), and on the other side will be the RRC3+ Sport and the 3.7V LiPo battery (for the Telemetrum). Placing each battery directly underneath (on the opposite side of the sled) its corresponding altimeter allows for the battery wires to need to cross as short a distance as possible. Also, without any cutouts that risk structural failure in the parts, these parts weigh a combined 186.65g, thirty grams less than the total of the equivalent parts in the old design. The only con to this configuration is how there is no longer a place to put the antenna holder for the Telemetrum altimeter (originally on the separate battery compartment). However, this drawback can be accepted because the Telemetrum is known to be perfectly capable of functioning without anything holding its antenna in place inside the avionics bay.

Additionally, transmitter #4 (the RRC3+ Sport altimeter) was removed from the flysheet as well as the RF Transmission Frequencies table below. This was due to the team having decided to use the Telemetrum altimeter as the sole transmitter of avionics telemetry even though the RRC3+ Sport altimeter has the ability to transmit with an additional component, allowing for overall simplicity.

2.2. Changes Made To Payload Criteria

Since PDR, several modifications to the design of the payload system have been made.

The UAV’s airframe has undergone a number of small geometric changes to better accommodate the tight envelope inside the payload bay and to save as much weight as possible. Such changes impacted the UAV’s internal electronics plates, its landing structures, and the construction of its armatures. Material was removed where possible to reduce weight. The addition of an aerodynamic cover is also new since PDR, added to protect the expensive electronic components housed in the UAV. The lunar ice mining and procurement system has had minor modifications to its integration with the UAV’s airframe as well as its electronic drive circuit.

The electronics and control systems on-board the UAV have been further developed as well. The power distribution system now incorporates a limit switch for powering on the UAV automatically after it has been deployed. Further power systems analysis has been conducted to better understand the available flight time for the UAV. Navigation and landing algorithms have been developed for controlling the flight of the UAV throughout the course of the mission. The design of the ground control station has also matured significantly since PDR. Preliminary user interfaces have been designed alongside buttons, switches and indicators that give payload personnel greater control over the mission. The electronic backbone of this system, including a printed-circuit-board has also been designed.

The retention and deployment system design has also matured significantly since PDR. New bearings for controlling UAV orientation were added for finer control. The sled on which the UAV sits in the bay has undergone significant maturation with the addition of a servo-actuated rack and pinion system for completely constraining UAV motion. The electronics associated with this system have matured as well. New power electronics for controlling the stepper motor have been designed and a sophisticated software package has been developed for controlling the system from the GCS.

2.3. Changes Made to Project Plan

Since PDR, the team has gone through and added specific sections to talk about each of the social outreach events, add requirements and verification plans, refined the overall timeline for the year, and added a team wide testing section where both avionics and payload testing can be done.

3. Launch Vehicle Summary

3.1. Design and Verification of Launch Vehicle

3.1.1. Mission Statement and Mission Success Criteria

It is the goal of the PSP-SL team to design, build, test, and fly a launch vehicle that carries a functional payload to a predetermined altitude of 4325'. This payload will be a ground-deployable, autonomous lunar-ice sampling drone. Upon successful flight and recovery of the launch vehicle, this payload will be safely ejected from the launch vehicle and will move a set distance away from its landing point to collect a soil sample. A successful mission will satisfy the following criteria:

- The vehicle flight is stable off the launch rail and throughout ascent
- The vehicle reaches the desired altitude of about 4325' AGL
- All recovery gear is successfully deployed at the appropriate altitudes
- The vehicle lands safely within the recovery zone boundaries
- The vehicle can be flown again without need for repairs or alterations
- The payload active retention system fully separates the fairing and rover from the payload bay after the vehicle returns to ground level
- The rover successfully reaches a distance of at least 10' away from any part of the vehicle and collects a soil sample

3.1.2. Chosen PDR Design

The 2020 PSP-SL launch vehicle chosen for PDR had a 6" inner diameter body tube. The team chose a 5:1 Von Karman nose cone design. The launch vehicle was 125" long (tip to tail), including a 2" long switch band, a 48" long upper airframe, a 38" long lower airframe, a 1" long retainer, and a 36" long nose cone. The fins were trapezoidal with a max height of 6.25" from the exterior of the airframe, a tip chord of 4", a root chord of 12", and a fin sweep angle of 50.5 degrees. The overall predicted mass of the launch vehicle was 53.4lbm. About 16lbm of this mass came from the nose cone, body tubes, three fins, and the switch band, which were all composed of G12 filament-wound composite fiberglass. The rest of the weight came from internal components, such as a UAV payload system, parachutes, a recovery system, the avionics bay, and a camera payload. Individually, the nose cone weighed 2.8lbm, the upper airframe weighed 5.61lbm, the lower airframe weighed 9.96lbm, and the fins weighed 2.85lbm. Meanwhile, the payload bay and components weighed 11.7lbm, the main parachute weighed 4lbm, the drogue parachute weighed 0.375lbm, and the avionics bay and components weighed 6.34lbm.

3.1.3. Chosen CDR Design and Justification

The launch vehicle design chosen for CDR consists of three main parts: the nose cone & upper airframe, avionics bay, and lower airframe.

The total length of the launch vehicle is 125.0”, with a nose cone length of 36.0”, an upper airframe length of 48.00”, an avionics switch band connecting the upper and lower airframe that is 2.00” long, and a lower airframe which is 38.00” long. The overall length of the launch vehicle offers a good distribution of mass - the average Center of Mass (CM) is located at 76.34” and moves the fin set far enough aft to have good stability, as having the fin set far aft moves the Center of Pressure (CP) far aft.

The launch vehicle has a maximum outer diameter of 6.15” (upper and lower airframe body tube sections), and a maximum inner diameter of 6.00” (upper and lower airframe body tube sections). This diameter was chosen to maximize the available interior tube cross sectional area (and thus volume) for the payload and avionics section.

The fin set of the launch vehicle consists of three fins, equally spaced at 120 degrees from each other. The fin set of three fins that has been chosen creates less drag than a set of four fins would, and also helps the launch vehicle gain more altitude, as three fins weighs less than four. Fewer fins also help the motor accelerate the launch vehicle more easily, because of the reduced mass and drag compared to four fins. All three fins are estimated to weigh 48.1oz including their 1.5” high fin tabs and 1.5” radius root fillets made from epoxy.

The fin shape that has been chosen is a trapezoidal “almost-clipped” delta fin shape. The chosen fin shape has a root chord of 12.00”, a tip chord of 4.0”, a height of 6.25”, a sweep length of 7.58” and a sweep angle of 50.5 degrees. The thickness of the fin is approximately 0.1875”. The fin tab height is 1.5”. The fin also features a beveled leading edge for reduced dynamic drag and for better streamlining. The shape has been designed to optimize stability and manufacturability while keeping dynamic and induced drag reduction in mind.

The chosen motor is the Cesaroni Technology Inc. L1115 Classic 4 Grain Solid Rocket Motor (SRM). This will be discussed more in Section 5.3.

3.1.4. Dimensional Drawings

3.1.4.1. Assembled Launch Vehicle

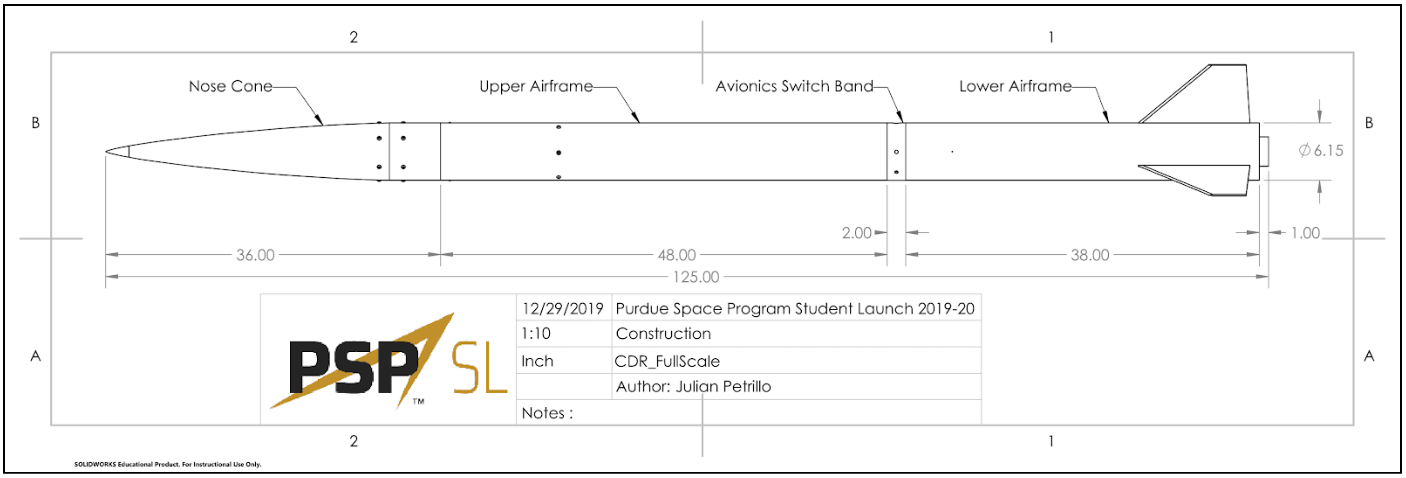


Figure 3.1: Full-Scale Launch Vehicle Dimensional Drawing

The assembled launch vehicle, as shown above, includes all metal hardware and fiberglass structural components that make up the body of the rocket and its subsystems, but does not include the recovery gear such as parachutes, fireproofing, tethers, or linkage. The assembled launch vehicle consists of three main parts: the nose cone, upper airframe and lower airframe. The launch vehicle consists of an outer diameter of 6.15", and an inner diameter of 6.00".

3.1.4.2. Lower Airframe Subsystem and Components

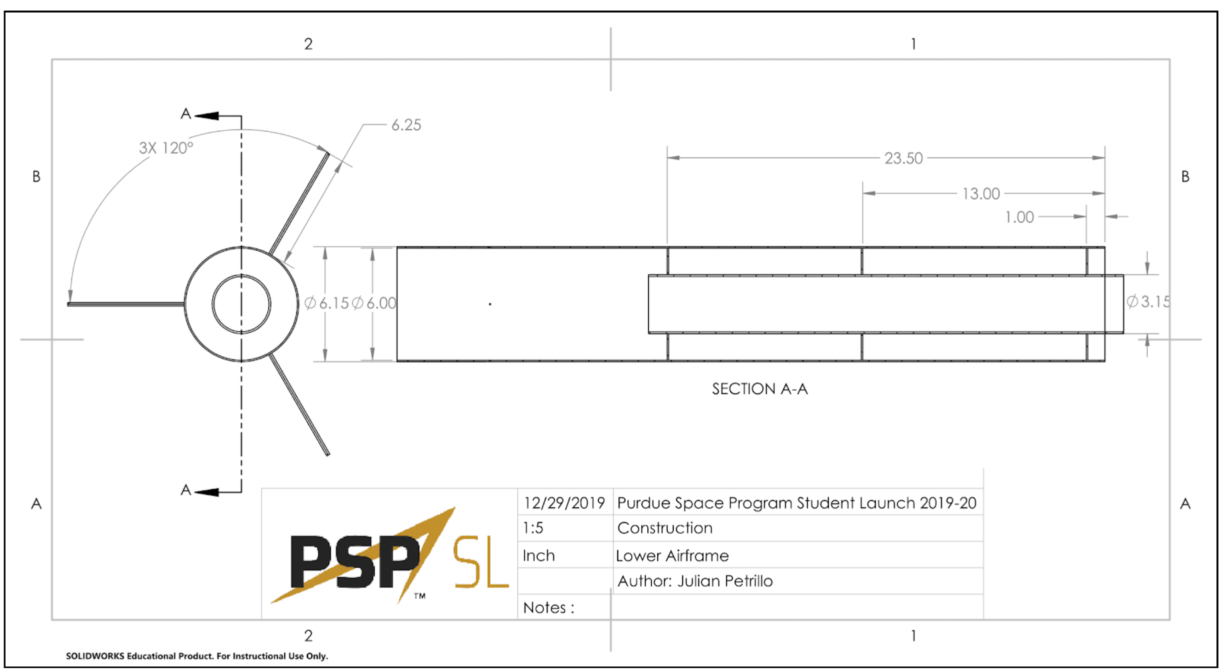


Figure 3.2: Full-Scale Lower Airframe Dimensional Drawing

The lower airframe section will have an outer diameter of 6.15" and an inner diameter of 6.0". The lower airframe has a length of 38" and will contain three slots to align the three fins of the launch vehicle. The lower airframe is tasked with housing the motor tube and other necessary components of the motor.

3.1.4.3. Avionics Bay Subsystem and Components

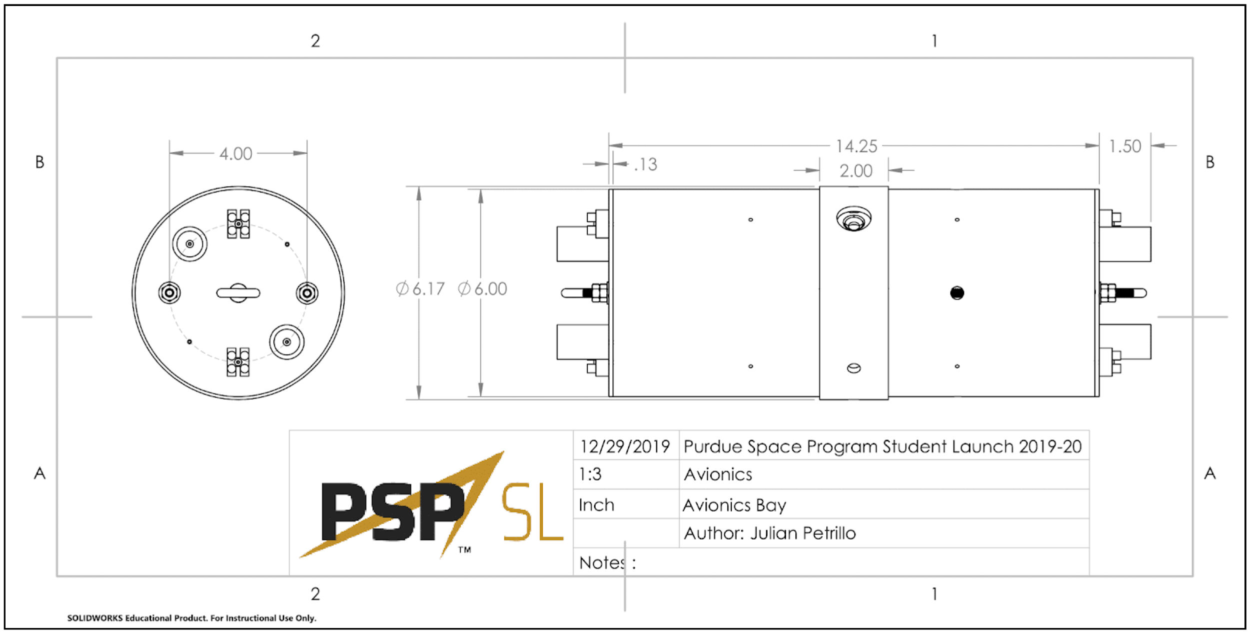


Figure 3.3: Full-Scale Avionics Bay Dimensional Drawing

The avionics bay has an outer diameter of 6", coupler length of 14", and total length of 17.25" when including the black powder canisters. It also has a switch band around the middle with a width of 2". The primary purpose of the avionics bay is to house the primary and redundant altimeter/ejection systems, provide an attachment point for the drogue and main parachutes, and house the secondary payload (camera) system.

3.1.4.4. Upper Airframe Subsystem and Components

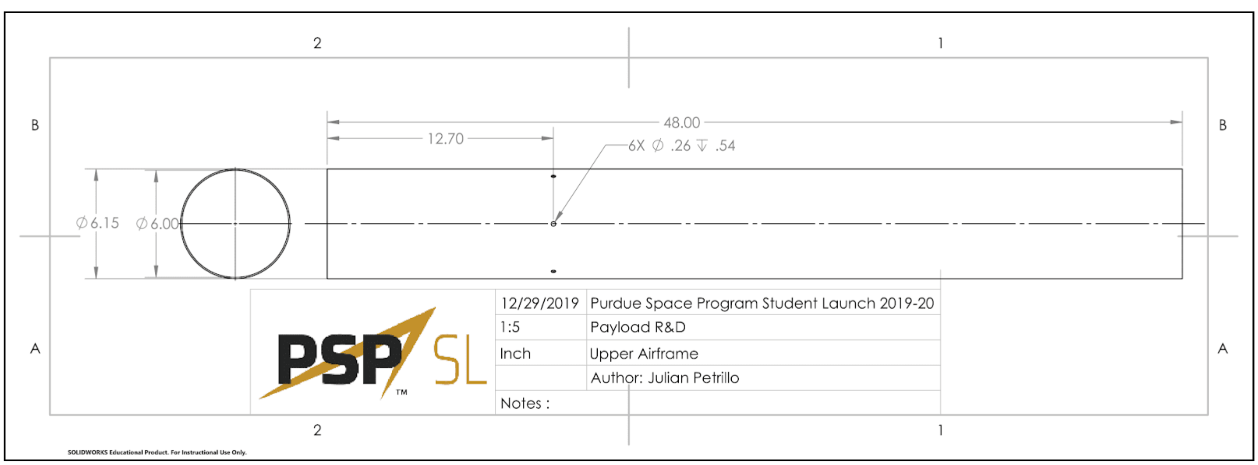


Figure 3.4: Full-Scale Upper Airframe Dimensional Drawing

The upper airframe of the launch vehicle is 48” long. The outer diameter of the upper airframe is 6.15” and the inner diameter is 6.0”. The upper airframe is tasked with containing the main recovery gear of the launch vehicle, as well as interfacing with the payload and avionics bays.

3.1.4.5. Nose Cone and Payload Subsystem and Components

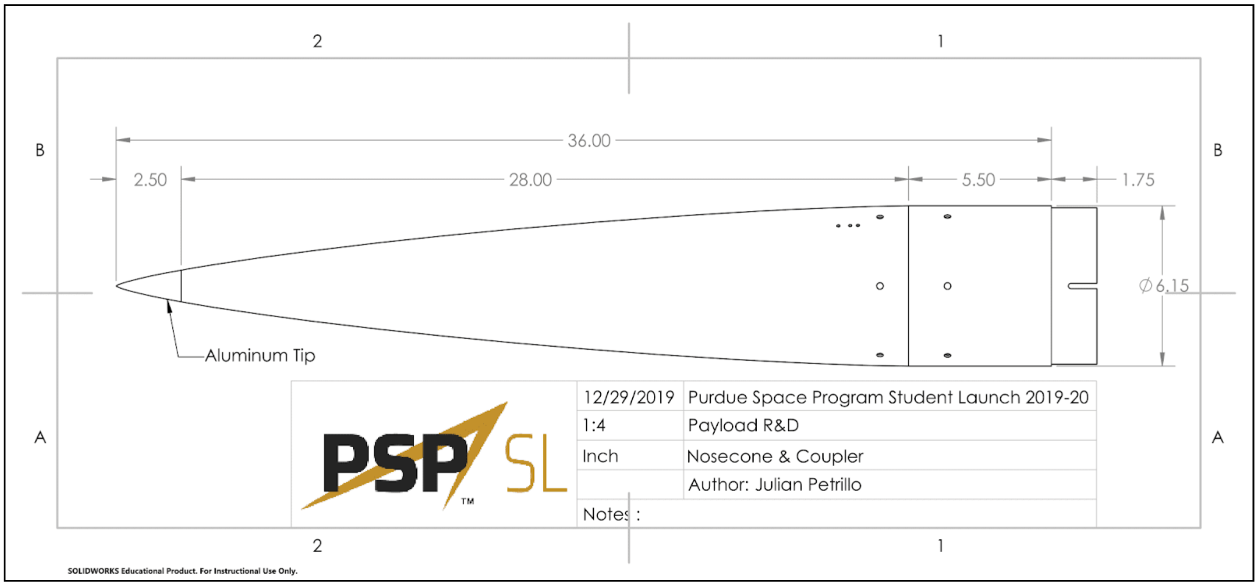


Figure 3.5: Nose Cone and Payload Coupler Dimensional Drawing

The nosecone of the launch vehicle is 36”, with an outer diameter of 6.15” and inner diameter of 6.0”. The nosceone is designed to reduce drag, features an increased interior volume for future payloads or avionics, and interfaces with the upper payload coupler.

3.1.5. Manufacturing Readiness

Most of the baseline parts for this year’s vehicle have been commercially bought through various high-power rocketry vendors. The fiberglass body tubes, and motor tubes were bought from Wildman Rocketry. The couplers and centering rings were also bought from Wildman Rocketry. The material that most purchased parts are made of is G10 fiberglass. Fiberglass is strong enough to last through the aerodynamic forces of flight but cheap enough for the team to buy. The parts purchased that did not use G10 fiberglass are the motor retainer, various eye bolts, screws, rail button, and threaded rods used in the launch vehicle. These materials are all made of commercially purchased aluminum. The team has conducted various tests on these parts as well and made sure that these parts can last through the rigors of flight and recovery. The team is also able to verify, based on experience, that the G10 fiberglass will be more than sufficient for the flight profile and mission that our team is looking at.

Currently, the team has personally manufactured the main parachute bulkplate and the subscale trapezoidal fins. The team constructed the fins via a water jet located on Purdue’s campus. The main parachute bulkplate was manufactured using a HAAS 3+2 axis CNC on campus as well. The team has also used a drill press to drill in all the vent holes needed for the body tubes and the pressure bleed hole needed for avionics, the holes for the shear pins, and the holes for the payload team's bulkplate. The team also cut the fiberglass body tube down to match the dimensions needed from the OpenRocket design, which was done with a circular saw. For the team's subscale launch vehicle, the fiberglass bulkheads were cut by hand using a bandsaw. In order to prep for each vehicle's construction, each of the launch vehicle's attachment joints - joints at which one part is epoxied onto another part - are sanded down, either by hand or by electric sanding machine. This is to make sure the epoxy has firmer hold on the attachment joint.

Overall, the team is confident that it has the proper resources and abilities to properly construct each section of the launch vehicle. The team currently has all components purchased and received for our 2020 launch vehicle.

3.1.6. Design Integrity

3.1.6.1. Launch Vehicle Fins

A fin set has been chosen that consists of three fins made of G10 fiberglass. The set of three fins weighs less and creates less drag than a set of four fins would, and thus helps the launch vehicle gain more altitude. All three fins together are estimated to weigh a total of 48.1oz including their fin tabs and 1.5" radius root fillets made from epoxy.

3.1.6.1.1. Design and Shape (Material)

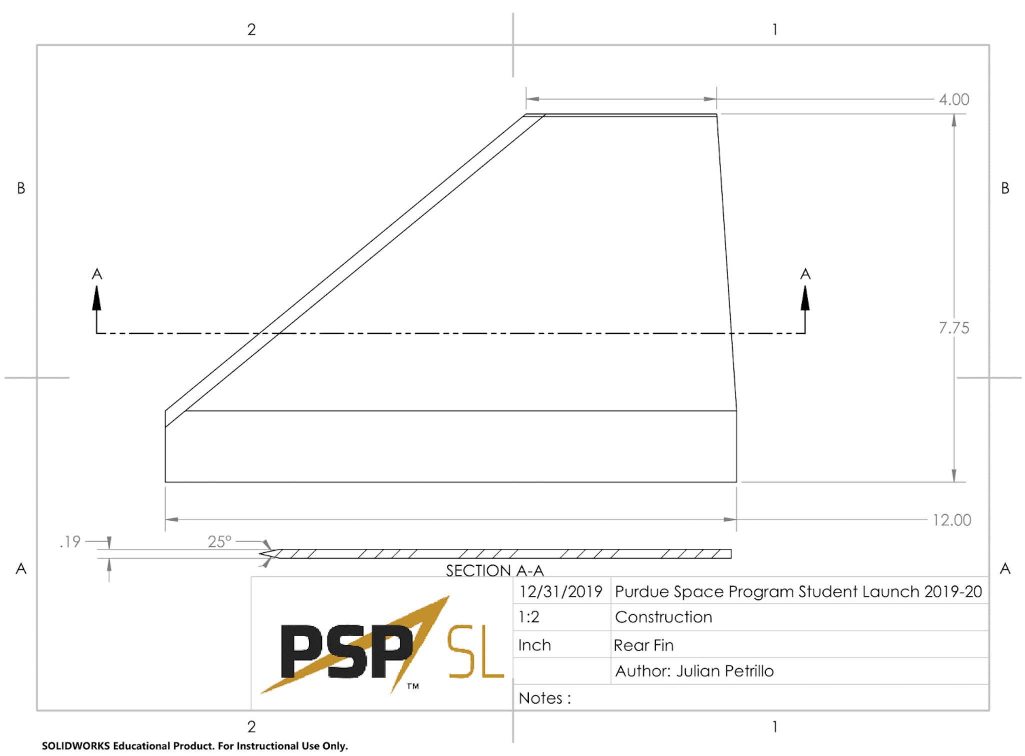


Figure 3.6: Full-Scale Fin Dimensional Drawing

For this year, the decision has been made by the team to adopt a trapezoidal, near clipped delta, fin shape. This fin shape is a common style in model rocketry, and provides good stability. The chosen fin design has a root chord of 12.00”, a tip chord of 4.00”, a height of 6.25”, a sweep length of 7.58” and a sweep angle of 50.5 degrees. The thickness of the fin is approximately 0.1875”. The fin tab height is 1.5”. The fin also features a beveled leading edge for reduced dynamic drag and for better streamlining.

The shape has been optimized to provide high stability, manufacturability, and at the same time reduce drag. For optimization of the fin design parameters, OpenRocket was primarily used. The team first decided on having a 12” root chord , as a relatively large surface area provides a good force of aerodynamic lift ($L = \alpha * P * A$, where L is the aerodynamic lift force, α is angle of attack, *P* is dynamic pressure and *A* is the surface area of the fin). If the launch vehicle tries to deviate from its course, the fins will correct it. The larger the lift force on the fins, the higher the stability that the fins provide. In order to have a large area, the team also decided to have a fin span (or height) of 6.25”. This fin span, along with the 12” root chord, provided a large enough surface area to gain the necessary stability. At the same time, however, the larger the surface area of the fin, the larger the dynamic drag component of the aerodynamic force, which makes the vehicle lose energy and thus lowers the altitude. The team tried different root chord lengths for the same fin span and different fin spans for the same root chord, using OpenRocket. The root chord of 12” was large enough to give a reasonable surface area, but not too large to weigh down the vehicle and increase dynamic drag , which was the case with larger root chords. The fin span of 6.25” was large enough to also provide a useful surface area, and is what the team found to be the optimal fin span, as it is roughly the same as the airframe diameter (6.15”) and it is not too large to increase dynamic drag to a point where a lot of energy is lost and altitude is significantly lowered, which is the case for larger fin spans.

As for the sweep angle and the fin shape, the team experimented with different angles in OpenRocket, and ended up with 50.5 degrees as the best, since it moved the surface area of the fin further aft than smaller sweep angles, thus moving the center of pressure for the launch vehicle further aft, which increases the stability. Smaller sweep angles did not move the CP as far back as the one we chose, and larger angles (swept back designs) ended up decreasing stability.

Another important part of the fin design is the chosen tip chord, which is 4.00” long. The tip chord needs to be small, in order to reduce induced drag and save energy, which can help the vehicle gain altitude. At the same time, however, it can decrease the overall surface area of the fin, reducing the lift component of the aerodynamic force and thus reducing vehicle stability. Indeed, when changing the tip chord in OpenRocket, tip chords smaller than 4.00” reduced the surface area and thus the stability of the fin, while tip chords larger than 4.00” reduced altitude slightly, and did not provide any significant and useful increase in stability.

With regards to the thickness of the fin, it was decided to have a thickness of 0.1875”, as this thickness has been adopted in previous designs and proved to be effective with the chosen material that the team is using, which is fiberglass. The thickness also proved to be effective on the subscale launch vehicle fins, as they were in perfect condition after the flight. Verification of the effectiveness of this thickness and how much bending force the fin can withstand with this thickness and the chosen material is done with Finite Element Analysis (FEA) on Solidworks version 2020. It is necessary to note that the edge of the fin will be beveled, providing a more aerodynamic shape than a square edge, which can help decrease dynamic drag. The beveled fin shape was tested in the subscale rocket and proved to be successful.

The chosen material for the fin is G10 fiberglass, which is light, can resist bending forces and high temperatures, and is easy to shape with belt sanders and saws to obtain the shape that is needed for the fins.

When it comes to mounting the fins on to the lower airframe, each fin is inserted through a friction fitting slot through the lower body tube and glued onto the lower body tube, motor tube and centering rings (which “sandwich” each fin from the leading and trailing edge of the fin tab). The epoxy is shaped into round fillets, to provide more structural strength and resistance to stress and strain. The correct mounting of the fins is ensured by using a fin mount that stays in place until the epoxy completely dries.

3.1.6.1.2. *Flight FEA*

FEA was used to aid in verification of the full-scale fin design. The team analyzed the effect of dynamic drag forces on the fin edge and the result of the bending moments caused by lift and drag.

For both the lift force (bending) FEA and dynamic drag force FEA , the same beveled fin model was used, with the same material properties defined in both cases (G10 fiberglass) and the same fixing locations on the fin tab.

Lift Force FEA Results

FEA for lift forces on the fin was done in Solidworks with an applied force of 50N at the tip of the fin, on a trapezoidal surface area of 0.2524in^2, which was created by sketching a parallel line to the tip chord, 1/16” away from the fin tip. The pressure on the specified area on the fin tip is 44.5397psi, which is almost 3atm of pressure. The fin was fixed at the fin tab surfaces before the application of the force on the fin tip. The fixtures can be seen as yellow vectors and the applied force can be seen as purple vectors on all lift force FEA figures. The fixing that was done simulates the way the fin is glued onto the motor tube, lower body tube and centering rings with epoxy, which is shaped in fillets.

This provides us with a good safety factor for verification of the structural integrity of the fins, as it is highly unlikely for the fin to experience such enormous pressures on the fin tip.

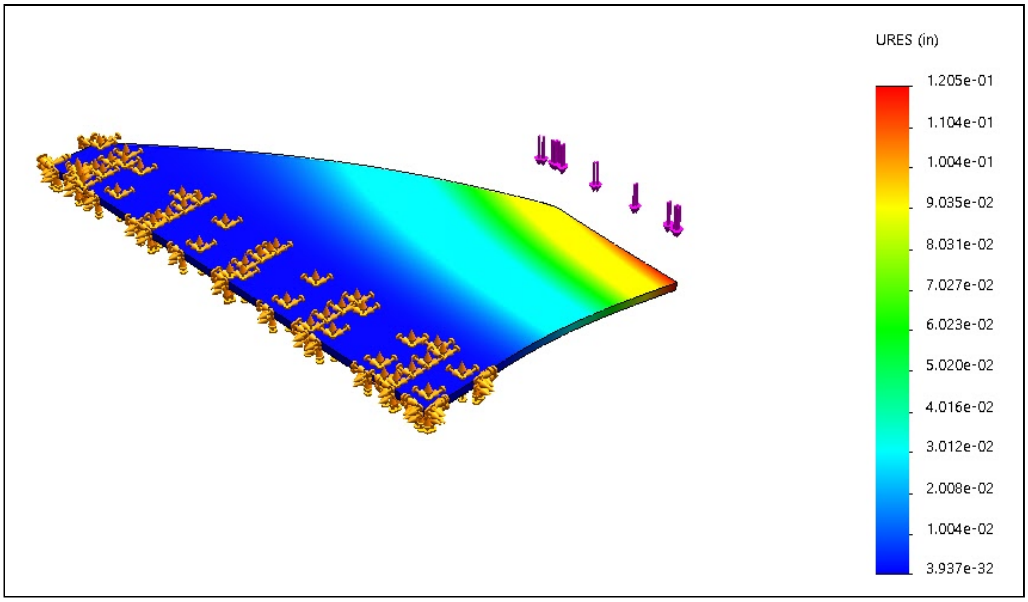


Figure 3.7: Fin Lift Force FEA Displacement Results

Figure 3.7 above shows a diagram with displacement caused by the bending force, with a maximum displacement of 1.205e-01" and a minimum displacement of 3.937e-32". The displacement is constant from fore to aft, with the smaller values towards the fin root , colored blue, and gradually larger values towards the fin tip, colored orange and red.

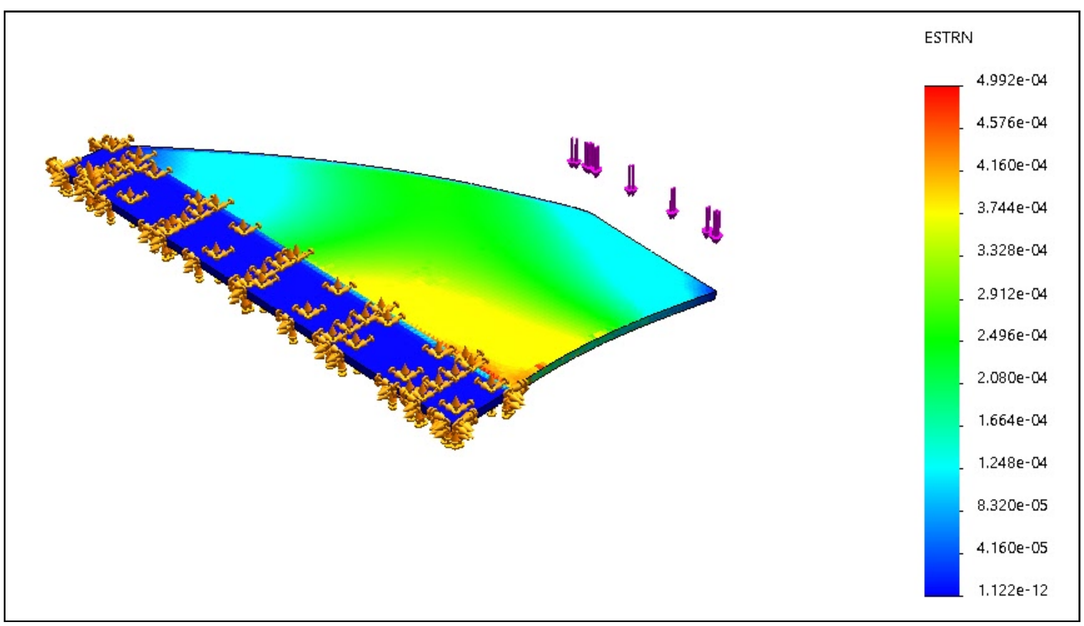


Figure 3.8: Fin Lift Force FEA Strain Results

Figure 3.8 above shows a diagram of the strain experienced by the fin under the same conditions, with the small red coloured area on the rear corner of the root of the fin having the maximum strain, which is 4.992e-04, then gradually smaller values of strain are experienced at the centre of the fin (green coloured areas) and the minimum value is experienced at the rear corner of the fin tip as well as the front corner of the root of the fin (blue coloured areas). The minimum strain value is 1.122e-12.

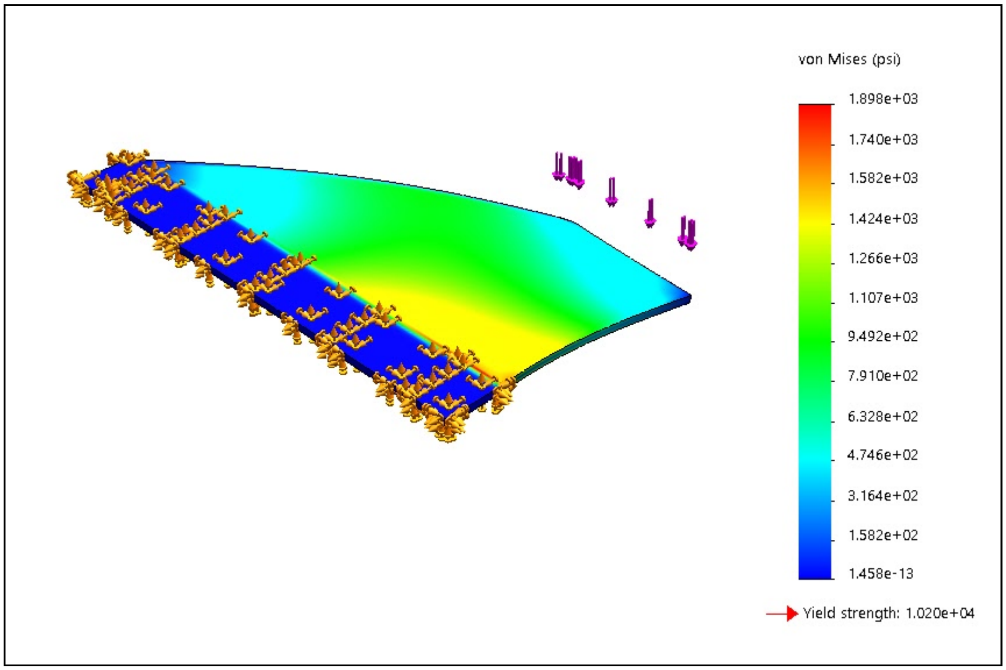


Figure 3.9: Fin Lift Force FEA Von Mises Stress Results

Figure 3.9 above shows a diagram of the Von Mises stresses experienced by the fin under the same conditions. The Von Mises stress diagram looks almost identical to the strain diagram, with the small red coloured area on the rear corner of the root of the fin experiencing the maximum stress, which is 1.898e+03psi and is well below the yield stress of the material, which is 1.020e+04psi. This simulation verifies the structural integrity of the fin, as there is no point exceeding the yield stress value by at least an order of magnitude. The values keep becoming gradually smaller values towards the centre of the fin (green coloured areas) and the minimum value is experienced at the rear corner of the fin tip as well as the front corner of the root of the fin (blue coloured areas). The minimum stress value is 1.458e-13psi.

Dynamic Drag Force FEA Results

FEA for the dynamic drag forces on the fin was done in Solidworks with an applied force of 800N distributed along the leading edge of the fin, on a beveled surface area of 5.6588in^2. The pressure on the specified area on the fin tip is 31.7819psi, which is approximately 2atm of pressure. The fin was fixed at the fin tab surfaces before the application of the force on the fin leading edge. The fixtures can be seen as green vectors and the applied force can be seen as yellow vectors on all Dynamic Drag force FEA figures.The fixing that was done simulates the way the fin is glued onto the motor tube, lower body tube and centering rings with epoxy, which is shaped in fillets. This provides us with a good safety factor for verification of the structural integrity of the fins, as it is highly unlikely for the fin to experience such enormous pressures on the fin leading edge, as with the maximum velocity below Mach 0.5 and the maximum coefficient of drag below 0.5 (according to Open Rocket simulations), the drag force is an order of magnitude larger than the maximum expected one on each fin [26N, calculated using a Python script with the following formula for dynamic drag force:

$F = Cd * 1/2 * \varrho * v^2 * A$, where *F* is the dynamic drag force, *Cd* is the drag coefficient (0.5 was used even though this is slightly above the maximum simulation value), ϱ is air density, *v* is velocity and *A* is the surface area of the leading edge of the fin].

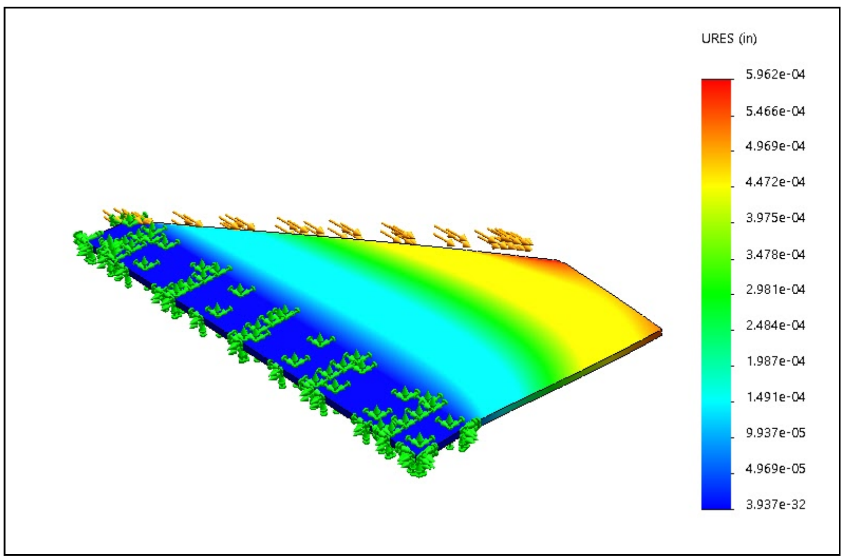


Figure 3.10: Dynamic Drag Force Displacement on Fin

Figure 3.10 above shows a diagram with displacement that is caused by the dynamic drag force, with a maximum displacement of 5.926e-04”, while the minimum displacement is 3.937e-32”. The displacement is even throughout the fin, with the smaller values towards the fin root , colored blue, and the gradually larger values towards the fin tip, colored green, then larger ones orange, and the maximum values colored red at the fin tip.

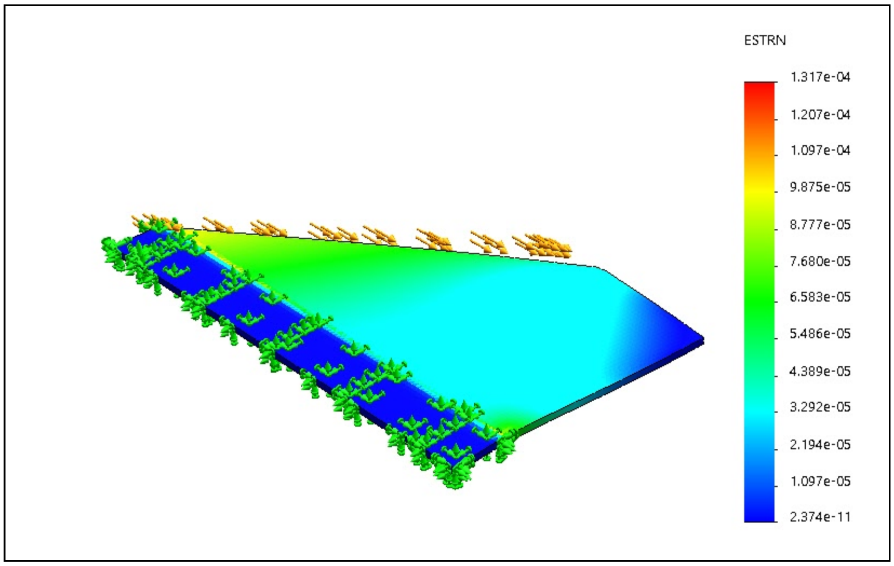


Figure 3.11: Dynamic Drag Force Strain on Fin

Figure 3.11 above shows a diagram of the strain experienced by the fin under the same conditions, with a barely noticeable red coloured area near the front corner of the root of the fin having the maximum strain, which is 1.317e-04, then gradually smaller values of strain are experienced at the centre of the fin (green coloured areas) and the minimum value is experienced at the rear corner of the fin tip (blue coloured areas). The minimum strain value is 2.374e-11.

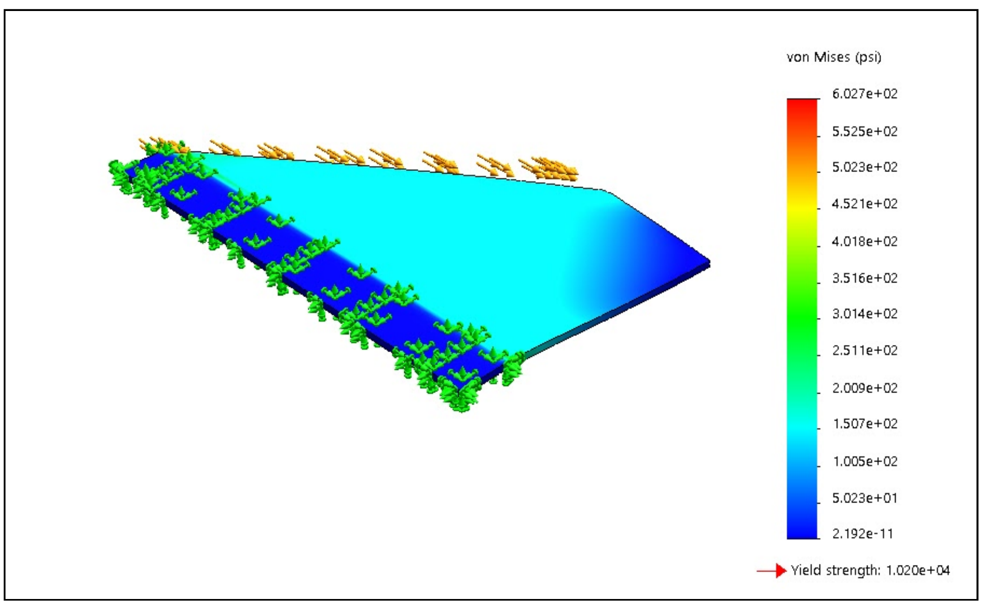


Figure 3.12: Dynamic Drag Force Von Mises Stress on Fin

Figure 3.12 above shows a diagram of the Von Mises stresses, from two different points of view, experienced by the fin under the same conditions. The Von Mises stress diagram looks similar to the strain diagram, with the barely noticeable red coloured area on the front corner of the root of the fin experiencing the maximum stress, which is 6.027e+02psi and is well below the yield stress of the material, which is 1.020e+04psi. This simulation verifies the structural integrity of the fin from this aspect as well, as it is unlikely to exceed the yield stress value by at least two orders of magnitude. The values keep becoming gradually smaller values towards the centre of the fin (green and cyan coloured areas) and the minimum value is experienced at the rear corner of the fin tip (blue coloured areas). The minimum stress value is 2.192e-11psi .

3.1.6.1.3. Fin & Lower Airframe Flow Analysis

Fin flow analysis is another important part of fin design verification, and is done using Solidworks version 2020. Fin flow analysis is used to verify the fin flow over the fin and how the fin is going to perform aerodynamically during flight. Fin flow analysis helps verify the advantages the chosen fin shape provides.

Beveled-Edge Fin & Lower Airframe Flow Analysis Results

Flow simulation of the airflow around the lower airframe of the launch vehicle was conducted using Solidworks version 2020. The lower airframe 3D model was used, with the addition of a top lid on the body tube and a lid on the motor tube, in order to restrict the airflow around the lower airframe model. The lower airframe is one of the most significant parts which we would like to simulate air flow upon, since the lower airframe contributes the most to the launch vehicle's stability with the triple fin set, as this moves the CP on the aft end to provide the required aerodynamic stability. For this flow simulation, a velocity setting of -560ft/s on the Y axis was used. This velocity magnitude is equal to 0.5 Mach at sea level, which would be slightly larger than the maximum velocity magnitude the launch vehicle could possibly reach. The negation is done so that the air velocity has an opposite direction to the velocity of the launch vehicle, and this velocity setting is at a zero angle of attack. Gravity was also set to be in the same direction (Y axis) as the air flow, representing the direction the launch vehicle will be oriented at the time of launch. These settings should provide an accurate flow simulation for the launch vehicle, based on what we expect from the OpenRocket simulations and the verifications we have conducted with RASAero and hand/code calculations as well.

The results of the flow simulation - shown below in Figure 3.13 - are as expected, as the flow around the lower airframe is smooth and mostly non-turbulent, as can be seen from the streamlines and the velocity contour.

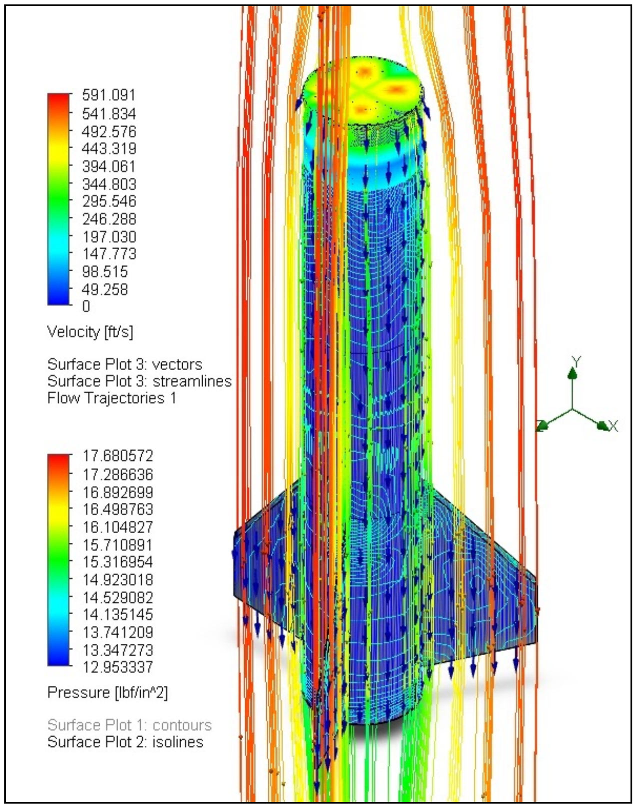


Figure 3.13: Full-Scale Lower Airframe Flow Analysis

Velocity Vectors & Streamlines Surface Plot:

In the isometric view of Figure 3.14 below, the airflow can be seen moving downwards in smooth streamlines. The surface plot shows the color of the streamlines and air velocity vectors, which are colored dark blue on the fins, as well as the whole lower body tube, meaning the air velocity near the surface of the launch vehicle is in the range of 0 to 98ft/s. This is expected, as the air closest to the surface of the fins and the lower body tube the air forms a boundary layer, due to the air's viscosity, where the velocity is close to 0ft/s.

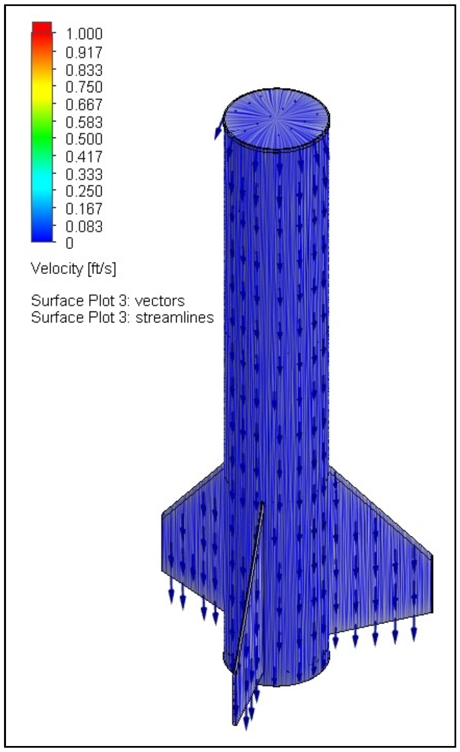
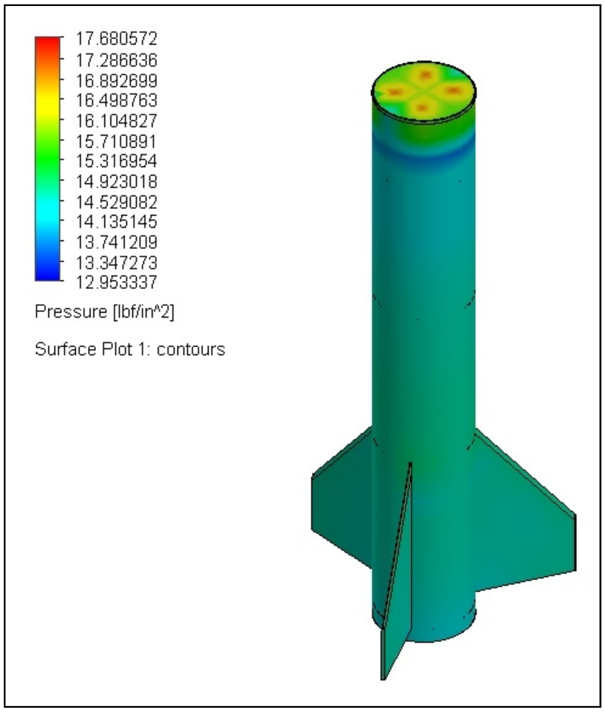


Figure 3.14: Velocity Vectors and Streamlines on Lower Airframe

Pressure Contour Surface Plot

In the isometric and side view pictures below, the pressure of the incoming air can be seen from the surface plot contour. As seen from the plot, the pressure of the incoming air, while at a high value at the top of the lower airframe (green colored region) decreases significantly while progressing downwards (from cyan to dark blue color, which is the minimum pressure of 13.0psi) and then increases again, progressing from cyan blue to green color (from 14.1psi to 15.7psi), reaching a maximum at the leading edges of the fins, and at the roots of the fins (green colored areas, from 14.9psi to 15.7psi) and then decreasing across the fin surface, as the flow progresses towards the trailing edge of the fin (color becomes cyan near the trailing edge) and the end of the lower body tube (cyan colored). In the isometric view, the top lid sealing the part can also be seen taking the maximum pressure, although this will not be the case with the real launch vehicle, as the launch vehicle will have an upper airframe which will be much more aerodynamic, since it features a nose cone.



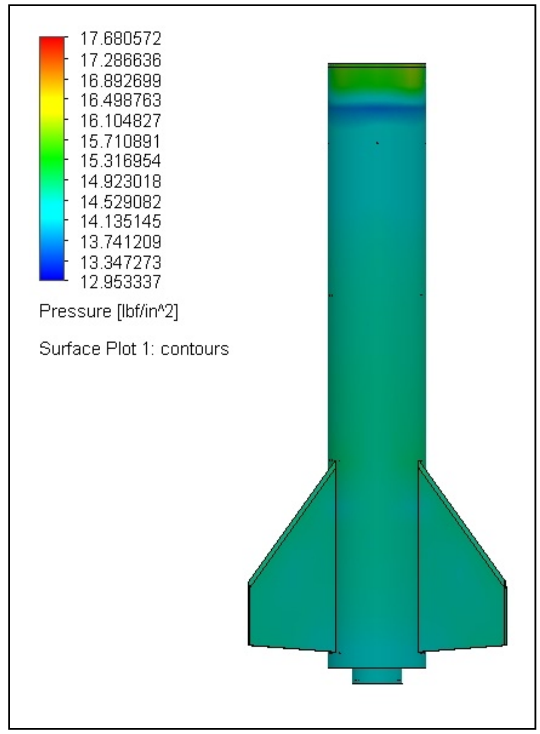


Figure 3.15: Isometric View of the Pressure Contour Surface Plot on the Lower Airframe (LEFT)
Figure 3.16: Side View of the Pressure Contour Surface Plot on the Lower Airframe (RIGHT)

Pressure Isolines Surface Plot:

In the isometric picture below, the pressure isolines of the incoming air can be seen from the surface plot isolines contour. As seen from the plot, the pressure isolines of the incoming air, while at a high value at the top of the lower airframe and while very dense (green colored region, dense isolines), which means there are several different levels of pressure in the same region and thus a rapid change in pressure, progressively become less dense downwards, which means there is a more smooth change in pressure (from cyan to dark blue color, which is the minimum pressure of 13.0psi). The pressure values increase again, progressing from a cyan blue to green isoline color (from 14.1psi to 15.7psi), reaching a maximum at the leading edges of the fins, and at the roots of the fins (green colored areas, from 14.9psi to 15.7psi), where the isolines also become very dense again, meaning there is a rapid increase in pressure locally, and then decrease in density and value across the fin surface, as the flow progresses towards the trailing edge of the fin (color becomes cyan near the trailing edge and isolines become less dense) and the end of the lower body tube (cyan colored). In the isometric view, the top lid sealing the part can also be seen taking the maximum pressure, as the isolines are very dense and are coloured red, although this will not be the case with the real launch vehicle, as the launch vehicle will have an upper airframe which will be much more aerodynamic, since it features a nose cone.

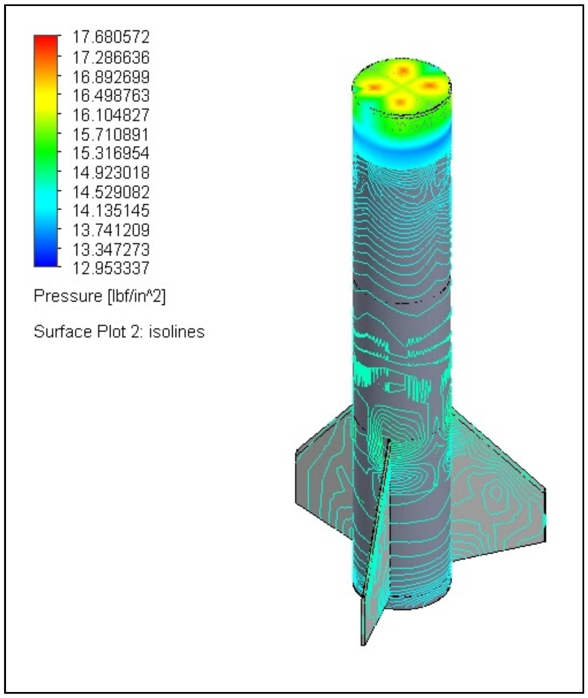
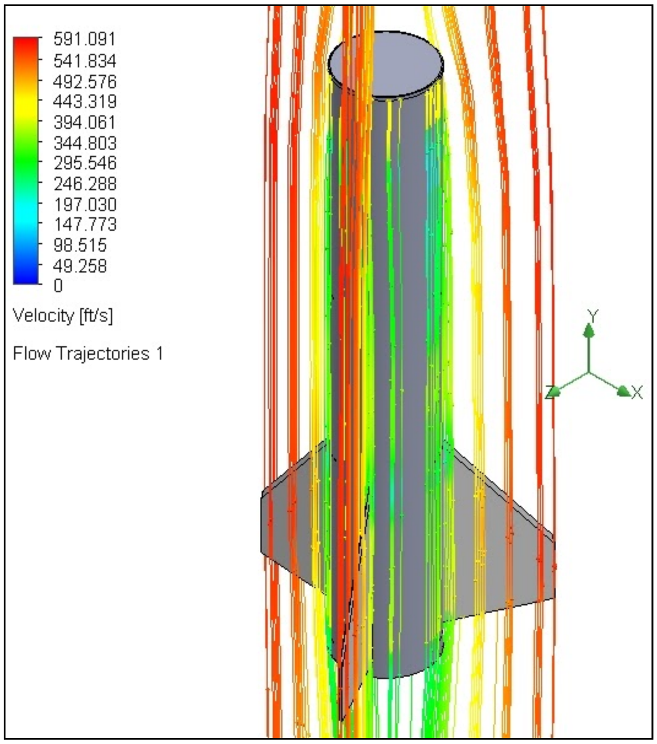


Figure 3.17: Isometric View of the Pressure Isolines Surface Plot on the Lower Airframe

Flow Trajectories with Velocity Contour:

In the picture below, flow trajectories can be seen, which have been contoured according to velocity magnitude, with a velocity setting at -560ft/s on the Y axis (the negation indicates the direction of the velocity). The maximum velocity of the air can be seen at the outer streamlines, which are red-colored, and has a magnitude of 591.1ft/s . The air velocity can be seen becoming gradually smaller in value the closer they are to the vehicle body tube and surfaces. The minimum air velocity values on the vehicle's surface can be seen at the upper part of the airframe and near the upper part of the fin root, where the air velocity streamlines are colored cyan (reaching magnitudes as low as 147.8ft/s). The global minimum air velocity values can be found inside the airframe (the global minimum is 0ft/s inside the airframe), where there is an airflow with very small flow velocities due to pressure differentials caused by the vent holes in the airframe. Below, both an isometric and a side view of the streamlines can be seen:



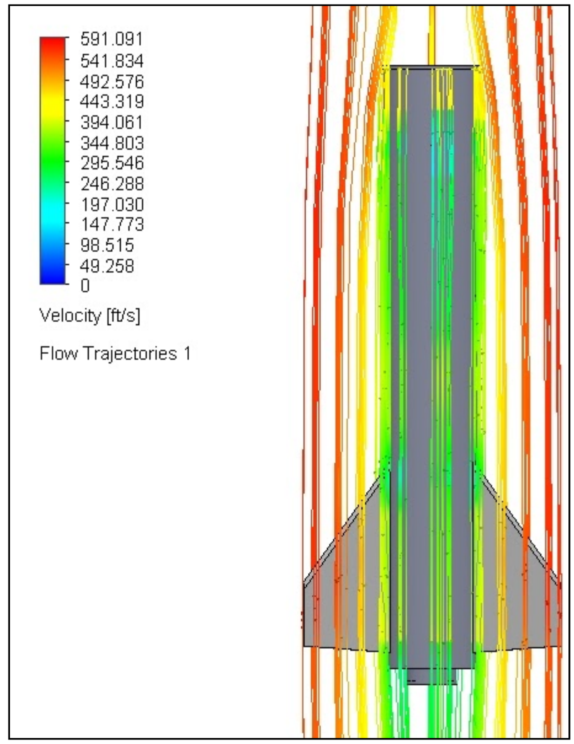


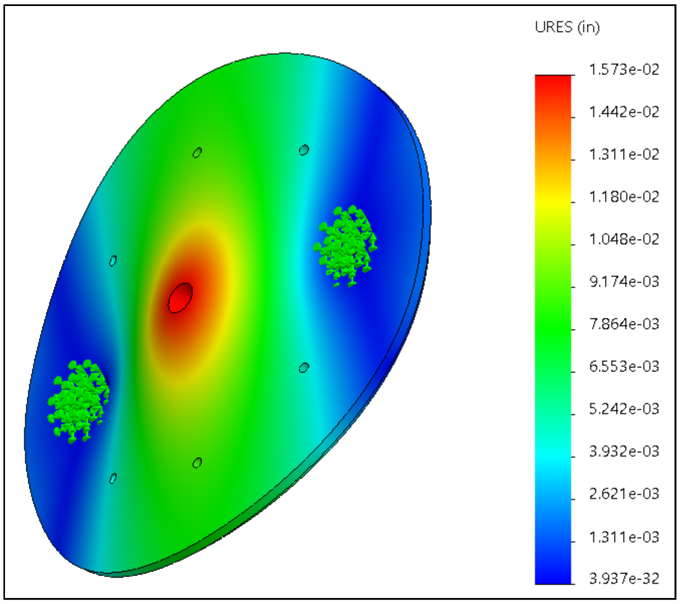
Figure 3.18: Isometric View of the Flow Trajectories on the Lower Airframe (LEFT)
Figure 3.19: Side View of the Flow Trajectories on the Lower Airframe (RIGHT)

3.1.6.2. Bulkheads

3.1.6.2.1. Avionics Bulkhead

The leading bulkhead design will feature a circular plate with varying diameters. The larger disk will have a diameter of 5.998" and the smaller disk will have a diameter of 5.775". The thickness of each respective disk will be 0.125". This is to allow the bulkhead to fit perfectly into the coupler. The bulkhead will have one 0.3" diameter hole in the center to fit an I-bolt, two 0.3" diameter holes located 2" away on either side to fit the threaded rods, and six additional holes with diameters suitable for 440 screws spaced evenly about the same circumference as the threaded rod holes. These holes will be used to secure two 8g capacity black powder canisters and two terminal blocks, which will be placed on opposite sides from each other on the bulkhead. The remaining two holes will be used to feed the e-matches from the interior of the coupler to the terminal blocks.

The material used on the bulkheads will be A-glass fiber (a type of fiberglass). The appropriate material properties (such as elastic modulus, Poisson's ratio, shear modulus, and tensile strength) were retrieved from Matweb.com. A 1500lbf simulation was run to simulate the Von Mises forces acting on the bulkhead and the resulting displacement due to the main parachute opening (see section 3.1.6.3). The force was simulated to be acting on the area on the inside of the bulkhead where the washer that holds the I-bolt in place makes contact. Fixtures were placed on the areas on the outside of the bulkhead where the washers that hold the threaded rods in place make contact.



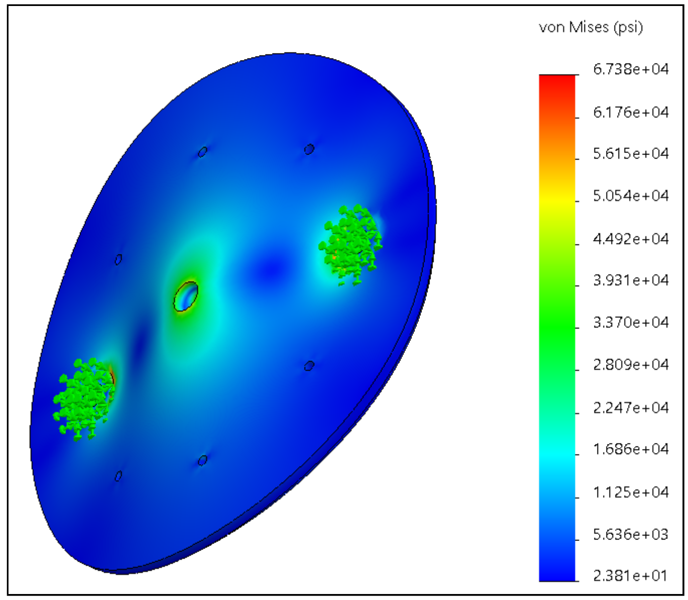


Figure 3.20: Displacement of Avionics Bay Bulkhead (LEFT)
Figure 3.21: Von Mises Forces on Avionics Bay Bulkhead (RIGHT)

Figure 3.20 shows a screen capture of the translational displacement of the bulkhead. The maximum experienced translational displacement is 0.01573”, located around the I-bolt hole in the center of the bulkhead. Figure 3.21 shows a screen capture of the Von Mises forces experienced by the bulkhead. The maximum experienced stress is 0.0006738psi, located around the inside of the threaded rod holes as well as around the I-bolt hole. This is far below the yield strength of the material, meaning that the bulkhead will not experience any deformation beyond the elastic range. This demonstrates that the team has chosen the proper material and thickness for this component to withstand the expected flight forces, particularly the deployment of the main parachute, where the most force is expected.

3.1.6.2.2. *Payload Bulkhead*

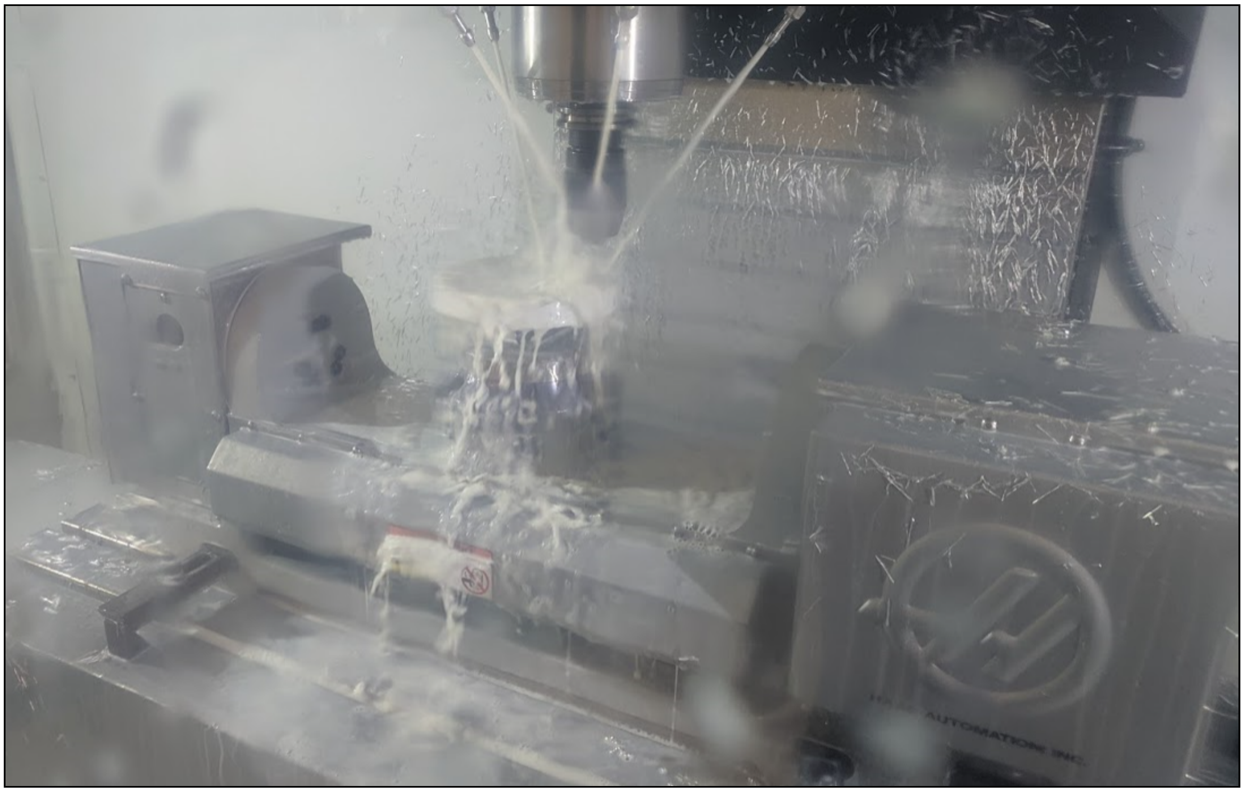


Figure 3.22: Machining the Main Parachute Bulkplate

Unlike the traditional bulkheads used in the avionics bay, the location and forces endured by the Main Parachute Bulkhead necessitate machining of the part from aluminum. The avionics bulkheads use clamping around a coupler to maintain their position inside the rocket, but that method is not an option at the forward of the rocket due to the payload bay. The use of aluminum enables the creation of structures not manufacturable out of plate fiberglass. These structures are a weight reducing web, and holes drilled around the radius of the part, allowing it to be mounted into the rocket directly. In order to simulate the worst-case scenario, the following analyses were performed with a 1800lbf load placed on a surface of equal diameter to flange on the chosen bolt that attaches to the main parachute. The material for each study was 6061 T4 Aluminum.

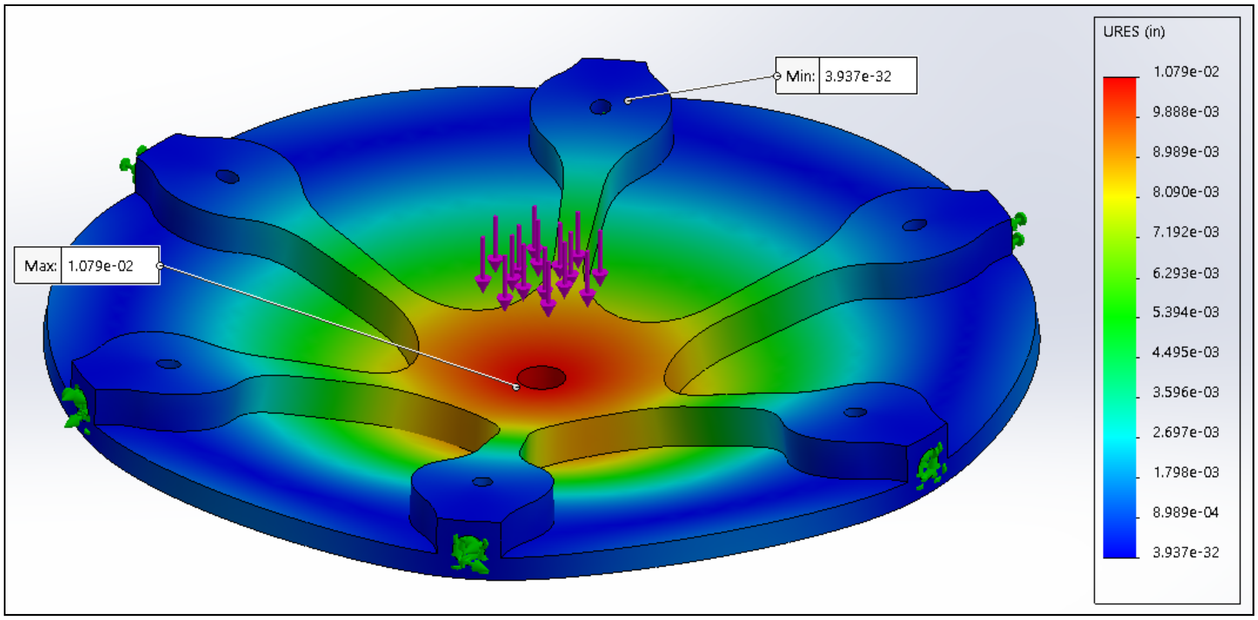


Figure 3.23: Displacement of Main Parachute Bulkplate (Scale: 55.6)

During the forces of main parachute deploy, the above displacement is expected to be present on this bulkplate. With a maximum displacement of .011", the plate can be considered to experience little to no change in shape during deployment.

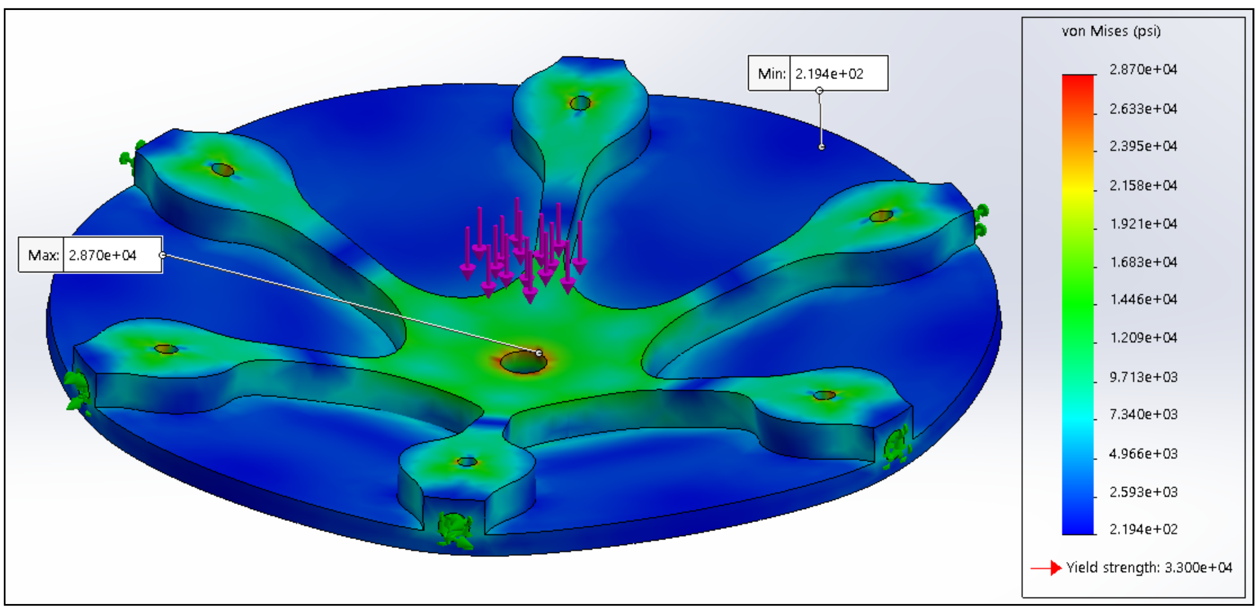


Figure 3.24: Von Mises Stresses within the Main Parachute Bulkplate (Scale: 55.6)

By iterating the part through a series of Von Mises FEAs the team has decided upon the above design to ensure that loads are accounted for with a meaningful factor of safety. The highest load on the part (2.87e+04psi) is below the yield strength of the material, and only appears as an edge effect around the attachment hole of the main parachute. The majority of the part is well below the yield stress of 6061, and as such can be considered reliable and safe to fly - even overbuilt.

3.1.6.3. Main Parachute Shock/Tension Study and Calculations

The 2020 PSP-SL team wanted to expand the team's understanding of the forces on the launch vehicle as a result of the parachutes used. For this study, it is assumed that both parachutes are deployed. The body of the vehicle at this moment can be segmented as shown below:

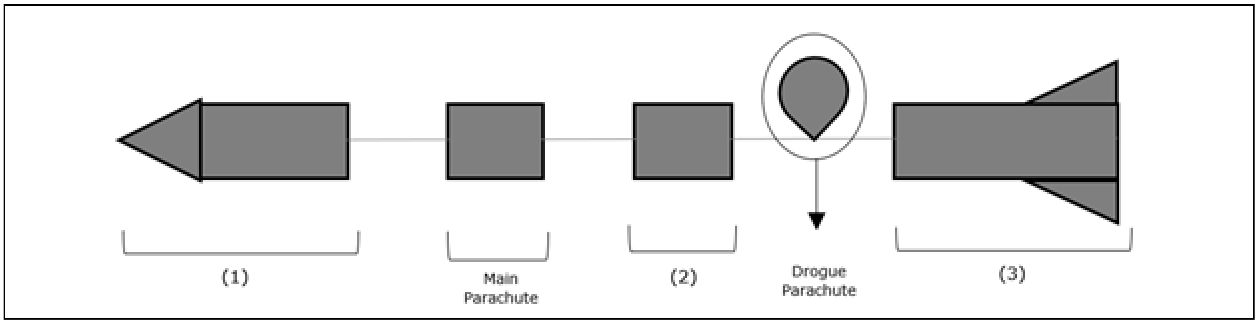


Figure 3.25: Separation of Vehicle into Bodies for Analysis

(1) Nosecone and upper airframe
(2) Avionics bay
(3) Lower airframe, nozzle, and fins

The calculations for the tension force consider the tension acting on the bulkheads by two different bodies. The calculations and diagrams for each body are as follows:

Body 1 – Vehicle

The vehicle consists of masses (1), (2), and (3). For this calculation it is assumed these masses act as a single body without fuel (53.2lbm). The diagram below illustrates the forces acting on it:

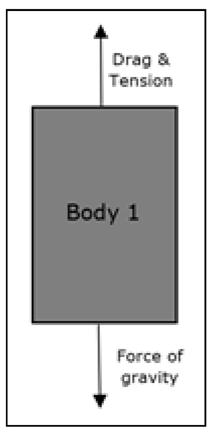


Figure 3.25a: Body 1 Free Body Diagram

As seen in Figure 3.25A, the drag and tension force acts upwards towards the main parachute and the force of gravity (F_g) acts downwards. According to Newton's second law, the force equals the object's mass times its acceleration. Hence the forces on body 1 can be written as:

$$Drag + Tension - F_g = (Mass_{Total} * Acceleration)$$

Therefore,

$$Tension = (Mass_{Total} * Acceleration) - Drag + F_g$$

The total mass of the body, drag force, and the force of gravity can be calculated as shown below. The free stream density and velocity values are calculated with a MATLAB script and found to be 0.002387slugs/ft^3 for free stream density and 41.27ft/s for free stream velocity. The coefficient of drag varies from one part of the body to the other and can be obtained from product specifications:

$$Mass_{Total} = Mass_{(1)} + Mass_{(2)} + Mass_{(3)} + Mass_{Drogue} + Mass_{Shock\ cord\ of\ drogue} = 49.20lbm$$

$$Drag = Coefficient_{Drag} * 0.5 * Density_{Free\ stream} * (Velocity_{Free\ stream})^2 * Area$$

$$Drag_{Total} = Drag_{(1)} + Drag_{(2)} + Drag_{(3)} + Drag_{Drogue} + Drag_{Shock\ cord\ of\ drogue} = 340.97lbf$$

$$Force\ of\ gravity = 32.2 * Mass_{Total} = 1582.96lbf$$

In order to calculate the acceleration, it is assumed that all three masses, including the drogue parachute, experience the same magnitude of acceleration. By using the graph of acceleration over time from the trajectory simulation code one can approximate the value of acceleration. In this case the acceleration of the body right before the parachute opens is considered so that the force it exerts on the bulkheads can be computed. This was approximated to be 0ft/s^2.

Hence the tension force acting on the bulkhead by body 1:

$$Tension\ 1 = 1241.99lbf$$

Body 2 – Main Parachute

Body 2 calculates the tension force the main parachute exerts on the bulkheads.

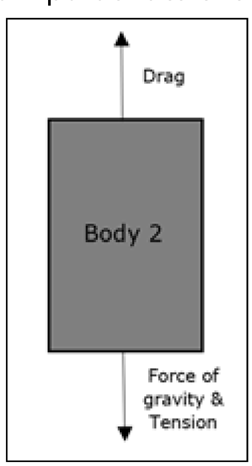


Figure 3.25b: Body 2 Free Body Diagram

Most of the calculations are the same as with body 1. The main difference is that the tension acts downwards and so the calculations are as follows:

$$Drag - Tension - F_g = (Mass_{Main} * Acceleration)$$

Therefore,

$$Tension = Drag - (Mass_{Main} * Acceleration) - F_g$$

The total mass of the body, drag force, and the force of gravity can be calculated as shown below. The values for the free stream density (0.002387slugs/ft^3) and free stream velocity (41.27ft/s) are the same as with body 1:

$$Mass_{Total} = Mass_{(1)} + Mass_{(2)} + Mass_{(3)} + Mass_{Main} + Mass_{Shock\ cord\ of\ main} = 51.56lbm$$

$$Drag = Coefficient_{Drag} * 0.5 * Density_{Free\ stream} * (Velocity_{Free\ stream})^2 * Area$$

$$Drag_{Total} = Drag_{(1)} + Drag_{(2)} + Drag_{(3)} + Drag_{Main} + Drag_{Shock\ cord\ of\ main} = 24730.76lbf$$

$$Force\ of\ gravity = 32.2 * Mass_{Total} = 1658.97lbf$$

In this case the acceleration of the body right before the parachute opens is considered so that the force it exerts on the bulkheads can be computed. This was approximated to be 0ft/s^2.

Hence the tension force acting on the bulkhead by body 2:

$$Tension\ 2 = 230.72lbf$$

The sum of these forces gives the total tension force acting on the bulkhead:

$$Total\ Tension\ Force = Tension\ 1 + Tension\ 2 = 1472.71lbf$$

3.1.6.4. Motor Mount Assembly and Motor Retention

The following section is dedicated to motor design and thrust FEA and shows Computer Aided Design (CAD) images of the full-scale launch vehicle. The motor is designed to be compatible with the current diameter and weight. It will also provide enough thrust to reach the target altitude as given by the derived requirements. The motor assembly consists of the motor tube, motor retainer, motor case, motor mount/cap and 3 centering rings to help hold the motor in place. The thrust FEA part of this section is dedicated to prove the safety of the motor and how the motor will not break the lower airframe or shoot through the inside of the airframe at launch. It will also provide the estimated stresses the launch vehicle will experience.

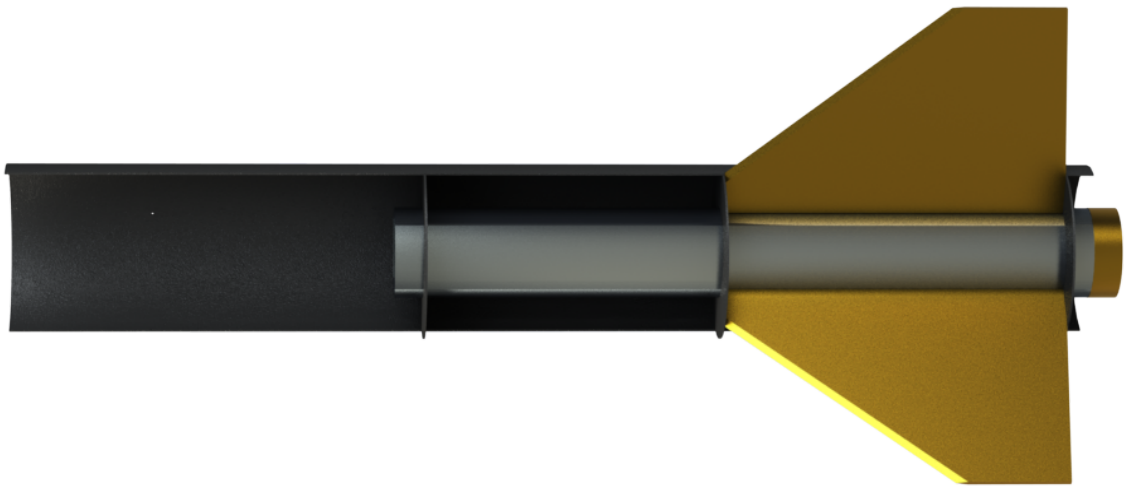


Figure 3.26: CAD of the Lower Airframe

The motor is 75mm by 621mm. It has a loaded weight of 154.14oz with a propellent weight of 83.79 oz, using Ammonium Perchlorate as the fuel, and a burnout weight of 67.48oz. It has an impulse of 1128.38lb-s, a max thrust of 385.48lbf and an average thrust of 251.78lbf. The burn time of the motor is 4.48s. Three centering rings are used to hold the motor in place between the three fins, as shown. The fin can is capped at the aft end with a motor retainer to ensure the retention of the motor. The motor assembly allows the motor to be held in place at launch with airframe, so that the motor will not shoot through the inside of the rocket. The fins are also connected to the motor with epoxy to provide better structural integrity and support.

3.1.6.5. CDR Mass Margin

The lower assembly, as shown above in Figure 3.26, is composed of fiberglass and has a complete weight of 9.95 lbm. The airframe only weighs at 4.44lbm. The motor below the lower airframe weighs at 9.73lbm. The 3 fins plus the 1.75” fillet weighs at 3.02lbm. Each fin is modified to provide the best stability in launch and during flight. This was done through simulations and analysis of fin designs like elliptical square and trapezoidal. All the data pointed at trapezoidal being the best at stability and reliability. The motor cap helps the motor stay in place from the other end, so the motor does not fall out of the launch vehicle due to the centering rings not doing their job. It also adds another layer of safety.

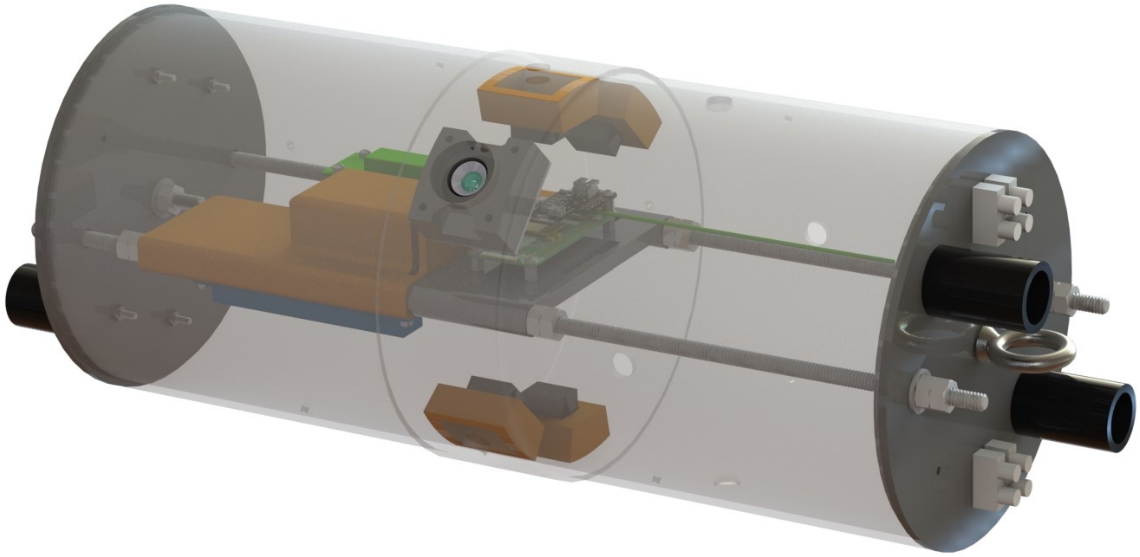


Figure 3.27: CAD of the Avionics Assembly

The avionics assembly contains equipment that helps in recovery and data gathering of the launch vehicle. The avionics assembly weighs 6lbm. It also has the drogue parachute and the main parachute near it. The drogue parachute weighs at 0.375lbm and the main parachute weighs at 4lbm. There are also shock charges fitted along the places where the drogue and main parachute are situated. These charges will allow the parachutes to deploy and allow the launch vehicle to land safely.

Figure 3.28: CAD of the Upper Assembly

The upper assembly is also composed of fiberglass and contains the payload bay and the drogue parachute. It is mainly a body tube connected by a coupler to the nose cone. Right below the couple is where the payload bay is situated. The total upper assembly weighs 24.4lbm and the airframe weighs 4.44lbm.

The payload assembly contains the UAV which nets a total weight of 12.5lbm. It also has a safety case which provides shock absorption to the payload.

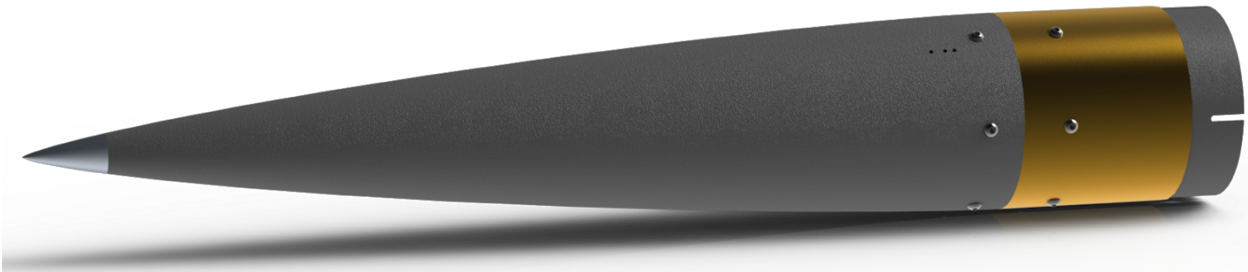


Figure 3.29: CAD of the Nose Cone

The nose cone has a total weight of 2.8lbm. It has a metal tip and is mainly composed of fiberglass as well. It can contain ballast to increase weight if needed.
The mass of the launch vehicle has not significantly changed. The hard cap of the launch vehicle weight is 55lbm, in case the mass increases. This gives a safety margin of 2.8%. This allotment allows an increase of weight from unseen extra Epoxy or other structural strengthening.

The subscale version of the launch vehicle was projected to be 5.42lbm with no motor and the actual weight slightly exceeded that. The motor tube weighed 0.8lbm and housed the AeroTech H-148R motor. The nose cone weighed 0.321lbm and the upper assembly weight was 2.81lbm. The lower airframe and the avionics bay were simulated to weigh 0.729lbm, where the airframe was the avionics bay. The payload bay came in with a weight of 1.44lbm.

3.1.6.5.1. Thrust Finite Element Analysis (FEA)

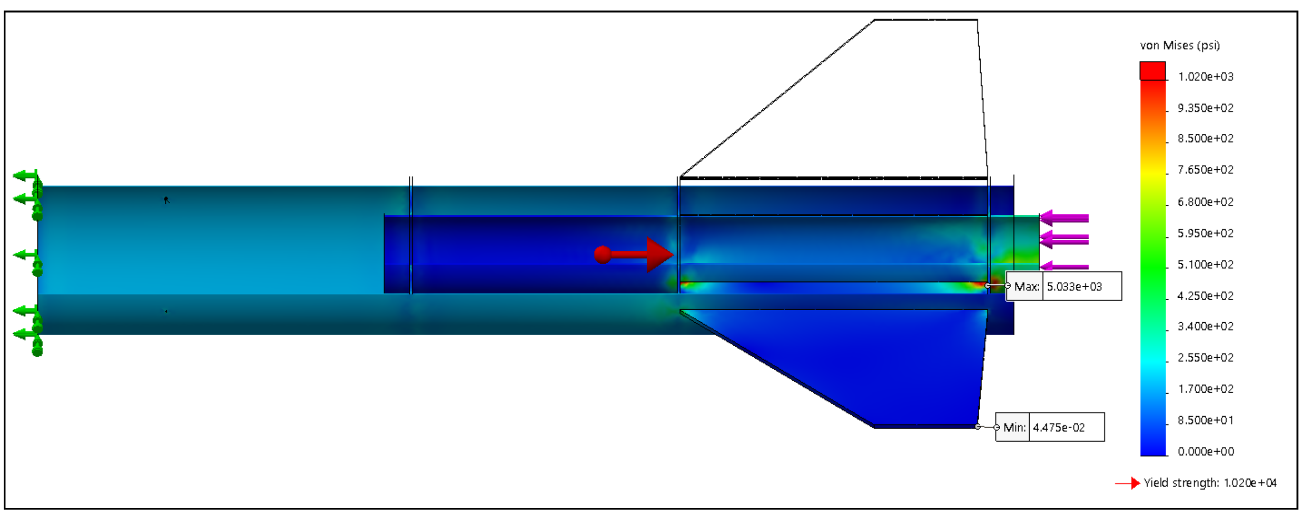


Figure 3.30: Lower Airframe Stress FEA Results

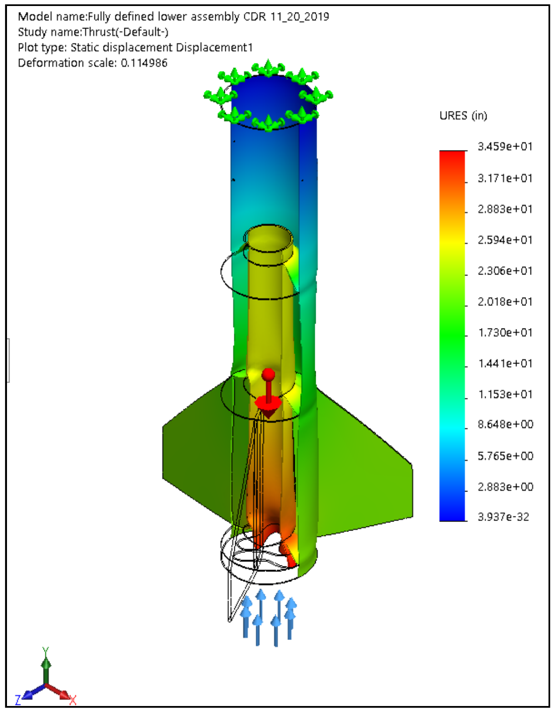


Figure 3.31: Isometric View of the Displacement FEA Results on the Lower Airframe

Figure 3.30 in this section depicts the stress plot in psi on a scale from 0 to 10% of the material yield strength. Given the relatively low-intensity gradient, this simulation proves that the selected material can withstand the force of the vehicle motor even while fixed at the top of the tube within a very generous safety margin. The maximum value reached was 5033psi, while our yield value is 10,200psi. These simulations were carried out using the maximum thrust of a Cesaroni L1115 rocket motor applied to the bottom of the motor tube, which secures it. In Figure 3.31, the vehicle was fixed about the top of the object, or where the lower airframe would connect to the payload. This displacement plot shows a 0.11 scale deformation of the solid body. The full scale results are correct in their parameters but incorrect in their final result, with values that far exceed any reasonable amount. For this reason, we have included the small scale image and plan to rectify this FEA before FFR. These plots were calculated in order to determine the feasibility of using our chosen motor in the vehicle, given the application point of the force and the material used.

3.2. Subscale Flight Results

The 2020 PSP-SL team, as per the rules of the 2020 NASA Student Launch Competition, designed, analyzed, assembled, and tested a subscale version of the team's expected launch vehicle. When going through the design process, the team wanted to make sure that the following topics were properly represented: near true half scale vehicle, flight trajectory analysis, and allowing new member understanding of vehicle construction.

3.2.1. Scaling Factors

3.2.1.1. Constant Factors

The outer diameter of the subscale rocket is constructed following the 0.50:1 ratio to the full-scale.

3.2.1.2. Variable Factors

- The motor's diameter was scaled 0.51 : 1 and its length was scaled 0.24 : 1, as the motor was chosen according to the permissions the team had for the launch field, the allowable altitude given to the team by Indiana Rocketry, Incorporated (IRI), and what was commercially available. The motor retainer and tube were also variable factors due to this.
- The avionics bay switch band followed a 0.75:1 scale. This change was made in order to shorten the overall length of the rocket, while leaving room for any toggle switch access ports that would be necessary in order to control the power of the avionics.
- Many small changes were made to an exact 0.50:1 scale. The length of the rocket followed a 0.52:1 scale. The fins followed a 0.49:1 scale. The avionics bay and threaded rods followed a 0.49:1 scale. A ballast of 0.5lbm was added to the nosecone for increased stability. The avionics bay was also moved forward in the rocket. All of these changes were made to account for inherent differences between the full scale launch and the subscale launch, most notable the lack of payload. In making these changes, the rocket was able to retain similar values of stability and the overall geometry of the rocket was retained.

3.2.2. Launch Day Conditions

The team's subscale launch vehicle was launched on the 24th of November, in a farm field near Pence, IN which the IRI organization has authorization to use up to 6,000' at any point. The overall launch conditions were favorable. The day was partially cloudy with temperatures ranging from 33F to 50F. Wind ranged from 5 to 13 miles per hour (mph) from the southwest. At the moment of launch, close to 3:30P.M. central standard time, the wind was estimated to be 12mph from the southwest. No precipitation was recorded.

3.2.3. Flight Analysis

3.2.3.1. Predicted Vs. Recorded Flight Data

The subscale rocket was launched using the AeroTech H148.
This engine has a diameter of 38mm and a length of 15.2mm. It carries 122g of redline propellant with a total weight of 309.1g. It is a regressive burn engine with a total burn time of 1.4s. The following table shows a comparison of the predicted and recorded flight results.

Metric	Predicted (OpenRocket)	Recorded
Wind Speed	15mph	12mph
Launch Angle	5 degrees	5 degrees

Apogee	677’	617’
Max Velocity	194ft/s	164ft/s
Ascent Time	7.1s	5.25s
Drift	47’	95’

Table 3.1: RRC3+ Sport Subscale Launch Flight Data (24 Nov 2019)

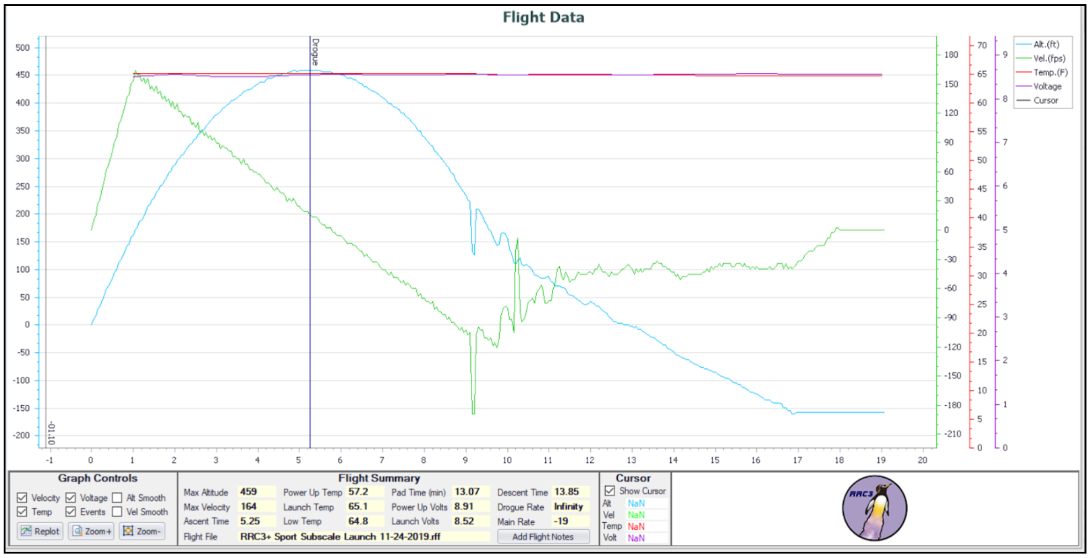


Figure 3.32: RRC3+ Sport Subscale Launch Flight Data (24 Nov 2019)

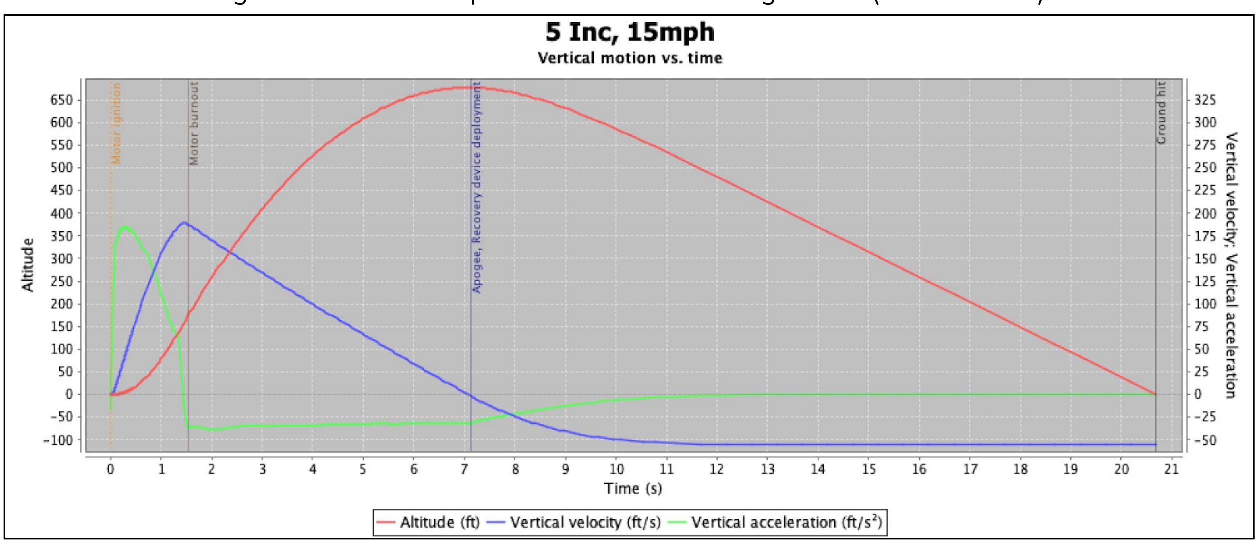


Figure 3.33: OpenRocket Simulation with Similar Conditions to Subscale Flight

Multiple OpenRocket simulations were performed with varying inclination and wind. As shown in Table 3.1, the flight results were relatively close to the simulations. Apogee was slightly above the expected (677’ vs 617’) and the maximum velocity achieved was nearly the same as the simulated one (164ft/s vs 194ft/s). Using OpenRocket, each time the simulation is run, the results are within a

+/- 40’ (6%). Therefore results from the simulation have a small degree of error and the team is happy with the results of the simulation compared to the flight results.

3.2.3.2. Errors and Discontinuities

The small differences between the simulation and recorded data reaffirms the accuracy of our simulation tool. The small difference in apogee of 60’, which accounts for less than 10% error, may be explained by the weight distribution of the subscale launch vehicle, with a dummy weight included which may have not been entirely accurate. Additionally, the small variations in fin weight, specifically the fin’s fillets may explain the minor differences in values. The difference in drift is explained by the ascent time difference, and thus by the total flight time difference.

3.2.3.3. Estimated Full Scale Drag Coefficient

By adjusting the OpenRocket simulation to the exact conditions of the launch day (12mph winds and a tilt of the launch rod of approximately 5 degrees into the wind). From this simulation an average coefficient of drag for the subscale rocket was calculated by the program to be around 0.49. As the full-scale rocket’s exterior is exactly similar to that of the subscale, the team estimates the full-scale vehicle’s coefficient of drag will be around 0.49 on average as well.

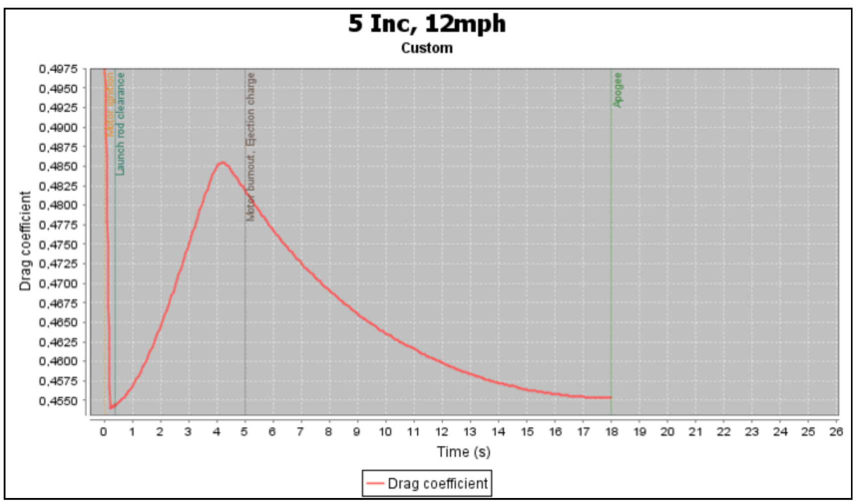


Figure 3.34: Exact Subscale Launch Day Simulation

3.2.4. Impact to Full-Scale Design

The subscale launch reassured the team that the chosen design fulfilled its purpose. A stable, safe launch vehicle capable of performing the mission it was designed for. The team has considered the possibility of increasing the lower airframe from 38’’ to 40’’ taking into account that an increase in the launch vehicle’s minimum stability off the launch rail would give the team a safety margin. The easiest way to do this would be to increase the overall stability of the launch vehicle, which can be done by increasing the length of the lower airframe. Overall, the success of the subscale launch proved the validity of the model and reassured the team that no major changes were needed.

4. Avionics, Recovery, and Mission Performance

4.1. Avionics & Recovery CDR Design & Justification

Coupler

A coupler that is 14” long will slide between the upper and lower airframe sections. It will have an outer diameter of 5.998” and an inner diameter of 5.775”. It will contain four holes with diameters suitable for 4-40 shear pins offset 3” from either side of the middle of the coupler. Depending on the ejection charge sizing calculations and ground separation testing, all four holes may not need to contain shear pins if this results in the amount of black powder required for successful separation being too high. Four 0.375” diameter static port holes will be located in the same circumference as the forward shear pin holes, offset 45 degrees. This diameter was calculated to be ideal considering the overall volume of the avionics bay.

Switch Band

The switch band will have an outer diameter of 6.17”, an inner diameter of 6”, and a width of 2”. It will have two centered holes on either side of it with diameters of 0.375” in order to access the switches from the outside of the vehicle. It will also contain a 1” diameter hole offset 45 degrees from one of the switch holes for the camera to see outside of the vehicle. The coupler will contain identical holes underneath each of these holes in the final assembly. The switch band is primarily used to separate the upper and lower airframe so that they can function as two separate entities during launch and deployment.

Primary and Redundant Altimeters

Both the Altus Metrum Telemetrum and the Missile Works RRC3+ Sport will be used for the recovery subsystem. The Telemetrum was chosen as the primary altimeter, while the RRC3+ Sport will be the redundant altimeter. These two altimeters were chosen due to their high score on the decision matrix. While the Telemetrum had a higher cost, it had a significantly higher maximum height and smaller area than the other options. Additionally, the Telemetrum contains a GPS and telemetry system which put it over the other options. The RRC3+ Sport was the second highest ranking altimeter due to its cheaper price and its ability to store large amounts of flight data, making it a great alternative altimeter despite its large size. Also, in case of a failure in the primary altimeter, it would be less likely for the same failure to occur to a redundant altimeter of a different make and model.

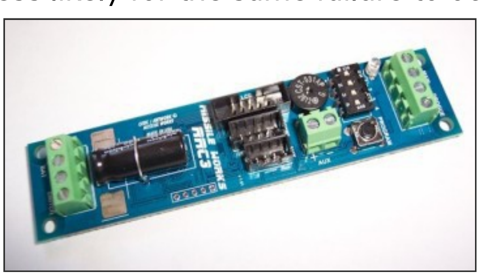


Figure 4.1: Missile Works RRC3+ Sport

Figure 4.2: Altus Metrum Telemetrum

Ejection Charges

On the ejection charge type decision matrix, the black powder scored a total of 245 points, whereas the CO_2 ejection device received a score of 195. Comparing the two total point values, FFFFG black powder was the clear choice for the type of ejection charge that should be used. Specifically, black powder was chosen because it occupies less volume and leaves less residue. These advantages outweigh the advantage of the CO_2 canister being lighter. Black powder has also been used in the past, and has been found to be reliable. In terms of design, two 8g capacity black powder canisters (one primary, one redundant) in combination with masking tape, aluminum foil, and a section of latex glove will be used to contain the black powder on each end of the avionics bay.

Switches

The switch design that will be used is the rocker switch, as shown in Figure 4.3. This choice represents a middle ground between mechanical complexity and reliability. In addition to these design factors, this is the switch type that has been used on previous projects, and previous experience will help to ensure a reliable design. Two 0.827” X 0.591” X 0.543” rocker switches will be utilized (one for the primary altimeter system and one for the redundant altimeter system). 3D-printed switch holders will secure the switches to the inside of the coupler while still allowing them to be accessed from the outside of the vehicle. They will be sized to house the switches with as little excess material as possible and will have a curve that conforms to the inner diameter of the coupler to ensure a secure fit.

Figure 4.3: Avionics Bay Rocker Switch

Other Custom-Designed and 3D Printed Parts

The designs of the altimeter sled, battery compartment, battery guard, and Telemetrum antenna holder (which are all of the parts that will be custom-designed and 3D printed by the team, plus the switch holders) have completely changed since PDR. The reasons for these changes include the fleshing out of the design of the camera system in the avionics bay as well as concerns that the overall weight of the 3D-printed parts was too high. A sled 2” in width will slide onto the threaded rods and sit directly underneath the switch band so that the electronics on the sled can be easily wired to the camera. The camera will be attached to the inside of the coupler and will look through the switch band. With the original separated configuration, both the altimeter sled and the battery compartment would not be able to fit in the avionics bay on one side of the camera sled. However, placing them on either side would necessitate crossing battery wires over the camera sled. Also, many cutouts had to be made in these two separate, bulky parts in order to get them to be an acceptable weight, and eventually it became clear that this design was no longer viable. While the battery guard will remain approximately the same, the avionics sled and the battery compartment will now be combined into a single part. On one side will be the Telemetrum and the 9V battery (for the RRC3+ Sport), and on the other side will be the RRC3+ Sport and the 3.7V LiPo battery (for the Telemetrum). Placing each battery directly underneath (on the opposite side of the sled) its corresponding altimeter allows for the battery wires to need to cross as short a distance as possible. Also, without any cutouts that risk structural failure in the parts, these parts weigh a combined 186.65g, thirty grams less than the total of the equivalent parts in the old design. The only con to this configuration is how there is no longer a place to put the antenna holder for the Telemetrum altimeter (originally on the separate battery compartment). However, this drawback can be accepted because the Telemetrum is known to be perfectly capable of functioning without anything holding its antenna in place inside the avionics bay. The altimeter sled/battery compartment combined part will be located aft of the camera sled, with the Telemetrum antenna pointing up as well as the opening to the batteries (to ensure they stay in place during launch). This part will be 4.5" in length, 4.55" in width, and a maximum of 1.78" in height. The battery guard (which will serve to hold the batteries in place when the launch vehicle is upside-down in the air while still allowing the batteries to be wired) will be a maximum of 0.25" in length, 4.55" in width, and a maximum of 1.78" in height.

4.2. Parachutes and Attachment Hardware

4.2.1. Parachute Choices

The parachutes that have been selected for this year's design are the 24" Fruity Chutes Classic Elliptical for drogue and the 120" Skyangle Cert-3 XXL for main. The Fruity Chutes drogue parachute was selected because it has a much higher drag coefficient (1.5 - 1.6) and a much lower weight (2.2 oz) than the 24" Skyangle Cert-3 Drogue that was used last year, so it will be much more optimized for this year's design. Generally, it scored very highly in the decision matrix. The Skyangle Cert-3 XXL main parachute was selected, even though it was not originally considered in the decision matrix, because it has a high enough carrying capacity to allow the heaviest section of the launch vehicle to land with a kinetic energy of less than 75ft-lbf, as specified in the NASA requirements. Originally, the Skyangle Cert-3 XL was chosen for the main parachute because it was the top scorer in the decision matrix and it has proven to be successful in past launches. Sizing up from the XL to the XXL allows for the beneficial properties of this series of parachutes to be maintained while making sure the one that will be used is large enough for the vehicle's needs.

4.2.2. Attachment Hardware and Heat Shielding

The drogue parachute will be attached to a 20' long, ½" tubular nylon shock cord, while the main parachute will be attached to a 40' long, ½" tubular nylon shock cord.

Harness/airframe interfaces include ¼" Stainless Steel (SS) quick links through the looped tether ends of the parachutes attached to ¼" SS I-bolts through the bulkheads on either end of the avionics bay.

To protect the parachutes from hot ejection charge gases, an 18” to a side square Nomex blanket will wrap around each the drogue and main parachutes when they are packed inside the airframe.

4.3. Electrical System and Schematics

4.3.1. Electrical Components and Redundancy

The electrical system in the avionics bay was designed using a parallel system, employing complete electrical and physical redundancy as a means of achieving higher reliability, predictable performance, and remarkable safety. Each subsystem in the parallel system, which is completely separated from the other, features an altimeter, battery, switch, drogue ejection charge (release), and main ejection charge (release). Moreover, separating the parallel system entirely from the main system guarantees that the parachutes will be released at the correct times even under the circumstance of multiple failures occurring within the primary system, therefore avoiding compromising the mission. The redundant altimeter is of a different make and model to ensure that there is minimal likelihood for the same type of error or failure to occur across both systems under the circumstance of an altimeter-specific failure occurring within the primary altimeter. Also, the redundant main charge will be programmed to go off 100’ lower than the primary main ejection charge, the redundant drogue ejection charge will be programmed to go off one second after the primary drogue ejection charge, and the redundant ejection charges for both parachutes will contain one more gram of black powder than the primary ejection charges. These factors will all work together to contribute to the overall reliability and success in the processes of separation and deployment as prescribed in the mission requirements.

4.3.2. Wiring Diagram (Schematic)

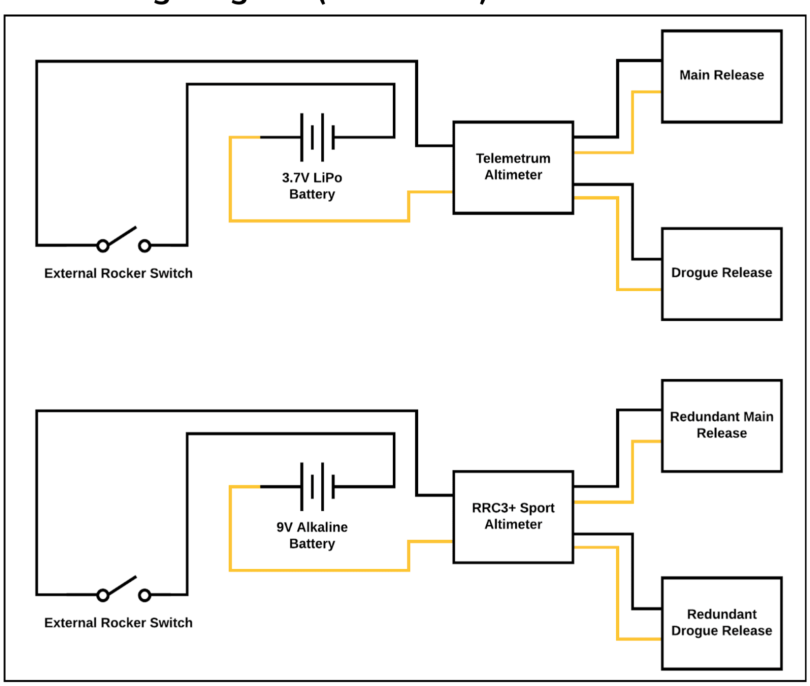


Figure 4.4: Avionics Wiring Diagram

As seen above, each of the altimeters is wired to a corresponding battery to receive power. The Telemetrum altimeter, used for the primary main and drogue release, is connected to a 3.7V LiPo battery and the RRC3+ Sport altimeter, used for the redundant main and drogue release, is connected to a 9V alkaline battery. Furthermore, each altimeter is connected to its own external switch so that it can be turned either on or off from outside the vehicle.

The primary altimeter, the Telemetrum, is wired to the primary main and the primary drogue ejection charges. The ejection charges are used to release a parachute, and, in this case, both the drogue and the main parachute have their own mechanism for release. Similarly, the redundant altimeter, the RRC3+ Sport, is connected to the redundant main and drogue ejection charges. These redundant ejection charges are used to release the corresponding parachute if the primary ejection charges have failed to do so, with a one second delay in release. In the event where there are no failures, the redundant charges will go off harmlessly into the open air. Therefore, all four ejection charges will go off at some point during the fight.

4.4. CAD and Dimensional Drawings

4.4.1. Avionics Bay Assembly and Sub-Assemblies (CAD)

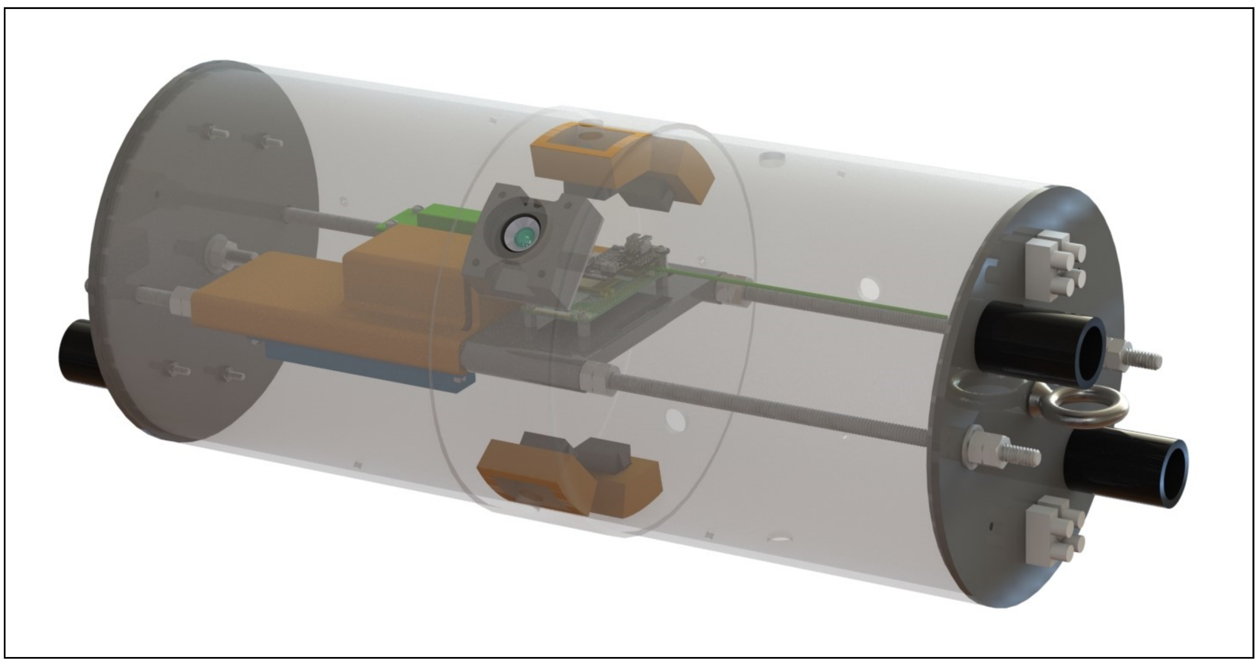


Figure 4.5: Full Avionics Bay Assembly

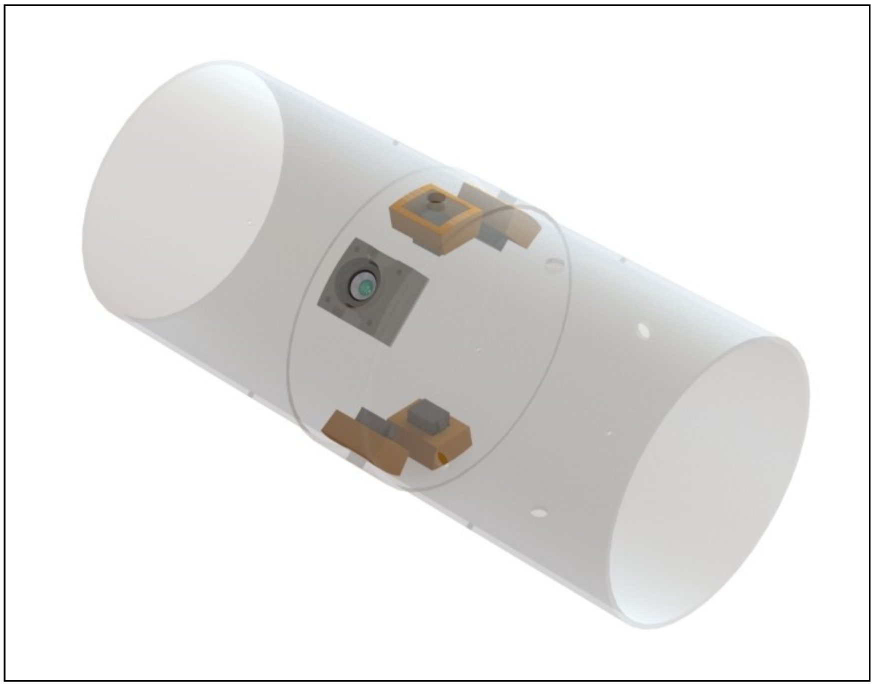


Figure 4.6: Avionics Coupler Assembly

Figure 4.7: Avionics Bulkhead Assembly

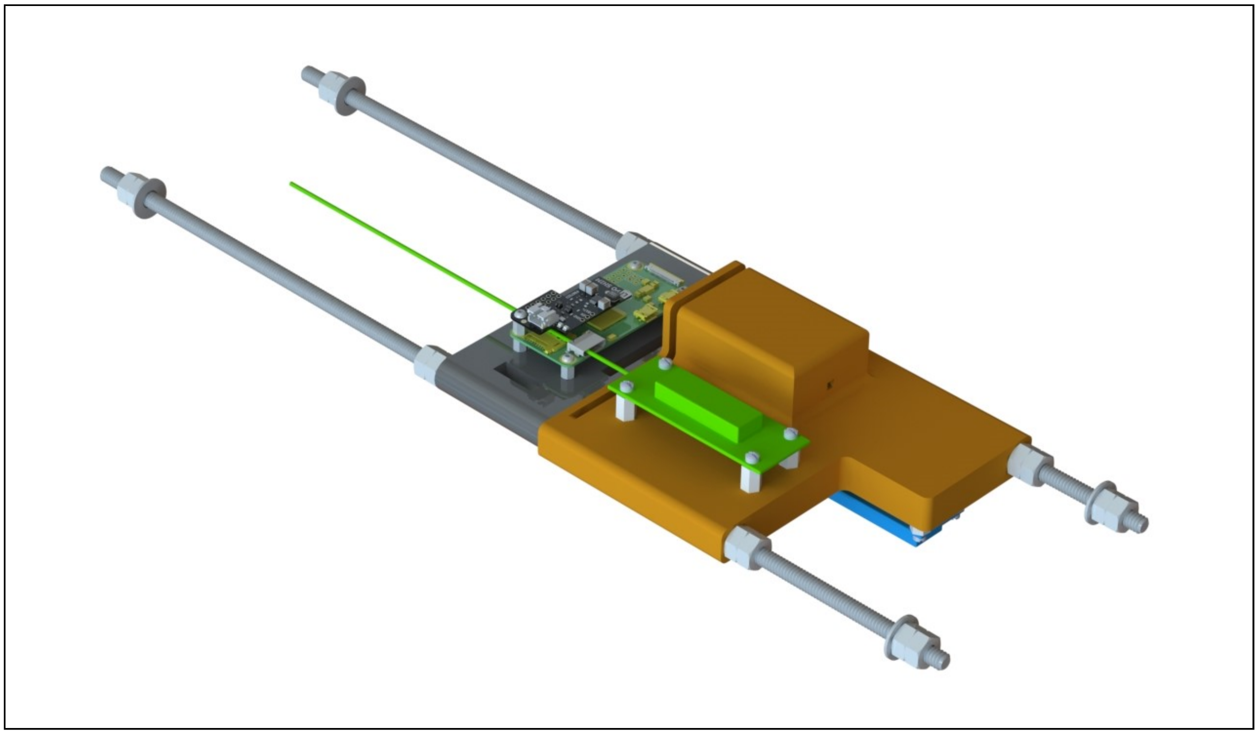


Figure 4.8: Avionics Sled Assembly

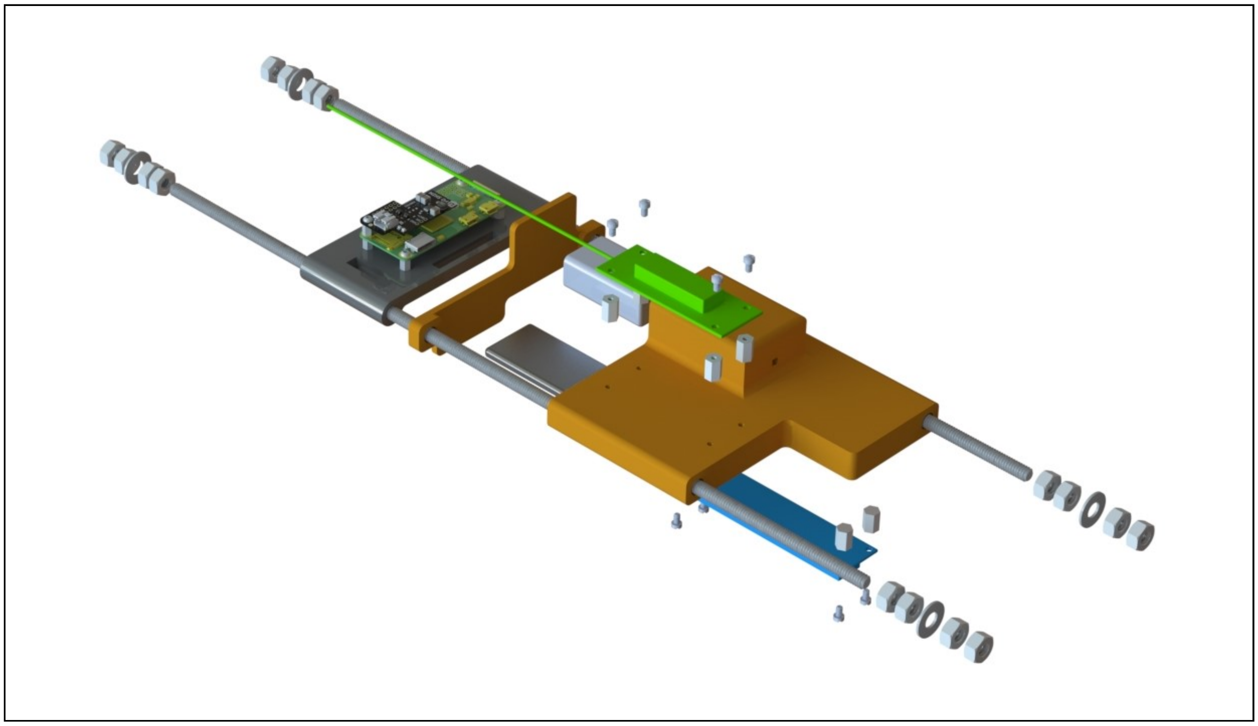


Figure 4.9: Exploded Avionics Sled Assembly

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

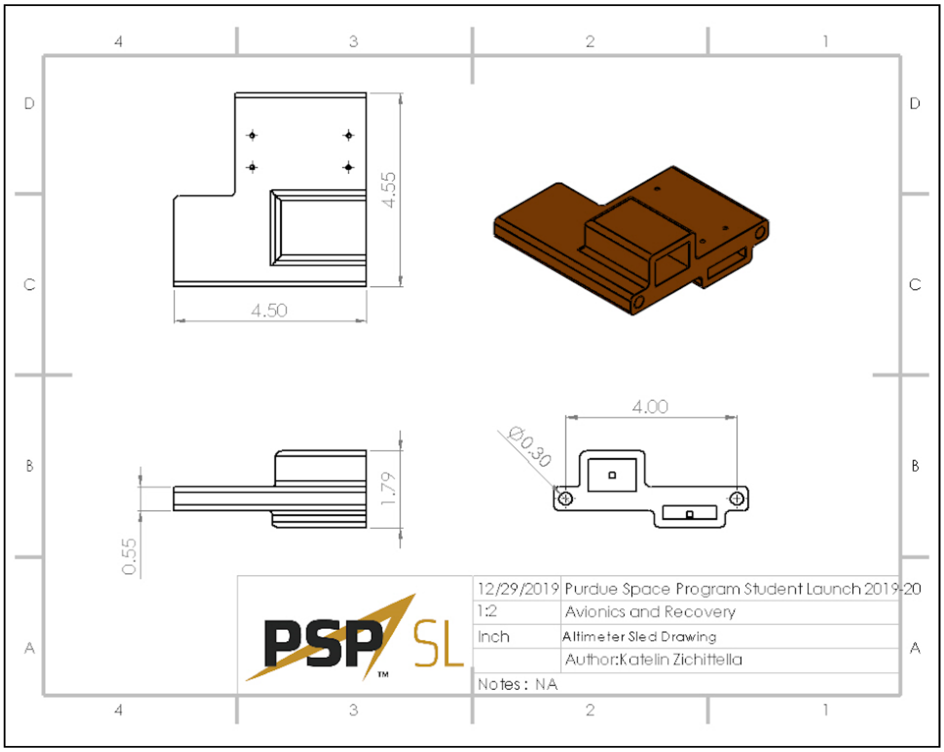


Figure 4.10: Altimeter Sled (With Built-In Battery Compartment) Dimensional Drawing

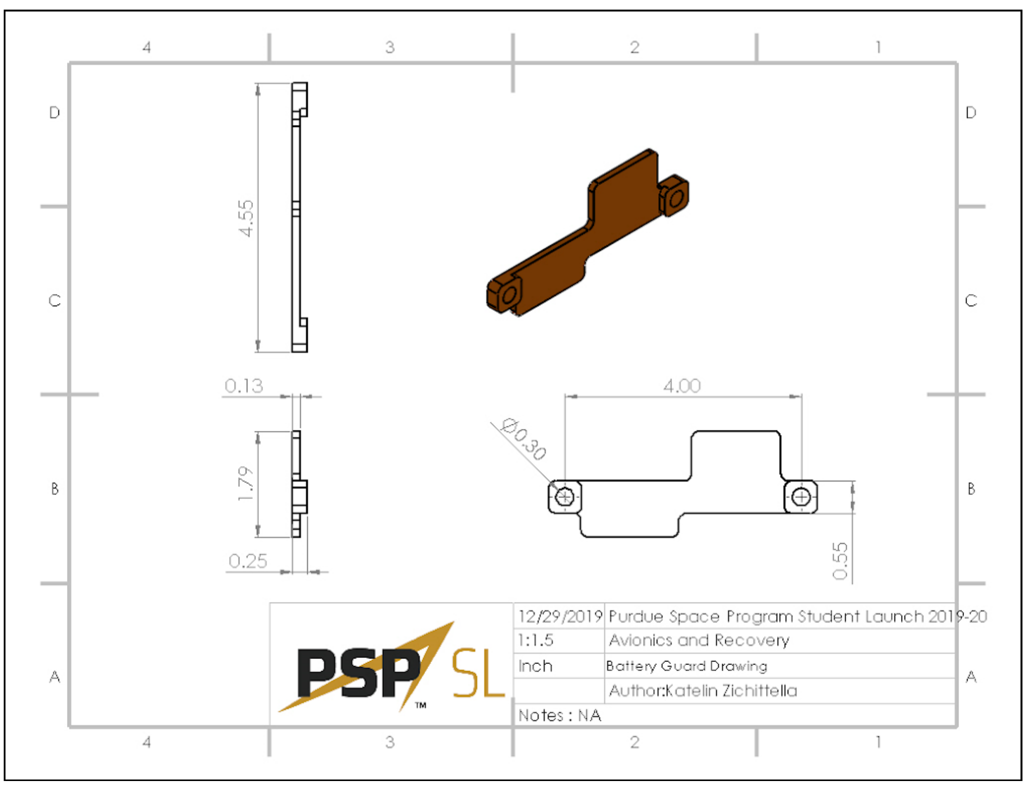


Figure 4.11: Altimeter Battery Guard Drawing

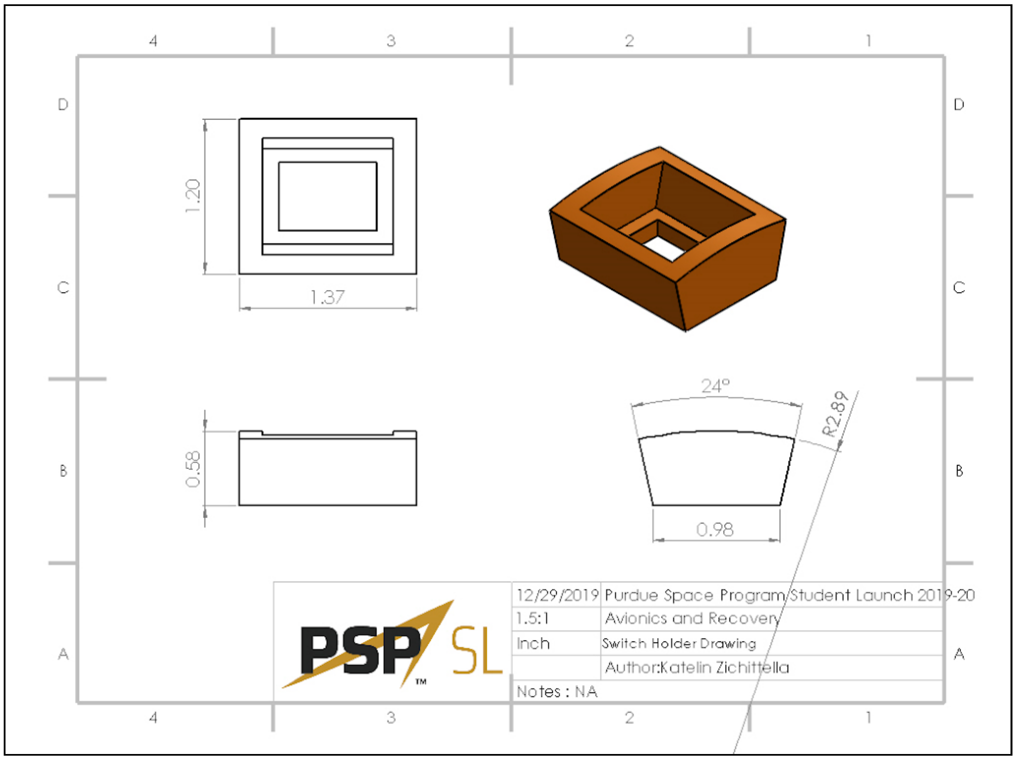


Figure 4.12: Avionics Bay Switch Holder Drawing

4.5. Ejection Charge Sizing and Airframe Pressurization

Both parachutes will be deployed via the use of black powder charges initiated by redundant altimeters. The primary drogue charge will ignite at apogee with the redundant at apogee plus one second, and the primary main charge will ignite at 800' AGL with the redundant at 700' AGL. The primary main charge will contain 5g of FFFFg black powder, and the redundant main charge will contain 6g of black powder (5g + 1g) in order to absolutely ensure complete separation. Similarly, the primary drogue charge will contain 2g of black powder and the redundant drogue charge will contain 3g of black powder.

By calculating the cross sectional area of a single 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.

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

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

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

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

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.

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

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

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

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

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 “open” 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 of the nylon shear pins can be calculated. The final value is multiplied by 1.2 and always rounded up as a safety factor.

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

Upper Airframe Side (Main) Primary Ejection Charge

$$G = Mass_{BP} = 0.006 * (6\ in)^2 * (29\ in - 12\ in) * 1.2 = 4.406 g \approx 5\ grams\ of\ black\ powder$$

(12” is subtracted from the open length of the upper airframe because the large main parachute located there significantly reduces the open volume of the upper airframe)

Upper Airframe Side (Main) Redundant Ejection Charge

$$G = 5g + 1g = 6\ grams\ of\ black\ powder$$

Lower Airframe Side (Drogue) Primary Ejection Charge

$$G = Mass_{BP} = 0.006 * (6\ in)^2 * 4" * 1.2 = 1.037 gs \approx 2\ grams\ of\ black\ powder$$

Lower Airframe Side (Drogue) Redundant Ejection Charge

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

4.6. RF Transmission Frequency of Tracker

The following table lists the RF transmission frequencies used in each of the altimeters.

Telemetrum	435 MHz
RRC3+ Sport	None

Table 4.1: Avionics RF Transmission Frequencies

During the flight, location and altitude telemetry data will be transmitted in real time from the Telemetrum altimeter to a laptop at the ground control station via a TeleBT, TeleDongle, and Yagi Arrow 3 element antenna. The avionics bay is designed to allow RF to transmit through it in that no carbon fiber, which is RF blocking, is used in its construction. Instead, the coupler will be composed of more RF-compliant fiberglass. In addition, no other electronics that could interfere with the signal will be placed in the vicinity of the Telemetrum transmitter.

5. Mission Performance Predictions

5.1. Trajectory Analysis

In the 2019 competition, a large amount of consideration was given towards varying wind speeds. However, the team failed to incorporate launch rail inclination into their simulation data. In this year's competition, the team gave more consideration to launch rail inclination while deciding to spend less time on the in-depth discussion of each wind case. Instead, there are two in-depth cases described while the other cases are tabulated for ease of understanding. The first case is the ideal case of no wind and no rail inclination. Obviously, this is not feasible, but it gives the perfect performance of the team's launch vehicle. Following that case is a 10mph average wind speed case with 10 degrees of launch rail inclination. Looking up average wind speed in early April in Huntsville, Alabama, the expected winds should be between 7mph and 9mph. Using a slightly higher speed should give the team more room for error. In addition, the rail is likely to be inclined away from the spectators, so the team decided that 10 degrees is the most likely case.

5.1.1. OpenRocket Simulations

10 Degree Incline, 10mph

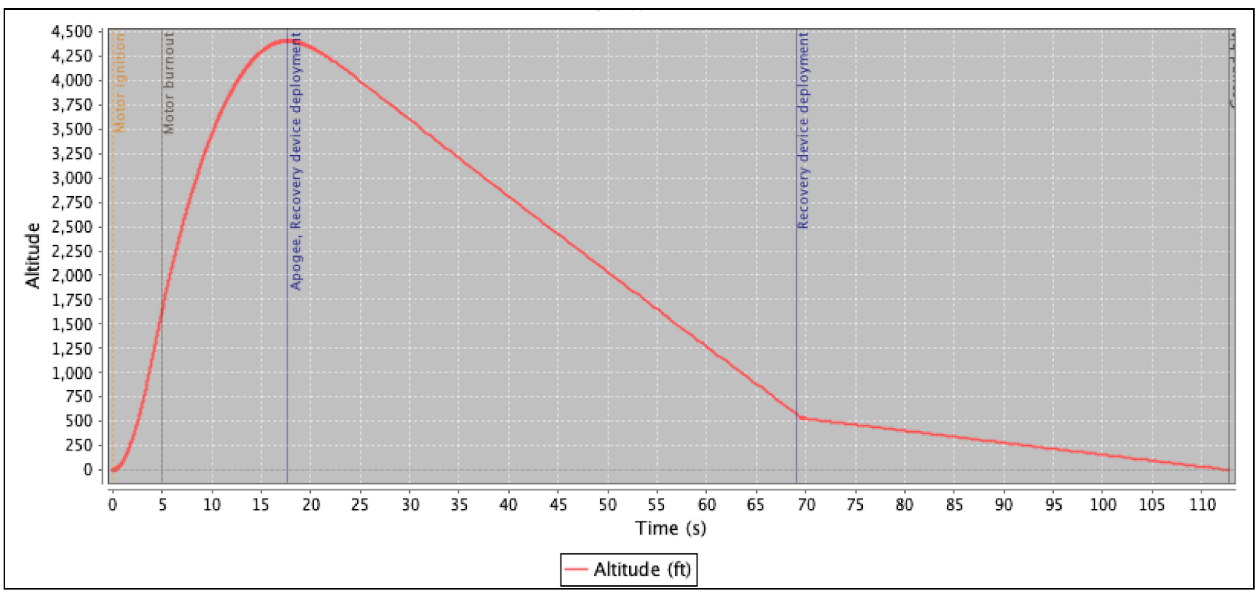


Figure 5.1: OpenRocket Simulation of Hypothesized Scenario for Huntsville Launch

As can be seen from the graph above, our rocket is simulated to reach a maximum altitude of 4406' above ground level. This is slightly above our target altitude of 4325' above ground level. Once our group has constructed our flight vehicle, we will have a more accurate weight measurement for the rocket that can then be entered into the simulation program. Because the rocket is anticipated to weigh more than the simulation shows, as weight is added into the computer model our ballast will decrease. From previous builds, the team has learned that the epoxy used to attach the separate components of the launch vehicle adds a considerable amount of weight which will change the launch altitude of the vehicle.

15 Degree Incline, 20mph

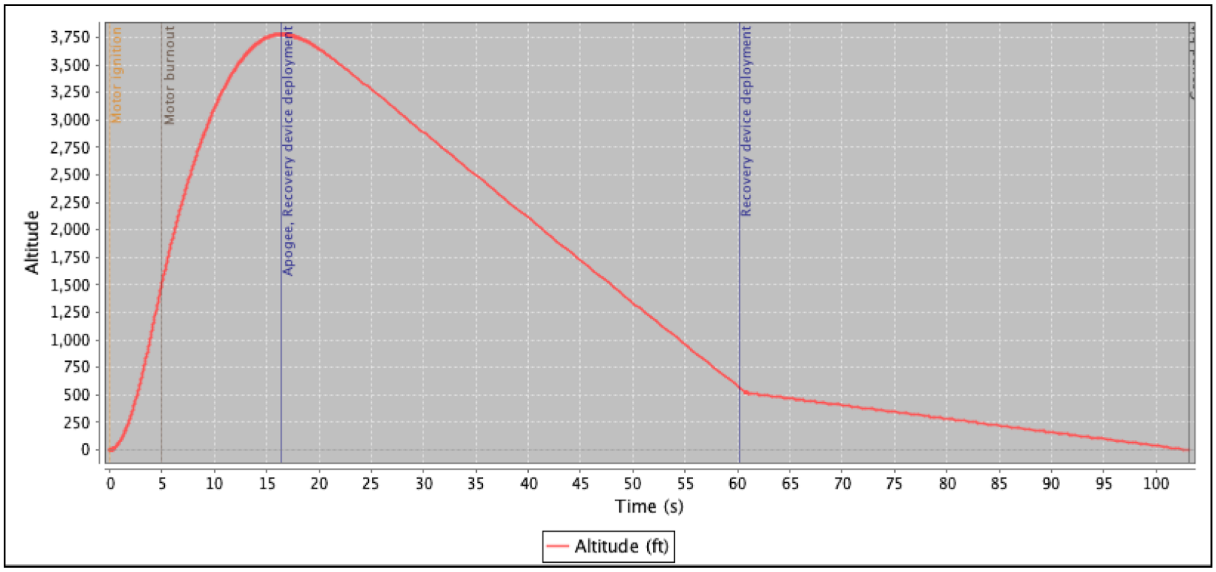


Figure 5.2: 15 Degree Inclination, 20mph Winds OpenRocket Simulation

The case of 15 degree inclination and 20mph winds is the worst case scenario that the team simulated. In this case, the max altitude the vehicle would reach is 3778' which is considerably lower than the team's target altitude. This case was simulated in order to assess the worst performance of the vehicle depending on the circumstance. This case is very unlikely to occur as the current prediction of launch day winds is around 5-10mph.

0 Degree Incline, 0mph

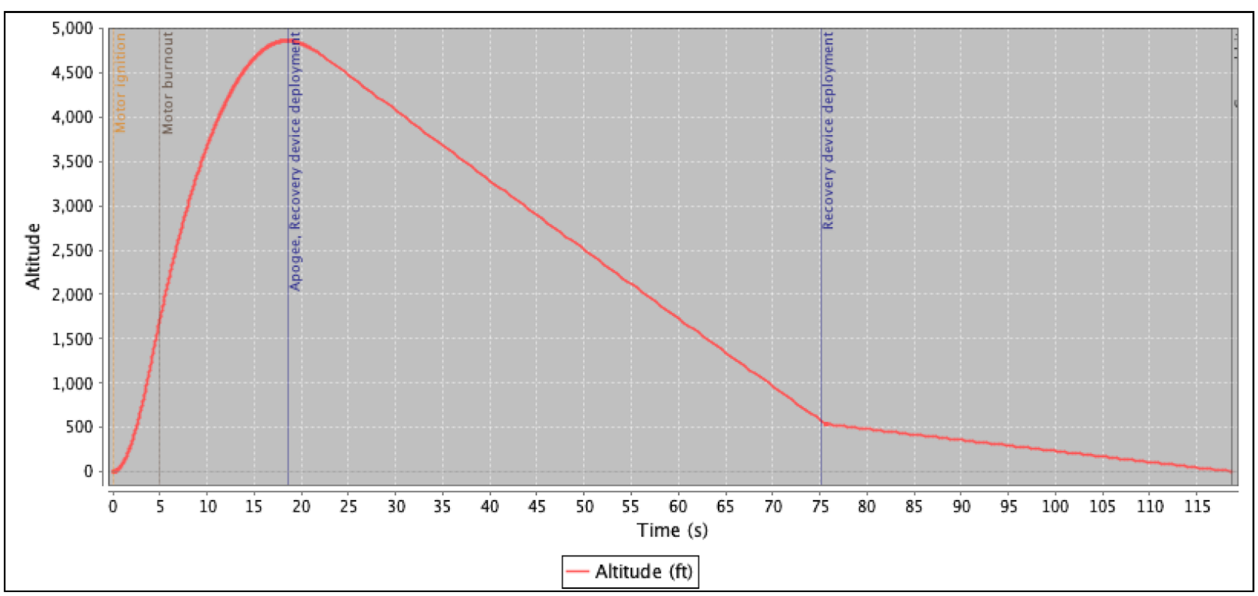


Figure 5.3: OpenRocket Simulation of Ideal Scenario for Huntsville Launch

The case of 0 degree inclination and 0mph winds is the best case scenario in terms of maximizing altitude. In this case, the max altitude the vehicle would reach is 4866' which is considerably greater than the team's target altitude. This case was simulated in order to assess the best performance of the vehicle depending on the circumstance. This case is very unlikely to occur for the same reasons as the 20mph case. However, it is important to know the range of performance for the launch vehicle as the launch day conditions have a certain degree of uncertainty.

5 Degree Incline, 10mph

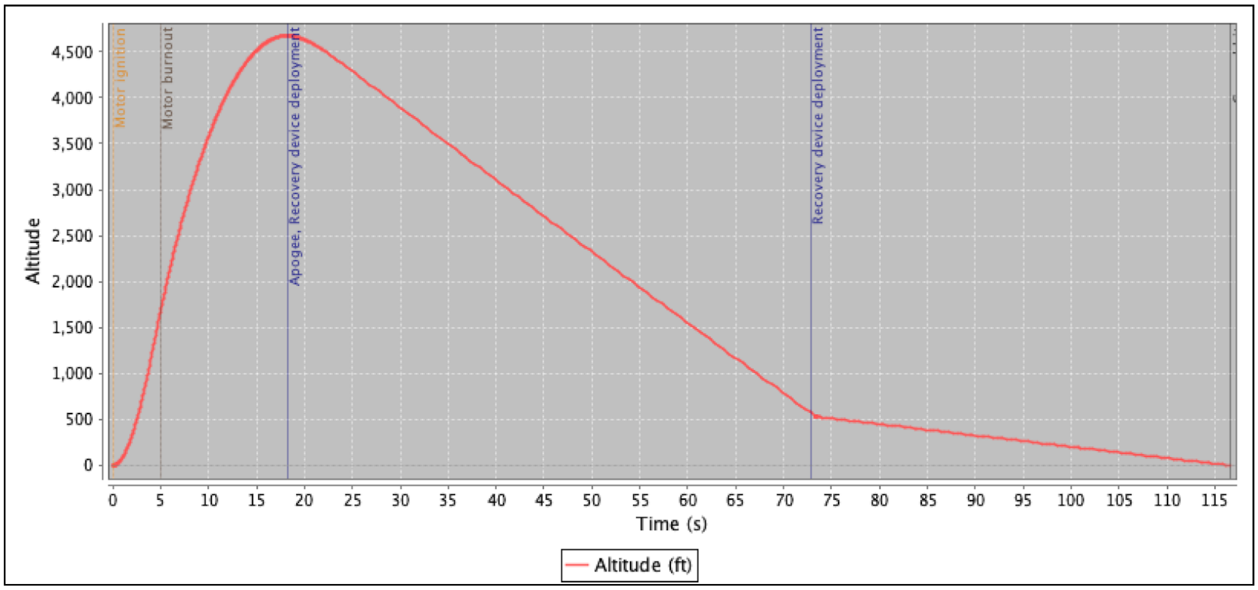


Figure 5.4: 5 Degree Inclination, 10 mph Winds OpenRocket Simulation

The case of 5 degree inclination and 10mph winds is a case that is more realistic than the ideal scenario that the team simulated. In this case, the max altitude the vehicle would reach is 4672' which is considerably greater than the team's target altitude. This case was simulated in order to assess the best performance of the vehicle depending on the circumstance. This case is more likely to occur for the same reasons as it is closer to the ideal case that we are predicting for the launch day.

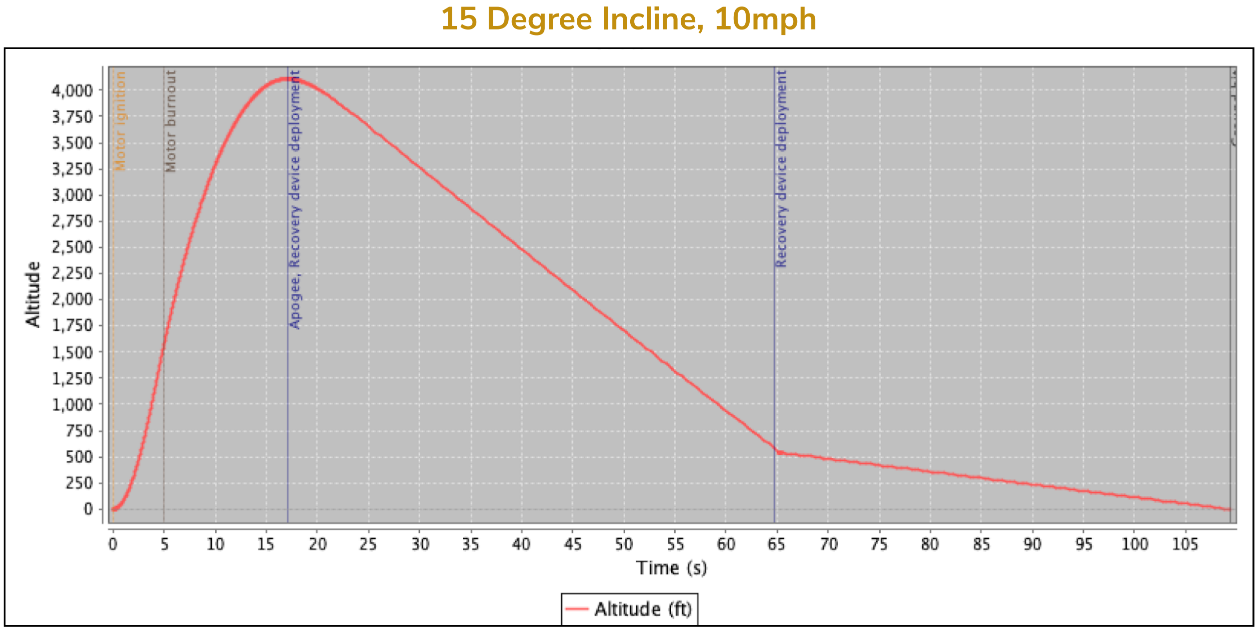


Figure 5.5: 15 Degree Inclination, 10mph Wind OpenRocket Simulation

The case of 15 degree inclination and 10mph winds is a case that is worse than the ideal case scenario that the team simulated. In this case, the max altitude the vehicle would reach is 4108' which is considerably greater than the team's target altitude. This case was simulated in order to assess the best performance of the vehicle depending on the circumstance. This case is more likely to occur for the same reasons as it is closer to the ideal case that we are predicting for the launch day. This case also demonstrates how the inclination can affect the apogee on launch day which provides an insight into how small changes in the launch day condition can create a large change in the apogee of the launch vehicle.

Other factors, such as surface finish and the cross sectional airfoil of the fins, are variables that we do not have implicit control over. Our team cannot accurately measure surface smoothness to compare the real and digital models, which will account for some difference in our actual and expected altitudes. In addition, the only choices presented to us when varying the fin's cross section are "square, rounded, or airfoiled". There is no direct input for edge thickness or taper length, further limiting our simulations.

All altitude simulations from which the graph above is derived were accomplished using OpenRocket 15.03 using the extended Barrowman calculation method and a six degree of freedom Runge Kutta 4 simulation method. Geodetic calculations were evaluated using spherical approximation, and a 0.02s time step for simulation calculations was used. Further altitude calculations will be done in RASAero II using similar parameters, and will be discussed in the next section.

5.1.2. RASAero Verification

To further the analysis of inclinations and varying wind speeds, RASAero was used to verify the results. This year, it was seen as very important to look at the results of the given instances, so ensuring accuracy is of utmost importance. The results yielded from the incline and wind speed stimulations in RASAero found consistencies with OpenRocket as different conditions led to a higher altitude, with a slight amount of error. As mentioned previously, a better understanding, and more accurate simulations will be run when a launch rail is assembled, so there is a better understanding of the weight.

10 Degree Incline, 10mph

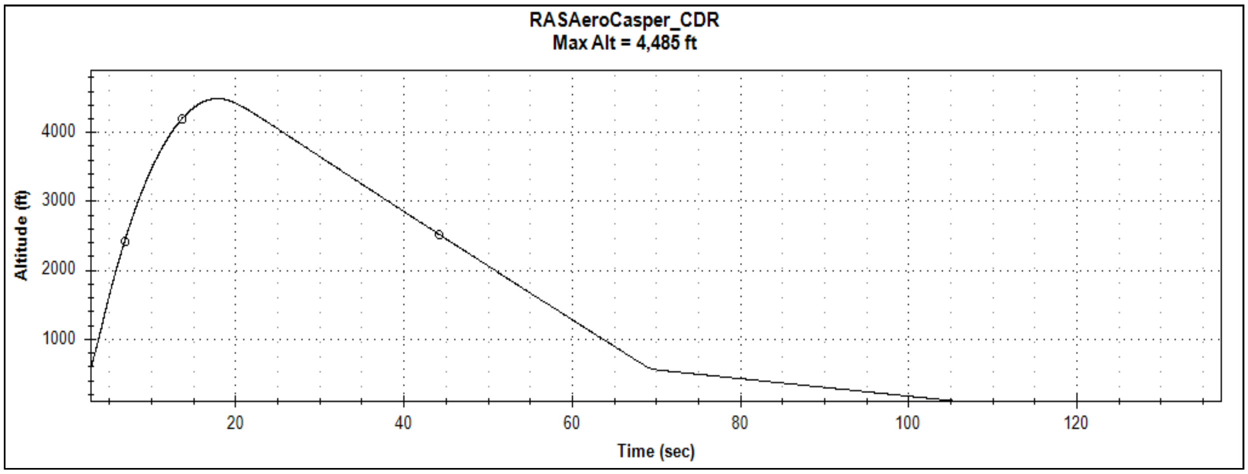


Figure 5.6: RASAero Simulation of Hypothesized Scenario for Huntsville Launch

The above tested case used a launch rail incline of 10 degrees, with an anticipated wind speed of 10mph. The simulation would run a less than ideal scenario, with an increased wind speed affecting the final altitude. With a launch rail incline of 10 degrees, and with 10mph winds, a maximum altitude of 4485' is anticipated.

15 Degree Incline, 20mph

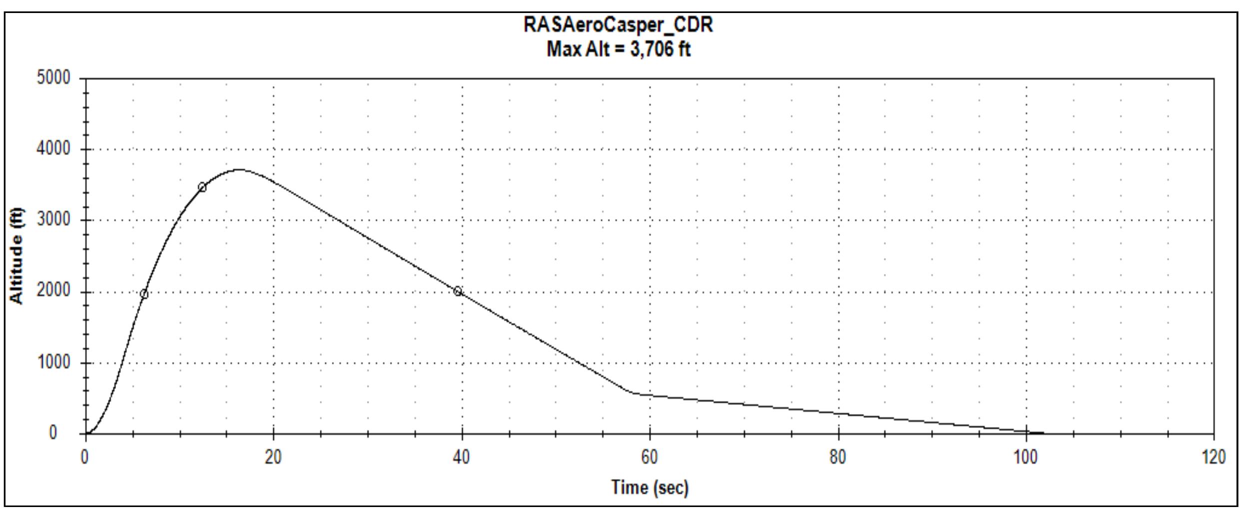


Figure 5.7: 15 Degree Inclination, 20mph Winds RASAero Simulation

Upon testing a 15 degree incline with 20mph winds, an altitude of 3706' was yielded. This was a worst case scenario for the launch conditions of the rocket as set by the competition. As expected, the conditions led to the lowest altitude of any of the cases.

0 Degree Incline, 0mph

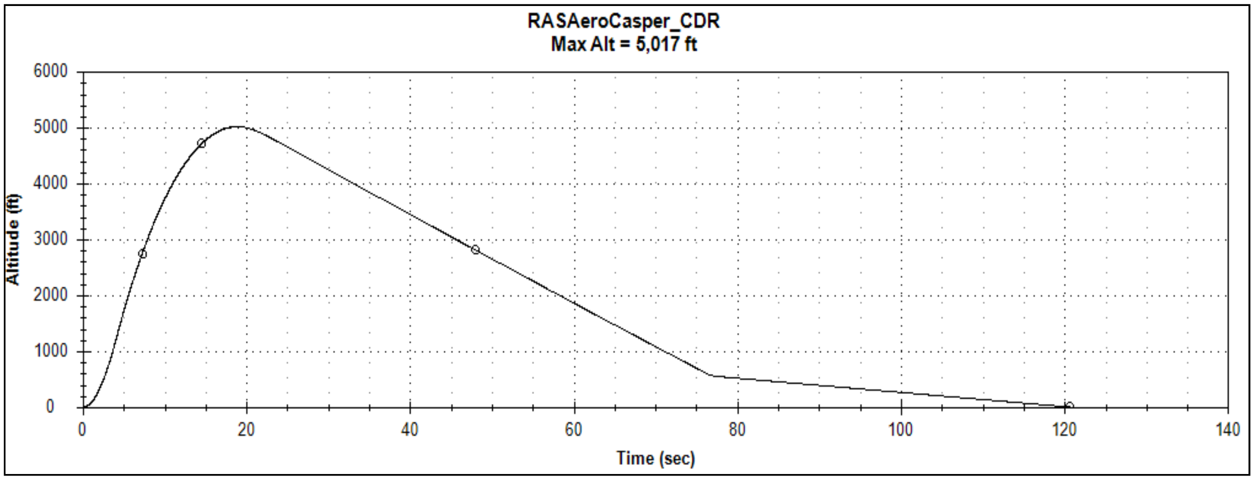


Figure 5.8: RASAero Simulation of Ideal Scenario for Huntsville Launch

For a 0 degree launch rail incline, and the absence of any winds, as expected the highest altitude is yielded. The altitude is significantly higher than the teams targeted, but, as mentioned above, this is the most ideal situation which is incredibly unlikely to occur on launch day. The stimulation had an altitude of 5017'. A consistency among the simulation software was that this case led to the highest altitude despite software nuances leading to a deviation.

5 Degree Incline, 10mph

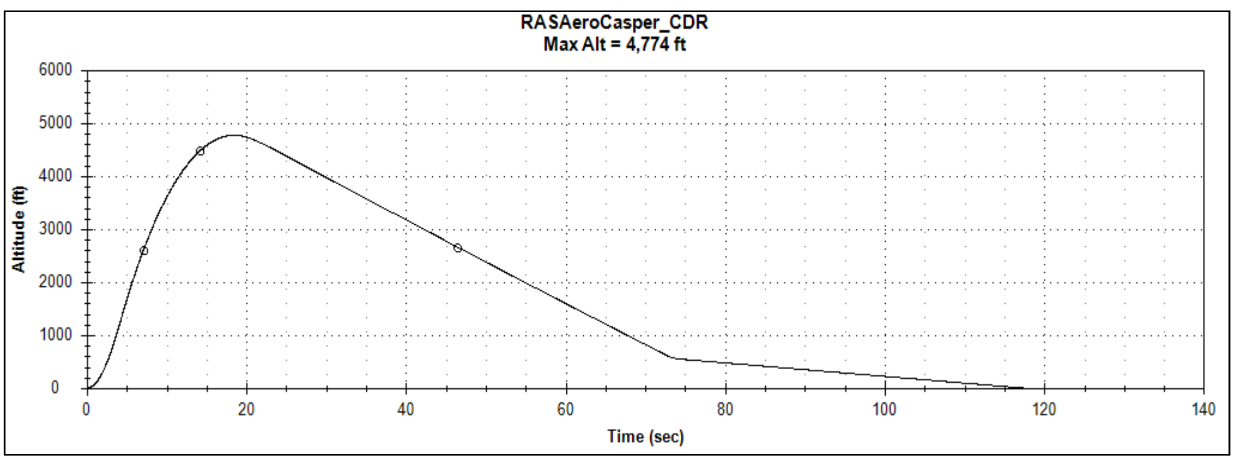


Figure 5.9: 5 Degree Inclination, 10 mph Winds RASAero Simulation

The situation of a 5 degree incline with 10mph winds led to an altitude of 4774'. This is one of the more likely situations to occur for the launch day, so keeping in mind the results of the simulation will give a good understanding of the rockets launch behavior being above the targeted altitude.

15 Degree Incline, 10mph

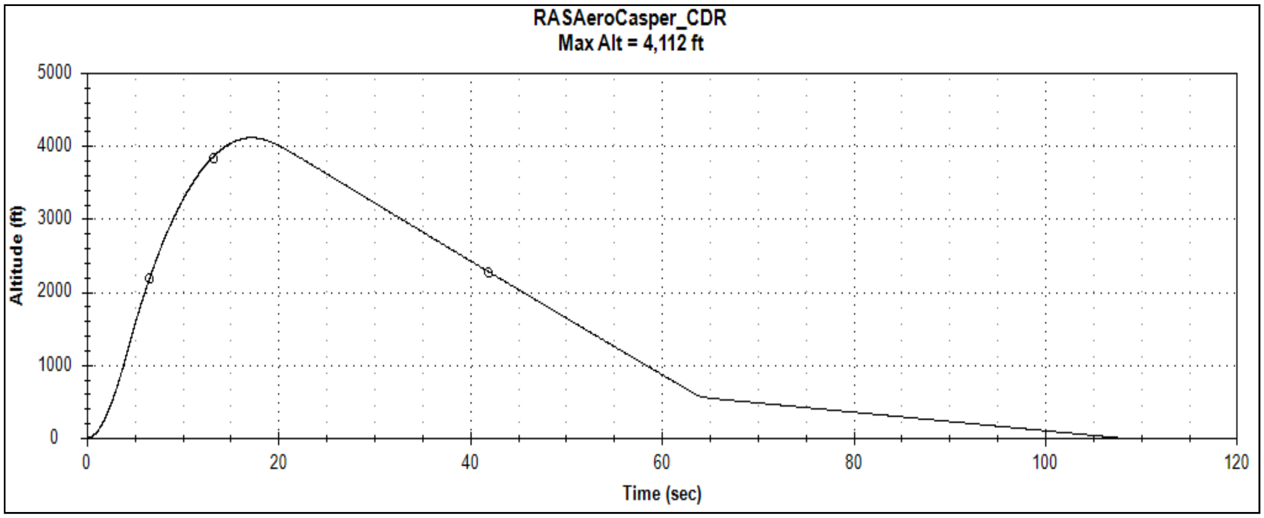


Figure 5.10: 15 Degree Inclination, 10mph Winds RASAero Simulation

This case would anticipate a less than ideal scenario for the rockets anticipated launch. The altitude would come to 4112', which slightly below the targeted altitude.

Using both simulation software to test the conditions of differing inclines, and wind speeds helps to provide the team with a broader understanding of the results that would be yielded for the cases. The differences among each case would tend to be within 100', and RASaero would always create higher results. This can be seen by comparing the data generated for the 15 degree incline and 10mph wind speed beng 4112', while open rocket had an altitude of 4108'. Of altitude, apogee, and drift, altitude had the lowest deviation with software being around 10', then drift fell within 50', while apogee tended to 250'.

Upon going and exploring this deviation, a few people on a rocketry forum found that an apogee deviation of 6% can be found, which comes from the differing ways that each software takes into account weights of components like body tubes.

Exploring the results of different software is always important, as relying on just one source to provide feedback doesn't assess all of the possible outcomes that a certain case can generate. It is always important to cross reference the results of software to ensure that each quirk a software may have to obtain results are understood and compared.

5.1.3. Trajectory Code

A MATLAB program was created to verify the values obtained from the OpenRocket simulations. The code also accounts for different launch angles and different wind speeds. Three different launch angles (0, 5, and 10 degrees) and three different wind speeds (0, 5, and 10 mph) were all tested. The code takes into account the temperature at launch (all of the tests below were run at 25C (77F)) as well as the change in air density with altitude. The thrust curve that was found from previous experimental data was also utilized, and the values were linearly interpolated to get a continuous graph.

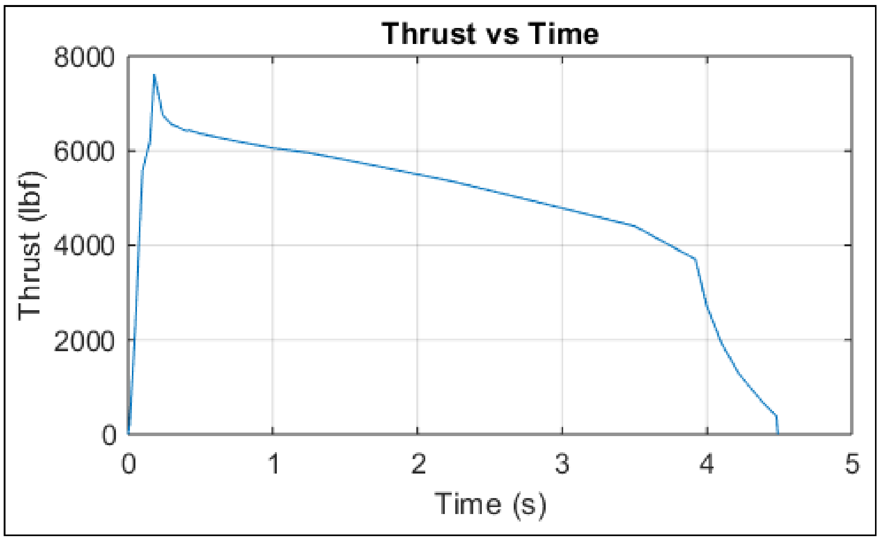


Figure 5.11: Interpolated Motor Thrust vs Time Plot

The code is split into four parts for the four phases of the flight: boosting phase, coasting phase, drogue parachute phase, and main parachute phase.

Boosting Phase: During the boosting phase, the launch vehicle's position is updated for a time step of 0.1s. Only the forces of gravity, drag and thrust are acting in this phase. This phase ends when the motor stops burning.

Coasting Phase: During this phase the launch vehicle does not experience thrust from the motors, so the only forces that the vehicle experiences are drag and gravity. Once the velocity becomes 0ft/s, the code moves onto the next phase.

Drogue Parachute Phase: In this phase, the drogue parachute is deployed, and the drag from the drogue parachute is also added to the other forces. This phase ends when the launch vehicle reaches an altitude of 800'.

Main Parachute Phase: In this phase the main parachute is deployed. It was assumed that the main parachute's radius increases linearly with time, and the total time it takes for the parachute to open was estimated to be 0.2s. With this, the drag from the main parachute is also calculated until the altitude reaches 0'.

Altitude vs Time Plots

0mph Wind Speed, 25C

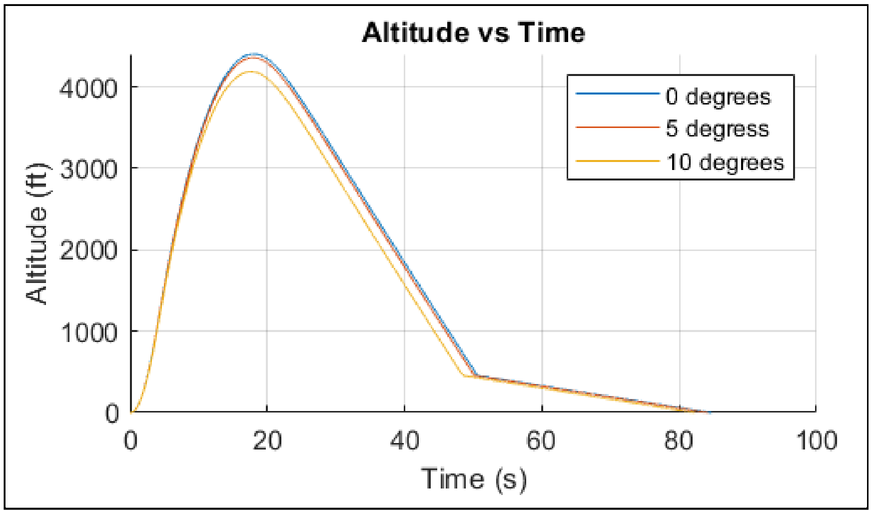


Figure 5.12: 0 mph Wind Speed Altitude vs Time Plot

The apogee is 4461' with a 0 degree launch angle, 4360' with a 5 degree launch angle, and 4189' with a 10 degree launch angle.

5mph Wind Speed, 25C

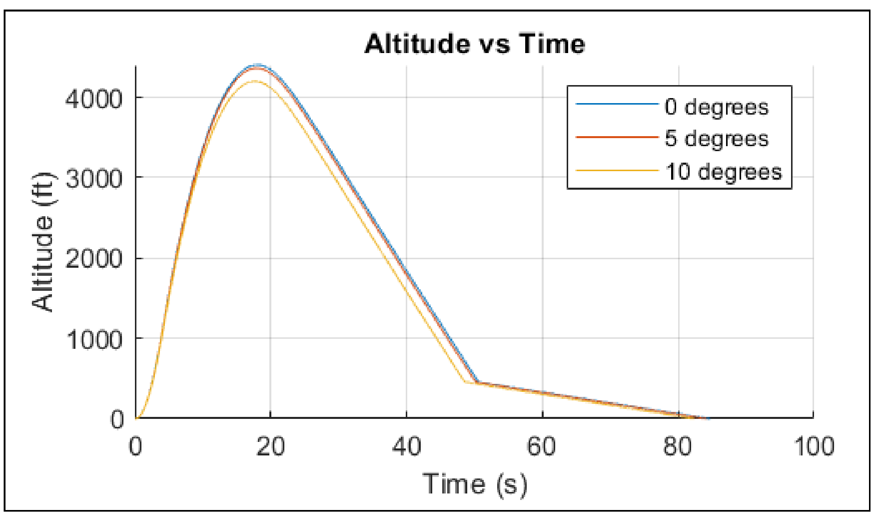


Figure 5.13: 5mph Wind Speed Altitude vs Time Plot

The apogee is 4406' with a 0 degree launch angle, 4365' with a 5 degree launch angle, and 4202' with a 10 degree launch angle. The apogees of the 5 and 10 degree launches are higher for 5mph wind than 0mph wind. This could be due to the vehicle getting pushed by the wind instead of being opposed by it.

10mph Wind Speed, 25C

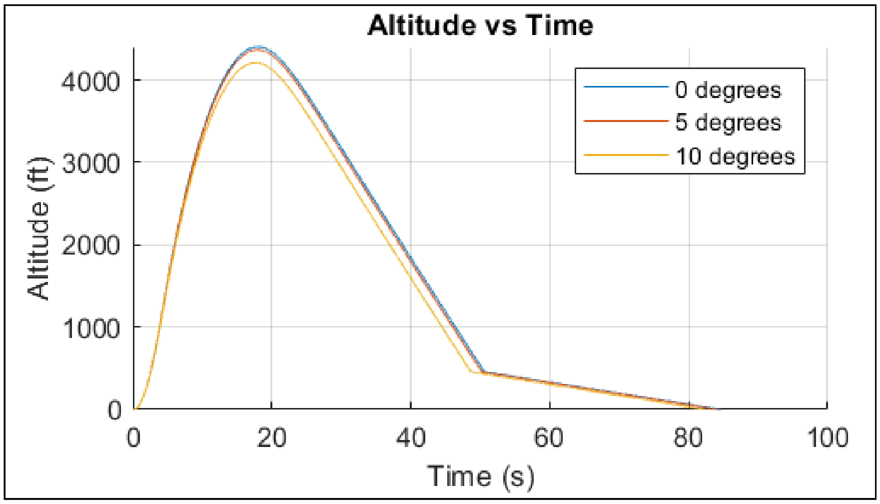


Figure 5.14: 10mph Wind Speed Altitude vs Time Plot

The apogee is 4406’ with a 0 degree launch angle, 4369’ with a 5 degree launch angle, and 4213’ with a 10 degree launch angle.

5.1.3.1. Subscale Launch Verification

The subscale launch was used to verify the accuracy of the MATLAB code.

Actual Subscale Data

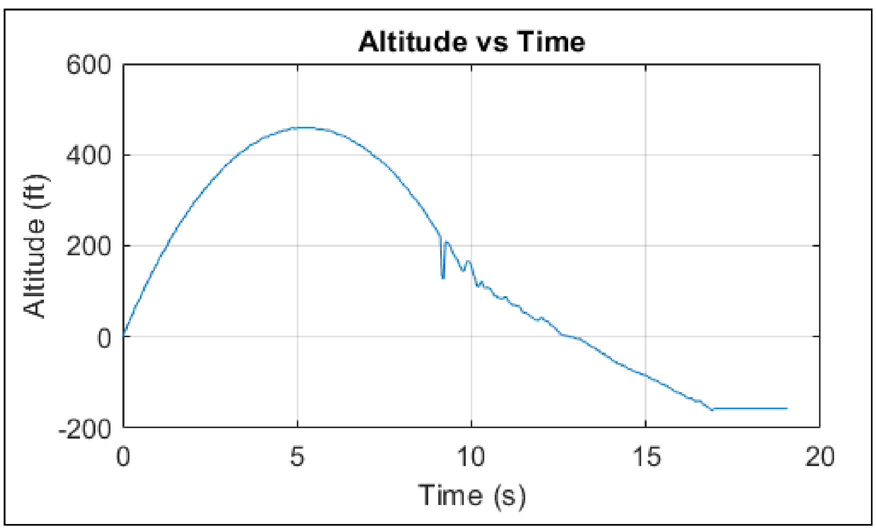


Figure 5.15: Recorded Subscale Altitude vs Time Plot

The subscale launch took place at an elevation of 698’, with 12mph winds, with a 5 degree launch angle into the wind, and at a temperature of 42F. The H114 motor thrust curve was found using experimental data.

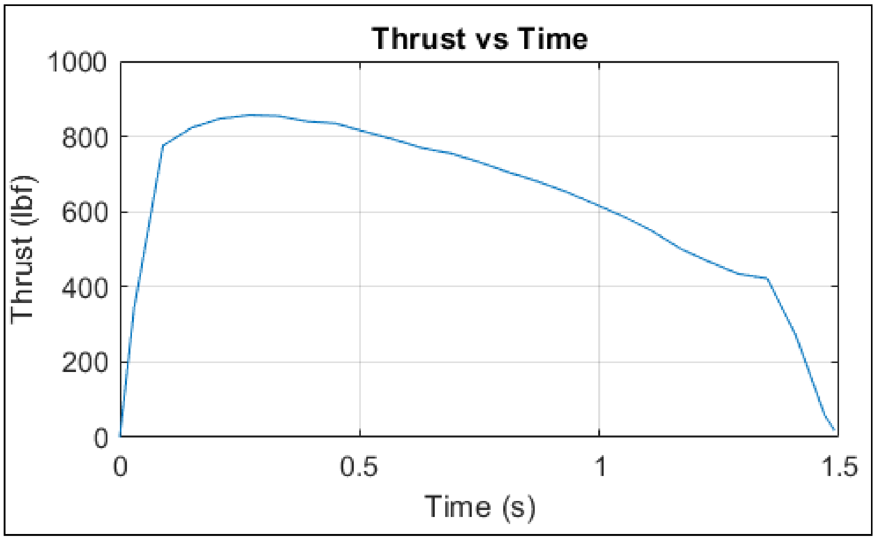


Figure 5.16: Subscale Thrust vs Time Plot

The apogee recorded by the altimeter was 617’.

Using these as the launch conditions, the MATLAB code was able to predict the flight path.

Predicted Subscale Data

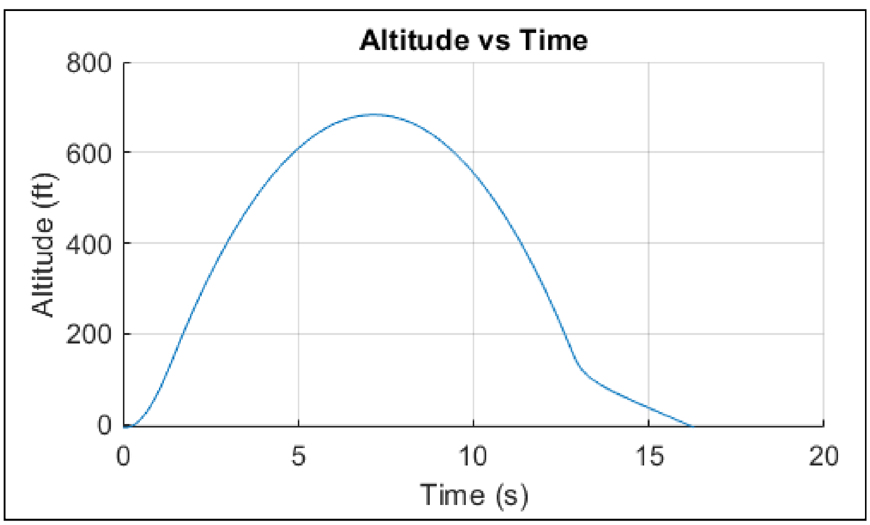


Figure 5.17: Predicted Subscale Altitude vs Time Plot

The predicted apogee is 683’. There is a 10.6% error between the actual and predicted value for the apogee. This could be because of the way the angle is calculated. The MATLAB code assumes a constant angle, only depending on the initial launch angle. The code does not account for the drag and rotation caused by the launch vehicle being at an angle. Once these errors are rectified, the predicted results should be more accurate compared to the actual results.

5.2. Vehicle Characteristics

5.2.1. Stability Versus Time

5.2.1.1. OpenRocket

0 Degree Incline, 0mph

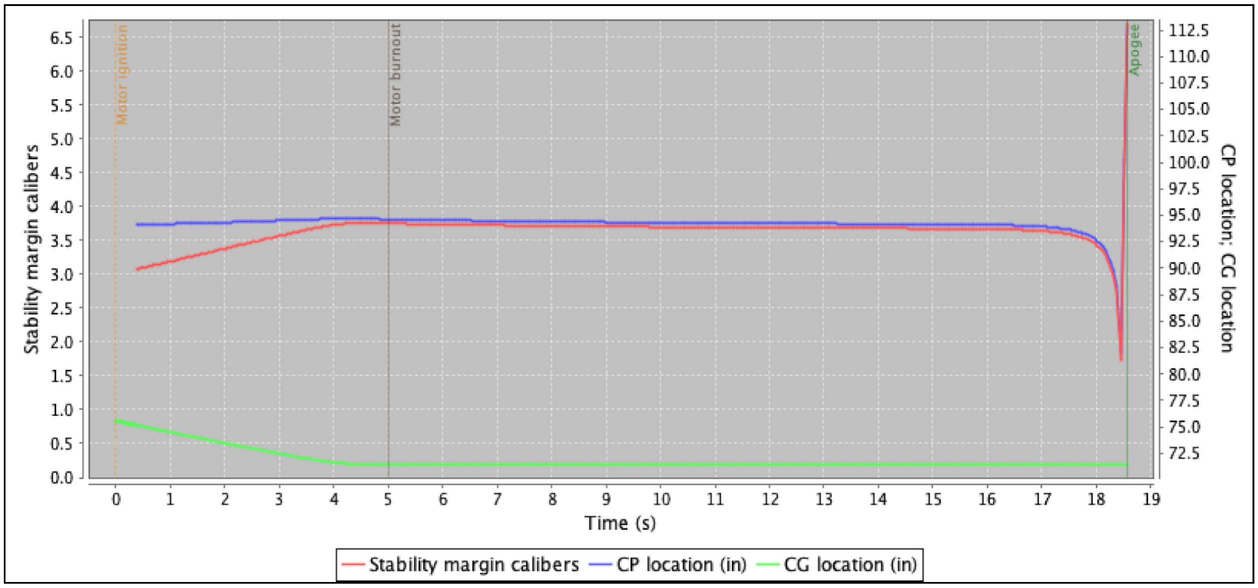


Figure 5.18: OpenRocket Stability vs Time Simulation of Ideal Case

As seen from the graph above, the launch vehicle case exits the 144” launch rail with a minimum stability margin of 3.08cal, meeting the minimum requirement of two calibers. During the ascent phase, the launch vehicle does not experience a significant drop in stability until it reaches a low enough velocity that the fins cannot maintain aerodynamic stability. At this point, the launch vehicle begins slowing down significantly due to drag and gravity and starts arcing over as it approaches apogee. Despite this, the launch vehicle maintains above 3.5cal for nearly all of the boost and coast phase.

The center of pressure, the node where the total sum of all pressures acts on the vehicle, starts at a distance of 94.348” from the datum, which is deemed to be the tip of the nose cone. The center of gravity, a node where all moments about an axis of rotation equally oppose each other, begins at a distance of 75.569” from the datum of the launch vehicle, placing it 18.779” ahead of the center of pressure. During the burn time of the motor, the center of gravity moves forward at a constant rate due to the constant burn rate of the solid propellant. The total shift is 4.101”, or almost one full caliber.

5.2.1.2. RASAero Verification

0 Degree Incline, 0mph

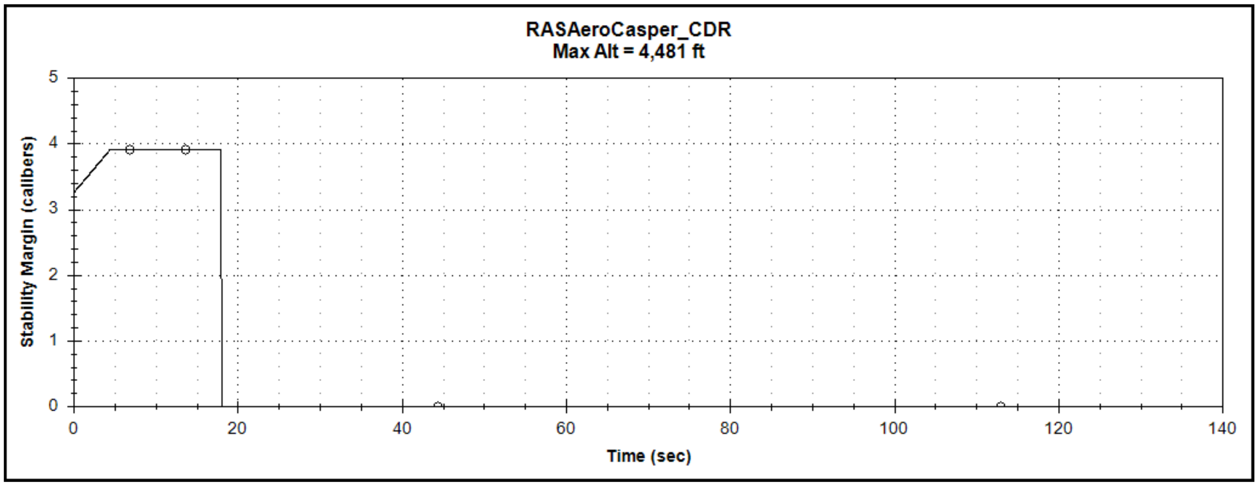


Figure 5.19: RASAero Stability vs Time Simulation of Ideal Case

The RASAero verification shows that the launch vehicle is never below the required 2.0cal of stability required off the launch rail. The vehicle starts at 3.29cal according to RASAero, implying a slight differential between its results and that from OpenRocket. However, both programs show the stability margin requirement is met.

Over the course of the flight, the stability margin increases as expected, to a value of 3.91cal. To get these values, RASAero needs to input the center of gravity. This value was taken from the OpenRocket model, implying some error. With this in mind, the center of pressure was calculated within RASAero, therefore it must be accurate to its means. CP from RASAero was calculated to be 95.04” from the aft end whereas OpenRocket found it to be 94.348” from the aft end. This slight difference begins to explain the error between the initial stability margin in both programs.

5.2.1.3. Avionics & Recovery Code

A stability vs time plot was generated from the MATLAB trajectory code to verify the results obtained from OpenRocket.

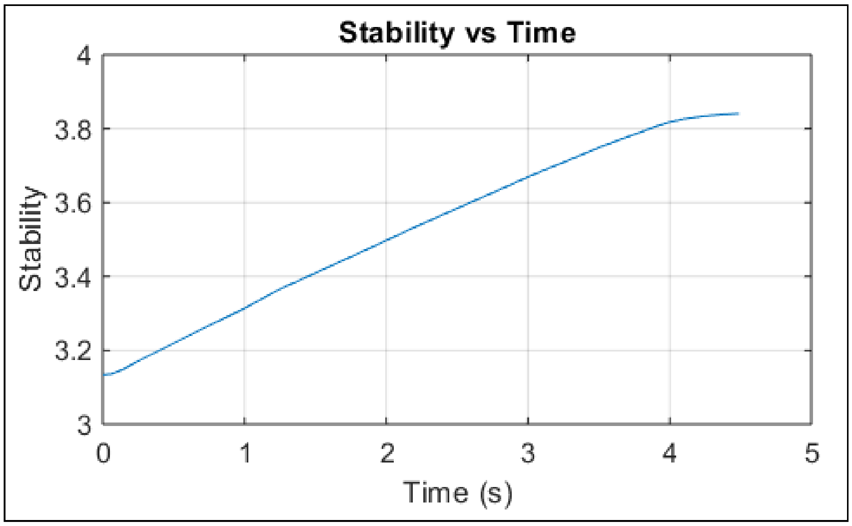


Figure 5.20: MATLAB Trajectory Code Stability vs Time Plot

The plot above shows the stability vs time of the vehicle from launch until the time when the motor shuts off. The plot takes into account the non uniform change in mass of the rocket while ascending and assumes that the center of pressure stays constant throughout this phase. The stability starts at 3.09cal, and, at the end of this phase, reaches almost 3.8cal.

5.2.2. Drag Versus Time

0 Degree Incline, 0mph

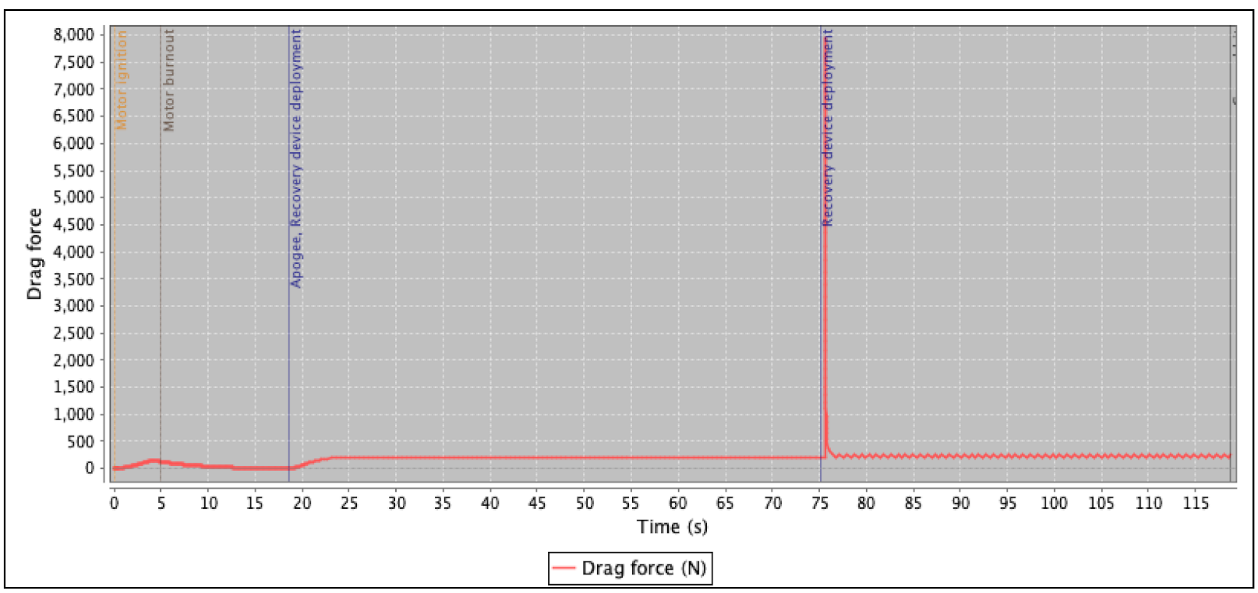


Figure 5.21: OpenRocket Drag vs Time Simulation of Ideal Case

As shown by the figure above, the drag force stays around 213N during the majority of the mission. The drag spikes when the parachute is deployed which is expected given the nature and purpose of the parachute. According to OpenRocket, the wind speed is considerably less than the velocity of the launch vehicle so the different cases with wind speed and inclinations have little impact on the drag force according to the simulation.

5.2.3. Drift Distance Estimations & Hand Calculations

To calculate drift distance, the team used the equation stating drift distance equals the vehicle's descent time multiplied by the wind speed. This equation assumes that the wind blows in only one primary direction during descent. The 20mph case is over the allowed 2500' drift distance of the competition, but the team also recognizes that the chance it is both allowed to launch in 20mph winds is low, and the distance is under 2500' for 19mph wind speeds.

5.2.3.1. OpenRocket Drift Estimation

0 Degree Incline, 0mph

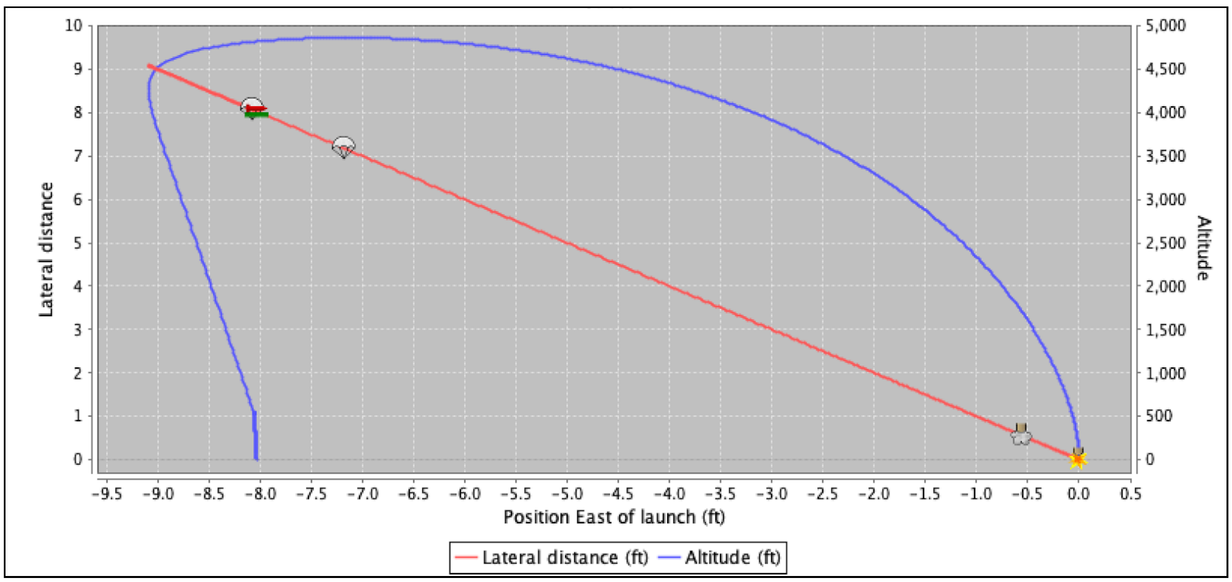


Figure 5.22: OpenRocket Simulation of Ideal Drift Distance Case

With an average wind speed of 0mph with 0.5mph standard deviation and 10% turbulence intensity, our simulated maximum drift distance during flight is roughly 9'. The launch vehicle travels nearly 9' west of the launch site as it tilts into the wind.

0 Degree Incline, 5mph

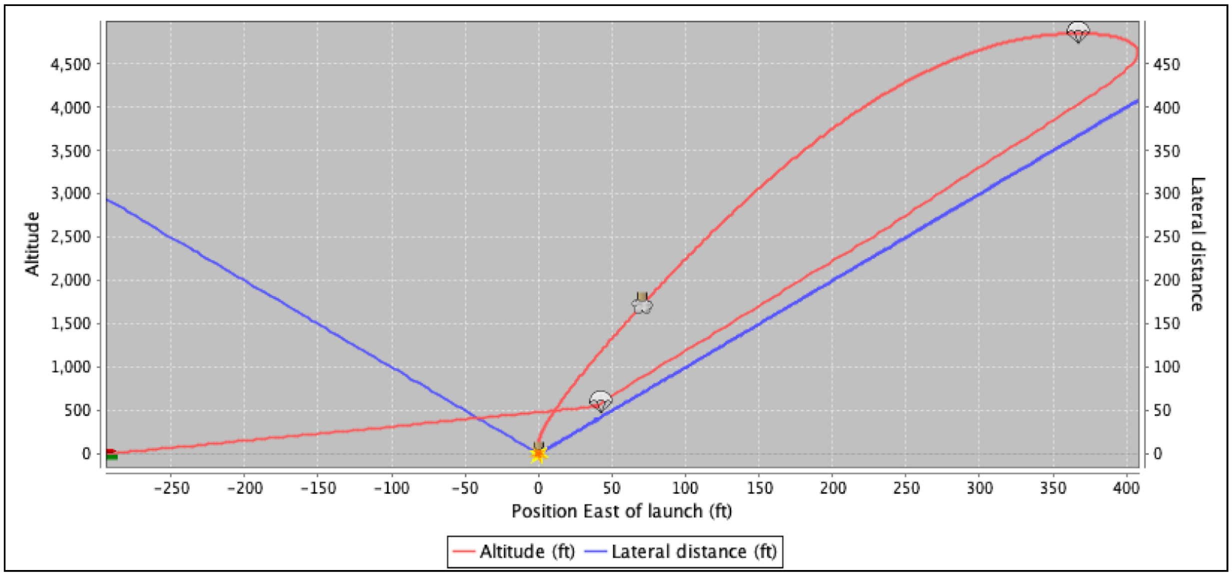


Figure 5.23: 0 Degree Inclination, 5mph Winds OpenRocket Drift Distance Simulation

With an average wind speed of 5mph with 0.5mph standard deviation and 10% turbulence intensity, our simulated maximum drift distance during flight is roughly 410’. The launch vehicle travels nearly 410’ east of the launch site as it tilts into the wind, then drifts back over the launch site during recovery and continues heading west until touchdown 275’ west of the launch position.

0 Degree Incline, 10mph

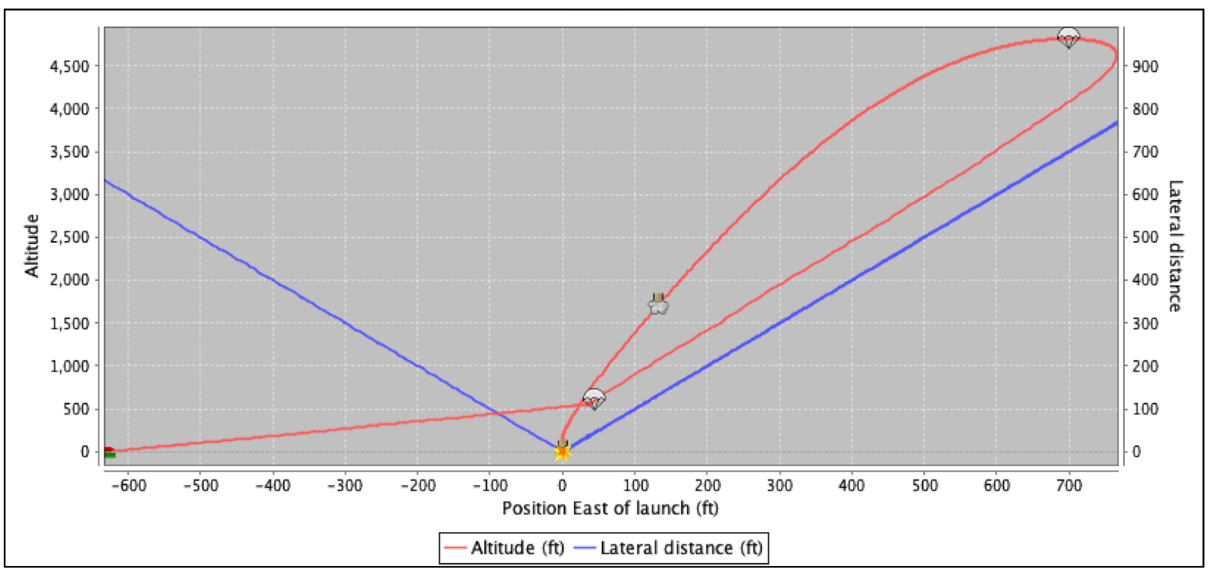


Figure 5.24: 0 Degree Inclination, 10mph Winds OpenRocket Drift Distance Simulation

With an average wind speed of 10mph with 0.5mph standard deviation and 10% turbulence intensity, the simulated maximum drift distance during flight is roughly 750’. The launch vehicle travels nearly 750’ east of the launch site as it tilts into the wind, then drifts back over the launch site during recovery and continues heading west until touchdown 650’ west of the launch position.

0 Degree Incline, 15mph

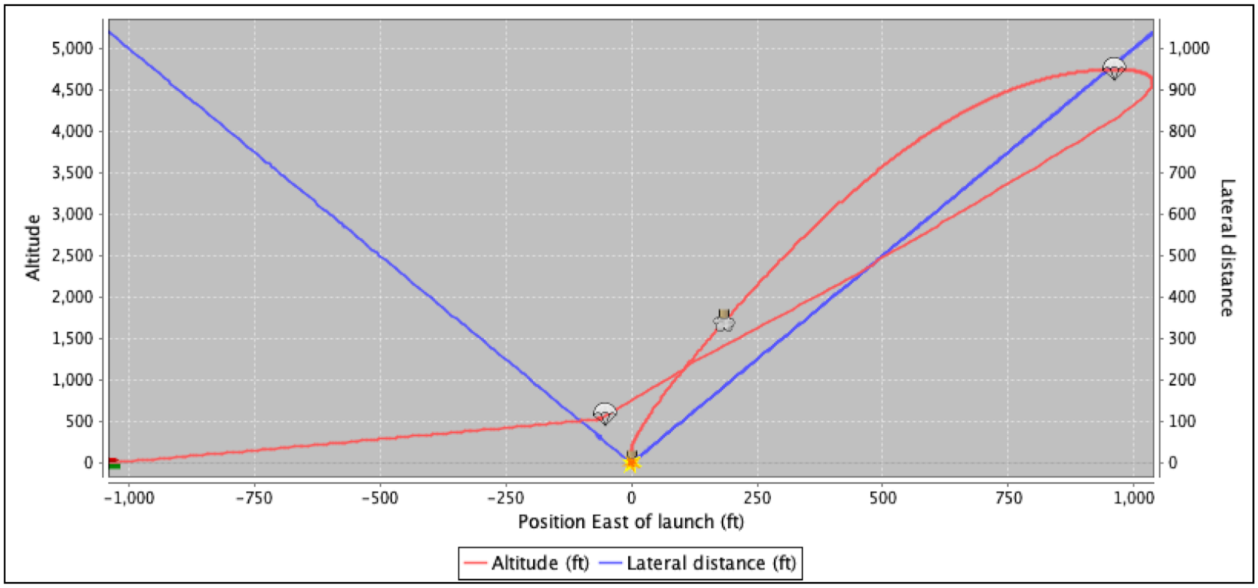


Figure 5.25: 0 Degree Inclination, 15mph Winds OpenRocket Drift Distance Simulation

With an average wind speed of five miles per hour with 0.5mph standard deviation and 10% turbulence intensity, the simulated maximum drift distance during flight is roughly 1150’. The launch vehicle travels nearly 1150’ east of the launch site as it tilts into the wind, then drifts back over the launch site during recovery and continues heading west until touchdown 1050’ west of the launch position.

0 Degree Incline, 20mph

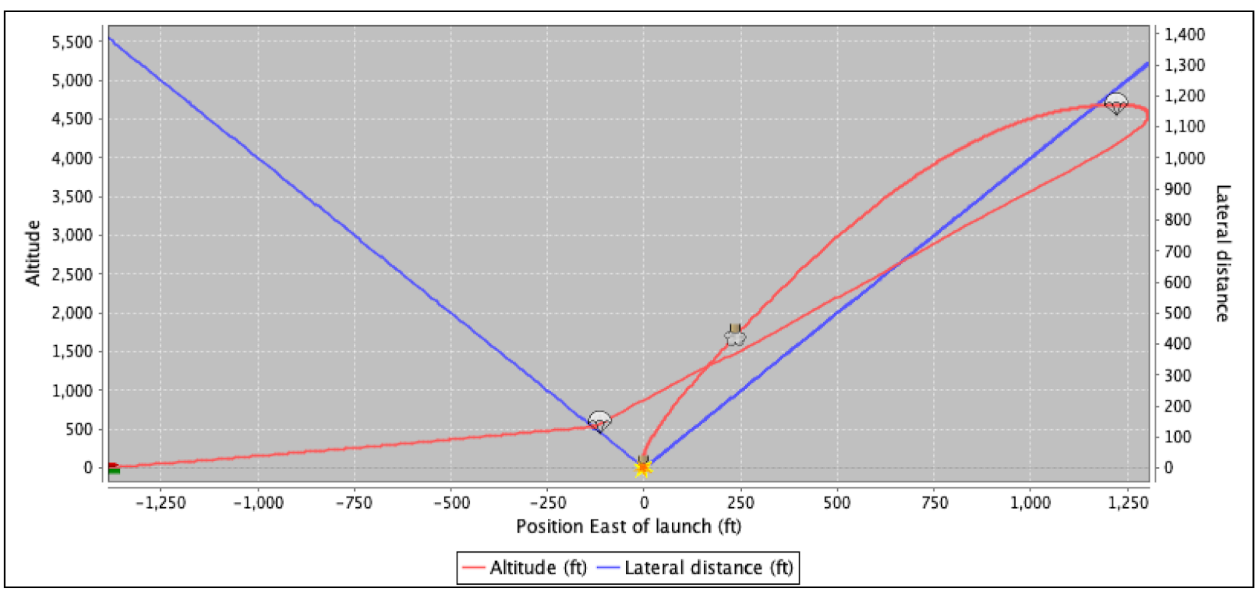


Figure 5.26: 0 Degree Inclination, 20mph Winds OpenRocket Drift Distance Simulation

With an average wind speed of 5mph with 0.5mph standard deviation and 10% turbulence intensity, the simulated maximum drift distance during flight is roughly 1350'. The launch vehicle travels nearly 1275' east of the launch site as it tilts into the wind, then drifts back over the launch site during recovery and continues heading west until touchdown 1350' west of the launch position.

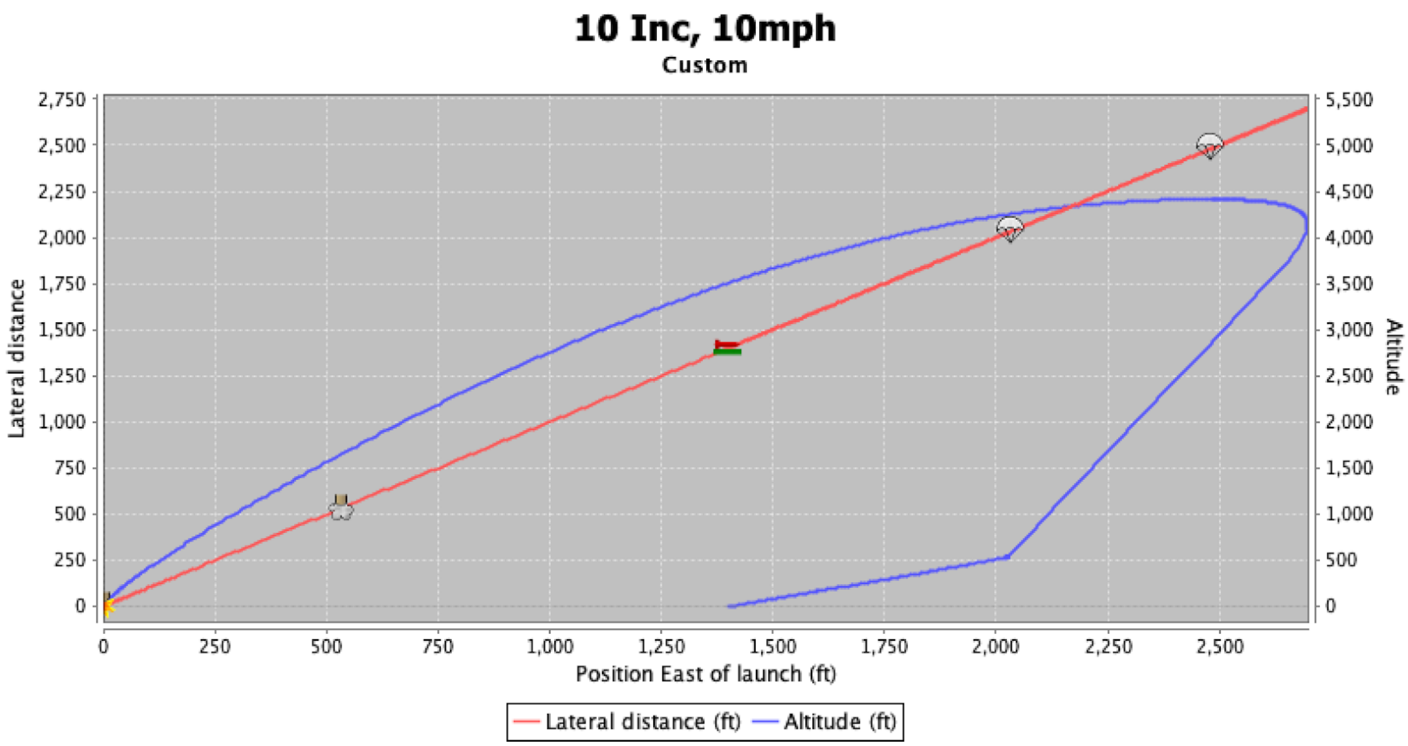


Figure 5.27: 10 Degree Inclination, 10mph Winds OpenRocket Drift Distance Simulation

In this ideal case for the launch vehicle on the day of launch, the vehicle has a max lateral distance of roughly 2750' east of the launch site but only lands 1350' east of the launch position. This simulation of 10 degrees of inclination and 10 mph winds show that the vehicle has its maximum drift distance while in the air but returns to the ground within the restriction of 2500' set by the NASA SL handbook.

5.2.3.2. RASAero Drift Calculation

0 Degree Incline, 0mph

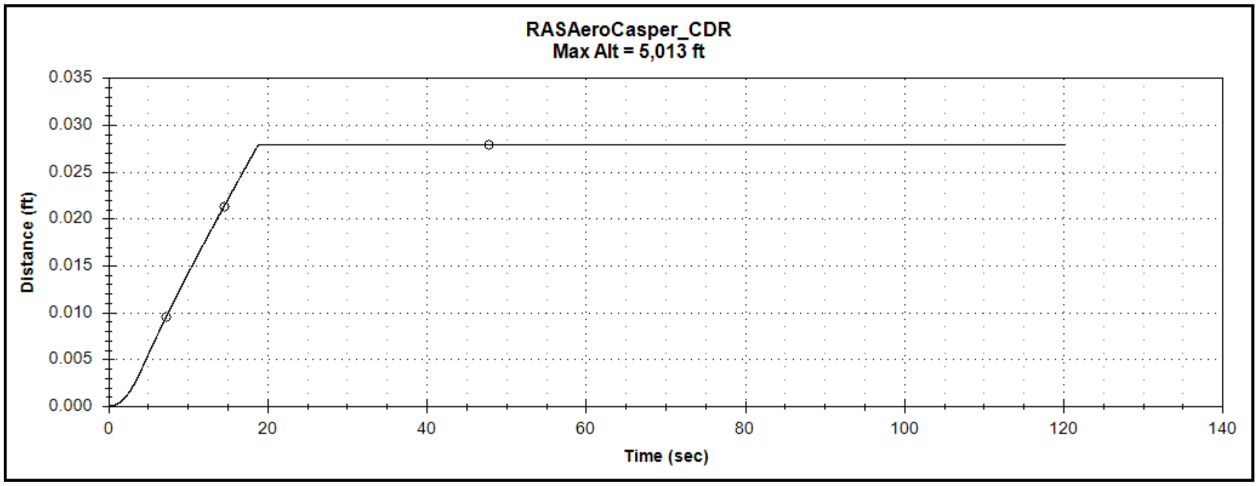


Figure 5.28: RASAero Simulation of Ideal Drift Distance Case

As expected, the drift for a launch vehicle under no crosswind or rail inclination has nearly no drift. The launch vehicle does drift minimally prior to motor burnout, but the value is less than 0.1'. Compared to OpenRocket, the value is about 9' less, but that value is fairly negligible.

0 Degree Incline, 5mph

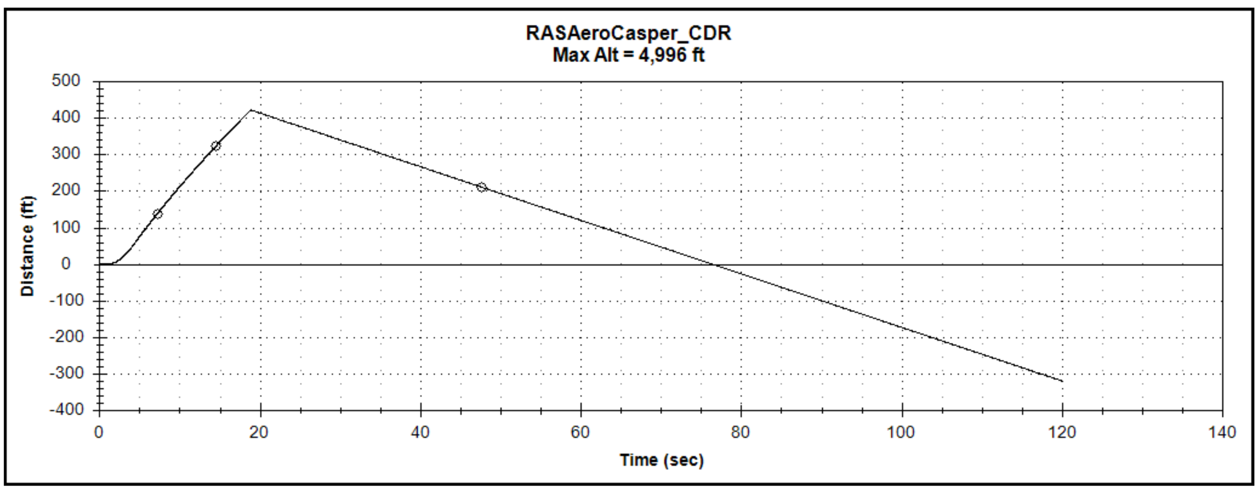


Figure 5.29: 0 Degree Inclination, 5mph Winds RASAero Drift Distance Simulation

The 0 degree inclination angle with 5mph winds gives much more meaningful results. It is important to note that the graph shows displacement. Therefore, by the time the launch vehicle lands, it will have moved approximately 300'. OpenRocket had a value of 410' of drift in this case for comparison.

0 Degree Incline, 10mph

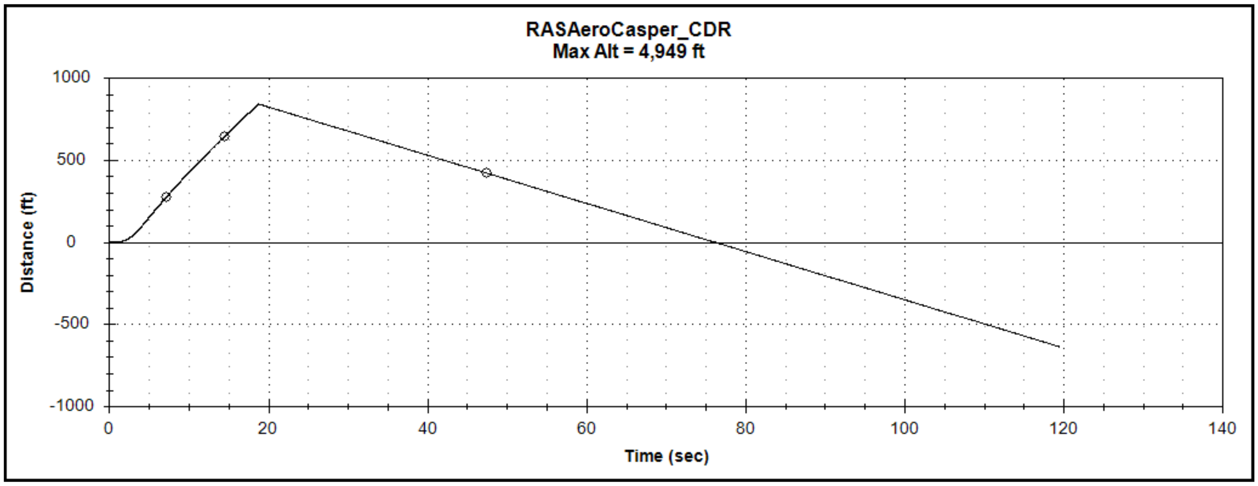


Figure 5.30: 0 Degree Inclination, 10mph Winds RASAero Drift Distance Simulation

In this case, the launch vehicle drifts around 700’ by the time it reaches the ground. From burnout, the drift is approximately 1600’. This drift is still well within the limits of the competition. OpenRocket stated a value of 650’ of drift, much closer to the 700’ value given from the original displacement.

0 Degree Incline, 15mph

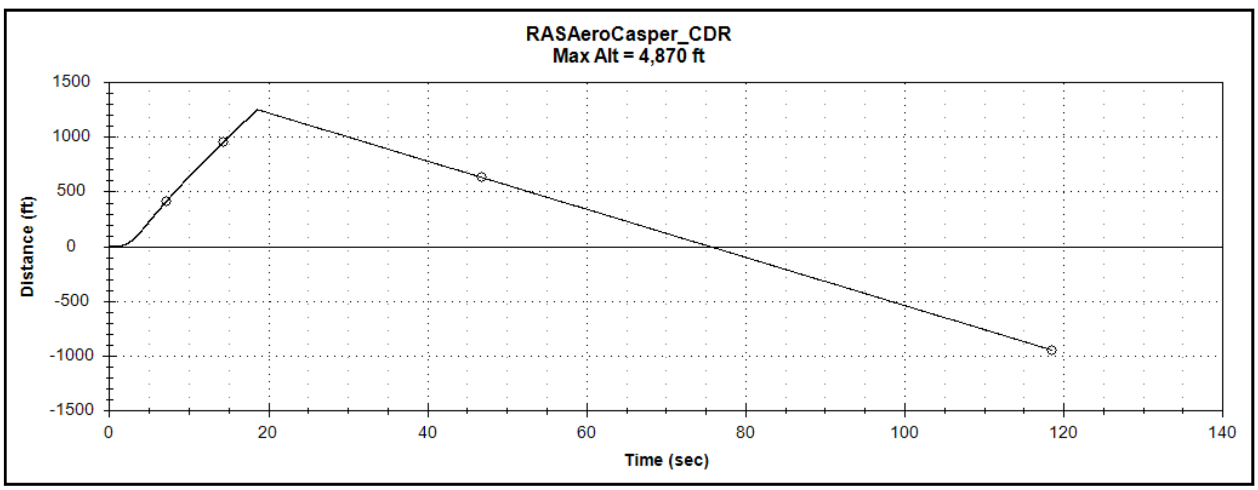


Figure 5.31: 0 Degree Inclination, 15mph Winds RASAero Drift Distance Simulation

The drift for 15mph winds is nearly 1000’ of displacement. However, the drift from burnout is approximately 2250’ which is much closer to the limits of the competition. OpenRocket determined a value of 1050’ of drift for this combination.

0 Degree Incline, 20mph

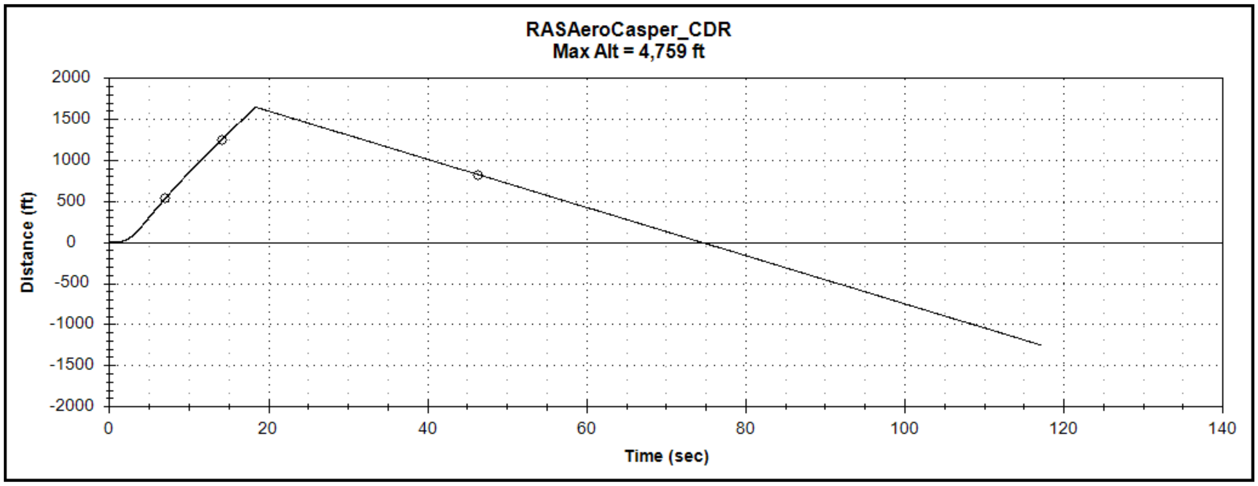


Figure 5.32: 0 Degree Inclination, 20mph Winds RASAero Drift Distance Simulation

The displacement was about 1300' when under 20mph winds. The drift from burnout was nearly 2900' which is much more concerning. However, the team is confident that this is not a reasonable case to expect for launch day, and that more reasonable results can be calculated as shown in the next section. Finally, for comparison, OpenRocket determined the drift for this case to be 1350ft. Therefore, in every case, the displacement values between the two programs was within 50ft, implying a level of precision. The team believes this precision will also translate to accuracy in the real-world.

10 Degree Incline, 10mph

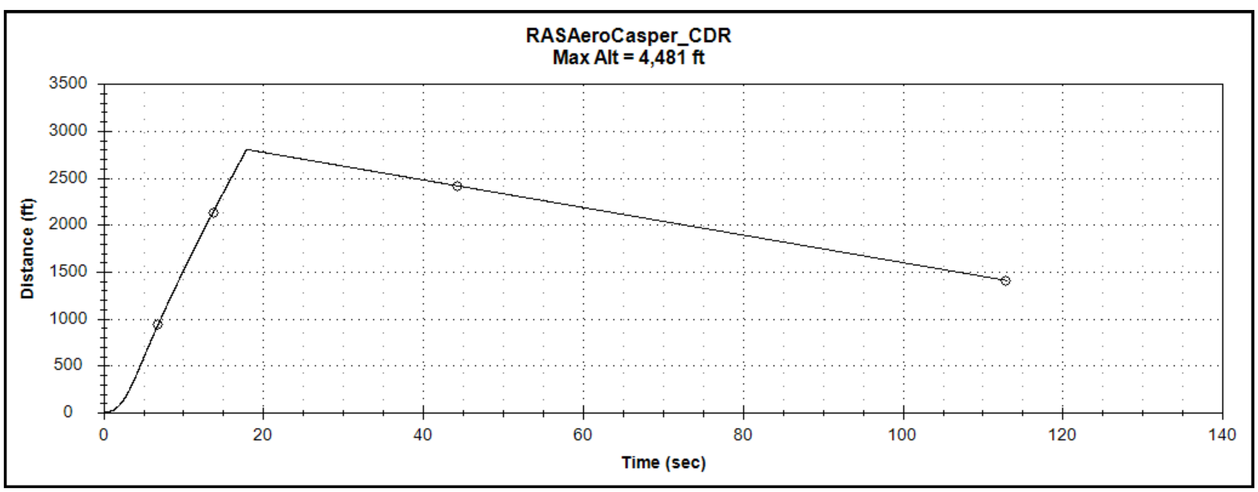


Figure 5.33: RASAero Simulation of Hypothesized Drift Distance Case

The team has determined that the 10 degree inclination angle with 10mph wind case is the most realistic case for launch day. As a result, this graph should hold the most meaningful results. It states that there will be approximately 1500’ of displacement. Taking the distance from burnout, there will be around 1200’ of drift. Both of these values are well within the drift requirement.

5.2.3.3. Hand-Calculations / Code Estimation

To calculate the drift distance of the launch vehicle by hand, the team used the following equation:

$$Drift = t_f * V_\infty * sin(\theta)$$

In the equation above, t_f is the total flight time (the entire time the wind causes lateral movement), V_∞ is the free stream velocity, and Θ is the launch angle. This equation assumes that the wind blows in only one direction during the launch vehicle’s ascent. It also assumes that the launch rail is canted at five degrees during the launch and that the apogee occurs directly above the launch rail. Although the 20mph case results in a drift distance of over the allowed 2500’, the team recognizes that the likelihood of 20 mph winds at the time of launch is low. In addition, the drift distance for wind speeds under 20 mph are within the allowed 2500’.

Launch Rail Angle [deg]	Wind Speed [mph]	Drift Distance From Pad [ft]
0	0	~0
0	5	684
0	10	1340
0	15	1995
0	20	2605

Table 5.1: Drift Distance Calculations for 0 Degree Inclination Angle and 0-20mph Wind Speeds

Although this equation provides a good estimate of the drift distance at each wind speed, the assumptions made by this equation result in a higher degree of error. As shown below, OpenRocket estimates the drift distance for the case closest to our most likely case, with a 0 degree inclination and 10 mph winds, to be 750’ at the most, with the launch vehicle coming back toward the launch site until it touches down at approximately 0 ft. This estimation by OpenRocket assumes, like the equation used for hand calculations, that the wind is constant in both speed and direction from the time of launch until the launch vehicle touches down. Since the OpenRocket simulation works with more variables than the equation for the hand calculations, the two estimates will have slightly different results.

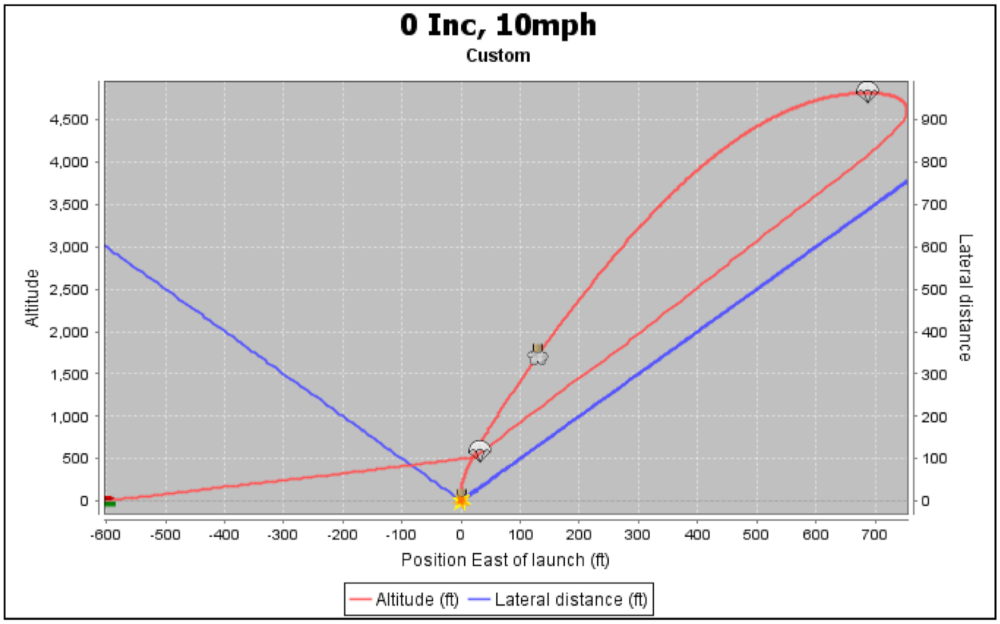


Figure 5.34: OpenRocket Drift Estimation vs Hand Calculations with 0 Degree Inclination, 10mph Winds

However, the discrepancy between the OpenRocket simulation and hand calculations is not always constant. As shown in the figure below, the worst case scenario done by hand calculations, the 0 degree inclination and 20mph winds case, is well within the maximum allowed drift distance for the competition, according to OpenRocket, while the hand calculations estimate the drift distance to be much greater.

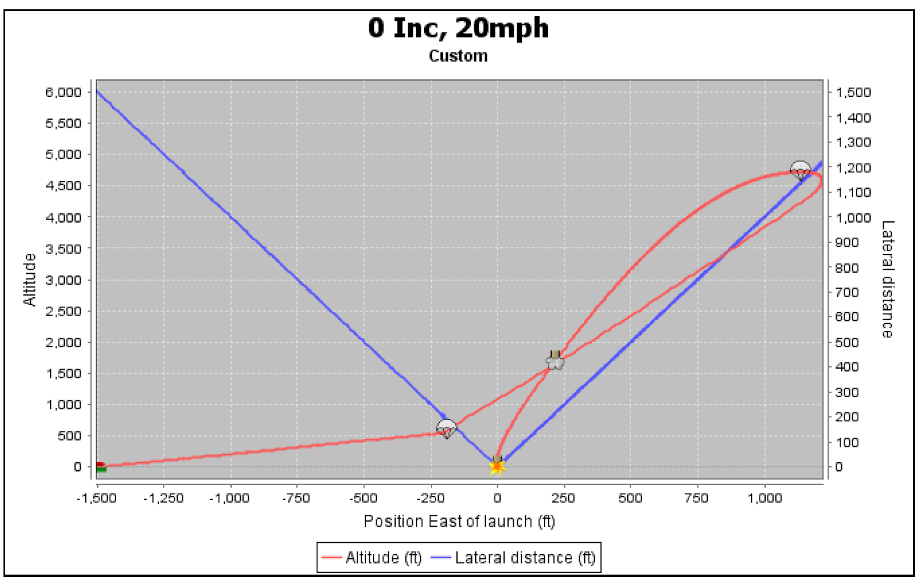


Figure 5.35: OpenRocket Drift Estimation vs Hand Calculations with 0 Degree Inclination, 20mph Winds

As shown above, the hand calculations provide a good estimate of the drift distance of the launch vehicle. However, the team will be using multiple sources to check the simulations and calculations, since the assumptions made by each affect the end result.

5.3. Motor Characteristics

The propulsion system that the 2020 PSP-SL full scale launch vehicle will utilize is a Cesaroni Technology Incorporated (CTI) Classic L1115 4-grain SRM. This motor has a Specific Impulse (I_{SP}) of 213.46s from analysis compared to a commercial I_{SP} of 214s with a regressive motor thrust profile. This SRM provides approximately 5,015N-s impulse over a burn time of 4.48s. Data available online for the CTI L1115 SRM was used to gain a first order approximation for different propellant and motor characteristics. Motor test data gave a rudimentary thrust versus time curve along with motor inert and propellant masses. Using this data, a MATLAB script was started to help the team better understand the mass flow rate over the burn time and run the ballistics analysis during flight. A spreadsheet producing polynomial curve fits of thrust data and other engine parameters derived from the motor test data was created to aid in developing the MATLAB propulsion analysis (for FRR the team's goal is to have this implemented into MATLAB).

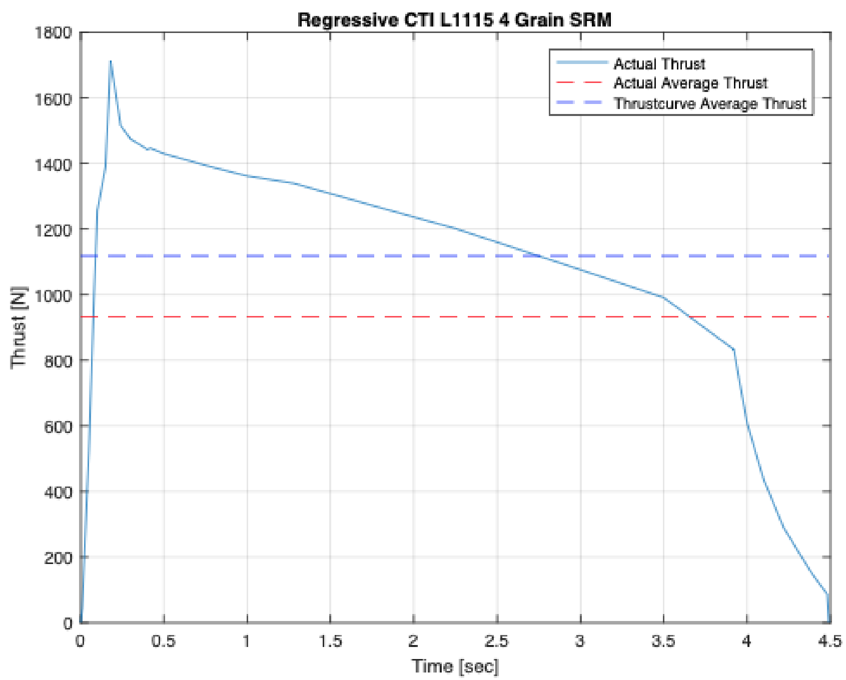


Figure 5.36: Full-Scale Thrust vs Time Plot

The CTI L1115 SRM is considered to have classic propellant, this means that the motor has a combination of Ammonium Perchlorate (NH_4ClO_4, AP) for the oxidizer, atomized aluminum for the fuel, and has several other components such as: curative, R45M Hydroxyl-Terminated Polybutadiene (HTPB) rubber binder, and various bonding agents. This year the team is expecting about ~12.5%

aluminum and about ~72.5% AP with the remaining ~15% consisting of the binder, bonding agents, additives, and curatives.

The MATLAB script begins with initialization of known parameters. Unknown values that had to be found or estimated included nozzle area ratio, throat regression rate, case thickness, length of the grain inside the case, and the inner diameter of the grain. The throat regression rate was based on the value provided in previous homework assignments. Using the diagram in Figure 5.34, the case thickness, area ratio, length of grain in the case, and the inner diameter of the grain were all estimated. These values can be found in Table 5.2.

Parameter	Value			
Estimated Case Thickness	0.0015	m	0.059	in
Grain Inner Diameter	0.0238525	m	0.938	in
Grain Outer Diameter	0.0651256	m	2.564	in
Grain Length	0.1331976	m	5.244	in
Nozzle Throat Diameter	0.009525	m	0.375	in
Nozzle Exit Diameter	0.01905	m	0.750	in
Expansion Ratio	4	-	4	-
Throat Erosion Rate	0.1	mm/s	0.003937	in/s

Table 5.2: Estimated Values for Initial Mass Flow Calculations

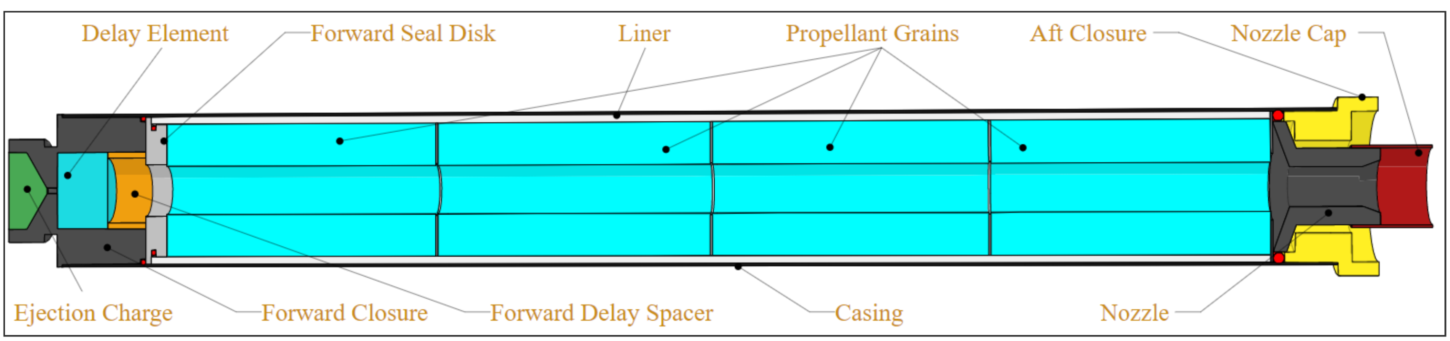


Figure 5.37: Cutaway of CTI Model Rocket Motor Used for Approximating Engine Internal Dimensions

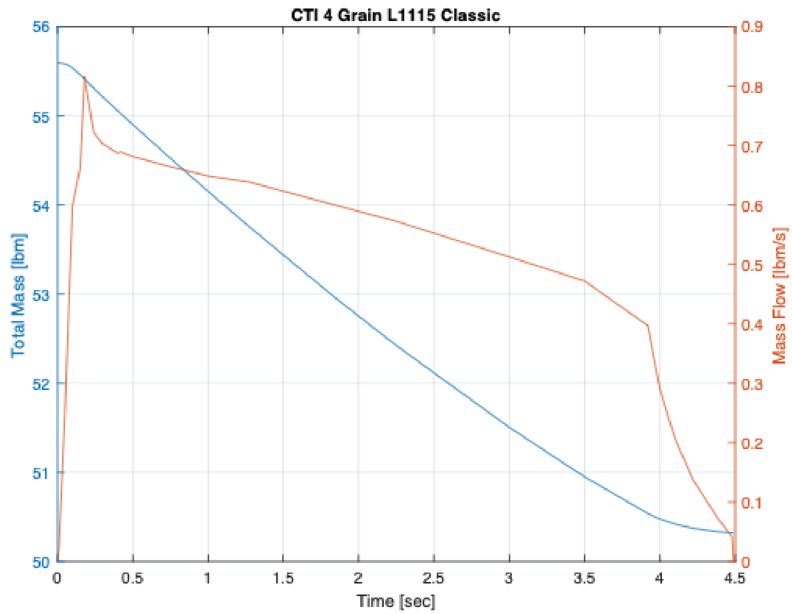


Figure 5.38: Full-Scale Launch Vehicle Mass vs Time Plot

The plot above shows the change in mass of the vehicle during the boosting phase, as described earlier. It also shows the change in the propellant mass flow which follows the same trend as the thrust as they are directly proportional. The only change in mass across the system is the propellant. The values of masses were taken from previous experimental data at irregular intervals of time. The data was then linearly interpolated between each of the data points to get a smooth curve. The initial mass of the launch vehicle is 55.6lbm, and the mass of the propellent is 2394g (5.277lbm).

Using these values, the total volume of propellant was calculated using a simple geometry to calculate the volume of an annulus (the propellant grain was assumed to have a circular port geometry based on motor's simplicity and size). Using this volume, and the given initial propellant mass, the density of the propellant was found to be ~1624.53kg/m^3. This propellant density is reasonable because ammonium perchlorate (known component of the motor) has density < 2000kg/m^3 and this density would include some packing inefficiency in the motor itself. Due to this similarity, the geometry estimates were deemed acceptable.

In order to calculate mass flow rate per time, an estimated I_{SP} was first calculated. It was found that the motor I_{SP} is roughly 213.45s, based on the motor test data and using the equation below.

$$I_{sp} = \frac{F_{avg}}{g\,\dot{m}_{avg}}$$

In this equation, F_{avg} is the average thrust provided by the motor data sheet, and $\dot{m}_{avg}$ is the average mass flow across the burn, obtained by dividing initial propellant mass by total burn time (both supplied on the motor data sheet). To obtain ṁ over burn time, the same equation was rearranged in the equation below, instead using the force per second data provided by the motor data sheet.

$$\dot{m} = \frac{F(t)}{g\, I_{sp}}$$

At this point, both the I_{SP} and the ṁ as a function of time had been calculated using this script. Throat erosion was also ready to be incorporated and would be implemented after the ballistic calculation was performed. In order to complete the ballistics calculation, an equation for burn area as a function of web distance was first calculated. This relationship can be seen in the equation below.

$$A_b = \pi\,(R_1 + 2\,web)\,L_g + 2\,(R_0^2 - (R_1 + web)^2)$$

In this equation, A_b is burn area [m^2], R_1 is the inner diameter of the grain [m], R_0 is the outer diameter of the grain [m], web is the web distance [m], and L_g is the length of the grain [m]. This equation will later be used in the iteration to calculate ballistics. In order to compare the modeled the regression rate chamber pressure, the experimental chamber pressure over time was first calculated. This was done by using the equation shown below.

$$P_c = \frac{F(t)}{cf\;A_t}$$

This equation uses experimental force, a calculated throat area (A_t) based on the throat erosion rate, and a thrust coefficient. The thrust coefficient (cf) was determined based on the increasing area size. This was done by taking a couple of $cf_{optimal}$ values for given area ratios and setting up a linear interpolation to find the specific cf value as the throat area changed. The $cf_{optimal}$ values were considered accurate for this application, as the nozzle was most likely designed for perfectly expanded flow at sea level. After finding $cf_{optimal}$ for an area ratio of four, the c* value was calculated using the equation shown below.

$$c^* = \frac{I_{sp}g}{cf}$$

To more accurately model the performance of the launch vehicle, a non-constant I_{SP} as a result of throat erosion was considered in the code. The erosion rate was set as a constant 0.0001m/s (0.1mm/s). In the "while" loop in the propulsion code that calculates the new burn area with small time steps, a correction is also made to the throat diameter. The code takes the previous throat diameter and adds two times the erosion rate to the diameter for every time step. This has a relatively small effect on the I_{SP}, since only about 0.208mm of the throat diameter erodes away during the burn time.

$$r_b = a * (p_c)^n$$

This c* was assumed to be constant across all cases at 1710.5m/s. Using this value, and the chamber pressure equation above, the experimental chamber pressure was found as a function of time. Next, the ballistics analysis was performed on the rocket. To run this analysis for the first time, a burn rate coefficient and burn rate exponent were randomly picked in a reasonable range, based on classic SRM burn rate coefficient and exponent value ranges. Inside a loop in the MATLAB code, the time step was increased, the current burn rate was calculated using the previous iteration's chamber pressure, the web distance was increased based on burn rate, and geometries were updated accordingly. Once these values were obtained (using equations previously described), the equation below was used to calculate the current timestep's chamber pressure.

$$P_c = \left[\frac{a\rho_b A_b c*}{g A_t}\right]^{1/(1-n)}$$

Using the chamber pressure at a specific time, the mass flow rate and force provided by the motor were calculated as well. This was accomplished using the two equations below.

$$\dot{m} = \frac{g\, p_c A_t}{c*} \qquad F = I_{sp} * \dot{m}$$

Now, the force, mass flow rate, and chamber pressure were obtained for both the experimental data and ballistic analysis. The burn rate coefficient and burn rate exponent were obtained from a plot of the natural log of burn rate versus natural log of chamber pressure. Mass flow rates during flight as a function of burn time based on motor test data and calculated propellant density were used to calculate the volume of propellant consumed during a given length of time. This propellant volume was then combined with the assumptions about initial chamber diameter and length to derive an approximation of the burn rate at a given time during the motor burn. From there, burn rate was used with the chamber pressure (also derived from the data) to give the information necessary to compute the burn rate coefficients.

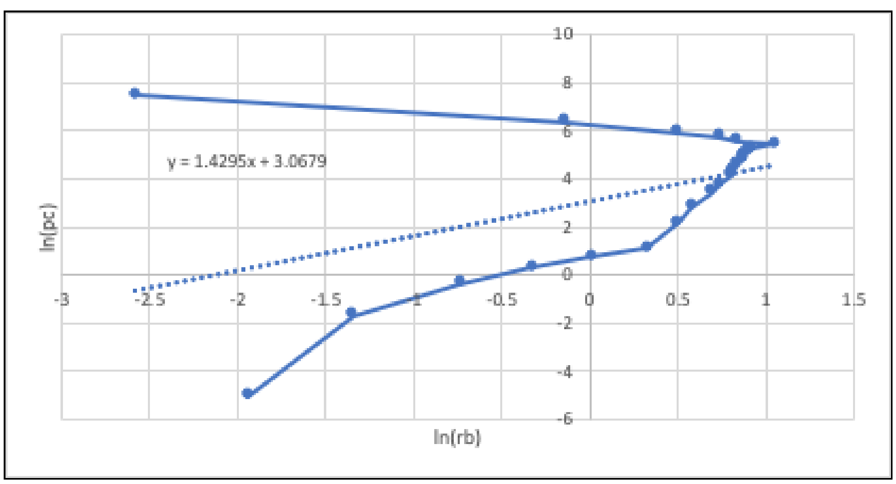


Figure 5.39: Log-Scale Evaluation of Burn Rate vs Chamber Pressure

At this point the team is still working on refining the process and results of this analysis, but the team's current results have been attached. The team knows that these values are wrong due to the values being significantly over the typical values for SRMs, typically SRMs of this size should fall between $0.2 < n < 0.6$ and $0.5 < a < 1.5$. With that in mind, Figure 5.36 shows the burn rate coefficient and exponent that were calculated using the linear best fit line coefficients where the intercept represents ln(burn rate coefficient) and the slope of the line directly represents the burn rate exponent. From the best fit line for the data, the burn rate exponent the team found was about 1.4295 (n) and the burn rate coefficient (a) = exp(3.0679) = 30.256cm/s. As stated before, the team understands that these results are inaccurate and plans on refining the analysis.

5.4. Kinetic Landing Energy

Using the MATLAB simulation, which records the velocity at every time step and the mass at every time step, the kinetic energy of the vehicle can be calculated. It assumes that the velocity of the vehicle and the two parachutes are equal to each other. It also assumes that when the drogue parachute deploys, the horizontal component of the velocity becomes zero. This means that wind speed does not matter for the landing kinetic energy.

During drogue deployment, the vertical velocity, and thus the total kinetic energy, are at their maximum when the launch angle is 5 degrees. However, there is minimal difference between the values at 5 and 10 degrees. Wind speed has no impact on either values.

Wind Speed [mph]	Launch Angle [deg]	Total Kinetic Energy [ft-lbf]	Vertical Velocity [ft/s]
0	0	764.04	31.26
5	5	787.91	31.69
5	10	784.54	31.68
10	5	787.91	31.69
10	10	784.54	31.68

Table 5.3: Drogue Deployment (Time = ~Apogee + 1s) Kinetic Energy and Vertical Velocity

During main deployment, the total kinetic energy and vertical velocity are relatively the same regardless of launch angle or wind speed.

Wind Speed [mph]	Launch Angle [deg]	Total Kinetic Energy [ft-lbf]	Vertical Velocity [ft/s]

0	**0**	13,500	131.31
5	**5**	13,500	131.30
5	**10**	13,400	131.07
10	**5**	13,500	131.30
10	**10**	13,400	131.07

Table 5.4: Main Deployment (Altitude = ~500') Kinetic Energy and Vertical Velocity

During landing, the total kinetic energy and vertical velocity are at their maximum at a launch angle of 10 degrees. Generally, as launch angle increases, the vertical velocity and total kinetic energy also increase. Wind speed had no impact on either set of values.

Wind Speed [mph]	Launch Angle [deg]	Total Kinetic Energy [ft-lbf]	Vertical Velocity [ft/s]
0	**0**	142.86	13.51
5	**5**	143.41	13.54
5	**10**	148.87	13.62
10	**5**	143.41	13.54
10	**10**	148.87	13.62

Table 5.5: Landing (Altitude = ~0') Kinetic Energy and Vertical Velocity

6. Payload System

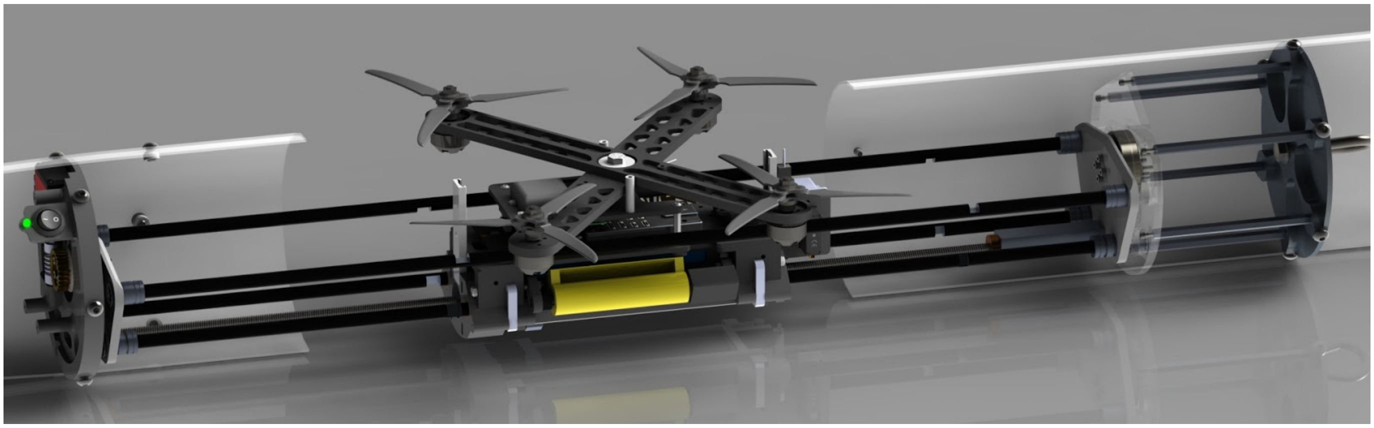


Figure 6.1: Render of Payload UAV and its Retention and Deployment System

6.1. Mission Statement and Success Criteria

6.1.1. Mission Statement

The mission of the UAS is to safely identify, extract, and recover a simulated lunar ice-sample with a UAV deployed from a high-powered rocket. The UAV will be mechanically retained in the payload bay of the launch vehicle during flight in a fail-safe Retention and Deployment (R&D) system. The UAV will be deployed from the launch vehicle after its has completed its flight. The UAV will be capable of semi-autonomous flight, navigation and ice sample recovery. Any and all phases of the mission will have pre-programmed contingencies along with the option of immediate manual override and mission termination.

6.1.2. Mission Profile and Success Criteria

The mission is divided into five distinct phases with each phase decomposed into functional events that describe the operational mission path along with alternative (contingency) mission paths. The five phases are Vehicle Launch and Recovery, UAV Deployment and Integration, Signaled Takeoff, Recovery Area Search and Ice Procurement, and Sample Recovery. The mission and phases will be decomposed into a series of functional flow block diagrams. This also includes contingency planning in the case the autonomous mission planner reaches the necessary criteria to terminate the mission. In order for the UAS to successfully complete its mission, it must complete all functions and phases detailed below.

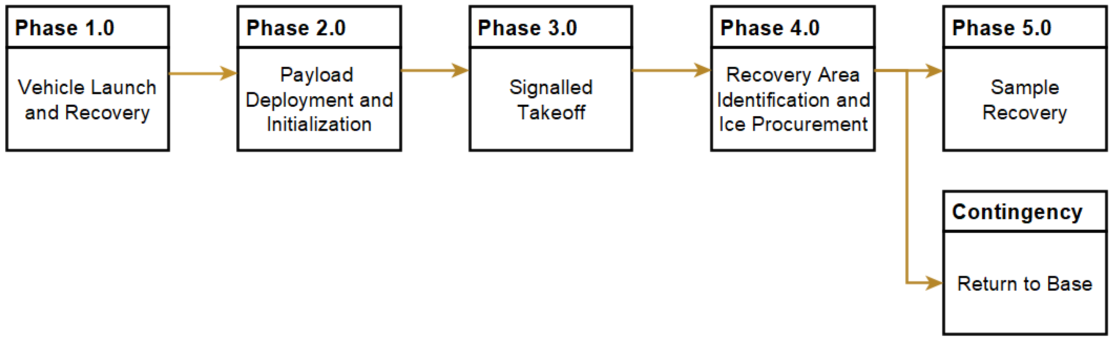


Figure 6.2: Payload Mission Profile

Phase 1: Vehicle Launch and Recovery

The vehicle launch and recovery phase will include limited action from the UAS. The UAV will not be turned on to meet requirements set forth by the FAA, and therefore the GCS will not need to be active nor controlled by an operator. Instead, the R&D payload retention system will mechanically hold the UAV through a failsafe mechanism. Upon successful launch, vehicle touchdown, and permission from the Range Safety Officer (RSO), the mission may proceed to Phase 2.

Phase 2: UAV Deployment and Integration

Subsequent to the vehicle recovery, the GCS will send a signal to the R&D system to begin the deployment of the UAV from the launch vehicle body. The R&D system will separate the nose cone from the upper airframe of the launch vehicle, exposing an opening in which the UAV will be held on to a sled. Once fully deployed, the UAV will correct its orientation by rotating along the axis of the launch vehicle body. Following the successful deployment and orientation of the UAV, a limit switch will engage allowing the UAV to be powered. Once powered, the UAV will begin automatically integrating all of its components and await a successful GPS link. Upon obtaining a valid GPS link and successfully integrating all components on the UAV, the mission may proceed to Phase 3. A functional flow diagram describing Phase 2 of the mission is seen in Figure 6.3 below.

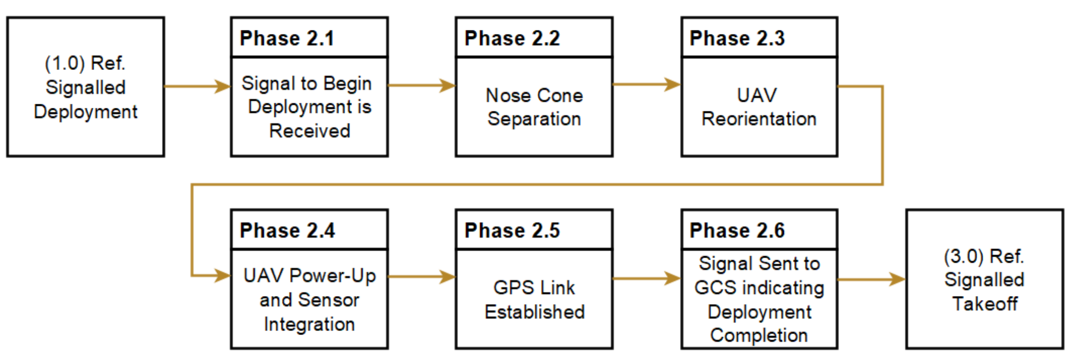


Figure 6.3: Payload Mission Phase 2 Flow Diagram

Phase 3: Signalled Takeoff

The objective of the signalled takeoff phase of the mission is to safely lift the UAV out of the payload bay, placing the vehicle in a position to begin its search for an ice recovery area. The signalled takeoff phase will begin with a signal sent from the UAV to the GCS, informing the ground support team that it has obtained a valid GPS link and has integrated all of its components. The ground support team will then send a signal from the GCS to the UAV, activating the UAV's takeoff sequence. The Flight Control Computer (FCC) will control the takeoff, lifting the vehicle out of the payload bay approximately 100' above ground level. Once the UAV has stabilized at this position, the mission may proceed to Phase 4. A functional flow diagram describing Phase 3 of the mission is seen in Figure 6.4 below.

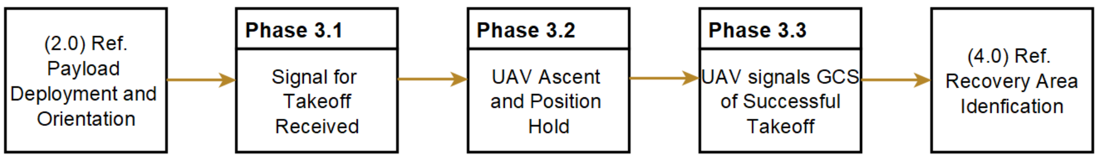


Figure 6.4: Payload Mission Phase 3 Flow Diagram

Phase 4: Recovery Area Identification and Ice Procurement

After the completion of Phase 3 of the mission, the UAV will begin its search for an ice mining recovery area. This search will occur autonomously, employing a constant-altitude grid search strategy. The UAV will utilize a pre-programmed map of the recovery field with approximate Global Positioning System (GPS) data of the recovery areas to establish its position in 3D space and follow a grid-like trajectory to search for a recovery area near its approximate position. Throughout this process, the UAV's vision system will capture and process images of the ground beneath the vehicle to determine if a recovery area has been identified. Once this identification has occurred, the UAV will enter a controlled descent, utilizing a stream of data from its vision system to align itself with the center of the recovery area. The UAV will descend to 1-2' above the recovery area, at which point the FCC will slowly land the vehicle. Once the vehicle has landed at the recovery area, the mission may proceed to Phase 5. A functional flow diagram describing Phase 4 of the mission is seen in Figure 6.5 below.

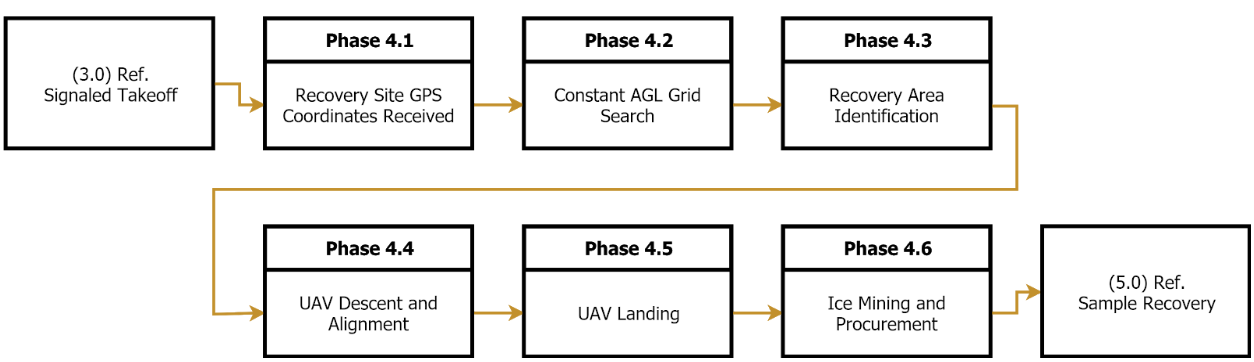


Figure 6.5: Payload Mission Phase 4 Flow Diagram

Phase 5: Sample Recovery

The fifth and final phase of the mission is the recovery of the lunar ice sample. Upon landing at the recovery site at the end of Phase 4 of the mission, the ice mining and procurement system will be engaged. This system's rotating cylindrical scoops will begin to actuate, collecting pieces of lunar ice material. After the completion of this operation, the GCS will send a take-off signal to the UAV. Upon receiving this signal, the UAV will gently lift itself 10-20' above the ground, and fly 50' away from the recovery area. Finally, after executing this maneuver, the GCS will send a signal to the UAV, executing a final landing sequence. Upon landing, the mission will be complete.

6.2. System Overview

6.2.1. UAS Overview

The UAS is responsible for autonomous flight, navigation to an ice mining recovery area, and procurement of the lunar ice sample. The system is broken down into two primary subsystems: the UAV and GCS. The UAV employs a quad-rotor design with an innovative folding method, allowing the vehicle to fit inside the airframe of the launch vehicle and interface with the R&D system. The UAV also has a sophisticated flight control system and mission management system for controlling autonomous flight, including a computer vision system for autonomous recognition of recovery areas. An ice mining and procurement system is integrated into the lower section of the UAV, allowing for ample recovery of simulated lunar ice material. The UAV is estimated to weigh 2.7lbm and consume approximately 152W of power. This leads to an estimated total flight time of 10.1min. Further analysis of this is given in Section 6.3.7. The diagram below shows how these subsystems are integrated together to accomplish the requirements of the mission.

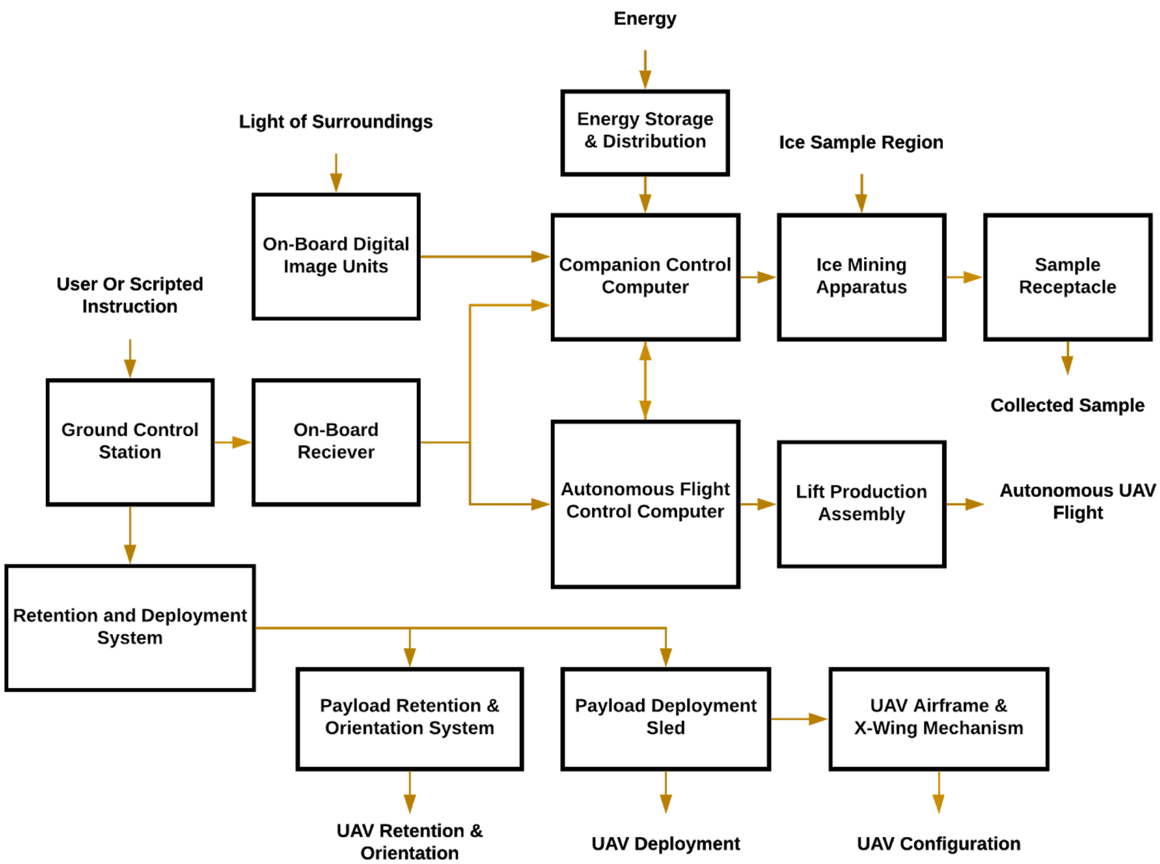


Figure 6.6: Payload UAS Functional Block Diagram

6.2.2. R&D Overview

The R&D system serves to provide a usable take-off surface to the UAV system, while ensuring the safety of the payload during flight. The system has two active components, a dual shaft stepper motor, and a continuous rotation servo. These components use a combined 30-40W, respectively, so a large 3S Li-Ion battery is used for power. In order for the UAV to function it must be as close as possible to vertical. This reorientation is achieved through the continuous servo and a gyroscope attached to sled. Additional orientation information may be retrieved from an encoder on the sled. The dual shaft stepper motor controls the position of the nose cone, upper airframe, and sled after landing. The total system weight is approximately 8lbm. More information about the mechanical and electrical operations of the R&D System can be found in the relevant section.

6.3. UAS Detailed Design

6.3.1. UAV Airframe Design

Since the rules were released, the payload team was keen on using an aerial system as opposed to a system capable of traversing through the land like previous years. When looking for inspiration, the team referenced many of the commercially available drones on the market, observing their form factors as well as their built in solutions to weight and space problems. There were three overall designs the team worked through in the process of developing the UAV: One design consisted of 4 moving arms that would rotate outwards in pairs of 2 using motors. The second design, with a pipe-like structure, mainly emphasized modularity and landing safely. This consisted of pipes with two legs sitting below the bottom plate of the UAV. This design solved the issue of electronics placements as the electronics could be mounted within the frame volume of the pipes, however as discussed in PDR, the associated manufacturing issues kept us away from following up on this approach.

The third design, our final and main design, took a slightly different approach from our previous ideas, intending to change design features to increase functionality and uniqueness while maintaining simplicity that boosts manufactuability and prevents catastrophic failure. Instead of having pivots for each armature, this design focused on creating a single pivot point between just two armatures at the center while still giving us the benefit of using 4 propellers for more lift. This was achieved by using a torsion spring at the pivot point. The structural frame of this design consists of only two plates separated by vertical standoffs, taking inspiration from the pipe design's modularity. These attachment plates provide as much space as we require to lock electronics in place while maintaining high manufacturability and protective capability. Under the bottom plate of the UAV is the 3D-printed battery mounted between two legs that also hold the mining systems.

It was decided that the ice mining system, the function of which is the ultimate goal of the UAS, should have priority on ground accessibility. Throughout development, the ice mining system has remained a top constraint on the vehicle's shape. While the rest of the airframe facilitates flight, the lower structure is designed to provide ample points of attachment and flexibility for the purposes of ice mining systems.

This airframe was designed to be compatible with the R&D systems, and as such, provides places of attachment for the R&D sled for constraint in all directions of motion during flight and landing. The form and function of the airframe also yields to the constraints of the R&D system during deployment, utilizing interactions between the UAV and the sled to enable the UAV to reorient and take flight when instructed.

A decent amount of analysis has been put into the selection of materials and the location of components to best serve the UAV's purposes and meet all requirements. Below is a discussion of each of the assemblies, structures, and parts involved in the UAV airframe. This includes their locations and functions, how they have changed, and our future plans as the design phase comes to a close while testing and iteration begins.

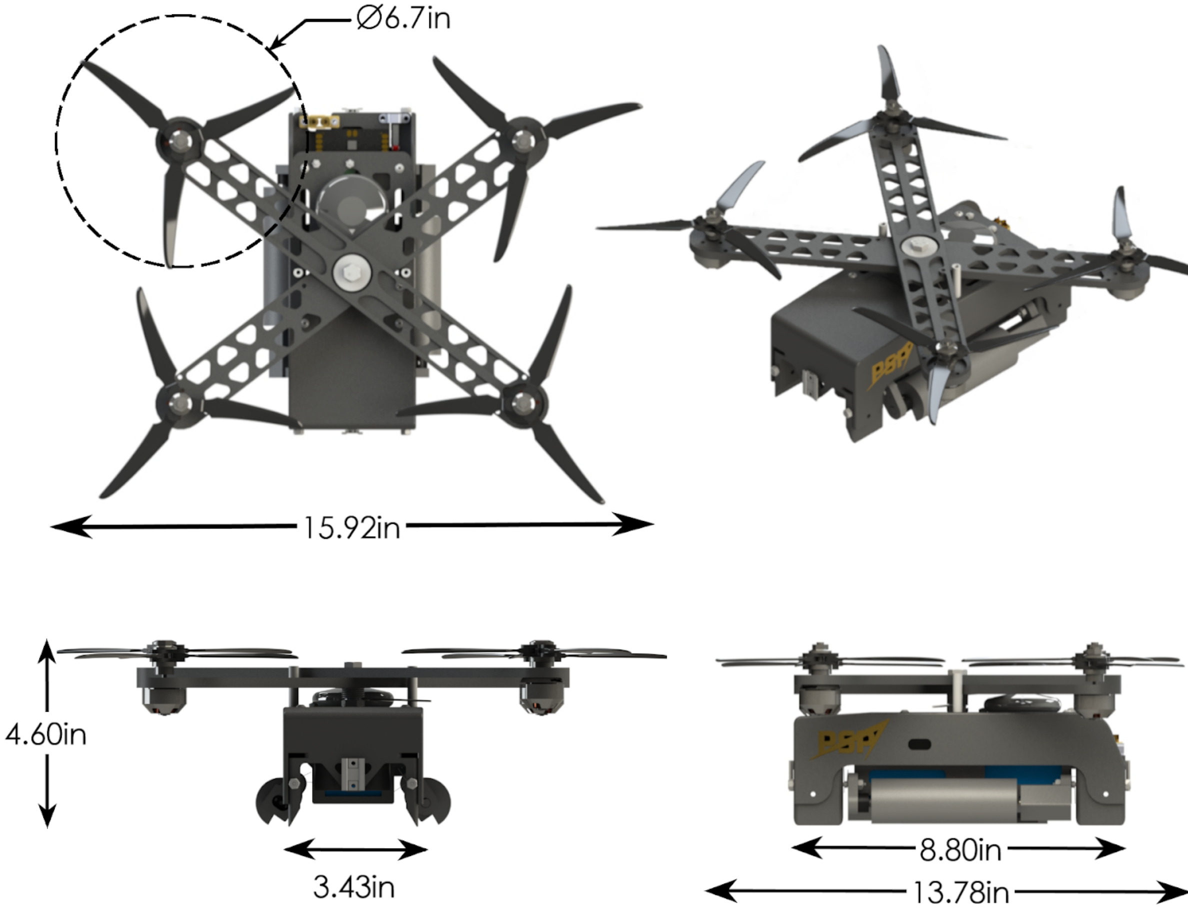


Figure 6.7: Full Payload UAV Isometric and Orthographic Views

6.3.1.1. Frame Design

Our overall frame design includes a central plate skeleton which houses electrical components and provides structure. Upon these plates sits the X-Wing Mechanism, which moves via spring actuation after activation from the R&D system's sled. In order to avoid any damage to the battery, it is installed underneath the bottom plate along with two leg structures that also hold the ice mining system. Covering the entire UAV is a protective shell known as the Aero Package. This section is an overview of all airframe structures, beginning with the development of its overall design before breaking down into assemblies that enable the airframe to perform its desired jobs.

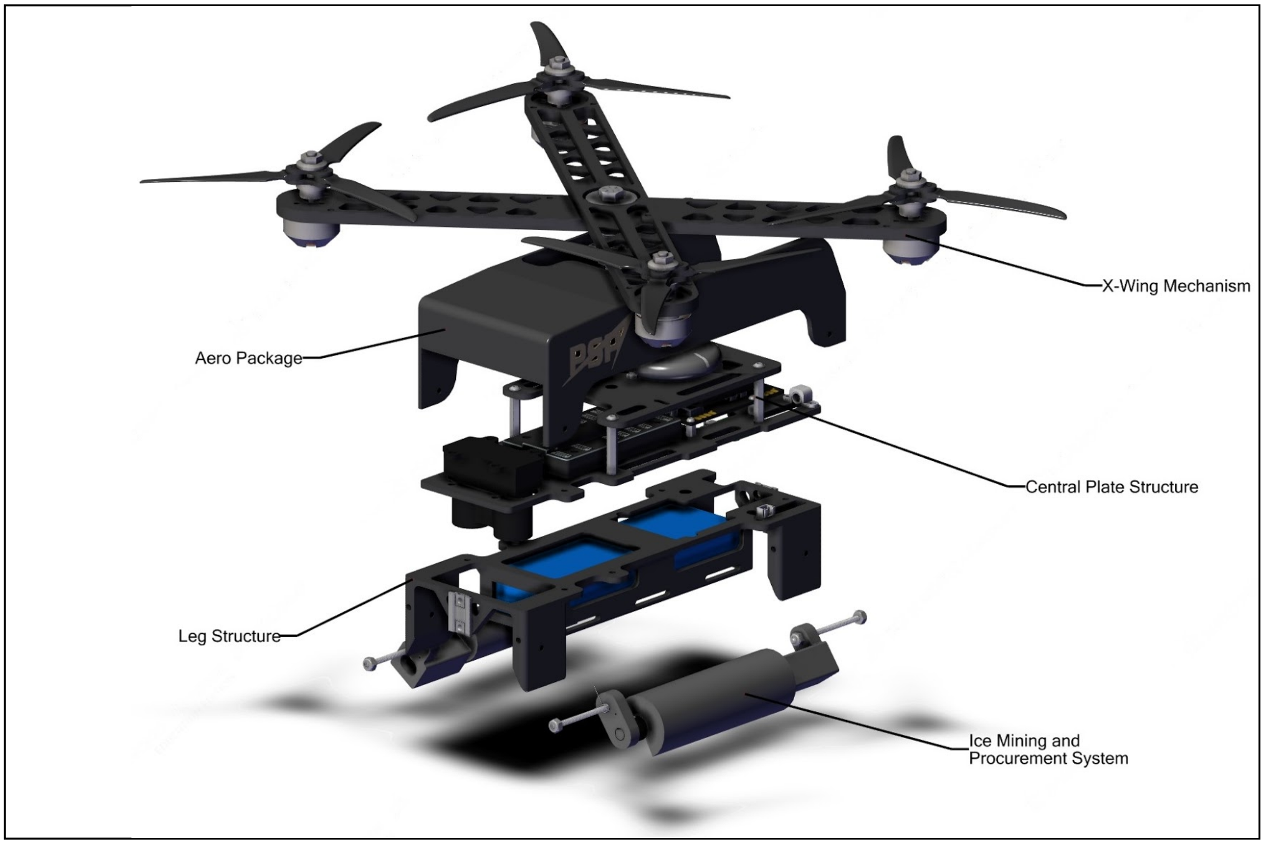


Figure 6.8: Payload UAV Exploded View Render

Central Plate Structure

All hardware and electronic components need some form of structural support in order for the UAV to be functional. The airframe needs to support the weight of all the sub-components and along with that, the design needs to consider center of mass as one of the important factors. More importantly, the airframe also needs to be contained within the constraints such as the inner diameter of the couplers which is 5.77”, the horizontal distance of 3.7” between the guide rods, and the vertical height of 3.4’’ from the top of the R&D sled and the inner surface of the rocket tube. Since PDR, the decision was made to allow the airframe team to define the constraints of the R&D sled design, which has allowed the space and functionality needs of the airframe to dictate its form.

For the design of the UAV frame, the team pursued the use of plates as main structural components and for housing electronics; in hobby flying, it appears that many drones take this approach to minimize weight while increasing the ease of manufacture—two very highly sought after properties in our design. Due to limited space, the UAV frame needs to fit all the electronics in an organized manner to make use of space as efficiently as possible. The frame is designed in a way that considers the accessibility of the electronics while remaining modular enough to add new components as necessary. The main plates are separated vertically to have access to the space in between. Since electronics are primarily housed between the central plates, we felt that they are sufficiently protected from damage in the event of a crash. This was confirmed during our first test flight when the UAV was dropped from around 10'. Further collision testing will be necessary in order to confirm the ability of the UAV to handle falls with all components attached.

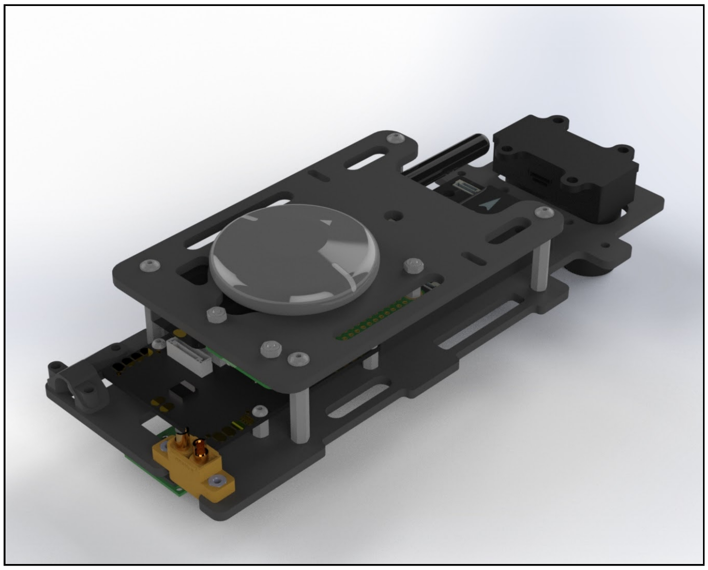


Figure 6.9: Render of Payload UAV Frame Plate Structure Assembly

The current design consists of two horizontal plates, made out of Nylon 6, that are separated by nylon vertical standoffs. It is designed to utilize the space between the guide rods while allowing the drone to take off freely. Through the center of the upper plate is where two armatures are mounted in an "X" formation, loaded by a torsion spring; this is referred to as the "X-Wing Mechanism" and will be discussed in a later section. Space between the bottom face of the lower arm of this mechanism and the upper surface of the upper plate is utilized to mount the telemetry and the GPS module in a compact manner without interfering with any other parts. Since PDR, the need to reduce weight and increase component compactness has led to the upper plate being shortened in length. While this raised questions on the team of whether the top plate can handle the forces of flight, through analysis and field testing with a 3D-printed PLA prototype plate, it appears that this part will function just fine; Nylon 6 has much greater strength characteristics to boot.

A lower plate is installed about 1.25" below the upper plate and houses the most essential components to the operation of the UAV. In this area, components are attached to both the top of the lower plate and bottom of the upper plate to get the best use out of limited space. In order to structurally connect these two plates, four vertical standoffs are attached between them with two bolts each. This will allow the corresponding bolt to be taken off for ease of access to other parts.

As the design process continued, worries about excess weight began to surface, this eventually resulted in the removal of much material from each electronics plate as possible. At this point, many more holes than originally designed are present on both plates, meaning more machining will be required, however the associated savings in weight will be much appreciated as more parts are loaded on. Again, since Nylon 6 is stronger than our prototype Polylactic Acid (PLA) material, we expect safety performance to be about the same or better.

Attached to the bottom plate of the UAV is the battery in an underslung 3D-printed PLA housing unit as part of the landing structure. Since the battery is the most volatile component of the UAV, it requires a sufficient level of protection to survive a collision without becoming punctured, becoming potentially dangerous to nearby users and the environment. This housing unit provides protection from four directions and the battery will be fastened with additional protective material and marked for ease of sight.

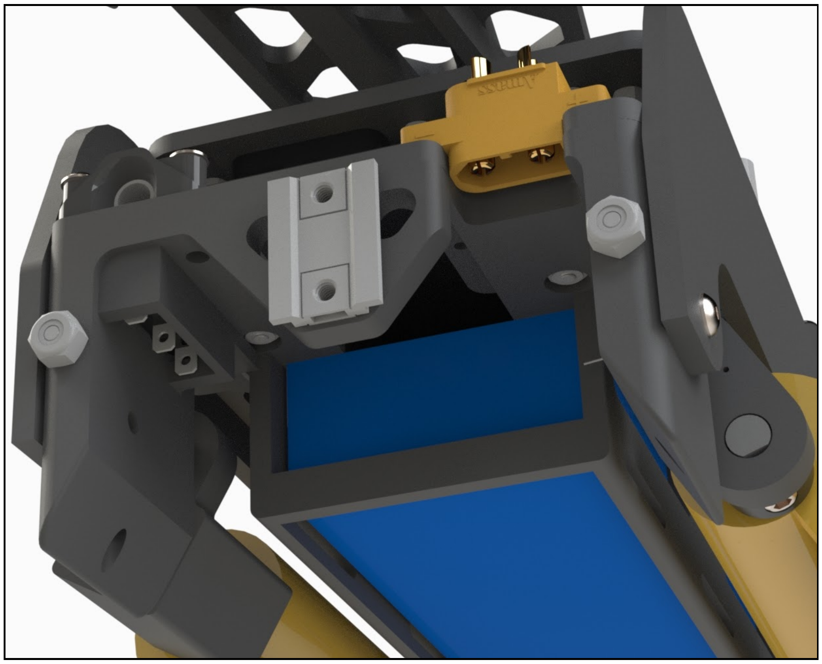


Figure 6.10: Render of Payload UAV Battery Housing from Below

Landing Structure

To support the landing of the UAV and facilitate attachment with the R&D sled, two sets of legs (one at each corner) have been fitted beneath the lower plate. These legs will also attach to the ice mining system, slung underneath the airframe. In order to produce the required complex shape to meet all attachment constraints, a 3D model has been developed for printing in PLA. This shape will allow for all of the possible attachment methods to be further considered and tested. Nothing will be directly underneath the Light Detection and Ranging (LiDAR) sensor or the camera to prevent obstruction. Currently, this has resulted in a design that can sit on the angular sides of the R&D sled while being pinned in place by the R&D sled locking mechanism. The legs have holes for attachment of the lower plate, ice mining assembly, as well as a small point of attachment for the UAV's Aero Package. The ice mining system will be attached by pins running through either end and will be positioned in such a way so as to fit on top of the sled and be able to move out of the way of the launch vehicle structural rods as the UAV exits the vehicle.

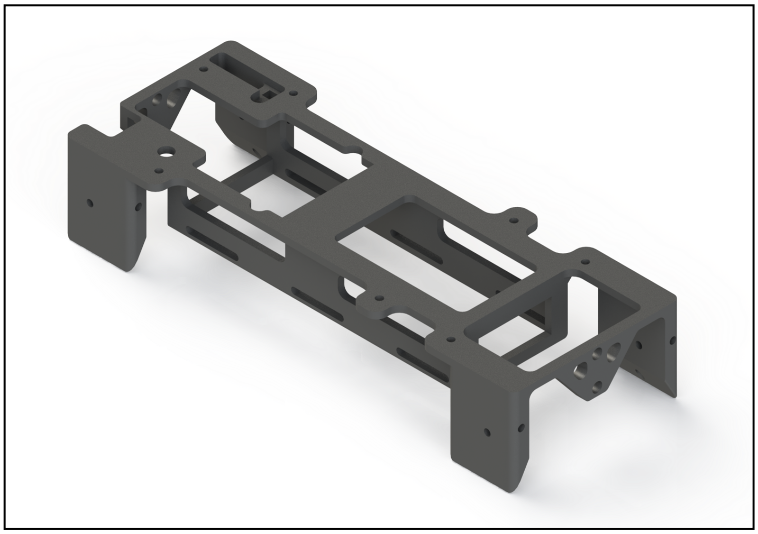
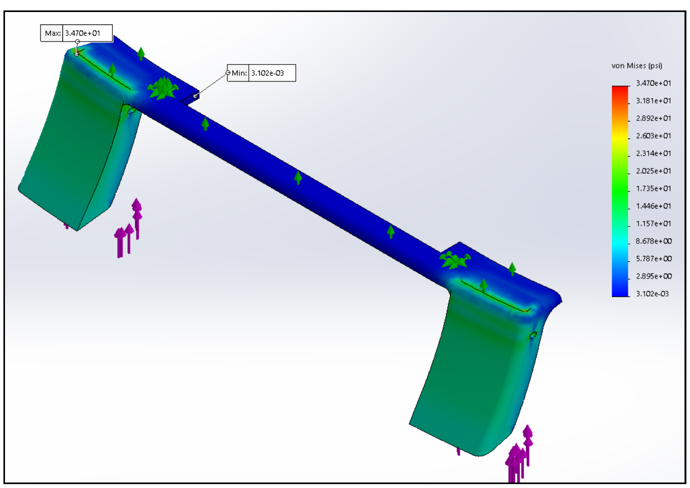


Figure 6.11: UAV Landing Leg Structure

This part requires many constraints to come together all at once, also including being the first load-bearing part to come in contact with the ground upon landing. Therefore, much development has gone into its shape and structural integrity. While this part is intended to break apart or crush on landing before any other parts do, it still needs to be structurally sound enough to perform its job without incurring excess weight upon the UAV. In the future, as the UAV design continues to iterate, plans for reinforcement of the legs have been considered through FEA and topology optimization. These tests are in their early stages and will be implemented through new research done by our 3D printing sub-team. The main goal of this optimization is to maximize the stiffness to weight ratio and will utilize overlapping infill technology to increase strength in designated areas. Below is an early study of stresses on a single set of legs as well as a further developed topology optimization mesh generated to reduce the overall mass of the model by 80%.



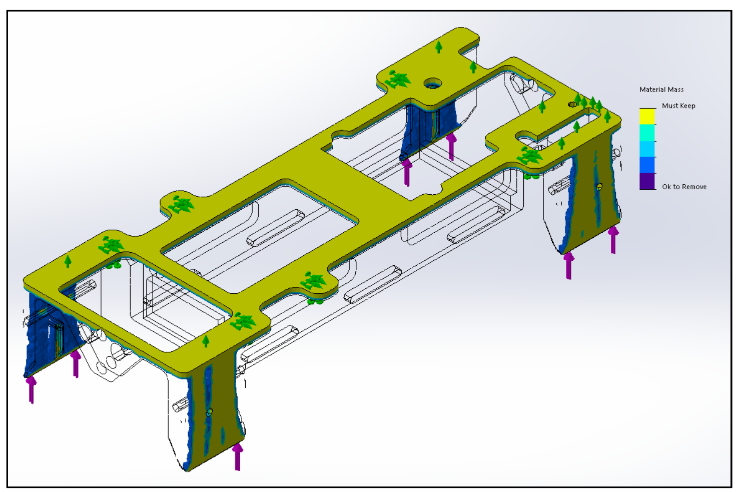


Figure 6.12: Payload UAV Landing Leg Stress Analysis (LEFT)
Figure 6.13: Payload UAV Topology Optimization FEA (RIGHT)

6.3.1.2. X-Wing Mechanism

In order to allow the propellers to provide adequate lift, they need separation from one another by as much distance as allowable without intersecting with any flight components. Since the internal diameter of the launch vehicle limits the maximum separation distance while the UAV is contained within the R&D system, the lift production units will need to be moved into position once released. Since PDR, the system providing this motion has been reworked to be active, rather than passive; the activation of the retention system releases the UAV's outer constraints, but the UAS maintains its own separate activation method.

The dimensional boundary of the UAV is required to be contained within the 6" diameter launch vehicle airframe section. With internal components, the UAV will need to be further constrained to approximately 5.77" in diameter. With the height constraints imposed by the sled of the retention system and a 3.7" width constraint between retention system structural rods, the maximum height from the top of the sled to the inner wall of the launch vehicle airframe is about 3.4"; much of this height has to be dedicated to electronics and ice mining. Using these values as a basis, the opening mechanism design involves the folding of lift unit "pylons" or "armatures" to physically shrink the "bounding box" of the lift system while inactive.

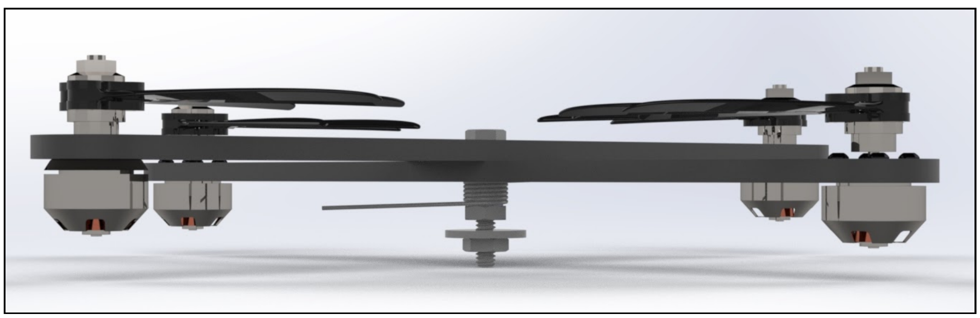


Figure 6.14: Orthographic Render of Payload UAV X-Wing Mechanism - Open Configuration

The current design, seen in the figures above and below, follows a seemingly uncommon concept for removing as many moving parts as possible in an effort to lower complexity in manufacturing as well as overall weight. The opening mechanism referred to as the "X-Wing Mechanism" includes two 12" overlaid armatures that can rotate about a single central axis running through the UAV. At either end of each armature is one of the UAV's four lift-producing motor units. While in its closed configuration, the armatures are within 30 degrees of each other, allowing the overall width of the assembly to be minimized under the target 3.7" width. Both armatures are installed with a passive opening actuator in the form of a coaxial torsion spring at 3.1in-lbf maximum torque. This spring will attempt to separate the two armatures to near perpendicular configuration, allowing for the lift producing units to maintain a maximum separation distance. Through observation, it was discovered that a full 90 degree separation would cause the lift-producing propellers to intersect the main rocket body while spinning, so this angle needed to be decreased to prevent damage and failure to deploy. In order to provide a stopping point to prevent further rotation, standoffs will be positioned in such a way that further than about 85 degree rotation is prevented. In order to facilitate adjustment of this degree measure as components come together, the UAV's top plate includes two slots that provide flexible positioning of these standoffs as deemed necessary by the design crew. In the future, if a final desired degree measure is determined, these slots may eventually be replaced by a pair of holes.

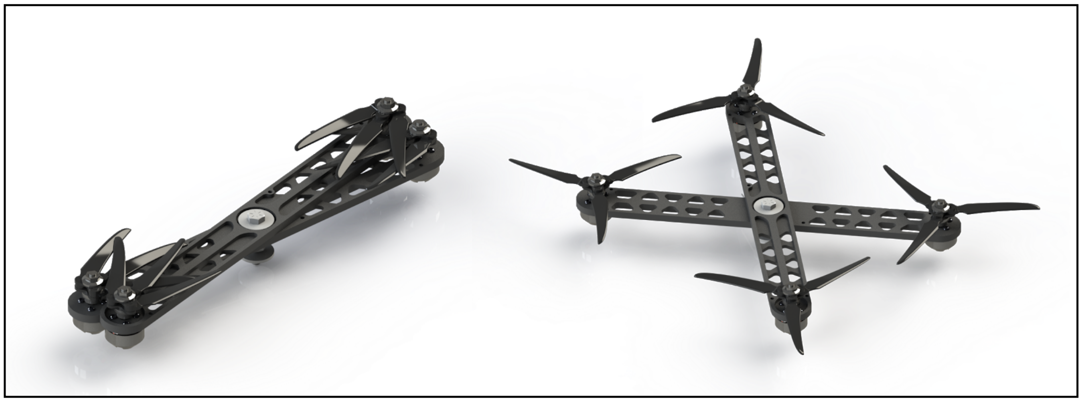


Figure 6.15: Isometric Renders of Payload UAV X-Wing Mechanism - Open and Closed Configurations

Inspecting this design, the forces applied during a flight are transferred through each individual armature pylon to the central washer and bolt, being further applied at the top of the airframe plate. The main concerns with continuing along with this design are whether its overall closed length, which is slightly less than 12", will be able to properly open once the R&D system deploys. At this time, the R&D sled contains a mechanism by which the X-Wing Mechanism will be allowed to employ its rotation maneuver once a pin is removed from the end of both armatures (this action coincides with the release of the UAV's legs). Considering that the X-Wing Mechanism will be constrained by the outer walls of the fuselage while in flight, it was decided that this design would be enough to prevent unnecessary opening and vibration during flight.

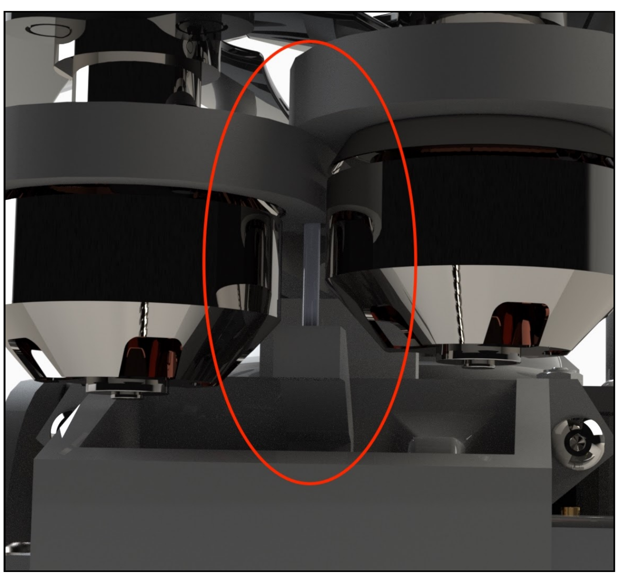


Figure 6.16: Render of Sled Pin in Payload UAV X-Wing Mechanism

Furthermore, an additional concern is whether the installed torsion spring can apply sufficient torque to open the X-Wing Mechanism; our current estimates assume to use a 3.1in-lbf torsion spring, however, this magnitude will be adjusted to fit the needs of the mechanism as prototype construction and testing continues. This quantity depends on a variety of factors including the coefficient of friction between the two armatures; while the currently selected torsion spring works for the materials used in our prototype, it is possible that the final product will have unseen differences that may hamper its function.

Finally, once again, as the design process has come to a close, weight optimization has begun to take over the team's concerns. Since the armatures of the X-Wing Mechanism have been analyzed to have much more than sufficient strength to carry the entire UAV, a few cutouts have been added to the Nylon 6 armatures to decrease their weight. FEA analysis appears to confirm that this form adjustment stays within our material's yield strength bounds for flight loading purposes; additional testing will be required to determine the structural integrity of these armatures in the event of a collision.

6.3.1.3. Aero Package

While the airframe's main plates appear to keep internal parts together quite well under collision, the team feared foreign objects poking inside, especially during an improper landing with the ground (i.e. upside down). As the design of the UAV approached its final form, it was decided that the UAV should be encapsulated by a shell that could provide additional protection during impacts and aerodynamic ability during flight. Inspired by the aerodynamic fuselages used by most modern moving vehicles, the UAV's simplistic 3D printed PLA Aero Package is designed to slip over the UAV body and cover its expensive internal components. At this point in time, the Aero Package is considered mostly to be aesthetic, though as testing continues there remain some potential plans to further develop the aero package to decrease its parasitic drag coefficient in the direction of motion through the elimination of sharp corners and rough surface; time permitting, this could be performed through Computational Fluid Dynamics (CFD), which a few members of the team have access to.

Figure 6.17: UAV Airframe Shell

6.3.1.4. Material Selection

Material selection for components on the airframe primarily involved considerations to cost, weight, manufacturability, and most importantly component application. The material considered for plate components that were structural and/or responsible for protecting electronics were Carbon Fiber Reinforced Polymer (CFRP) composites, Fiberglass Reinforced Polymers (FRP), and Nylon 6. Solidworks FEA was used to complete a static loading analysis to observe the material behavior on the X-Wing Mechanism arms under several load conditions. This analysis was completed in PDR and lead to the selection of Nylon 6 due to its nominal material behavior under maximum loading conditions, low risk to schedule, and low impact on budget.

During CDR, the X-Wing was optimized to reduce weight and improve structural integrity to ensure that Nylon 6 would be a favorable choice compared to the other choices. Seen below in Figure 6.18 is an FEA Simulation used to optimize the X-Wing to see if weight could be reduced while improving structure. With Nylon 6 as the chosen material, the optimized X-Wing at its maximum 2-G expected loading has a minimum Factor of Safety (FOS) of 4.2. The X-Wing is seen to approach failure at an 8-G case which the UAV will hopefully never endure outside the launch vehicle. Given the large FOS for the expected maximum flight loading, the Nylon 6 material was chosen for all airframe structural applications which include the X-Wing pylons and the plates on the central plate structure.

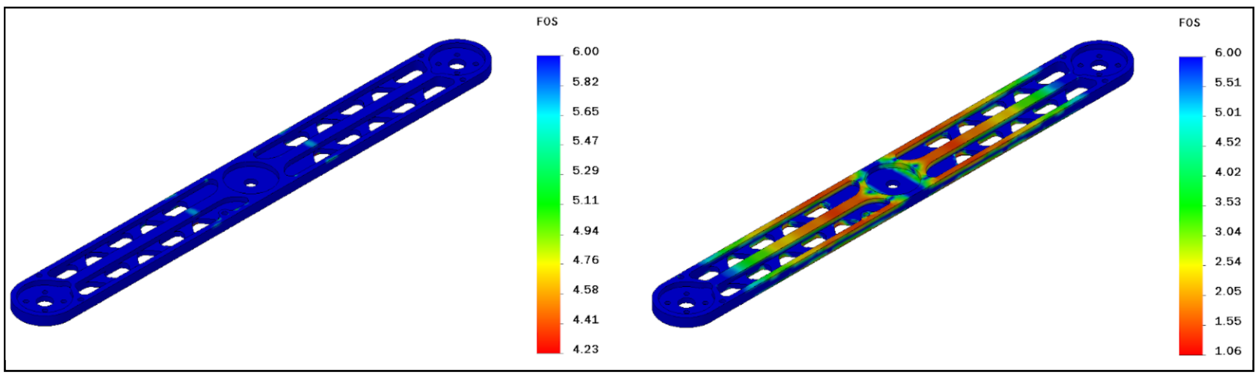


Figure 6.18: Payload UAV X-Wing Pylon FEA Simulation (2G Loading Left, 8G Loading Right)

The remaining components on the airframe are characterized by complex geometries that would be difficult to manufacture traditionally. It was decided that these components would be additively manufactured using fused deposition modeling giving the following material choices: PLA, Acrylonitrile Butadiene Styrene (ABS), Polyethylene Terephthalate Glycol (PETG), and Carbon Fiber (CF) reinforced PLA. CF reinforced PLA was chosen to be the material of choice for all 3D printed components on the airframe. Although CF reinforced isn't stronger than normal PLA, it has higher rigidity and would perform favorably for its application as a leg structure or the aero package. ABS and PETG were considered but it was determined that components made this material may be difficult to manufacture correctly and have unfavorable material properties.

6.3.2. Propulsion System

The propulsion system is a set of Electronic Speed Controllers (ESCs), brushless Direct Current (DC) motors, and propellers that work together to produce thrust that drive the UAV. Selection for each of these components primarily focused on the following objectives: minimizing weight, maximizing efficiency, and maximizing vehicle stability. Selection of these components were also be constrained by cost and size given the tight envelope inside the launch vehicle.

6.3.2.1. Motor Selection

Selection of the motor involved minimizing weight and size and matching power requirements. In order to minimize space, the motors were required to mount underneath its structural attachment so that the propeller shaft would run the structure and the propeller would lie just above it. Based on the maximum expected thrust-to-weight ratio of 2.0 and a maximum speed of 20mph, the motor was required to be able to output 102.5W of power. Based on these requirements, the lightest motor found that fit mounting, sizing, and space requirements was the Turnigy Aerodrive SK3 – 2822. The ESCs that were subsequently selected were based on the matching motor current requirement while minimizing weight. The ESC selected was the Turnigy Multistar ARM 21A since it more than doubles the 10A motor current requirement while being relatively lightweight and space efficient. The motor and ESCs can be seen in Figures 6.19 and 6.20 respectively.

Figure 6.19: Turnigy Aerodrive SK3 – 2822 Motor (LEFT)
Figure 6.20: Turnigy Multistar ARM 21A Electronic Speed Controller (RIGHT)

6.3.2.2. Propeller Selection

Given the spacing requirements necessitated by the UAV while stored in the launch vehicle and the space required when the UAV is deployed sitting on the sled, it was determined that the propellers needed to be foldable for compact storage and must be capable of unfolding when the motors are spinning at an idle throttle. In terms of efficiency, a larger propeller maximizes air contact and therefore efficiency, however, the propeller must fit within the stringent space requirements. In terms of flight stability, a propeller with more blades reduces the load on each propeller blade and provides a smoother flight but slightly decreases efficiency. The propeller selected that matched the criteria the best is the Lumenier 6.7x3x3 Folding Propeller seen below in Figure 6.21. This propeller provides a nice compromise between maximizing contact area with a 6.7" diameter and storage size. This propeller also has 3 propeller blades that yield smoother flight characteristics than commonly seen 2 blade folding propellers.

Figure 6.21: Lumenier 6.7x3x3 Folding Propeller

6.3.3. Ice Mining and Procurement System (IMPS)

6.3.3.1. Scoop Design

For PDR, there were two designs that the team had, a cylinder with multiple scoops around it and a cylinder with only one scoop across the whole side. Following PDR, the team was able to narrow the two choices down to one choice, the design with one scoop. When making the decision about which design, there were many considerations. The design employed one large cylindrical scoop that would rotate around an axis and collect the ice pieces as it spins. Below is an image depicting this design:

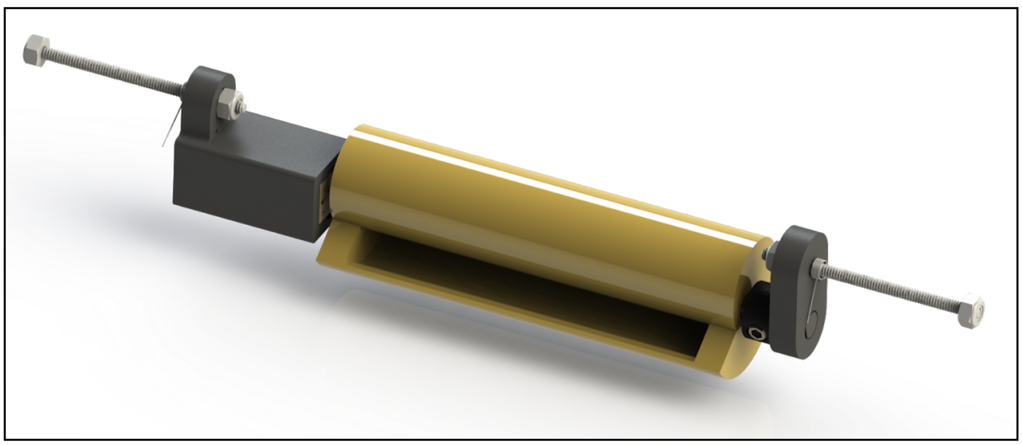


Figure 6.22: Complete IMPS Assembly

When making the decision about which design, there were many considerations. From discussions within the team able to decide that the one scoop option would be easier to manufacture. In addition to that, it was discovered that the single scoop would be able to pick up the lunar ice at a more efficient rate than the design that had multiple scoops. With the single scoop if material somehow gets jammed within the scoop the rest of the scoop would still be able to collect the material, but that would not be the case for the multiple scoop design as it would collect at a much slower rate due to a blockage in a scoop.

After refining the design, it was discovered that to make the manufacturing of the scoop would be extremely difficult if the scoop was 3D-printed in one piece. To avoid hours of cleaning out support from inside the scoop, there was a scoop cap created to make it so there would be no support that would need to be printed when printing both the scoop and the cap.

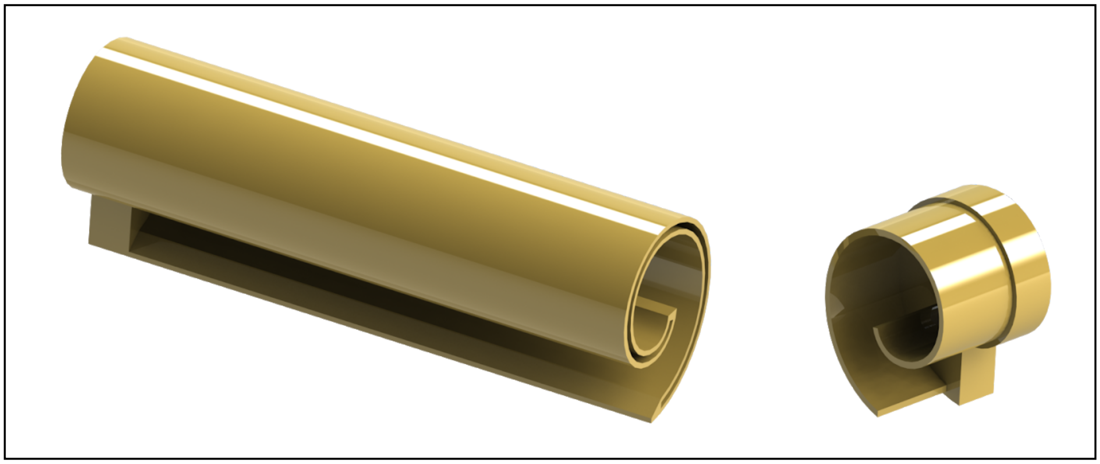


Figure 6.23: IMPS Scoop and Cap Assembly

Through testing, the team was able to confirm that our decision of the single scoop with the motor that we chose was the best option. In the PDR Q&A there was a picture of what the lunar ice so using the dimensions from the photo which were roughly 25mm across. In order to simulate the lunar ice, the team used plastic craft beads that were between 20-30mm across and placed them in a container so that the ice mining system could be tested. Using this testing system, the team was able to find that when the scoop and motor were held in place by the arms of the drone it was able to collect more that the 10mL of lunar ice consistently without assistance. However, if the beads were a uniform shape then they would form a type of lattice and scoop would have a much more difficult time trying to collect the beads. The team believes that this is only a problem with this test, since the real material appears to only have dissimilarly shaped particles that would not be able to form a lattice.

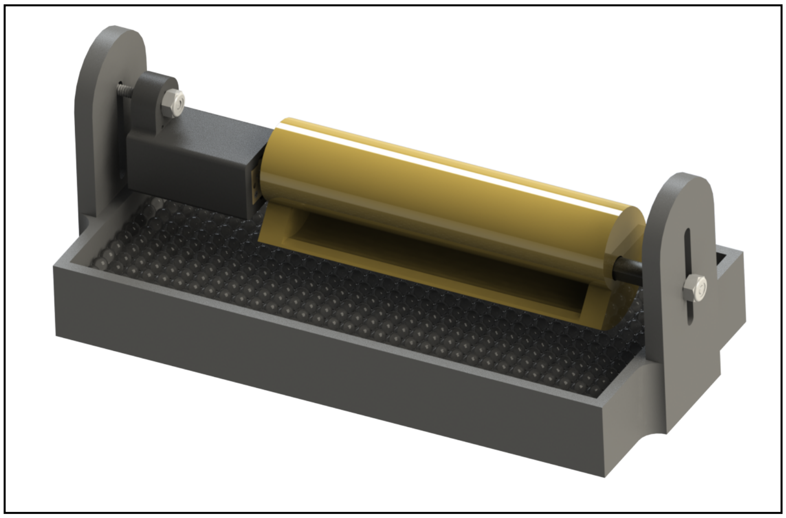


Figure 6.24: Payload Ice Mining Test Stand

6.3.3.2. Airframe Interface

The ice mining mechanism needed to fit on board the UAV in some fashion. This means all the same requirements and restrictions for dimensions applied. It must fit within the 5.77” inner diameter of the coupler. The team chose to place the system between the legs of the lander for a few reasons. The first being that the scoop had access to the ground and could mine for the granules. The next important reason was the fit of the UAV inside the airframe bay. This was one of the largest remaining spaces to place a payload on the UAV. The other options that were considered were under the battery tray and on the ends of the UAV. The first idea ran into components such as the lead screw on the bottom of the R&D system. Placing the system between the legs allowed for a long and spacious design to capture more granules compared to the spaces on the ends. The challenge with this decision were the guide rails. They presented an obstacle that would make a fixed design impeding to UAV take-off. That is why the design was placed on nylon all threads that allow for rotation of the whole ice mining system. This means, that when the UAV begins to takeoff from the sled, the ice mining system will rotate inwards under the UAV when it comes in contact with the guide rails. Torsion springs have been added to the design to encourage the system to spring back to the correct position after they have cleared the guide rails.

The ice procurement system attaches via two points of contact to the UAV airframe. The system fits in between the legs of the airframe. The motor that spins the clamshell scoop is encased by a mount that is 3D-printed to fit it exactly. The mount also extends the system out from the legs allowing to rest on the sled, rotate into the UAV, and rest on the ground when the UAV has landed.

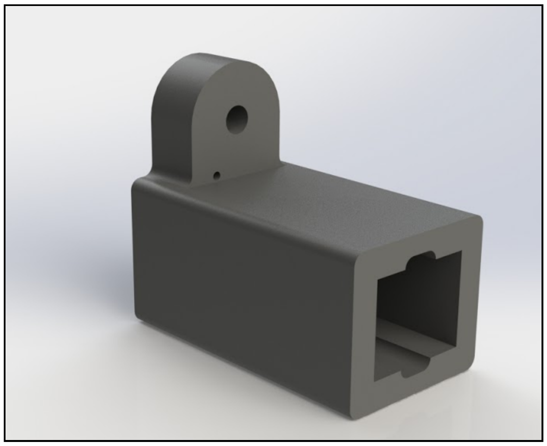
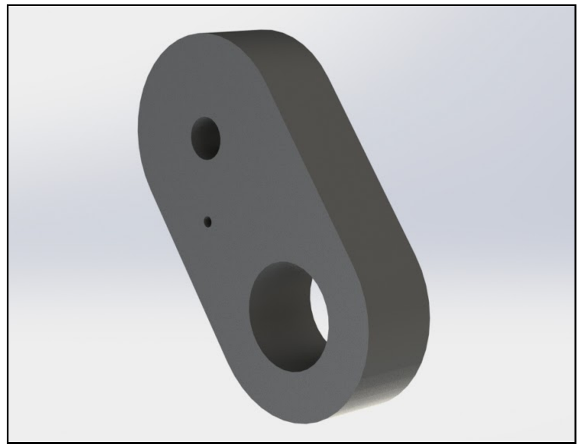
 

Figure 6.25: IMPS Motor Mount (LEFT) and Ice Clamshell Mount (RIGHT)

6.3.3.3. Electrical Design

The electrical design of the IMPS is tasked with controlling the actuation of the system. Brushed DC motors were chosen to drive each scoop in the system. Brushed DC motor technology was chosen due to its simplicity and its relative weight with respect to other types of motors stepper motors. The specific model of brushed DC motor that was chosen is pictured below in Figure 6.26.

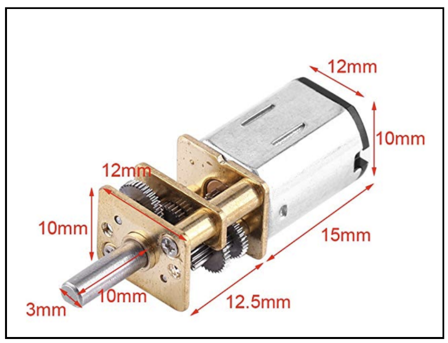


Figure 6.26: IMPS Brushed DC Motor

This motor was chosen primarily for its low mass and volume. Weighing in at only 10g, this motor would not significantly impact the overall weight budget of the UAV. This meant that an identical motor could be used for both of the scoops that comprise the IMPS. Preliminary developmental testing of the IMPS indicates that the gear system built into this motor can provide enough torque to actuate the system.

To drive this motor, a transistor-based drive circuit was employed. Figure 6.27 below gives the electrical schematic of this circuit.

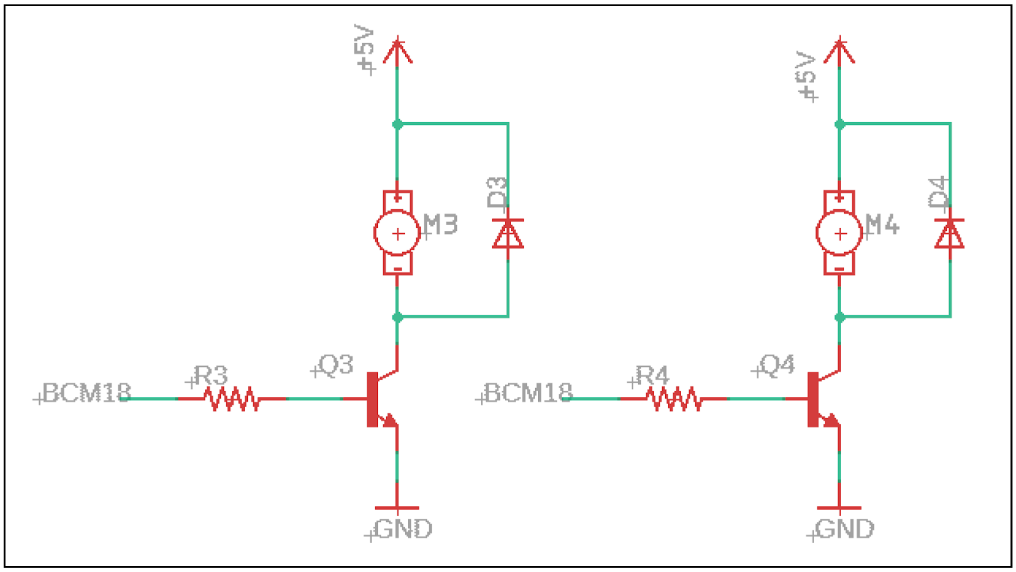


Figure 6.27: IMPS Motor Drive Circuit Schematic

A Bipolar Junction Transistor (BJT) allows the motor to be controlled by a digital logic pin on the Mission Control Unit (MCU). Each BJT is connected to the same digital pin on the MCU for simplicity and because both sides of the IMPS are to actuate at the same time. A fly-back diode is placed in parallel with each DC motor to protect the transistor from voltage spikes associated with changes in current through the motors. The motors themselves draw current from the 5V supply regulated by the Power Distribution Unit (PDU).

6.3.4. Flight Control and Mission Management

6.3.4.1. Flight Control System

The flight control system provides the UAV with attitude and altitude control and is comprised of a flight control computer and an integrated GPS and compass unit. The Flight Control Computer will require an integrated sensor set that provides data necessary for position estimation and also contains the control laws necessary for UAV flight. The GPS will augment the inertial data provided by the FCC and also provide GPS guidance capability for autonomous flight.

Flight Control Computer

In order to provide the autonomous capabilities necessary to complete the mission, the FCC requires sensors that provide barometric pressure, 6-axis acceleration, magnetometer data, and GPS data. The Pixhawk 4 flight controller was selected to fulfill this need due to its relatively small size and weight in comparison to other commercially available flight controllers. The FCC handles all low-level flight control tasks, allowing for higher-level, “off-board” control to be handled by another device or transmitter. The FCC is mounted to the central plate on-board the UAV airframe, facilitating easier electrical connections between itself and its peripherals mounted elsewhere on the UAV. The FCC is pictured in Figure 6.28 below.

Figure 6.28: Payload Flight Control Computer

GPS/Compass

The GPS receiver and compass electronics used for mini UAVs are typically integrated and are used by the FCC to collect positioning data for use in mission guidance and position estimation. The combined GPS/compass chosen for use in the Flight Control System (FCS) is the Pixhawk 4 GPS module. It not only includes the minimal functionality requirement of a GPS and compass, but also includes a safety switch and indicator Light Emitting Diodes (LEDs). The GPS receiver is pictured in Figure 6.29 below.

Figure 6.29: Payload GPS Receiver

6.3.4.2. Mission Management System

The Mission Management System (MMS) will handle the processing of incoming commands from the GCS and issue all setpoint altitude and setpoint position data to the FCS that will control flight of the UAV. The mission management system is primarily comprised of three components: the MCU, the AGL altimeter, and the digital imaging unit.

Mission Control Unit

The Mission Control Unit is the brain of the entire Mission Management System. The MCU needs to be able to communicate with two different systems (FCS and IMPS) as well as gather and parse information from the altimeter sensor and the Digital Imaging Unit (DIU). While the MCU is not directly responsible for UAV flight it will be responsible for sending the direction data needed to help navigate the payload toward the ice procurement area. Once at the ice procurement area the MCU will also be responsible for sending the signals to start and finish the ice mining.

The Raspberry Pi Zero microcomputer was selected to meet this need. A microcomputer was chosen instead of a simpler microcontroller due to the computationally intensive tasks required by the MCU. The computer vision algorithm that processes data from the DIU requires far more processing power than a device such as an Arduino would provide. The Raspberry Pi communicates over its serial port with the FCC, providing high-level, off-board control to the UAV. Further discussion of the MCU's integration into the electrical design of the UAV can be found in Section 6.3.7. The MCU is pictured in Figure 6.30 below.

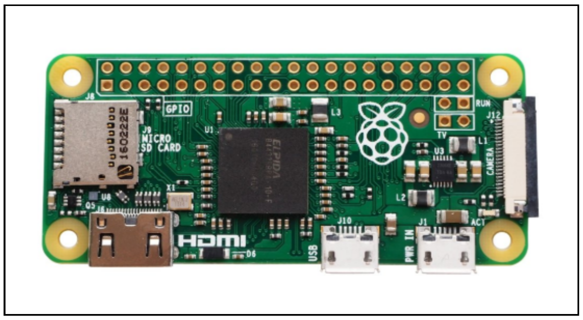


Figure 6.30: Payload Mission Control Unit

AGL Altimeter

The AGL altimeter will provide above ground level (AGL) data to the MCU that will be used for recovery area identification and validation. This information will be used directly for the inertial estimation of the UAV position and will also be used when descending towards and tracking the recovery system meaning that high accuracy is a requirement. By itself, the FCC estimates AGL altitude using a barometric sensor to measure relative altitude from its takeoff point along with Digital Terrain Elevation Data (DTED). In order to better control trajectory during descent and to better identify and discriminate the recovery area, it was determined a distance sensor would be required to augment the barometric sensor in use by the FCC.

The LiDAR lite V3 was chosen for the AGL altimeter for its high operational range. Since one of the primary uses of the AGL sensor is in the UAV's descent toward a lunar ice recovery area, it was important that the sensor could give precise data for distances exceeding 100'. The LiDAR lite V3 has a maximum range of 130', thus fitting this profile. The sensor is placed at the front of the UAV's airframe, pointed straight down at the ground. The sensor communicates with the MCU over the commonly-used I^2C communication protocol, limiting the need for additional General Purpose Input/Output (GPIO) pins on the Raspberry Pi Zero. The AGL altimeter is pictured in Figure 6.31 below.

Figure 6.31: Payload AGL Altimeter

Digital Imaging Unit

The DIU is the base of what will be used for locating the lunar ice. The DIU is a digital camera that will be connected to the MCU and will supply image data for subsequent processing. The DIU will be mounted to the bottom of the UAV and periodically take pictures of the ground. These pictures will be sent to the MCU which will then analyze them to guide the UAV to the lunar ice.

The Raspberry Pi Camera Module was selected for the DIU due its low power consumption and its ease-of-use with the Raspberry Pi Zero. The Pi Camera is capable of capturing 8 megapixel images, far exceeding the minimum resolution needed for object detection. The DIU is mounted to the underside of the center plate of the UAV airframe, pointing straight at the ground. A proprietary ribbon cable is used to send data from the cable to the MCU. The DIU is pictured in Figure 6.32 below.

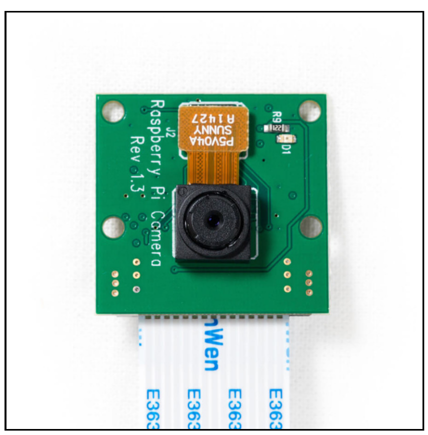


Figure 6.32: Payload Digital Imaging Unit

6.3.4.3. Software Design

Navigation Algorithm

After the UAV is successfully deployed, its next objective is to quickly and efficiently locate a lunar ice recovery site. General size and shape data for each recovery site is known, and approximate GPS coordinates for each site will be gathered in the hours leading up to the mission. These two types of data points, GPS location data and visual imagery data, make up the two primary inputs into the UAV's navigation algorithm. Figure 6.33 below gives a high-level visual representation of the navigation algorithm.

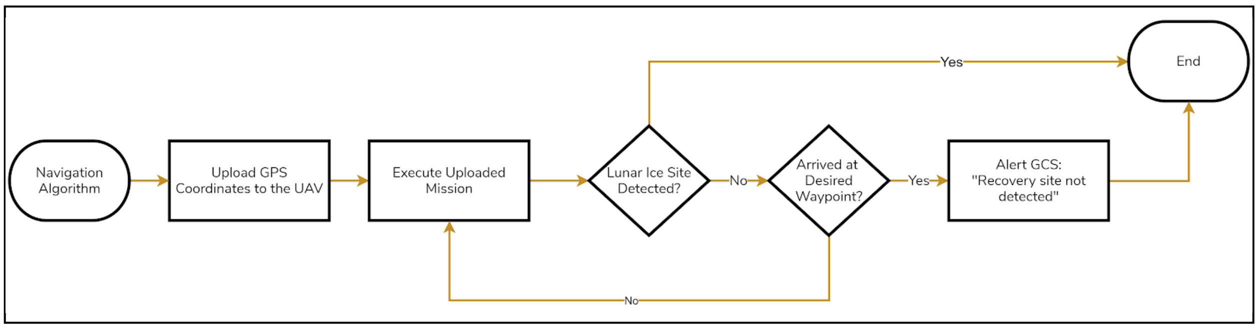


Figure 6.33: Top-Level Payload UAV Navigation Algorithm

The first step in the navigation algorithm is the selection of the desired recovery site. Since the starting location of the UAV is unknown, once the vehicle establishes a connection with the GCS, the coordinates of the closest recovery site will be uploaded in the form of a predefined flight plan to the UAV. The FCC supports a "Mission Mode" in which it will autonomously navigate to GPS waypoints built into such a flight plan. Throughout the UAV's flight to the recovery site, a recovery area detection algorithm will be run in parallel with the navigation algorithm. This allows the UAV to detect other recovery sites whose coordinates may not have been gathered before the mission. Due to the short amount of flight time available to the UAV before its batteries discharge, any time saved by stopping at a closer recovery site could be critical to the success of the mission. Once the UAV detects a recovery site, it executes its vision-guided landing algorithm, described below.

Recovery Area Detection

In order to detect lunar ice recovery sites while the UAV is flying, the DIU captures image data that is immediately processed on the MCU. OpenCV, an open source computer vision and image processing library, forms the backbone of this process. The software is designed to capture a new image with the DIU and then process it, looking for shapes in the image that could be a lunar ice recovery site. This shape detection is accomplished through a sequence of processes taken on the image, including resizing the image, converting the image to grayscale, and drawing contours around shapes in the image. If an object is detected in the image, the centroid of the object, with respect to the pixels of the image, is calculated. This forms the basis for the vision-guided landing algorithm described below. It is important to note that the software on the MCU is designed to alert payload mission personnel via the GCS when a recovery area has been detected. The UAV will not advance to the next stage of its mission until a confirmation signal is received from the GCS. This prevents any error that may present itself in the object detection algorithm from prematurely advancing the mission.

Vision-Guided Landing Algorithm

Once the UAV has navigated to one of the lunar ice recovery sites and detected the landing site with its camera system, a vision-guided landing algorithm will execute. This algorithm is tasked with bringing the UAV from an altitude above the landing site of about 100' down to a soft landing on the simulated lunar ice. In order to accomplish this task, a control scheme has been implemented using the off-board control function of the FCC. The control loops run on the MCU which sends commands over serial to the FCC.

The first controller manages the descent velocity of the vehicle. A desired set-point velocity of 5ft/s is selected while the UAV descends at an altitude of greater than 10'. Once UAV reaches an altitude of 10' AGL, the set-point automatically changes to 1ft/s, slowing the UAV down enough to facilitate a soft landing at the lunar ice site.

The second controller is more rigorous than the aforementioned velocity controller. Seen below in Figure 6.34, a PID control scheme is used to control the UAV's lateral offset from the center of the lunar ice recovery site. In order to keep the UAV aligned with the center of the lunar ice site throughout the descent, a data input with higher fidelity than GPS coordinates must be used. Image data from the DIU is this data source. Each image taken by the camera finds the centroid of the detected recovery area in terms of the pixels in the image. The set-point of the PID controller is therefore a centroid position at the center of the image. When an error is found with respect to this set-point, the PID controller outputs a lateral correction to be made by the UAV.

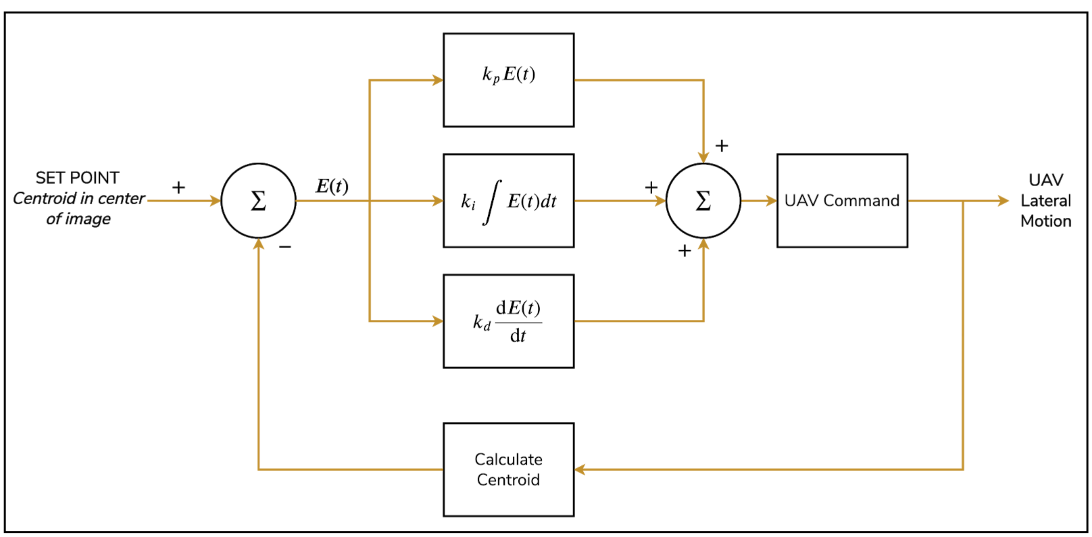


Figure 6.34: Vision-Guided Descent PID Controller for Payload UAV

Software-in-the-Loop Simulation Environment

A Software-in-the-Loop (SITL) simulation environment is being built to test the UAV behavior in a variety of different scenarios to aid in the development of the offboard control, navigational, and computer-vision software. The simulation environment is built using Gazebo: an open-source 3D robotics simulator that includes a high-performance physics engine that can accurately model the dynamics of the UAV. Gazebo also natively includes models for sensors that are found on the UAV such as a camera, LiDAR, magnetometer, GPS, and barometer. In addition to testing UAV software, the simulation can interface with GCS software and hardware to test the UAV's response to various commands and scenarios. The UAV model along with the payload bay in the simulation environment can be seen below in Figure 6.35.

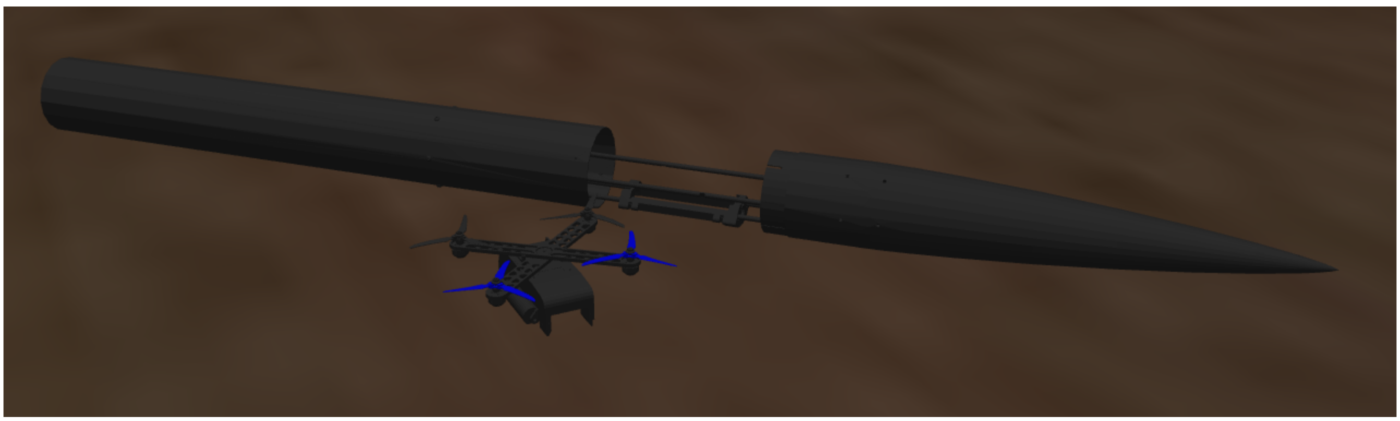


Figure 6.35: SITL Simulation Environment Models

6.3.5. Ground Control System

The mechanical and electrical design of the ground control station (GCS) have undergone slight modification and significant development. The control panel and displays have been simplified to improve the user interface and focus on providing feedback strictly relevant to safe and effective UAV operation. The electrical system has been designed to ensure extended operation of the GCS, and includes several system monitoring capabilities such as battery level monitoring. The software architecture on the GCS is modeled to be asynchronous and responsive to enable real-time command and control of the UAV and R&D systems. Internally, microcomputers and microprocessors on the GCS have been designed to communicate using command/status/data words similar to the MIL-STD-1553 message protocol and the Micro Air Vehicle Link (MAVLink) protocol in use by the FCS, but tailored and simplified for the application of the GCS.

6.3.5.1. Theory of Operation

The GCS provides the physical interface between the pilot in command and the UAV. It includes any and all communication interfaces between an operator, the UAV, and the payload retention and deployment system that will be necessary to complete the mission. Its functionalities include monitoring telemetry data, viewing image data, autonomous mission planning, monitoring mission status, and flight mode switching. In summary, the GCS will combine functionalities commonly seen in laptop GCS setups with an interactive mission control panel that will allow for quick and informative decision making. The GCS will run parallel to a Taranis Q X7 Radio Controlled (RC) transmitter that will act as a redundancy in case the GCS is unable to control the UAV and/or manual control of the aircraft is required.

Figure 6.36: Payload GCS Assembly Render

6.3.5.2. Physical Design

The physical design of the GCS is comprised of two components: the Display Head Assembly (DHA) and the Control Panel Assembly (CPA). The DHA is primarily a single monitor that will display real-time flight data, image data, and mission data relevant to safe and informed UAV operation. The CPA is a set of switches, buttons, LEDs, and a keyboard that provides complete control over the UAV during flight testing and general operation. The GCS may be mounted onto a tripod if necessary, or may be operated from a table. The physical design of the GCS in both the tabletop and outdoor configurations is provided in Figure 6.36 above and Figure 6.37 below.

Figure 6.37: Payload GCS Assembly Render with Tripod

Display Head Assembly

A majority of the feedback that the GCS operator will receive will be through the DHA. The DHA consists of two separate displays: a primary 17.6” display and a 2.8” battery display. The primary monitor will display up-to-date time-space-position information data in the form of a cursor on a satellite image, flight attitude data, AGL and moisture sensitivity level altitude data, onboard camera imaging data, and mission progress. The battery display presents data relating to GCS and UAV battery information such as amp hour consumption and cell voltages. A separate display was chosen to solely display battery information such that manual operation of the UAV would not necessitate an operational primary display. A set of speakers will also provide notable changes to the operation of the UAV which include flight mode switches, violations of pre-programmed flight paths, battery status, and any other faults that may threaten the safe operation of the UAV.

Control Panel Assembly

The CPA is a mechanical-electrical interface that allows for the direct control of the UAV mission and its operation. It consists primarily of push buttons and switches that can control flight mode, arming and disarming of the UAV, payload bay locking/unlocking, and entry/exit of specific mission phases. Feedback is included on the CPA through various LEDs that accompany buttons and switches and through a 20x4 character Liquid Crystal Display (LCD) referred to as the Vehicle Status Display (VSD). A guarded kill switch is also included that will disarm the UAV to enhance safety of operation. Various other external connections are available on the CPA including 2 Universal Serial Bus (USB) ports and an ethernet port that will allow testing of the GCS with SITL-type simulation.

Control of the mission is divided into five sets of switches that are used subsequent to recovery and are programmed to follow go/no-go requirements before the mission may proceed. These five switch sets are as follows: deployment of the UAV from the payload bay, orientation correction after a successful deployment then subsequent UAV unlocking, arming of the UAV after reaching a flight ready orientation and configuration and recovery area search, ice extraction, and finally sample recovery. Information related to the operation and control of the mission will be displayed on the 20x4 LCD monitor. If an operator attempts to proceed with the mission without meeting all of the entry/exit criteria, a message will appear on the 20x4 LCD monitor detailing why. Presented in Figure 6.38 below is the preliminary configuration of the primary switches and the vehicle status display that will be used during the mission.

Figure 6.38: Payload GCS Control Panel Mission Switches

6.3.5.3. Electrical Design

The design of the GCS electrical architecture is derived from the physical design of the GCS, mission requirements, and finally the wireless connection and data links. The GCS electrical design needs to be able to detect all user inputs, process user inputs, provide operator feedback through three different displays, receive and distribute power, and finally communicate externally with the UAV and retention system. In order to meet these requirements, the design of the GCS electronics was broken up into three components: the Mission Control Computer (MCC), the control panel board, and finally the Power Distribution Board (PDB). The MCC lies at the center of the GCS electronics and handles a majority of user input interpretation, data processing, and sending/receiving wireless messages. The CPB is the electrical interface between the MCC and the control panel, and is responsible for detecting user inputs and informing the MCC of the user inputs. The PDB is responsible for power distribution to all electronics. A functional diagram can be seen in Figure 6.39 below detailing the GCS electrical design.

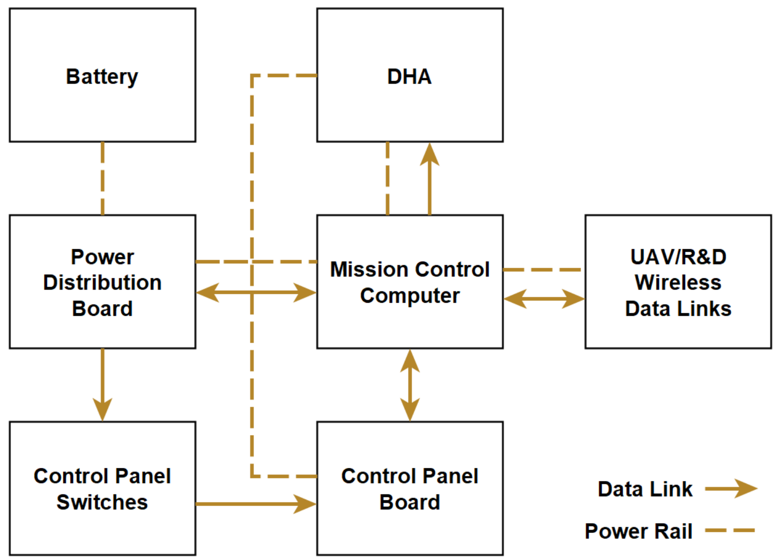


Figure 6.39: Payload GCS Electrical Design

Mission Control Computer

At the core of the GCS electronics is the MCC. The MCC handles all data processing, high-level decision making, and acts as a bus controller for the battery display and vehicle status display data links. The electrical device chosen to fulfill this roll is the Raspberry Pi 4 due to its ease of implementation and high-computational speed packed into a low-cost and lightweight package, all the while satisfying the number of serial/GPIO connections necessary to communicate with all other electronics.

Control Panel Board

In order to detect user input and provide feedback through LEDs, a control panel board with a dedicated microprocessor is used to monitor the position of all switches and push buttons and alert the MCC to any changes. The microprocessor chosen to monitor control panel inputs is the Teensy 3.6 due to its Input/Output (I/O) count, low cost and power consumption, and ease of implementation. The Teensy 3.6 is a small microprocessor packed with 42 easily accessible I/O pins and can handle the 38 connections required by the control panel. The teensy on the CPB connects directly to the Raspberry Pi 4 via its USB port and communicates serially across this port. A Printed Control Board (PCB) was designed to handle the high number of I/O connections and can be seen below in Figure 6.40.

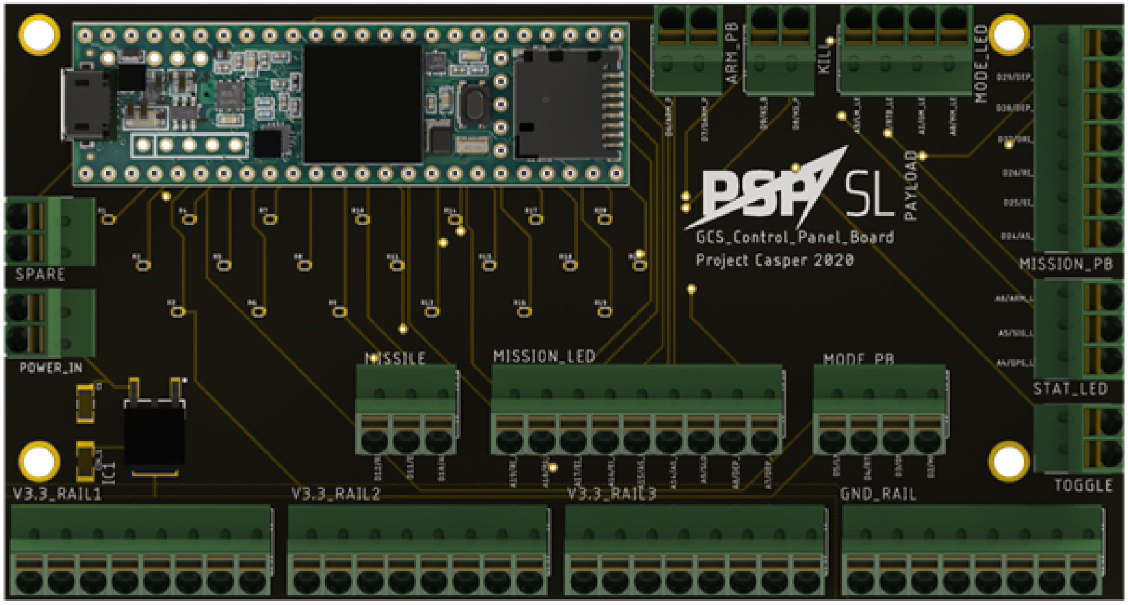


Figure 6.40: Payload GCS Control Panel Board

Power Distribution Board

The PDB serves several purposes related to power distribution, battery monitoring and battery display control, and sound system amplification. In order to complete these tasks, a custom PCB was designed with a dedicated microprocessor and can be seen below in Figure 6.41. The PDB interfaces with two 3 cell LiPo batteries such that one can be replaced at a time if battery levels are low. The PDB steps down the LiPo battery voltages to 5V that is then used by MCC and CPB along with other accessory components on the PDB. The microprocessor chosen to handle battery monitoring and battery display control is the Arduino nano due to its extremely cheap cost, high reliability, low power consumption, a USB connection for serial connection to the MCC, and enough I/O pins for battery monitoring and battery display control.

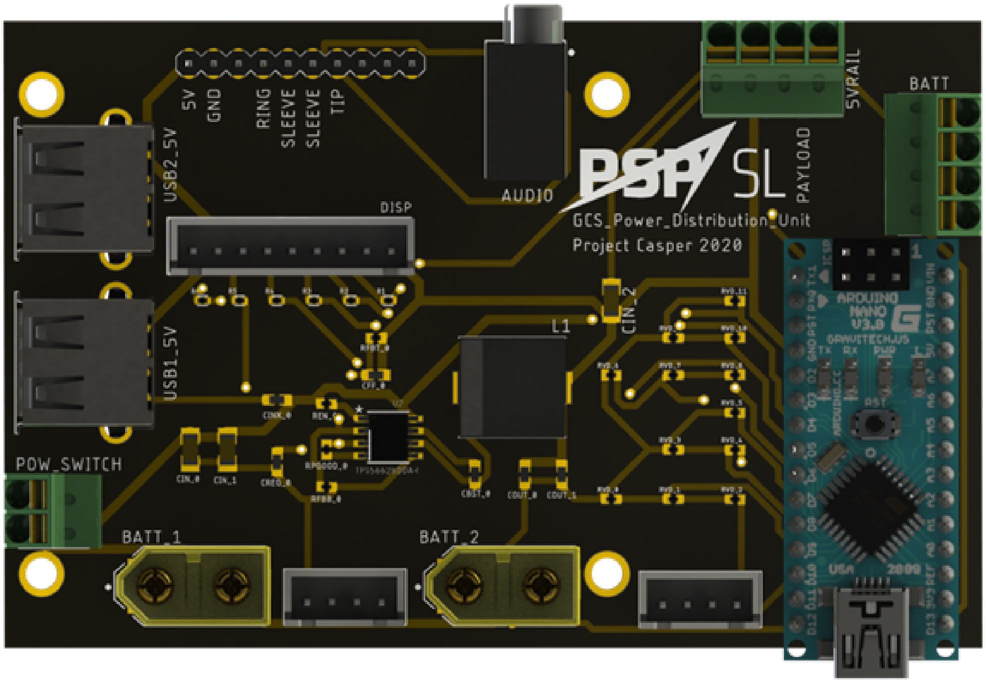


Figure 6.41: Payload GCS Power Distribution Board

The set of batteries chosen to interface with the PDB is the FLOUREON 11.1V 3S LiPo battery with a capacity of 5500mAh each giving a total capacity of 11Ah. A LiPo battery was chosen since they have a high energy density, are relatively cheap, and can easily be monitored. It is estimated that all display and processing electronics will consume a maximum of 4A continuously meaning an expected lifetime of 2.75hrs without battery replacement. To ensure that the GCS will remain operational during the mission, two spare batteries will be available to switch out giving a maximum operational time of 5hrs.

6.3.5.4. Software Design

Mission Control Software

The Mission Control Software (MCS) running on the MCC is designed to be asynchronous, responsive, and robust such that failures or software faults in any communication or data processing pipelines will not halt the operation of the MCS and can be restarted without relaunching the software. Implementation of the MCS will primarily be in Python with communication and telemetry monitoring using MAVSDK along with the front-end Graphical User Interface (GUI) being written with an open source Python library called Kivy. MAVSDK is a recently developed open source software development kit that enables applications to be written for MAVLink supported vehicles. MAVSDK enables real-time programmatic access to vehicle telemetry and the ability to generate mission profiles that can be uploaded to the UAV wirelessly.

Digital Bus Communication

The digital bus messaging protocol used by the MCC and microprocessors on the PDB and CPB uses command/data/status words to ensure the robust and efficient transfer of mission critical control commands and to reduce I/O computational requirements. Each of these words are 16-bit messages and contain data relating to word type, data depending on word type, and an odd parity bit to verify that the correct message was received. All transfer of data is initiated using a command word sent by any one of the processors. A data word is then sent subsequently after the command word to the device requesting or requiring data. Once that device has received data, a status word is returned to the other device informing that data has been transferred correctly. An example of this process can be found in Figure 6.42 below when the CPB informs the MCC of a change in the control panel configuration.

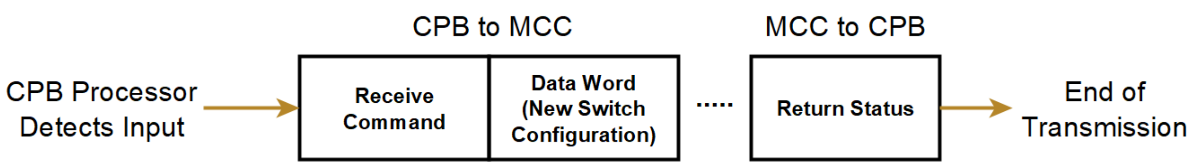


Figure 6.42: Payload GCS Data Transfer Example

This messaging protocol was chosen over a constant stream of data to reduce the I/O computational requirements for all processors. Each processor will send/receive data only if they need to. Also, this messaging protocol significantly increases the likelihood that the correct message is sent and that the message was received since the quality of the message is checked and a status word is transferred informing the sender that its message was received.

6.3.6. Communication and Data Links

Communication to the UAV is allowed through the GCS or an RC Transmitter. The UAV will have both an RC receiver and a telemetry unit that will be used to receive commands and send telemetry data. The RC transmitter will send data at a frequency of 2.4GHz, while the data links between the GCS and the UAV will communicate at a frequency of 915MHz. The diagram in Figure 6.43 below depicts the data-flow between all data-links of the UAS.

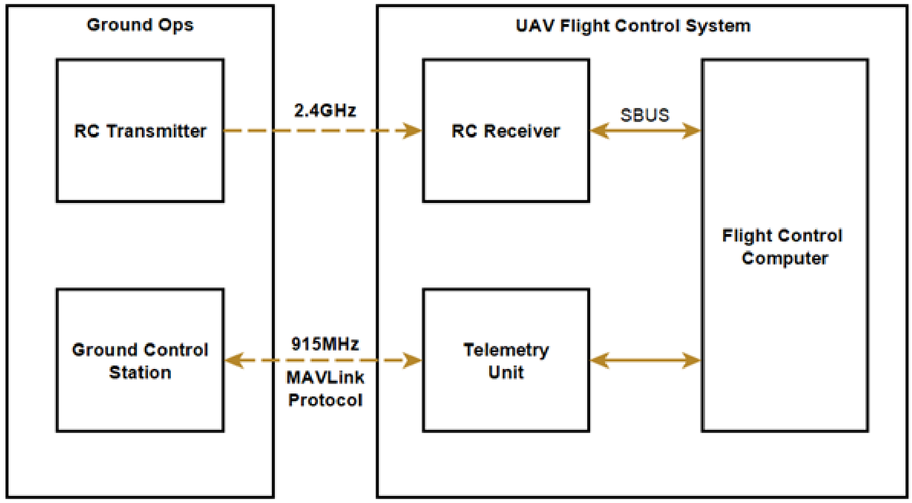


Figure 6.43: Payload GCS & Transmitter Communication / Data Link Example

6.3.6.1. GCS Data Link

The telemetry unit that will be installed on both the GCS and the UAV will be the 915MHz Readytosky 3DR Radio transmitter with a power of 500mW seen in Figure 6.44 below. Other options for telemetry radio operated on the same frequency band, but offered a reduced output power and therefore a reduced range. Based on the maximum expected operational ranges, the 500mW power output should be more than sufficient for BLOS range, however, testing will be conducted to verify this range. This power output was upgraded from 100mW due to concerns with range. Communication between the telemetry radios will use the open-source MAVLink protocol. The MAVLink protocol enables the sending of status data, command and control data, and image data.

Figure 6.44: Readytosky 3DR 915MHz Radio

6.3.6.2. RC Communication

An RC radio transmitter will be used for the manual and altitude stabilized control of the UAV in testing and general operation. The RC transmitter will be operated in parallel with the GCS but will not be required for mission operation; it will be used initially for flight qualities testing and as a backup when quick manual control of the UAV is required. Since the function of the RC transmitter only requires the manipulation of flight mode and 6-axis control, the most important consideration used in its selection is cost and availability. The selected RC transmitter is the Taranis Q X7 as it is relatively cheap but popular RC radio capable of controlling the UAV.

Figure 6.45: Taranis Q X7 RC Transmitter

The RC transmitter will provide one-way communication to the RC Receiver on the UAV primarily for manual and assisted UAV control. The selected RC receiver is the FrSky R-XSR receiver chosen primarily due to its low weight of 1.5g and compatibility with the chosen RC Transmitter.

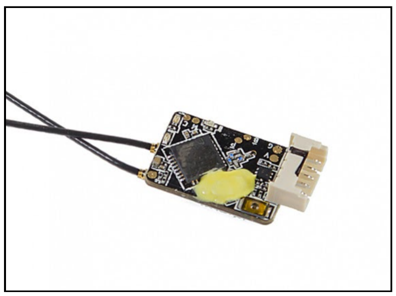


Figure 6.46: FrSky R-XSR Receiver

6.3.7. UAV Electrical Design

The UAV electrical design has two primary objectives in supporting the overall payload mission. First, the electronics on the UAV must be capable of providing proper voltages and currents to all electrical components in the system, including flight control, image processing, and ice mining components. This power system provides the sole source of energy for the UAV. For this reason, adequate energy capacity must be combined with intelligent power consumption to ensure the UAV is powered throughout the entirety of the mission. Careful analysis of this power consumption is essential to understanding the total amount of flight time available to the UAV.

Secondly, the electrical design of the UAV must facilitate the communication between the various components within the system. This includes the integration of sensors such as the AGL altimeter and the GPS, as well as serial communication between the MCU and the FCC. Details of the UAV electrical design, as well as an analysis of the UAV's power consumption, are given below.

6.3.7.1. UAV Battery and Power Distribution

At the heart of the power system on-board the UAV is an 11.1V lithium polymer (LiPo) battery. Lithium polymer battery technology was selected for this application for two primary reasons. First, LiPo batteries are ubiquitous in hobby UAV and RC applications due to their high energy density. LiPo batteries provide higher amounts of energy for less weight than other battery technologies such as nickel cadmium or nickel metal hydride. This is an essential feature for small, airborne applications such as a UAV. The second reason for choosing a LiPo battery to power the UAV is the technology's higher current discharge rate. In general, LiPo batteries are better for higher-current applications than competing technologies. Quadcopter UAV configurations are an example of such a high-current application. Each of the UAV's four motors could draw on the order of 2-5A of current. A LiPo battery is well suited to meet this power requirement.

The energy supplied by the LiPo battery must be safely distributed to the various electrical components on-board the vehicle. These components, such as the flight computer, the ice mining system, and the electronic speed controllers, operate with different power requirements. To fulfill this need, a power distribution system, based on Pixhawk 4 Power Management Board, is employed. This device was chosen because it is specifically manufactured for the purpose of distributing power safely to the Pixhawk 4 flight computer and associated UAV peripherals. Presented below in Figure 6.47 is a block diagram displaying a high-level view of the power distribution on-board the UAV.

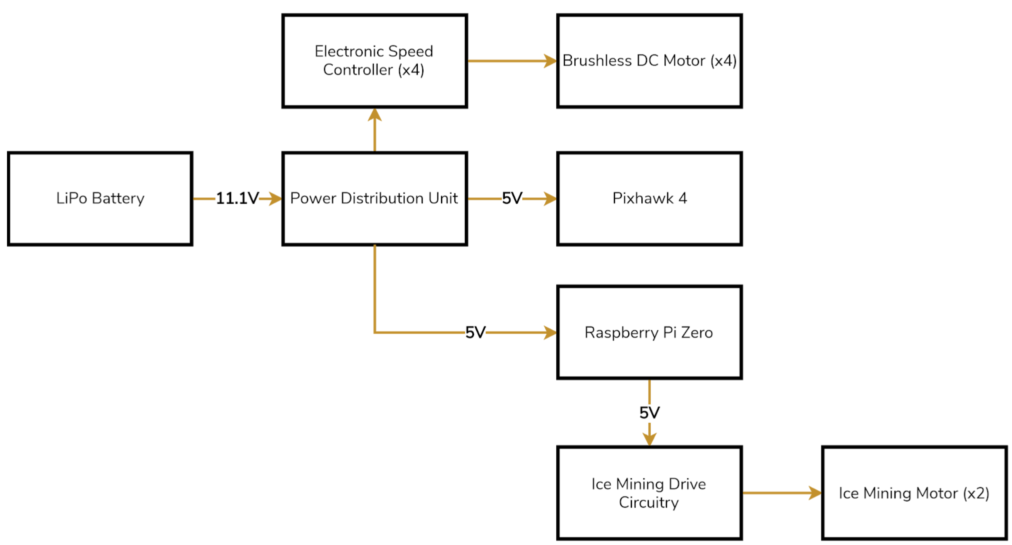


Figure 6.47: Payload UAV Power Distribution Block Diagram

This design makes use of the Pixhawk 4 Power Management Board's built-in voltage regulation, which yields a clean 5V signal that can be used to power other components on the UAV.

6.3.7.2. Power Consumption Analysis

One of the most important metrics associated with the electrical design of the UAV is its estimated power consumption. This number directly impacts the amount of time that the vehicle can stay in the air before depleting the energy in its battery. The power consumption of the UAV can be broken down into power consumed by the propulsion system and power consumed by the MCU. Power drawn by other components in the system, such as the FCC or the ice mining motors, was deemed negligible for this analysis, as these components either draw very little average power, or draw power for a very short amount of time.

Looking first at power drawn by the MCU, it is estimated that the system will draw, at maximum 1W of power, beginning when the UAV is deployed from the launch vehicle. This is a conservative estimate, based on oft-cited online data on Raspberry Pi Zero power consumption. According to this data, a Raspberry Pi Zero shooting 1080p video draws about 240mA of current at 5V. The Raspberry Pi Zero at the heart of the MCU will be capturing and processing images at a lower resolution and framerate than this. Therefore, this data can be used as an upper-bound on the power consumption of the MCU as a whole.

By far the biggest source of power consumption on the UAV is the propulsion system. A thorough analysis of the power consumption of such a system requires derivation of the relationship between electrical power consumption and a wide array of factors, such as mass, propeller size, propeller pitch, and battery properties, among others. The eCalc xcopterCalc online calculator, popular in the hobbyist RC community, was employed for this task. The tool utilizes a vast database of thousands of RC parts to estimate key metrics of any custom-built RC vehicle. Table 6.1 below outlines the data returned by this tool for this UAV.

UAV Power Consumption Data	
Hover Flight Time:	10.1 min
Single Motor Power Consumption (Hover):	50.5 W
Single Motor Power Consumption (Full Throttle):	102.5 W

Table 6.1: Payload UAV Power Consumption Data

The results of this analysis indicate the importance of minimizing the weight of the UAV. For example, subtracting just 100g from the mass of the UAV yields a flight time increase of roughly 90 seconds. During the fabrication of the UAV, this consideration will be critical in maximizing the amount of time the UAV can search for lunar ice.

6.3.7.3. UAV Circuit Design

In addition to supporting the power consumption and distribution requirements of the UAV, the electrical design must integrate the UAV's sensor package, facilitate communication between the FCC and MCU, and drive the IMPS. Figure 6.48 below gives the full system schematic for the electronics on-board the UAV.

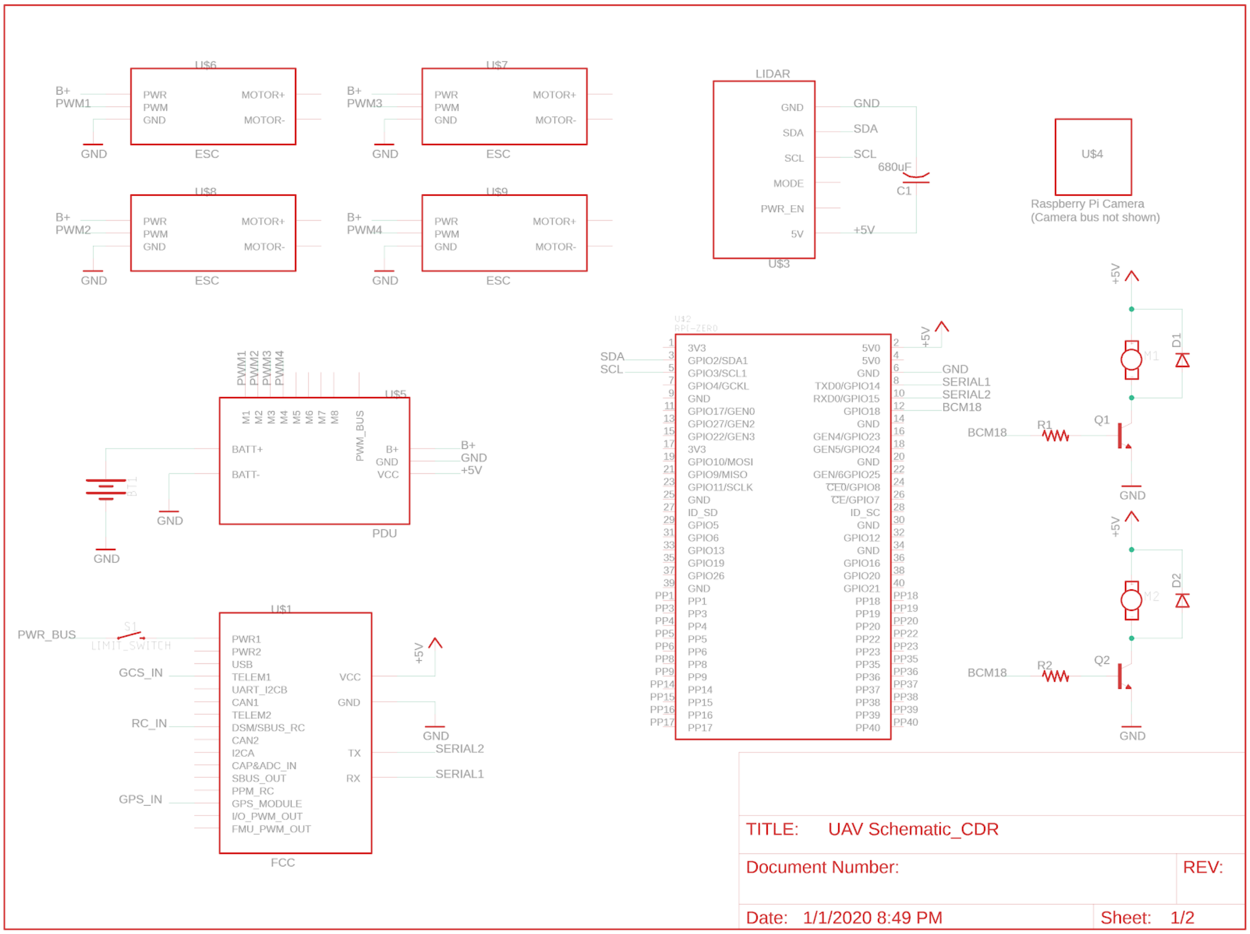


Figure 6.48: Payload UAV Electrical Schematic

Of particular importance to this design is the communication link between the Pixhawk 4 and the Raspberry Pi Zero. The system is designed such that Raspberry Pi can send high-level control commands via the MavLink protocol to the Pixhawk which in turn manages the low-level control functions to operate the UAV. To facilitate this communication, a serial connection between the Pixhawk 4 and the Raspberry Pi Zero was employed, making use of one of the two telemetry ports on the Pixhawk. This allows for a basic two-pin serial connection to be established, over which data can be sent using the MavLink messaging protocol. A diagram depicting this aspect of the electrical design is displayed below in Figure 6.49.

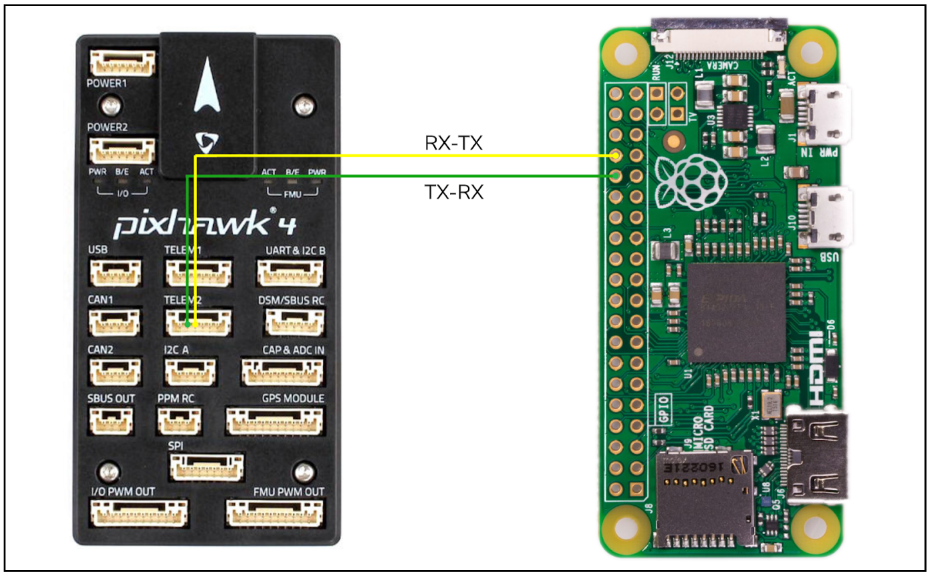


Figure 6.49: Payload UAV FCC-MCU Serial Connection

6.3.7.4. Switches and Indicators

The UAV is outfit with an array of switches and indicators, serving to increase its operational safety. Indicators on the UAV provide direct visual and auditory feedback related to the current state of the UAV. Switches on the UAV add additional layers of control and safety over the operation of the UAV. The specific switches and indicators found on the UAV are outlined below.

Pixhawk User Interface LED Indicator

The Pixhawk 4's primary means of signaling its current operational state is through an LED indicator located on its GPS sensor. This Red/Green/Blue (RGB) LED displays a number of different colors as well as solid vs. flashing states that provide critical information about the state of the UAV. Table 6.2 below gives the different LED states and their associated interpretation.

LED State	UAV State	Interpretation
Solid Blue	Armed, No GPS lock	The vehicle is armed but does not have a GPS lock. Cannot perform guided missions as a GPS lock is required.
Flashing Blue	Disarmed, No GPS Lock	The vehicle is disarmed, but there is no GPS lock. Always exercise caution when arming the vehicle.
Solid Green	Armed, GPS Lock	Vehicle is armed and has a valid GPS lock. Vehicle is able to perform guided missions.
Flashing Green	Disarmed, GPS Lock	The vehicle is disarmed and there is a valid GPS lock. Always exercise caution when arming the vehicle.
Solid	Fail Safe Mode	Vehicle will act according to fail safe

Purple		settings programmed into the vehicle safety. Includes Return To Base (RTB) or immediate landing - the latter is preferred.
Solid Amber	Low Battery Warning	Indicates the vehicle is low on battery. The vehicle should either RTB or land immediately.
Flashing Red	Error/Setup Required	Indicates that additional autopilot configuration is required such as calibration. Connect the flight computer to a GCS to troubleshoot.

Table 6.2: Pixhawk User Interface LED Indicator Interpretation

Pixhawk Status Buzzer

The FCC is equipped with a status buzzer that plays audible tones based on changes in the state of the UAV. Each state change is associated with a unique tone of duration 1-7 seconds. The tones are separated into two basic categories: startup tones and operational tones. Startup tones are triggered immediately when power is supplied to the FCC. These tones indicate whether or not the startup process was successful. Additionally, they signal the success or failure of firmware updates to the FCC. Operational tones have greater variability, but generally signal important warnings or give confirmation that some form of wireless data is received. For example, unique tones exist to indicate low battery or successful GPS lock. The Pixhawk Status Buzzer adds an essential layer of safety to the operation and development of the UAV.

Pixhawk Safety Switch

An additional layer of safety is provided by the Pixhawk Safety Switch located on the GPS receiver. This switch must be activated in order for the FCC to be armed. The Pixhawk Safety Switch is essential to mitigating safety risks associated with developing and testing the UAV. Additionally, arming this switch is an essential step in the UAV initialization procedure that is to be followed on the day of the mission.

UAV Power Switch

The UAV Power Switch is an essential feature of the UAV, allowing for automatic activation of the UAV's startup sequence after the vehicle has been successfully deployed by the R&D system. This safety feature of the UAV's design is a direct result of the requirement for the UAV to be completely powered off throughout the first phase of the mission. In order to automatically provide power to the UAV upon its deployment, a limit switch is added in series between the LiPo battery and the PDU. This switch is designed to be in an open state while the UAV is retained inside the launch vehicle, therefore opening the circuit and preventing power to the UAV. When the UAV is deployed, the UAV Power Switch is depressed, closing the circuit and supplying power to the UAV. Figure 6.50 below details the UAV Power Switch.

Figure 6.50: Payload UAV Power Switch

6.3.7.5. Handheld GPS Locator

One of the pieces of data critical to the success of the mission is the GPS coordinates of some or all of the lunar ice recovery sites. This is the primary data input to the UAV's navigation algorithm, but there are challenges posed to the acquisition of this data. The location of recovery sites is not readily available to payload mission personnel until the day of the launch, therefore a device capable of quickly gathering GPS data is needed. The Handheld GPS Locator (HGL) was designed to serve this purpose.

The HGL is comprised of three main components: an Arduino Pro Mini, A GPS break-out board built around the MTK3339 GPS chip, and a standard two-line LCD panel. The GPS sensor is capable of tracking up to 22 satellites with a maximum location update rate of 10Hz. The breakout board makes it easily compatible with the Arduino Pro Mini, a 5V microcontroller chosen for its small footprint, power consumption, and usability. The Arduino continuously polls the GPS sensor for new location data once the GPS has established a fix. A button on the HGL allows any instantaneous location data to be displayed on the two-line LCD panel. When gathering GPS data, personnel can therefore immediately record high-resolution location data that can be later sent to the UAV through the GCS. The schematic of the HGL is pictured below in Figure 6.51.

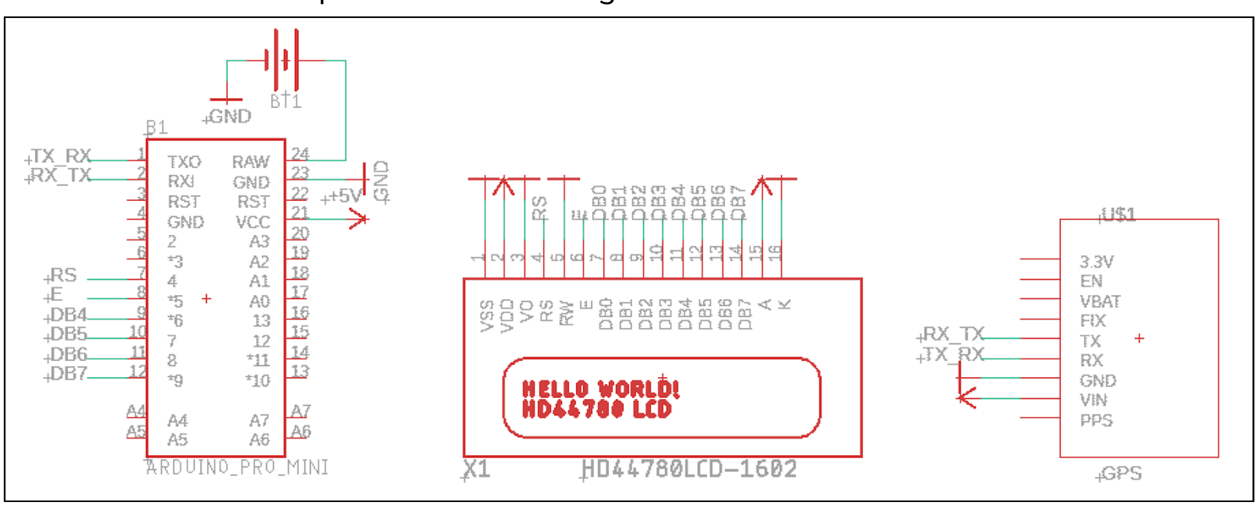


Figure 6.51: Handheld GPS Locator Schematic

6.4. R&D Detailed Design

6.4.1. R&D System Overview

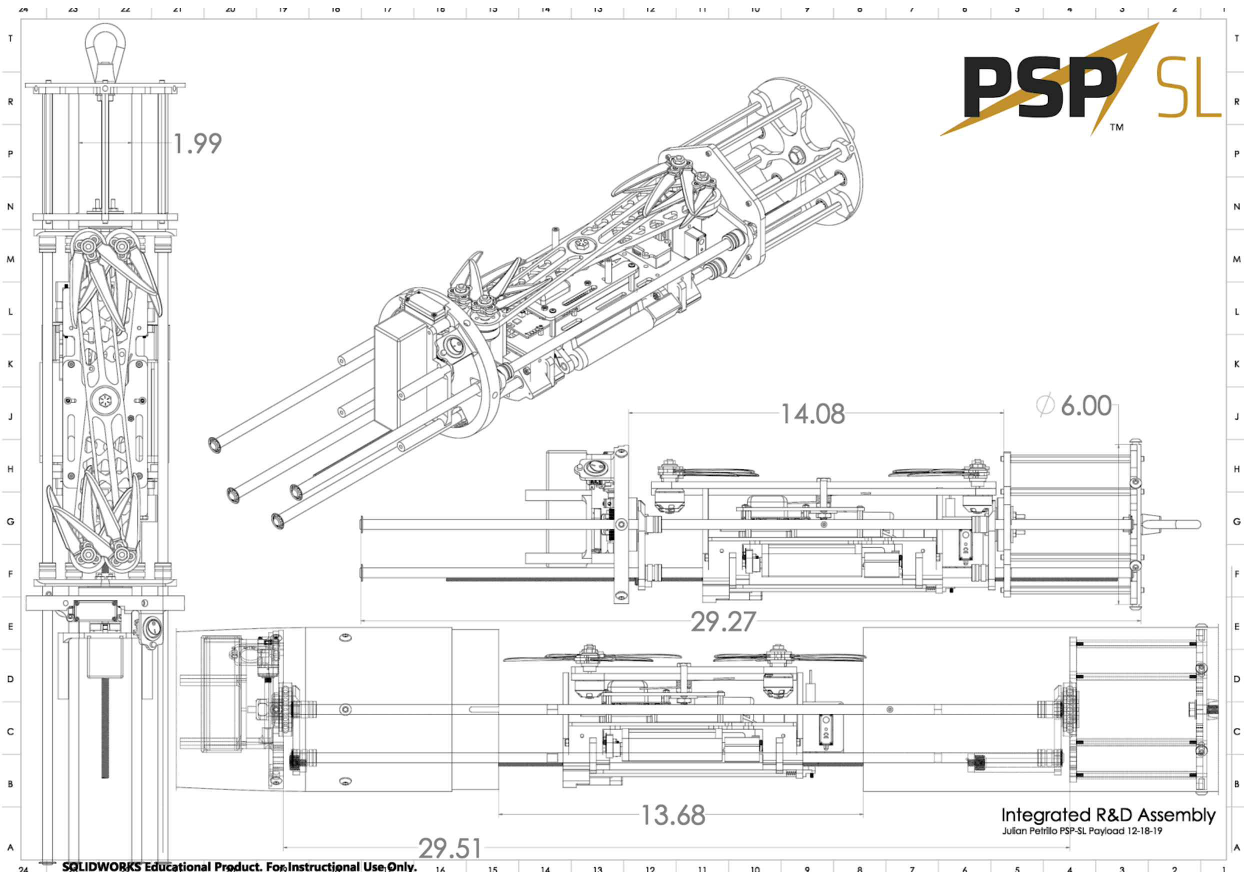


Figure 6.52: Integrated Payload R&D System Dimensional Drawing

The R&D system is comprised of the payload retention system, the sled deployment system, the payload orientation system, and the R&D electronics system. These systems work together to safely hold the UAV payload within the rocket throughout the course of the flight and to facilitate a successful deployment of the UAV after landing. Conceptually, the overall R&D system has been unchanged since PDR, but major design changes have been made within each subsystem due to assembly and manufacturability concerns as well as realizations from initial integration testing.

6.4.2. Payload Retention System

The payload retention system has undergone significant change and improvements since PDR to improve and extend its functionalities. The design of the sled focused primarily on the safe transportation of the UAV during launch, however, additional requirements derived from the design of the passive X-Wing deployment on the airframe and the electrical isolation of the flight computer necessitated an actuated system. The payload retention system, seen in Figure 6.53 below, uses a servo-controlled rack and pinion linear actuator to control the release of the airframe's passive mechanism, to prevent any vertical motion, and to isolate power to the FCC while the UAV is inside the launch vehicle. A single servo-controlled dual rack and pinion design was chosen as it fulfilled all the requirements discussed previously. Other methods, such as a barn-door mechanism that held the arms in and a spring-loaded latching mechanism were considered, but did not fulfill the requirements listed above and deemed to be too complex.

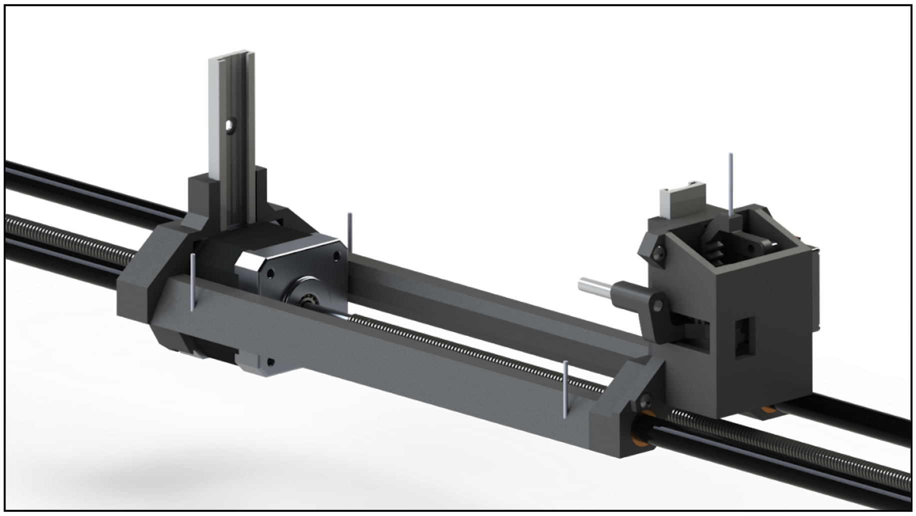


Figure 6.53: Payload Retention System

Locking and unlocking of the UAV is controlled by a servo-mounted pinion that is in contact with two racks. The locked and unlocked configuration can be seen in Figure 6.54 below. Both racks hold a rod that move perpendicular to each other simultaneously. The rack that moves laterally serves the purpose of preventing vertical motion during deployment and contacting a limit switch that electrically isolates the FCC. This rack will not be solely responsible for preventing vertical motion during flight; a mechanical tab inside the rocket body will contact the top of the UAV airframe to hold it down so the rack faces a reduced structural loading. As the UAV is being pushed outside of the launch vehicle, the tab slides off the top of the UAV and leaves vertical control solely to the sled. The X-Mechanism rack that moves vertically simply slides into a hole in the X-Mechanism arms and prevents it from opening. The majority of the sled will be 3D printed including the racks and pinion, but will undergo structural testing to ensure components will not be damaged during flight.

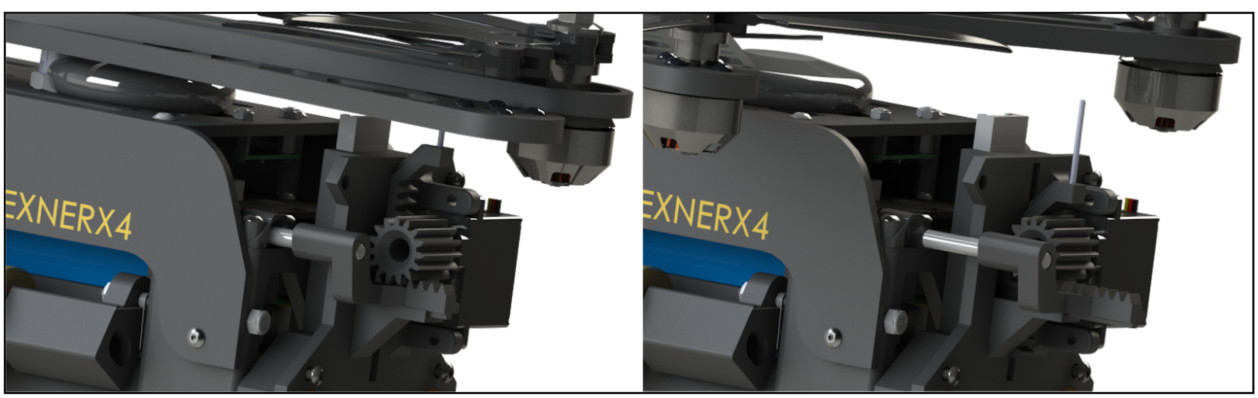
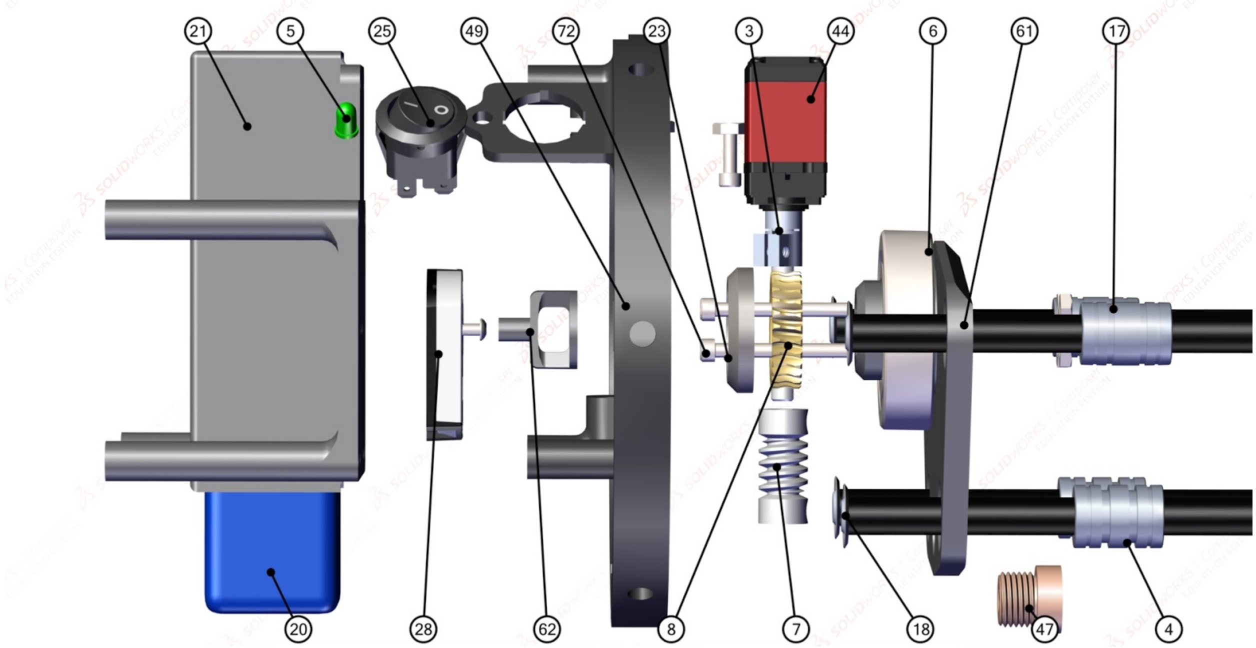


Figure 6.54: Payload Retention System Locked and Unlocked Configuration

The UAV is additionally guided using a set of low-profile guide rails that surround it on both ends. This solution was chosen over cylindrical guide rods as it would not allow the UAV to takeoff asymmetrically. A set of 4 dowel rods are also loosely inserted into each of the UAV's legs to aid with takeoff and prevent rotations.

6.4.3. Sled Deployment System



Description	BOM ID	Qty
525154.STEP	3	1
5986K672_FIXED ALIGNMENT LINEAR SLEEVE BRNG	4	2
5mm LED	5	1
60355K605_BALL BEARING	6	1
615462^615462 assembly_Worm Assembly-1	7	1
615464^Worm Assembly-1	8	1
9533T200_SELF-ALIGNING LINEAR SLEEVE BRNG	17	2
98430A134_EXTERNAL SELF-LOCKING RETAINING RING	18	8
Battery	20	1
Battery Holder	21	1

Description	BOM ID	Qty
Bearing Clamp Collar	23	1
Body1^RR812xxxx1 Switch(2)_RR812xxxx1 Switch-1	25	1
CUI_DEVICES_amt-102-base-rev-k	28	1
HSR-2648CR Servo	44	1
Left Nut	47	1
Nose Cone Interface Bulkhead	49	1
Rotary Nut Plate - Nose Cone	61	1
Sled - Encoder Adapter	62	1
socket head cap screw_ai	72	6

Figure 6.55: Deployment and Orientation Hardware and Electronics within the Nosecone

The linear rods of the sled deployment system have received substantial attention in both manufacturability and the smoothness of operation. The PDR designs called for retaining ring grooves to be cut into each end of the linear rods. After discussion with manufacturing experts at Purdue's Bechtel Innovation & Design Center, it was determined that this would not be possible to machine due to the length of the stock and Center's lack of a lathe with a secondary chuck. Instead, the clip retaining rings were replaced with "Push on Retaining Rings" which do not require a groove. The ring's lower grip strength was considered to be insignificant after analysing failure modes of the system.

A proposed modification to the Deployment system is the replacement of the rod sleeve bearings with kinematically proper linear bearings. It was noticed during testing that the initial Oilite sleeve bearings were prone to binding on the rods, as they were not designed for linear motion, and over constrain the system. The new solution uses 2 high tolerance fixed alignment bearings - #4 in the Bill of Materials (BOM) in Figure 6.55 - to provide alignment, and 2 self aligning bearings - BOM #17 - per rotary plate. This new solution would ensure elegant and reliable opening of the payload bay. It is currently being decided if the substantial cost of these linear bearings is worth the aforementioned improvements.

6.4.4. Payload Orientation System

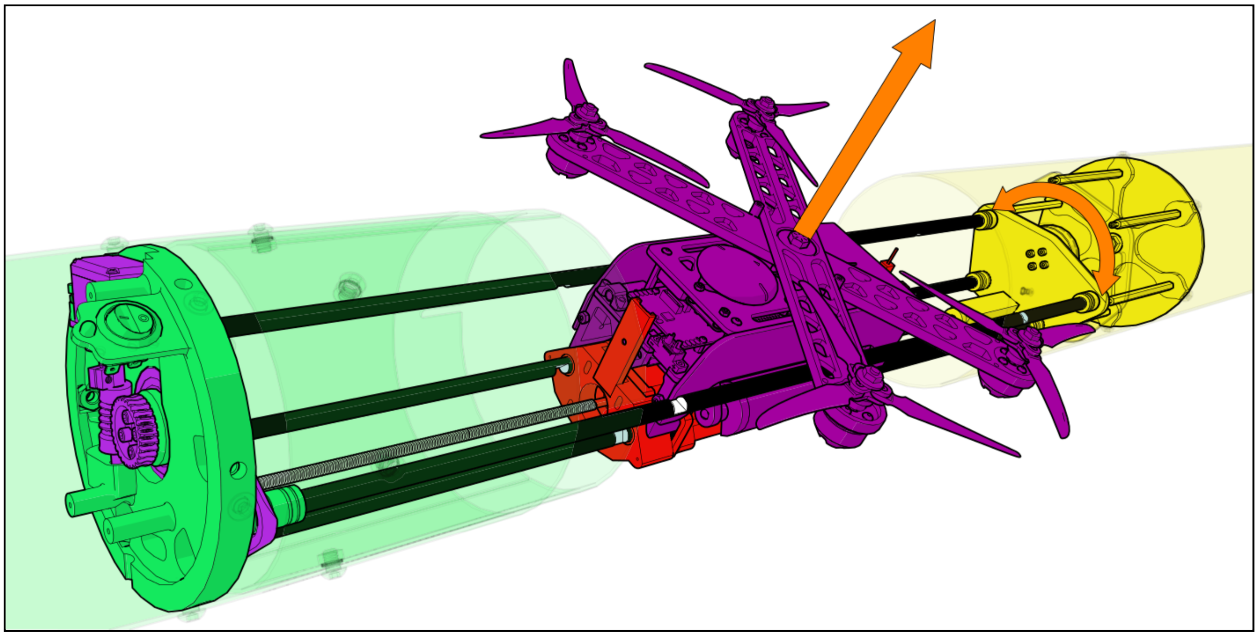


Figure 6.56: Payload UAV Deployment Diagram

The payload orientation system underwent substaintainal design modifications in the prior months due to manufacturability and assembly testing. Additional measuring systems are also being considered to aid in orientation and unlocking.

The single most influential modification to the entire R&D system is the replacement of the turntable bearings with high-precision sealed ball bearings. This modification was made in order to facilitate the use of the linear rod notch locking system, as discussed below. The initial turntable design had far too much slop, allowing the sled (and therefore the locking rods) to rotate more than 10° in either direction, preventing any sort of security. This problem was identified immediately during integration testing and was replaced with a light-weight plastic bearing. The plastic bearing, however had similar slop issues, also preventing trust-worthy locking. Finally it was decided that the weight of a metal bearing (BOM #6) would be a fair compromise to have a sturdy locking system. Both turntables were updated to these new bearings, and the parts that interfaced with the turntables were modified to fit the new bearings. The switch to central ball bearings also alleviated manufacturing issues with water jetting the turntables. Clamping collars (BOM #23), caps and epoxy ensure consistent positioning of components relative to the bearing and each other.

Another substantial design change is the replacement of the laser cut pivot bulkhead with the 3D-printed nose cone interface bulkhead (BOM #49). This change reduced part count by approximately 15 parts, as subcomponents such as the servo, PCB, battery, and switch mounting hardware were able to be integrated into a single component. This reduces weight, aids in assembly, and substantially reduces manufacturing time. This plate includes a location for captive nuts, which will be epoxied in place once alignment has been inherited from the nose cone curvature. Other notable features include switch mounts, standoffs for the battery mount, servo mount locations, and a stop for the central bearing to ensure that axial positioning is correct. All of these can be found on Figure 6.55.

An encoder system has been proposed and designed as a fall back. It is critical that the guide rods are able to fully slide through the linear bearings. This is only possible if alignment is perfect (otherwise the linear rod retaining rings may collide with the nose cone interface bulkhead). The servo may be able to achieve this level of precision, or it may drift during pad and flight time. In the event that the servo is not able to provide this alignment, an encoder will provide the data necessary to correct the problem.

6.4.5. Structural Integrity

While modification and concern was given to the material strength of various components, the focus of the CDR structural integrity upgrades was given to assisting motion and assembly.
The nose cone has a 7.5” coupler that ensures concentricity with the upper airframe. If this coupler was permanently epoxied in place it would require any hardware that belongs in the nose cone to fit through its reduced diameter, an unnecessary challenge for the already space lacking system. Instead, it was decided that the coupler would be cut in half into two semi-circular cylinders, allowing the coupler to be installed last in the assembly process. This coupler is then held in with button head screws and press-fit nuts.

During simulations and testing of the R&D system a critical assumption was realized. In order for the rod notch detent system to properly unlock, the nose cone and upper airframe must be rotationally matched. If they were not matched it would be possible for the nose cone to rotate instead of the sled, resulting in the sled being unable to deploy leading to a mission failure. In order to address this problem, 2 slots have been added to the coupler, in which screws in the upper airframe align. This ensures that throughout the “Bay Closed” operating phase, there would be no asynchronous rotation.

Additionally, machining has begun on an upgrade version of the main parachute bulkplate. More information can be found on this design in section 3.1.6.2.2 and 6.5.2.

6.4.6. R&D Electrical Design

While the general concept behind the R&D electrical system has stayed the same, it has seen many incremental changes since PDR, mostly in component selection. Aspects of the R&D electrical design that have not changed are:

- Teensy 4.0 microcontroller for data processing and control systems
- XBee PRO 900-HP for wireless communication with the GCS
- NEMA 17 Stepper motor to turn the lead screw for payload bay expansion
- HSR-2648CR Servo to drive the worm system for payload bay unlocking and orientation
- GY-521 (MPU-6050) Inertial Measurement Unit (IMU) to determine bay orientation relative to gravity

Any other components in the following design, shown in Figure 6.57 below, are new.

The most notable addition since PDR is the finished power system. A single Turnigy 2200mAh 3S 25C Lipo Pack supplies voltage at the common collector for the whole power distribution system. This power source was selected based on the following power requirements:

- ~3A@12VDC to drive NEMA 17 stepper motor for 1min
- At most 1A@5VDC to drive the orientation servo for 1min and UAV retention servo for 30s
- ~100mA@5VDC to power Teensy 4.0 microcontroller for 2hrs
- ~At most 290mA@3.3VDC to power XBee Pro 900 HP for 2hrs
- ~50mA@5VDC for all other low power devices combined for 2hrs

From these power requirements the calculated total energy needed (assuming only voltage conversion without current gain) is

$$3A*1min + 1A*1.5min + 0.1A*2hrs + 0.29A*2hrs + 50mA*2hrs = 0.955Ah$$

Using the 2200mAh battery results in a FOS = 2.3

To power all devices correctly, the following components are used:

- TB6600 Stepper driver to drive NEMA 17 stepper motor reliably without overheating (a problem encountered with the L298N)
- UBEC DC/DC Buck Converter to power both servos
- L7805 5V regulator to power Teensy 4.0, MPU-6050 IMU, and infrared break beam emitter
- LD1117 3.3V regulator to power XBee and infrared break beam receiver

All components that are not power-distribution specific are controlled using the Teensy 4.0 microcontroller, which receives instructions from the GCS via the XBee PRO 900-HP (more on this in the following section, 6.4.7). In order to orient the UAV vertically, the MPU 6050 IMU provides accelerometer and gyroscope data used to calculate the angle of the bay with gravity. Finally, in order to know how much to turn the bay for "unlocking," an infrared break beam emitter/receiver system indicates when the alignment is correct.

The two main options for assembly of the electrical subsystem components are PCB and hand wiring. Mounting to a PCB is ideal due to higher reliability and safety, but lacks flexibility for changes. As a result, all development up to this point has been hand wiring, however a PCB shall be the final deliverable once the subsystem reaches an appropriate readiness level. Currently, the electrical subsystem is proving out on the component level via hand wiring on a breadboard. The next step is proving out the design on the subsystems integration level, which requires validation with the retention and deployment subsystem. Therefore, the subsystem's readiness level is best assessed at technology readiness level 4. As a prototype will be needed for level 6 testing, a PCB has recently begun development.

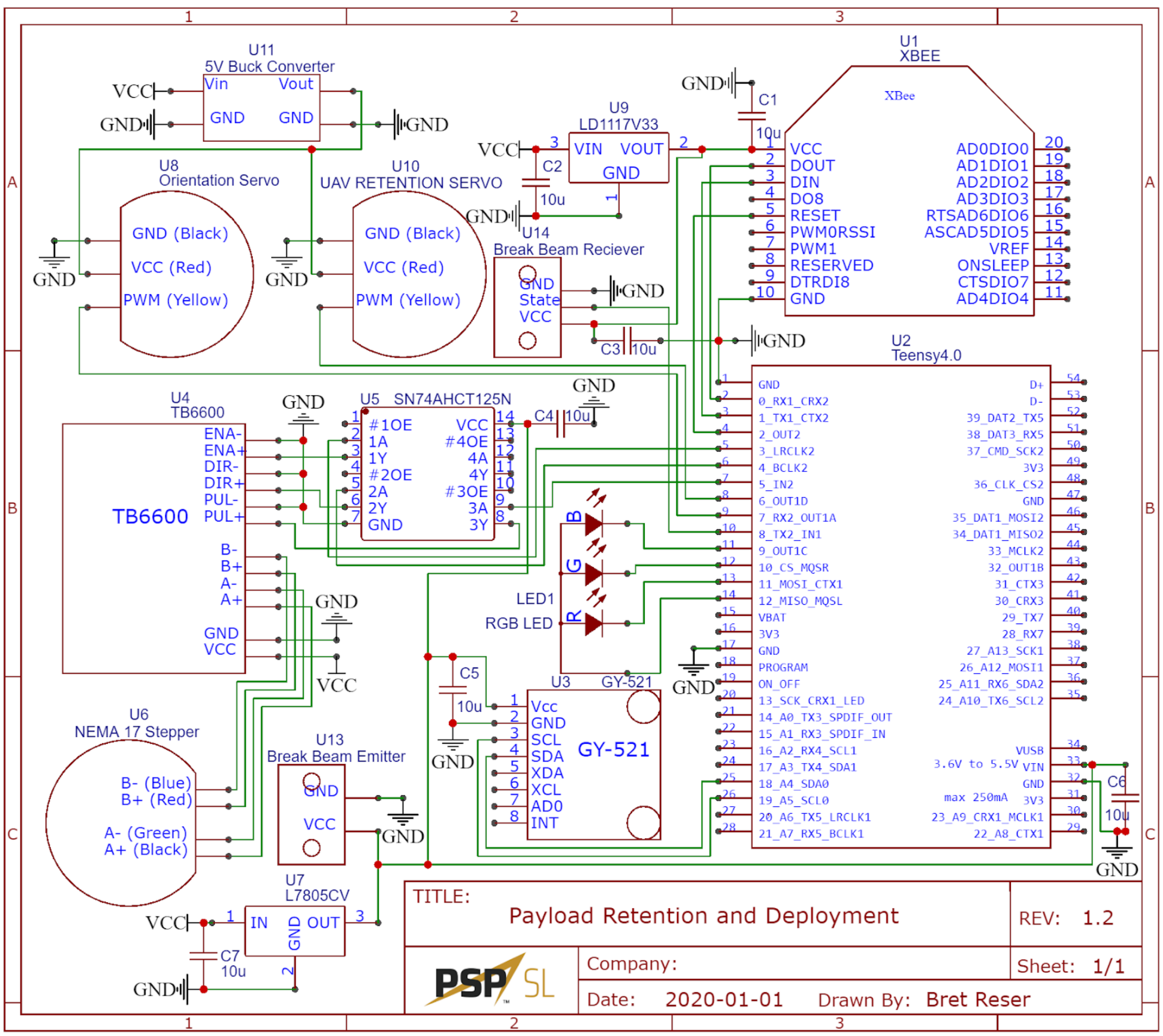


Figure 6.57: Wiring Schematic for Payload Retention and Deployment

6.4.7. Communication and Data Links

The software architecture of the retention and deployment system was designed to place as much control as possible in the hands of the GCS, rather than the launch vehicle being "smart." The benefit of interest here is flexibility when away from the launch vehicle. By putting low level code on the payload bay and high level code on the GCS, the complete retention and deployment software can be completed early on, with all high level decisions easily done by the GCS user on the fly.

The payload bay's software system is nearly entirely dependent on the GCS for instructions, as seen is Figure 6.58. Programmed into the microcontroller are a set of low level functions to control physical actuations and transmit data. These functions are only executed when a packet containing an instruction is received through the XBee PRO 900-HP (RF link to GCS). Possible instructions are described further in Table 6.3. By sending a sequence of these simple instructions, many complex actions can be created without the need to modify the payload bay code. While idle, the only work done by the microcontroller is constantly reading the bay orientation angle (required due to the use of a complementary filter to determine bay angle) and transmitting a periodic "heartbeat" (for GCS connection status).

Currently, the payload bay code is finished as originally designed, with an architecture in place to allow easy addition of instructions if needed. Development of a Kivy-based front-end GUI program for the GCS has recently begun, as the python-based back-end has so far been successfully tested for a large portion of its functionality.

Instruction	Input	Output
Set expansion stepper speed	Stepper speed in RPM	Acknowledgement with set speed
Step stepper	Signed number of steps	Acknowledgement when completed
Automatic orientation enable (orient the bay upright via orientation servo)	ON or OFF	Acknowledgement; when ON, outputs angle of bay periodically
Get bay angle	None	Bay angle relative to upright orientation
Get raw IMU data	None	Accel X, Y, Z and gyro X, Y, Z
Set UAV retention servo angle	Servo angle	Acknowledgement with set angle
Set orientation servo pulse width modulation	Microsecond value	Acknowledgement with set microsecond value
Unlock payload bay	None	Acknowledgement initially, then second acknowledgement when bay unlocked

Table 6.3: Payload Retention and Deployment Instructions Available to GCS

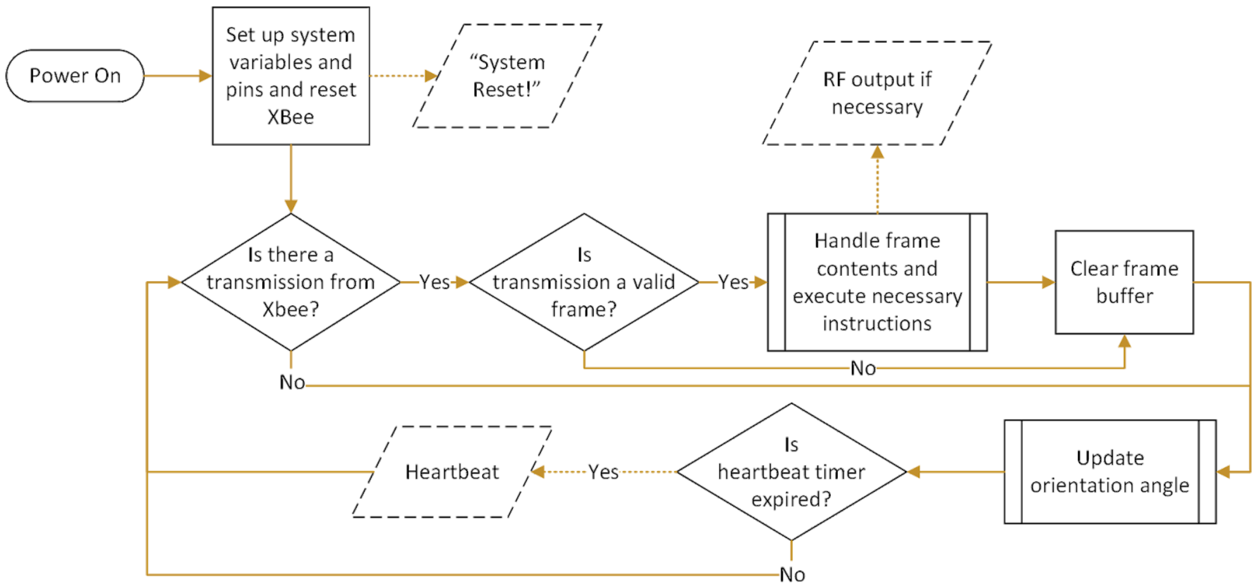


Figure 6.58: Teensy 4.0 Microcontroller High-Level Software Overview

6.5. Payload Systems Integration

6.5.1. R&D and UAV Integration

The UAV integrates with the R&D system through the payload retention that acts to protect and secure the UAV during flight and ensure that the UAV will unlock in a level configuration ready for flight. In the R&D deployed configuration, the UAV can be placed on the sled where it can be locked or unlocked independent of the R&D system. Locking the UAV on the sled isolates power to the flight computer and keeps it powered off. The UAV is kept secure in the deployed state using an actuated system that can be controlled independent of the R&D system. In the R&D closed configuration, the airframe is held in by the payload retention system and a lip on the inside of the launch vehicle body that assists with restricting motion of the UAV. The UAV secured to the payload retention system can be found in Figure 6.59 below.

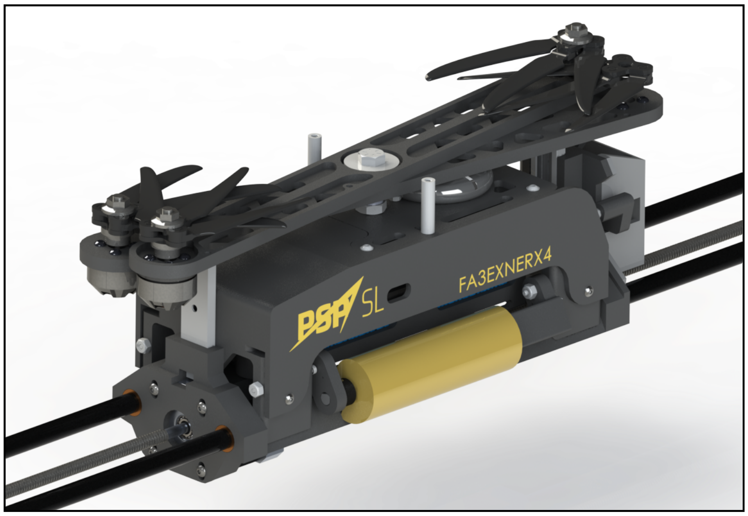


Figure 6.59: Payload UAV Integration with R&D System

The payload retention system is secured to the R&D structure on a set of shafts that the sleds slide on and a leadscrew that controls the deployment of the sled. The shafts and leadscrew interface with a set of bulkplates that allow the sled to be rotated once the payload retention system has been fully deployed.

6.5.2. Launch Vehicle Integration

Along with the role of containing the payload for flight, the R&D system also serves as a major structural component, integrating the main parachute, upper airframe, and nose cone elements. Locations for screws in both the nosecone interface bulkhead as well as in the main parachute bulkplate integrate the R&D system into the nose cone and upper airframe respectively. The nosecone screws will screw into captive nuts, which have the ability to pivot in such a way that the screw heads on the outside of the rocket will be able to maintain tangency with the nose cone, reducing aerodynamic drag. FEA has been used significantly throughout the system to ensure that the various components will structure as planned. The single most important structural element for the recovery of the rocket is the main parachute bulkplate. This piece has been redesigned multiple times to ensure that it can handle the high instantaneous loads of mains deploy while also being lightweight. Thorough FEA analysis with a built-in safety factor has been conducted to verify its structural integrity. This analysis can be found in Section 3.1.6.2.2.

On launch day, the payload team will have already pre-assembled the entire R&D system, and will merely integrate the system with the upper airframe and nosecone prior to launch. This will mean the installation of ~12 screws, and the verification of the functionality of the system. Once installed in the rocket, the electronic status of the system will still be able to be modified and diagnosed through a switch and RGB LED, accessible through holes drilled in the nose cone. This accessibility and pre-assembly will minimize the time spent assembling and verifying the rocket on launch day, reducing the chances of a time crunch leading to failures.

7. Safety

Critical to each successful launch is the safety and wellbeing of all involved, be they team members or passive spectators. Creating a safe launch environment consists of 3 key elements: having a consistent and predictable plan for launch, understanding the risks associated with said launch, and preparing and implementing plans to mitigate such risks. Not only do these decrease the likelihood of injury during launch, but they also increase the likelihood of a successful flight: predictability of a launch lends itself to repeatability; understanding the ways a launch can fail lends itself to preventing failure.

7.1. Operational Procedures

7.1.1. Launch Checklists

In preparation for actual launch procedures, checklists have been created to maintain continuity of launch operations, to alleviate possible ambiguity in launch operation, and ensure maximum personnel safety in any contingency. Alleviation of such conditions is critical not only to an efficient launch operation, but also a safe launch: continuity means predictability, and predictability not only reduces the number of launch elements that can go awry, but also allows bystanders to recognize and prepare in the unlikely and unfortunate event that the launch does go catastrophically wrong. 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 abate the danger associated with the latter, preparation and the creation of a contingency is the best method for reducing the risk of tragedy.

7.1.1.1. Before Launch Day Checklist

Actions to be performed before travel to launch site

- ❏ Follow packing checklist to ensure all required items are brought to launch site (See Appendix B)

Actions to be performed day before launch

- ❏ Charge avionics batteries and backup batteries
- ❏ Charge payload batteries and backup batteries
- ❏ Check altimeters for continuity and operation

7.1.1.2. Pre-Launch Checklist

General Safety

- ❏ Ensure that at least two people are using this checklist to prep for launch
- ❏ Ensure that a trained Range Safety Officer is present
- ❏ Have first aid equipment and at least one phone available for use nearby
- ❏ 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
- ❏ Have adequate fire suppression equipment available for use nearby
- ❏ Ensure a fire blanket has been placed under the pad if conditions at launch are dry enough to require it
- ❏ Inspect personnel for Pocket Safety Document possession and ready accessibility

Payload preparation and loading checklist

- ❏ Assembling the UAV
  - ❏ Install ice mining assemblies on either side of the airframe
  - ❏ Install LiPo battery
  - ❏ Install X-Wing assembly
  - ❏ Install propellers. Check for any structural defects
  - ❏ QUALITY WITNESS: Inspect assembled UAV for the following. If any of the following are missing, halt launch procedures and direct attention to the appropriate authority.
    - ❏ Ice mining assembly presence
    - ❏ LiPo battery presence
    - ❏ Electronic component presence
      - ❏ FCC
      - ❏ MCU
      - ❏ PDU
      - ❏ LiDAR
      - ❏ GPS
      - ❏ 915 MHz radio
      - ❏ GSFY-10 limit switch
      - ❏ Ice mining power electronics
    - ❏ Verify all electrical connections by tracing printed UAV electrical schematics
    - ❏ X-Wing assembly presence
      - ❏ Inspect X-Wing assembly for visible or tactile cracks, scratches, or irregularities
      - ❏ Test X-Wing folding action and inspect springs for visible distortion
    - ❏ Propeller presence.
      - ❏ Inspect propellers for visual or tactile cracks, scratches or irregularities
- ❏ Initialize GCS Software
  - ❏ Check GCS battery level. Battery shall be no less than 75% charged
  - ❏ Power on GCS
  - ❏ Run GCS startup script. Ensure all software applications are running nominally
- ❏ Arm the UAV
  - ❏ Inspect airframe mechanical and electrical connections

- ❏ Check UAV battery level. Battery voltage shall be no less than 12.1V
- ❏ Check RC radio battery level. Battery voltage shall be no less than 7.2V
- ❏ Verify that valid SD card is installed in FCC
- ❏ Power vehicle on. Verify that vehicle status lights and beep codes indicate no errors when plugged in to GCS
- ❏ Calibrate FCC sensors
  - ❏ Calibrate the compass
  - ❏ Calibrate the gyroscope
  - ❏ Calibrate the accelerometer
  - ❏ Perform "level-horizon" calibration
- ❏ Verify working camera stream between vehicle and GCS
- ❏ Verify working telemetry stream between vehicle and GCS
- ❏ Verify correct safety settings are installed on the vehicle
- ❏ Power vehicle down

❏ Arming R&D Electronics/Software

- ❏ Install a fully charged LiPo battery for the R&D electronics in the designated location in the nose cone of the rocket
- ❏ Verify that all electrical connections between the stepper motor, servo motor, stepper driver, XBee radio, microcontroller, and electronic indicators are secure. Trace connections on schematic print-out
- ❏ Verify that all electronics are securely mounted in place as designed
- ❏ Power on the R&D electronics via rocker switch, verifying that the correct LED color pattern is displayed on the LED, viewed through the side of the nose cone
  - ❏ Verify that "powered on" message is received at the GCS

❏ UAV and R&D Integration

- ❏ Install UAV on R&D sled. Ensure all four latches are secured to the legs of the UAV and the two guide rails are in the proper position on the forward and aft side of the UAV
- ❏ Verify that limit switch is placed in proper position with respect to the sled
- ❏ Fold propellers into launch configuration
- ❏ Fold X-Wing mechanism into launch configuration
- ❏ QUALITY WITNESS: Inspect installed UAV for the following. If any of the following are missing, halt launch procedures and direct attention to the appropriate authority.
  - ❏ UAV presence
  - ❏ 4 secured UAV latch connections
  - ❏ Limit switch presence
  - ❏ Test rotation of payload deployment system for any catches in rotation.
  - ❏ Inspect inside nose cone and upper airframe for possible obstructions to payload rotation
- ❏ Enclose the UAV in the rocket by mating the nose cone and the upper airframe

General launch vehicle construction (To be done after prepping avionics and reloads)

- ❏ Ensure computer simulations have already been run of the launch vehicle in its current construction state before launch to analyze both normal and ballistic scenarios
- ❏ QUALITY WITNESS: Inspect assembled launch vehicle for the following. If any of the following are missing, halt launch procedures and direct attention to the appropriate authority.
  - ❏ Check that all fins and lugs are secure and aligned
  - ❏ Check that the body tube is in good condition
  - ❏ Check that the motor and ejection system are in good condition, are functional, and are securely installed
    - ❏ Ensure the proper motor and ejection have been selected for the desired flight profile and that they are certified by NAR, Tripoli, or CAR
    - ❏ Check the reload motor for proper build-up, paying special attention to the O-rings
    - ❏ Ensure the ejection charge is properly installed, and is the proper amount according to calculations
    - ❏ Check that the motor mount is secure, is in good condition, and will not deflect motor thrust
  - ❏ Check that the recovery system is in good condition, is functional, is securely installed, and is strong enough to withstand recovery loads
    - ❏ Check that shock cords are securely attached and are not cracked, burned, or frayed
    - ❏ Check that shroud lines are not burned or tangled
    - ❏ Check that all hardware, such as snap swivels and screw eyes, is in good condition and secure
    - ❏ Check that parachute protection is installed properly and is in good condition
  - ❏ Check that the electronics bay is in good condition, is functional, and is securely installed
    - ❏ Ensure the altimeters are properly installed
    - ❏ Check that the avionics are initially disarmed and that an “Arm before flight” reminder is in use
    - ❏ Check that the electronics bay is properly vented and that wires do not cover any ports
    - ❏ Check that the drogue and main wiring are in good condition
    - ❏ Check that all electronics bay hardware and electrical connections are secured against acceleration forces
    - ❏ Check that all electrical components are free from condensation or moisture
    - ❏ If appropriate, check the settings of the mach lock-out / mach delay
    - ❏ Ensure the battery or batteries being used are charged and in operational condition, and secure battery positions with masking tape

- ❏ Check that the ejection charges are properly set up
- ❏ Close and secure the electronics bay

- ❏ Prep and install motor
  - ❏ Grease motor tube
  - ❏ Bolt on forward closure (with eye bolt attached)
  - ❏ Grease outside of grain one and place in motor tube
  - ❏ Insert RUBBER washer
  - ❏ Repeat last two steps for all motor grains
  - ❏ Apply lubricant as necessary
  - ❏ 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.

Final Flight Check

- ❏ Check the nosecone and any stage or payload couplers for a secure and proper fit
- ❏ Check for continuity, resistance, and cracks or flaws in the pyrogen of the igniters; all igniters must touch the propellant, have adequate electrical current flowing to them, and have no shorts
- ❏ Ensure that the center of gravity and center of pressure are in their expected positions
- ❏ Perform manufacturer's checking instructions on the avionics
- ❏ Check that shear pins are installed for main parachute compartment
- ❏ Ensure drogue ejection will not cause main to deploy

Pad Distance

- ❏ Only the minimum number of personnel are at the pad to prep for launch
- ❏ All team personnel and spectators are a safe distance from the pad based upon a minimum distance table (See Appendix B)
- ❏ Ensure barriers are in place to keep spectators away from the launch area

Pad Installation

- ❏ 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
- ❏ Ensure that enough electrical current will reach the igniters of the launch vehicle
- ❏ Verify that the launch vehicle moves smoothly on the launch rail; clean the rail and launch vehicle as necessary
- ❏ Install motor
- ❏ Arm the avionics system once the launch vehicle is on the pad
  - ❏ Ensure that the avionics systems are all turned on

- ❏ Ensure that the igniter clips are clean and secure them to the pad; install igniter into motor
- ❏ Connect launch leads to motor igniter

Flight Trajectory

- ❏ Ensure the launch and the flight will not be angled towards any spectators
- ❏ Double check that the launch vehicle will not fly higher than its permitted clearance waiver; know the expected performance of the model
- ❏ Check cloud ceiling and winds and make sure the skies around the launch area are clear
- ❏ If needed, use an anemometer to avoid launching during extremely windy intervals
- ❏ Ensure there are no obstructions or hazards in the launch area

Beginning the Launch

- ❏ Shortly before the countdown, give a **loud** announcement that the launch vehicle will be launched; if applicable to the situation, use a PA system
- ❏ Ensure that all spectators are aware of the launch and that parents are in close contact with all children
- ❏ When launching, give a **loud** countdown of "5, 4, 3, 2, 1, LAUNCH!"

7.1.1.3. Launch Checklist

- ❏ Ensure that at least two people are using this checklist to observe the launch
- ❏ Ensure the stability of the model is being monitored
- ❏ Ensure that the recovery system is successfully deployed.
- ❏ Carry out a safe recovery of the model
- ❏ If radio control is used for flight functions (e.g. recovery), check that the operating frequency is in the 27, 50, 53, or 72 megahertz bands. Use of 75 megahertz for flight functions is not permitted.
- ❏ Ensure launch vehicle trajectory is being tracked during flight. Be aware of tilt or drift from mass/aerodynamic imbalance, wind, or other sources. **Do not turn off the altimeters until data has been acquired from them after landing.**
- ❏ Ensure crosswind positioning of spectators and vehicles
- ❏ Ensure that the launch pad is being monitored after takeoff in case any dangers arise at the pad
- ❏ Ensure all passerby and spectators are aware of the launch
- ❏ 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.
- ❏ Monitor the flight path, using binoculars if necessary

- ❏ Make sure whoever is responsible for recovery is kept fully aware of the status of the launch vehicle (failed to launch, nominal in-flight, mid air failure, returning for recovery, etc.)
- ❏ Communicate launch progress effectively to NASA officials, if needed

In the case of a misfire:
- ❏ Wait a minimum of one minute
- ❏ Disarm launch controller and avionics
- ❏ Remove failed igniter and motor if needed

In the case of unintended ballistic trajectory
- ❏ Should the launch vehicle enter 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 in attendance of the launch are to immediately turn away from the direction of the launch vehicle and run for a minimum of 10 seconds

Note: when launching through an affiliate launch day, team emergency contingencies are superseded by launch day procedures.

7.1.1.4. Post-Launch Checklist

- ❏ Ensure that at least two people are using this checklist after launch
- ❏ Double check that there are no hazards which have gone unnoticed during the launch before approaching the launch pad or the launch vehicle for clean-up.
  - ❏ If there are hazards, notify emergency personnel
- ❏ Check all components of the rocket and payload for damage
  - ❏ If the payload is damaged, notify payload subteam and team leads
- ❏ Check that there are no obstacles around the landing site and that it is safe for the payload to be deployed
  - ❏ If obstructions are present, notify team leads
- ❏ Clear all personnel of launch site and return to the Hub
- ❏ Initiate Payload Mission

7.1.1.5. Post-Flight Checklist

- ❏ Let NASA officials verify the results of the launch, if necessary
- ❏ Ensure that at least two people are using this checklist after launch
- ❏ Sweep the area around the touchdown site to make sure there are no potential hazards around the rocket
- ❏ Make sure there are no live wires or sharp components on the rocket before handling

- ❏ Double check that all necessary data from the avionics bay has been retrieved
  - ❏ If so, disarm the avionics
- ❏ Disarm the launch controller
- ❏ Place cap on launch rods, if necessary
- ❏ Take down the launch pad, if necessary
- ❏ Retrieve the main launch vehicle body and all components which may have landed separately
  - ❏ Check them for any failed ejection charges
  - ❏ If there are failed ejection charges, save all ejection circuits and remove any non-discharged pyrotechnics
- ❏ Perform sweep of launch field to ensure no materials are unintentionally left behind

7.2. Hazard Analysis Methods

Critical to the success of any mission is a comprehensive understanding of the dangers involved therein. Threats to personnel, environmental factors, project dangers, and launch vehicle failure modes all contribute to the cumulative risk associated with the mission. While each team takes careful steps to minimize any such risks, and to not put team members in harm's way, the danger associated with missions such as these will likely never be fully abated. Acknowledgement of these dangers, and a thorough plan for their mitigation, is the only way to guarantee the safety of the team, and to maximize the likelihood of mission success.

For the purposes of this mission, risk analysis was broken into two categories: event probability and event severity. The respective definitions of these categories are listed below.

Category	Value	Gauge
Remote	1	Less than 3% chance of event
Unlikely	2	3-10% chance of event
Possible	3	10-25% chance of event
Probable	4	25-50% chance of event
Likely	5	Greater than 50% chance of event

Table 7.1: Event Probability Safety Matrix

In the consideration of severity, the team established several metrics for defining the associated levels, key among them being reversibility and remediation time. The highest level of severity corresponds to total irreversibility of the damage done; in various contexts, this may mean loss of life or permanent function, permanent environmental damage, or a complete scrub of the mission. Lower levels of severity are characterized by long term reversibility, and further delineated by the length of said remediation, ranging from an on-the-spot solution (first aid, consumable tool replacement, small leaks or spills) to the need for external assistance or long recovery (hospitalization, machine repair, primary environmental restoration).

Category	Value	Health and Personal Safety	Equipment	Environment	Flight Readiness
Negligible	1	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	2	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	3	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	4	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	5	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 7.2: Event Severity Safety Matrix

In the consideration of these definitions, the team focused on establishing quantitative metrics for each level of probability and severity. The establishment of measurable standards for these categories enables the team to accurately assess the risk of every considered event (listed in the following sections). Cumulative risk for each event is found by a cross examination of the likelihood and severity of each event (performed in this case by the multiplication of the assigned values in each of these categories). A table demonstrating this is given 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

Any event categorized as 'low' or 'negligible' risks are considered acceptable by the team's standards.

Category	Negligible	Minor	Moderate	Major	Disastrous
Remote	1	2	3	4	5
Unlikely	2	4	6	8	10
Possible	3	6	9	12	15
Probable	4	8	12	16	20
Likely	5	10	15	20	25

Figure 7.3: Total Event Risk Safety Matrix

Prior to a plan for risk mitigation, many of the events listed below fall outside of the acceptable tolerance. 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 within acceptable tolerance.

7.3. Personnel Hazard Analysis

Hazard	Likelihood (Cause)	Severity (Effect)	Risk	Mitigation	Verification	Post Mitigation Risk
Burns From Motor	2 (Proximity To Launch Pad, touching engine too soon after launch)	3 (Mild To Moderate Burns)	6, Low	Maintain minimum safe launch distances. Wait an appropriate amount of time after launch to retrieve.	200' border will be established after mounting of launch vehicle onto launcher as compliance to NAR safety standards.	3, Low
Contact with Airborne Chemical Debris	3 (Airborne particulate debris)	2 (Minor burns, abrasions)	6, Low	Wear appropriate Personal Protective Equipment (PPE) such as gloves, lab coats and breath masks, wash immediately with water if contact is made.	Safety Team will verify with each participating member that appropriate PPE is worn.	4. Low
Direct Contact with Hazardous	3 (Chemical spills, improper use	3 (Moderate burns, abrasions)	9, Medium	Wear appropriate PPE such as gloves or lab coats, wash	Safety Team will verify with each participating	6, Low

Chemicals	of chemicals)			with water.	member that appropriate PPE is worn.	
Dust or Chemical Inhalation	3 (Airborne particulate debris)	3 (Short to long-term respiratory damage)	9, Medium	Wear appropriate PPE or respirator, work in a well ventilated area.	Safety Team will verify with each participating member that appropriate PPE is worn.	6, Low
Dehydration	3 (Failure to drink adequate amounts of water)	3 (Exhaustion and possible hospitalization)	9, Medium	Ensure all members have access to water at launch.	Mandatory water breaks will be held every hour where no work may be done during that period.	3, Low
Heatstroke	3 (High temperatures on launch day)	3 (Exhaustion and possible hospitalization)	9, Medium	Wear clothing appropriate to the weather, ensure all members have access to water at launch.	Team members must have adequate clothing, safety team will report violators to the project lead to decide if the violator should be dismissed to a colder area; water will be provided.	3, Low
Hypothermia	3 (Low temperatures on launch day)	3 (Sickness and possible hospitalization)	9, Medium	Wear clothing appropriate to the weather, ensure all members have access to a warm area to rest at launch.	Team members must have adequate clothing, Safety team will report violators to the project lead to decide if the violator should be dismissed to a warmer area.	6, Low
Electrocution	2 (Improper use of equipment, static build-up)	4 (Possible explosion, destruction of electrical tools or components, possible severe harm to personnel)	8, Medium	Give labels to all high voltage equipment warning of their danger; ground oneself when working with high-voltage equipment.	Pre-operation inspection will be done by safety officer to insure no open electrical components prior to high-voltage event. Allow only one member to work on electrical components at a time with proper PPE and student	4, Low

					supervising.	
Entanglement with Construction Machines	4 (Loose hair, clothing, or jewelry)	5 (Severe injury, death)	20, High	Secure loose hair, clothing, and jewelry; wear appropriate PPE.	No physical contact allowed without call out before use to make sure PPE is worn. Make sure rules followed as set forth by machining rules.	5, Low
Epoxy Contact	3 (Resin Spill)	3 (Exposure to Irritant)	9, Medium	Wear appropriate PPE such as gloves or lab coats, wash with water.	Safety officer or approved safety team member will verify proper PPE is used before and during epoxy handling	6, Low
Eye Irritation	3 (Airborne particulate debris)	2 (Temporary eye irritation)	6, Low	Wear appropriate PPE or protective eyewear, wash with water if contact is made	Guaranty PPE worn at all times during manufacturing. Call out before use to make sure PPE is worn by surrounding team members	4, Low
Hearing Damage	4 (Close proximity to loud noises)	3 (Long term hearing loss)	12, Medium	Wear appropriate PPE such as ear muffs when using power tools.	PPE equipment check must be done by a safety team member before conducting construction.	6, Low
Kinetic Damage to Personnel	2 (Failure to take appropriate care around unburned fuel, post-landing launch vehicle explosion)	4 (Possible severe kinetic damage to personnel)	8, Medium	Extinguish any fires before recovering, wait for motors to burn fully before recovering, wear appropriate PPE when recovering.	Make sure the area is evacuated and designated individuals are to recover components at a designated time when determined to be safe; no contact allowed without call out before use to make sure PPE worn.	5, Low
Launch Pad Fire	2 (Dry Launch Area)	3 (Moderate Burns)	6, Low	Have fire suppression systems nearby and use a protective ground tarp.	Make sure the area is evacuated and designated individuals are to recover	3, Low

					components at a designated time when determined to be safe; ground area will be cleared per NAR launch standard	
Injury from Ballistic Trajectory	1 (Recovery System Failure)	5 (Severe Injury, Death)	5, Low	Keep all eyes on the launch vehicle and call "heads up" if needed, limit the number of people at launch. Emphasize importance of keeping eyes on the launch vehicle during flight.	Go through safety procedures before the launch occurs. Minimum of 2 spotter will be watching rocket trajectory at all times, and have authority to call "Scatter"	5, Low
Injury from Falling Components	2 (Failure to keep all components securely attached to the launch vehicle; result of improper staging constraints, part failure, or excessive vibration)	5 (Severe injury, death)	10, Medium	Keep eyes on the launch vehicle at all times; make sure all team members who cannot watch the launch vehicle have spotters nearby; alert others if the launch vehicle enters a ballistic trajectory.	Go through safety procedures before the launch occurs. Minimum of 2 spotter will be watching rocket trajectory at all times, and have authority to call "Scatter"	5, Low
Injury from Navigating Difficult Terrain	2 (Uneven ground, poisonous plants, fast-moving water)	4 (Broken bones, infections, drowning, etc.)	8, Medium	Do not attempt to recover the launch vehicle from atypically dangerous areas.	Set boundaries to not cross at the launch location before the launch occurs.	4, Low
Injury from Projectiles Caused by Jetblast	2 (Failure to properly clean launchpad, failure to stand an appropriate distance from the launch vehicle during launch)	3 (Moderate injury to personnel)	6, 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.	Verify that the launchpad is clean and clear of debris before launch occurs. Create launch checklist to be completed before the launch vehicle can be launched.	3, Low
Physical Contact With Heat Sources	3 (Contact with launch vehicle parts	3 (Moderate to severe burns)	9, Medium	Wear appropriate PPE, turn off all construction tools	Confirm that appropriate PPE is being used. Make	3, Low

	which were recently worked with, improper use of soldering iron or other construction tools)			when not in use, be aware of the safety hazard that parts which were recently worked with present.	sure that everybody is informed of the hazard.	
Physical Contact with Falling Construction Tools or Materials	3 (Materials which were not returned to a safe location after use)	5 (Bruising, cuts, lacerations, possible severe physical injury)	15, High	Brief personnel on proper clean-up procedures, wear shoes that cover the toes.	Clean workspace every time after use. Create a checklist of where to put items after use.	5, Low
Premature Ignition	2 (Short Circuit)	2 (Mild Burns)	4, Low	Prepare energetic devices only immediately prior to flight.	Place previously used materials in separate container than the unused materials.	2, Minimal
Power Lines	2 (Launch vehicle Becomes Entangled In Lines)	5 (Fatal Electrocution)	10, Medium	Call the power company and stand clear until proper personnel arrive.	Alert all team members of the hazard. Everybody is required to stand clear of the area until certified personal clean up and verify that the area is safe.	5, Low
Power Tool Cuts, Lacerations, and Injuries	3 (Carelessness)	4 (Possible Hospitalization)	12, Medium	Secure loose hair, clothing, and jewelry; wear appropriate PPE; brief personnel on proper construction procedures.	No contact allowed without call out before use to make sure PPE worn. Make sure rules followed as set forth by machining rules.	4, Low
Injuries from Quadcopter payload	2 (Injury from spinning rotors)	2 (Minor cuts)	4, Low	Stay clear of quadcopter while it is in operation. Only team members familiar with the payload will handle it.	Stay minimum safe distance at launch and remain at that distance until payload and other components have landed. Have member of payload team retrieve the quadcopter.	2, Minimal
Tripping Hazards	3 (Materials which were	3 (Bruising, abrasions,	9, Medium	Brief personnel on proper clean-up	Have a clean up sheet for work	6, Low

	not returned to a safe location after use, loose cords on or above the ground during construction processes)	possible severe harm if tripping into construction equipment)		procedures, tape loose cords or wires to the ground if they must cross a path which is used by personnel.	space occupants to confirm everything is placed where it should be.	
Unintended Black Powder Ignition	3 (Accidental exposure to flame or sufficient electric charge)	5 (Possible severe hearing damage or other personal injury)	15, High	Label containers storing black powder, one may only handle the black powder if he/she possesses a low-explosives user permit.	Have check in/out form to confirm only those permitted to handle materials are the only ones handling the material.	5, Low
Workplace Fire	2 (Unplanned ignition of flammable substance, overheated workplace, improper use or supervision of heating elements, or improper wiring)	5 (Severe burns, loss of workspace, irreversible damage to project)	10, Medium	Have fire suppression systems nearby, prohibit open flames, and store energetic devices in Type 4 magazines as stated in CFR 27 part 55.	Make sure all members are updated on the workplace fire safety protocols. Have lists of all required fire suppression system accounted for and found near the area of work.	5, Low

Table 7.4: Personnel Hazard Analysis

7.4. Failure Modes and Effects Analysis (FMEA)

Hazard	Likelihood (Cause)	Severity (Effect)	Risk	Mitigation	Verification	Post Mitigation Risk
Airframe Failure	1 (Buckling or shearing on the airframe from poor construction or use of improper materials, faulty stress modeling)	5 (Partial or total destruction of vehicle, ballistic trajectory)	5, Low	Use appropriate materials according to extensive mathematical and physical analyses of the body tube, bulkheads, fasteners and shear pins, make use of reliable building techniques, confirm analyses with test launches.	Use a construction checklist which ensures mathematical analyses match physical analyses, if the airframe does not perform well in test launches perform another test launch with a new airframe design before converting to	5, Low

					full-scale, and use the launch checklists to ensure both before and after launch that the airframe is in good condition.	
Failure To Launch	2 (Lack of continuity)	1 (Recycle launch pad)	2, Minimal	Check for continuity prior to attempted launch.	Include checking for continuity in launch checklist that is to be completed prior to launch.	1, Minimal
CATO	1 (Motor defect, assembly error)	5 (Partial or total destruction of vehicle)	5, Low	Inspect motor prior to assembly and closely follow assembly instructions.	Include motor inspection in pre launch checklist to verify this task is completed.	5, Low
Instability	1 (Stability margin of less than 1.00)	5 (Potentially dangerous flight path and loss of vehicle)	5, Low	Measure physical center of gravity and compare to calculated center of pressure.	Have measured physical center of gravity documented and compared prior to arriving at the launch site.	5, Low
Motor Expulsion	2 (Improper retention methods)	5 (Risk of recovery failure and low apogee)	10, Medium	Use positive retention method to secure motor.	Include motor securement into pre launch checklist to verify that this task is complete.	5, Low
Premature Ejection	2 (Altimeter programming, poor venting)	5 (Zippering)	10, Medium	Check altimeter settings prior to flight and use appropriate vent holes. Test altimeter in similar conditions to those to be experienced at launch.	Include checking altimeter settings to pre launch checklist to verify that this task is complete. Altimeter testing before launch.	5, Low
Loss of Fins or Damage	2 (Poor construction or improper materials used)	5 (Partial or total destruction of vehicle)	10, Medium	Use appropriate materials and high powered building techniques.	Conduct stress tests on fins to make sure they can withstand all forces present during flight.	5, Low
Loss of Nose Cone	2 (Poor construction or improper materials used)	5 (Partial or total destruction of vehicle)	10, Medium	Use appropriate materials and high powered building techniques.	Ensure that nose cone is secured well before ejection during test runs, otherwise alter.	5, Low

Loss of Parachute	3 (Poor construction or improper materials used)	5 (Partial or total destruction of vehicle)	15, High	Use appropriate materials and high powered building techniques.	Ensure that parachute is secured well before ejection during test runs, otherwise alter to lower speed.	5, Low
Ejection Charge Failure	3 (Not enough power, electrical failure)	5 (Ballistic trajectory, destruction of vehicle)	15, High	Ground test charge sizes at least once before flight.	Conduct voltage test readings on power source before launch to make sure appropriate power is present for launch.	5, Low
Altimeter Failure	3 (Loss of connection or improper programming)	5 (Ballistic trajectory, destruction of vehicle)	15, High	Secure all components to their mounts and check settings.	Include checking component securements to pre launch checklist to verify that this task is complete.	5, Low
Cracking of Altimeter Sled	3 (Dropping of sled, over torquing of fixture nuts)	4 (Possible damage to electronics, inability to track altitude)	12, Medium	Visually inspect sled for cracks prior to and after installation on threaded rod, avoid over torquing of fixture nuts.	Inspection of sled will be performed during and after installation in launch vehicle	4, Low
Payload Failure	3 (Electrical failure, program errors, dead battery)	4 (Disqualified, objectives not met)	12, Medium	Test payload prior to flight, check batteries and connections.	Keep fresh batteries separate from previously used batteries. Use fresh batteries for each launch.	4, Low
Heat Damaged Recovery System	2 (Insufficient protection from ejection charge)	5 (Excessive landing velocity, potentially ballistic trajectory)	10, Medium	Use appropriate protection methods, such as Kevlar blankets.	Check that proper protection methods are securely placed before launch.	5, Low
Broken Fastener	1 (Excessive force)	5 (Ballistic trajectory)	5, Low	Use fasteners with a breaking strength safety factor of 2.	Conduct stress tests on fasteners to confirm that they meet force requirements.	5, Low
Joint Failure	2 (Excessive force, poor construction)	5 (Partial or total destruction of vehicle,	10, Medium	Use appropriate joint design according to extensive mathematical and	Ensure by design and testing that secure.	5, Low

		ballistic trajectory)		physical flight analyses, make use of reliable building techniques, confirm analyses with test launches.		
Centering Ring Failure	2 (Excessive force from motor, poor construction)	5 (Partial or total destruction of vehicle, ballistic trajectory)	10, Medium	Use appropriate centering rings according to extensive mathematical and physical flight analyses, make use of reliable building techniques, confirm analyses with test launches.	Ensure by design and testing that secure.	5, Low
Battery Overcharge	3 (Unsupervised /undocumente d charge)	3 (Destruction of battery)	9, Medium	Ensure batteries are documented and supervised if charging.	Ensure alarms set and other individuals are aware batteries charging.	3, Low
Premature Blackpowder Ignition	2 (Accidental exposure to flame or sufficient electric charge)	5 (Partial or complete destruction of vehicle	10, Medium	Ensure design has sufficient distance / protection from outside, and motor, charges, and batteries.	Ensure by design and testing that secure from other systems or puncture.	5, Low
Charge Ignition Close to Motor	3 (Poor design location leads to damage)	5 (Partial or complete destruction of vehicle	15, High	Ensure design has sufficient distance / protection from motor, charges, and batteries.	Independently ensure design is safe; ensure by isolated testing charge may work.	5, Low
Destruction of Bulkheads	2 (Poor construction or improper bulkheads chosen which cannot withstand launch forces, faulty stress modeling)	5 (Partial or total destruction of vehicle, ballistic trajectory)	10, Medium	Use appropriate materials according to extensive high-stress mathematical and physical analyses, make use of reliable building techniques, run stability tests, confirm analyses with test launches.	Bulkheads will be visually inspected with flashlight when possible prior to launch.	5, Low
Damaged Nose Cone	2 (Poor construction, damage from previous flights, poor storage, or	3 (Lower launch vehicle stability, possible deviations	6, Low	Check the nose cone for damage before and after each launch, choose a nose cone which is strong enough to withstand launch	Nose cones will be inspected and repaired before and after each launch in order to make sure they are up to	3, Low

	transportation)	from flight path)		forces according to mathematical and physical flight simulations, confirm choice of nose cone with subscale launches.	launch standards.	
Motor Tube Angled Incorrectly	2 (Poor construction, damage from previous flights, poor storage, or transportation)	4 (Lower launch vehicle stability, launch vehicle does not follow desired flight path well)	8, Medium	Ensure proper measurements and alignments are made during construction, ensure there is no rush to attach the motor tube, double-check the alignment of the motor before each flight, test that the desired motor alignment is correct with subscale flights.	Measurements will be made at 4 rotational points around the motor tube to insure equal distance from edge to launch vehicle edge coupling.	4, Low
Motor Tube Comes Loose	2 (Poor construction, damage from previous flights, poor storage, or transportation, faulty motor preparation)	5 (Ballistic trajectory, catastrophic destruction of vehicle)	10, Medium	Check the motor and motor tube for damage before each launch, run mathematical and physical flight simulations to ensure the tube performs as planned, confirm simulations with subscale launches.	Stress test the motor tube connection to make sure it can withstand expected forces acting upon it.	5, Low
Premature Stage Separation	3 (Premature ejection, poor choice of shear pins or fasteners)	5 (Possible recovery failure and damage to or loss of vehicle, ballistic trajectory)	15, High	Check altimeter settings prior to flight, use appropriate vent holes, and run thorough analyses to determine which types of shear pins and fasteners should be used.	Redundant altimeters will be used, calibration will be checked and verified by separate individuals.	5, Low
Forgotten or Lost Components	3 (Carelessness with launch vehicle components, failure to take note of inventory before attempting to launch)	4 (launch vehicle does not launch at the desired launch time)	12, Medium	Have spares for components which are small and easy to lose, have an inventory of all launch vehicle parts to be checked before moving the launch vehicle to a launch site.	Make sure not to forget anything. Have a team of 2 members go through and double check that everything has been taken and is accounted for.	4, Low

Launch Vehicle Disconnects from Launch Rail	2 (High wind speeds, failure to properly use the rail buttons, faulty rail buttons)	5 (Partial or total destruction of vehicle, ballistic trajectory which endangers personnel, onlookers, and property on the ground)	10, Medium	Use mathematical and physical analyses 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 by two separate individuals prior to launch for cracks, misalignment, or other inaccuracies.	5, Low
Flightpath Interference	2 (Wildlife in the air, unforeseen obstacles such as a loose balloon)	5 (Minor to severe change in the vehicle's flightpath, possible ballistic trajectory)	10, Medium	Ensure there are clear skies above before launching, ensure an FAA waiver has been obtained for the designated launch area.	Visual inspection of the surrounding area to make sure no incoming wildlife or loose objects.	5, Low
Unplanned Amount of Friction Between Launch Vehicle and Launch Rail	3 (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)	2 (launch vehicle does not follow the designated flight path well, lower maximum height)	6, 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.	2, Minimal
Failure to Ignite Propellant	2 (Faulty motor preparation, poor quality of propellant, faulty igniter, faulty igniter power source, damage to motor)	5 (launch vehicle does not immediately launch and is a considerable hazard until it is confirmed that it will not launch, changes to igniters or launch	10, Medium	Purchase propellant and motors only from reliable sources, team members who prepare the motor and igniters must be supervised by at least one other team member, determine if the igniters chosen work well during subscale testing.	Make sure igniters are well tested and are extremely reliable.	5, Low

		vehicle required)				
Propellant Fails to Burn for Desired Duration	2 (Faulty motor preparation, poor quality of propellant, damage to motor)	3 (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)	6, Low	Purchase propellant and motors only from reliable sources, check the motor for damage prior to launching, team members who prepare the motor must be supervised by at least one other team member.	Team member will be designated to observe the motor preparation procedure, only approved propellant sources will be used.	3, Low
Propellant Burns Through launch vehicle Components	2 (Faulty motor preparation, poor quality of propellant, poor construction, damage to motor, damage to propellant casing)	5 (Ballistic trajectory, catastrophic destruction of vehicle)	10, Medium	Purchase propellant and motors only from reliable sources, check the motor for damage prior to launching, team members who prepare the motor must be supervised by at least one other team member, test propellant casing in subscale flights.	Double check bulkhead after every flight to make sure it is in good enough condition for it to sufficiently protect launch vehicle components from propellent exhaust.	5, Low
Propellant Explosion	1 (Faulty motor preparation, poor quality of propellant, damage to motor)	5 (Ballistic trajectory, catastrophic destruction of vehicle, possible harm to bystanders)	5, Low	Purchase propellant and motors only from reliable sources, check the motor for damage prior to launching, team members who prepare the motor must be supervised by at least one other team member.	Team member will be designated to observe the motor preparation procedure, only approved propellant sources will be used.	5, Low
Payload Computer Failure	3 (Electrical failure, program error, poor setup of wiring causes a connection to come undone,	5 (Disqualified, objectives not met, loss of electronic control)	15, High	Test payload prior to flight, check batteries and connections before flight.	Ensure by design and testing that components will not fail under extreme stress.	5, Low

	forgotten connection, battery failure)					
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)	5 (Disqualified, objectives not met, failure to correctly trigger ejection charges)	15, High	Test the reliability of the wiring and batteries through subscale flights, check batteries and connections before flight.	Continuity checks will be used, visible wires will be inspected for nicks or damage prior to launch.	5, Low
Avionics Bay Fire	2 (Faulty wiring, battery failure, poor setup of wiring, adverse weather)	5 (May be disqualified if objectives are not met, possible failure to trigger ejection charges, damage to internal launch vehicle components)	10, Medium	Thermal protection of avionics bay, do not overload avionics bay with wiring, only purchase avionics and payload equipment from reliable sources, check avionics bay and payload performance with test launches.	Make sure no wires are exposed and that the avionics bay is sufficiently protected from heat.	5, Low
Human Error When Arming Avionics and Payload	3 (Forgotten connection, forgetting to activate avionics bay components or payload prior to launch)	5 (Disqualified, objectives not met, failure to correctly trigger ejection charges)	15, High	Leave reminders in multiple places to check that the avionics bay and payload are armed and ready before launch, follow launch checklists closely.	All designated launch procedure observers will inspect avionics for charge and activation.	5, Low
Arming System Failure	3 (Faulty arming system, faulty wiring, battery failure, poor setup of wiring causes a connection to come undone, forgotten connection)	5 (Disqualified, objectives not met, failure to correctly trigger ejection charges)	15, High	Ensure the avionics bay is successfully communicating with the team prior to flight, test arming system through test launches.	Ensure by design and testing that communication between components is established and reliable.	5, Low
Poor	2 (Failure to	5 (Partial or	10,	Read all instructions	Dual analysis will be	5, Low

Spacing Between the Ejection Charge and the Parachute	properly consider the requirements of the recovery system, poor budgeting of space in launch vehicle, failure to read instructions that come with parachute and/or ejection charges)	total damage to the parachute, parachute does not launch from the launch vehicle, possible recovery failure)	Medium	which come with the parachute and ejection charges, establish clear requirements of the recovery system early in the design process, run mathematical and physical analyses on the design of the launch vehicle, ensure the parachute is spaced properly with subscale test flights.	performed to insure no damage occurs to the parachute, ejection charge testing to insure no parachute damage.	
Stage Fails to Separate	3 (Faulty ejection charge, excessive strength is used to hold stages together, altimeter failure)	5 (launch vehicle does not follow desired flight path, possible ballistic trajectory, lower maximum height, damage to the launch vehicle)	15, High	Any team member who loads the ejection charges must be supervised by at least one other team member, examine ejection charges for damage before launch, ensure proper functionality of the altimeters, ejection charges, and interstage joints and fasteners through test flights and mathematical and physical analyses, have a secondary ejection charge for each stage separation.	Ejection charge testing will be performed to insure charges can separate stages, dual altimeters will be employed to enable redundancy.	5, Low
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)	5 (Main parachute does not slow down the launch vehicle, recovery failure, ballistic trajectory)	10, Medium	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, run mathematical and physical analyses as well as subscale tests to ensure parachute is in the right position in the launch vehicle, have a secondary ejection charge in case	Ejection charge testing will be done to insure charge effectively deploys parachute.	5, Low

				of emergency which is larger than the first.		
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)	5 (Drogue parachute does not slow down the launch vehicle, recovery failure, ballistic trajectory)	10, Medium	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, run mathematical and physical analyses as well as subscale tests to ensure parachute is in the right position in the launch vehicle, have a secondary ejection charge in case of emergency which is larger than the first.	Double check that packing of the drogue parachute to ensure that it reliably deploys.	5. Low
Parachute Canopy Breaks or Tears	1 (Poor canopy materials, improper ejection of recovery system, damage from previous flights or transportation)	5 (Possible recovery failure, ballistic trajectory)	5, Low	Only buy parachutes from reliable sources, remove threats to parachute integrity from the parachute housing, test the recovery system through mathematical and physical analyses as well as subscale flights, check the recovery system for damage before launch.	Run simulations and mathematical analysis to ensure the acquired parachute is capable of withstanding forces to safely descend the launch vehicle.	5, Low
Parachute Shroud Lines Break	1 (Poor shroud line materials, improper ejection of recovery system, damage from previous flights or transportation)	5 (Possible recovery failure, ballistic trajectory)	5, Low	Only buy parachutes from reliable sources, remove threats to parachute integrity from the parachute housing, test the recovery system through mathematical and physical analyses as well as subscale flights, check the recovery system for damage before launch.	Ensure by design and testing that the shroud lines are strong enough to handle expected forces.	5, Low
Shock Cord Break or	1 (Faulty shock cord,	5 (Parachute disconnect	5, Low	Any team member who connects the	Test the shock cord to ensure it can	5, Low

Disconnect	damage to shock cord, poor connection to the launch vehicle)	from the launch vehicle, recovery failure, ballistic trajectory)		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.	withstand the forces acting upon it during descent.	
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, unstable or ballistic flight)	5 (Shock cord or parachutes may not fully achieve their goal, possible ballistic trajectory, possible failed recovery)	10, Medium	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 performance of parachutes and shock cord in test flights, appropriately follow recommended sizings for shock cord and parachutes.	Designated parachute packing observer will record the packing and make notes on operation, and have right to demand repacking.	5, Low
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)	5 (Recovery failure, ballistic trajectory)	10, Medium	Only buy parachutes from reliable sources, test the recovery system through mathematical and physical analyses as well as subscale flights, check the recovery system for damage before launch, double check that the recovery system is properly mounted before launch.	Ensure by design and testing that the parachute is attached well.	5, Low
Parachute or Shock Cord Catches Fire	2 (Not enough space given between ejection charge and	5 (Shock cord or parachutes do not fully achieve their goal, possible	10, Medium	Any team member who packs the parachute or ejection charges must be supervised by at least	Designated packing operation observer will document packing process to insure proper	5, Low

	parachute, poor insulation of parachute, poor parachute packing, faulty or poorly chosen ejection charge)	ballistic trajectory, possible failed recovery, damage to internal launch vehicle components)		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.	placement.	
Destruction Due To Drag Forces	1 (Poor construction or improper materials used)	5 (Partial or total destruction of vehicle)	5, Low	Use appropriate materials and high powered building techniques.	Test subscale in wind tunnel and use dynamic similarity of coefficient of drag to estimate drag on full scale.	5, Low
Airframe Zipper	2 (Excessive deployment velocity)	5 (Partial destruction of vehicle)	10, Medium	Properly time ejection charges and use an appropriately long tether.	Test and observe at full scale launch prior to Huntsville.	5, Low
GPS Lock Failure	2 (Interference or dead battery)	5 (Loss of vehicle)	10, Medium	Ensure proper GPS lock and battery charge before flight.	Check battery charge before flight to ensure it is capable of providing power during the duration of flight.	5, Low
Insufficient Landing Speed	3 (Improper load, higher coefficient of drag for the parachutes than needed, higher surface area of the parachutes than needed)	2 (Unexpected changes in flightpath and landing area, increased potential for drift)	6, Low	Use subscale flights to determine if the subscale parachutes were accurately sized, use recommended and proven-to-work parachute sizing techniques.	Dual simulations will validate theoretical parachute performance.	2, Minimal
Excessive Landing Speed	3 (Parachute damage or entanglement, improper load)	5 (Partial or total destruction of vehicle)	15, High	Properly size, pack, and protect parachute based on landing speed calculations	Confirm proper landing energy via testing and observation at full scale launch prior to Huntsville.	5, Low
Battery Leakage/ Combustion	2 (Battery compartment becomes punctured)	5 (Potential for ballistic trajectory)	10, Medium	Check battery integrity before each launch.	Include checking battery condition in pre launch checklist.	5, Low

Loose Payload in Payload Bay	3 (Damage to payload retention system, improper payload installation)	4 (Improper orientation at payload deployment, possible mass shift affecting flight trajectory)	12, Medium	Payload retention system will be checked for cracks before sealing of launch vehicle, payload installation will be inspected as part of pre-launch procedure.	Inspection steps will be listed in pre-flight checklist.	4, Low
Improper Payload Orientation	4 (Damage to orientation system, system unable to read payload orientation)	4 (Failure to perform payload mission)	16, High	Payload bay will be visually inspected before sealing of launch vehicle for damage to orientation motors and physical obstructions.	Inspection steps will be listed in pre-flight checklist.	4, Low
Payload Rotor Arm Deployment Failure	2 (Damage to payload deployment springs, payload deployment obstruction)	4 (Failure to perform payload mission)	8, Medium	Payload spring systems will be inspected for damage and replaced as necessary.	Payload team lead will be responsible for final payload inspection prior to launch day.	4, Low

Table 7.5: Failure Modes and Effects Analysis

7.5. Environmental Hazard Analysis

Hazard	**Likelihood (Cause)**	**Severity (Effect)**	**Risk**	**Mitigation**	**Verification**	**Post Mitigation Risk**
Landscape	3 (Trees, brush, water, power lines, wildlife)	5 (Inability to recover launch vehicle, payload UAV crash)	15, High	Angle launch vehicle into wind as necessary to reduce drift and avoid hazards.	Inspect launch site before launch to verify that it is a suitable area to launch.	5, Low
Humidity	3 (Climate, poor forecast)	1 (Rust on metallic components)	3, Low	Use as little metal as possible. Store indoors.	Check weather beforehand for ideal launch time.	1, Minimal
Winds	3 (Poor forecast)	4 (Inability to launch, excessive drift, payload UAV drift/control issues)	12, Medium	Angle into wind as necessary and abort if wind exceeds 20 mph.	Check weather beforehand for ideal launch time.	4, Low
Rain/Storms	3 (Poor forecast)	3 (Damage to electrical components)	9, Medium	Keep launch vehicle away from moisture/elements	Visual inspection of electrical components to	3, Low

				prior to launch. Cover any exposed electrical components.	ensure dry surface.	
Low Visibility	2 (Fog)	4 (Inability to maintain visual contact with the launch vehicle)	8, Medium	Postpone launch if horizontal visibility is less than 5 miles, or if there is a cloud ceiling below our expected apogee with a 20% safety factor.	Check weather forecast for visibility conditions.	4, Low
High Temperatures	3 (Poor forecast)	3 (Heat stroke or damage to electrical components)	9, Medium	Keep launch vehicle in shaded area until before launch. Provide shaded area to team members as needed.	Check weather beforehand for ideal launch time. Contact will be made with launch site ahead of time to ensure presence of shaded structure, or some structure will be brought by the team.	3, Low
Low Temperatures	3 (Poor forecast)	3 (Frostbite, frost on ground, ice on vehicle, clogging of vehicle ventilation, change in launch vehicle rigidity and mass, higher drag force on launch vehicle)	9, 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. Provide team members with warm environment above 60°F.	Ensure team is notified through email or instant message of all weather on day of launch or manufacturing to wear proper clothing. Do not launch if weather is below designed intent of launch vehicle. Team members exhibiting signs of frostbite will be escorted to designated warm space (heated interior space or vehicle)	3, Low
Pollution From Exhaust	5 (Combustion of APCP motors)	1 (Small amounts of greenhouse gasses emitted)	5, Low	Use only launch vehicle motors approved for use by the National Association of launch vehicle, Canadian Association of launch vehicle, or Tripoli	Launch vehicle motors in consideration will be checked by a safety team member to ensure compliance.	5, Low

				Rocketry Association.		
Pollution From vehicle	2 (Loss of components from vehicle)	3 (Materials degrade extremely slowly, possible harm to wildlife or water contamination)	6, Medium	Properly fasten all components. Scavenge for fallen parts after launch is completed.	Inspect the securements of components before launch. Have designated clean up team for each launch.	3, Low
Pollution from Team Members	2 (Failed disposal of litter, improper cleanup procedures, members walk through important plantlife, farming fields, sod, etc.)	4 (Litter may degrade extremely slowly, wildlife may consume harmful litter)	8, 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 site cleaner than was found. Make sure disposable equipment is kept track of and guaranteed to remain at designated locations, designated waste disposal will be provided.	4, Low
Collisions with Man-made Structures	2 (Failure to properly predict trajectory, failure to choose an appropriate launch area)	5 (Damage to public property or private property not owned by the team, damage to team equipment)	10, Medium	Do not launch under adverse conditions which may affect the course of the launch vehicle (See Wind). Run a large number of tests which analyze the launch vehicle's trajectory mathematically and physically. Choose a launch area which is not close to civilization. Follow launch procedures closely.	Run tests to analyze and estimate the launch vehicle's trajectory so that the launch vehicle's path is known to the team. Do not launch launch vehicle under adverse weather conditions (See Wind) and choose a launch location which allows for open space to avoid accidents.	5, Low
Wildlife Contact with Launch Vehicle	1 (Failure to accurately predict trajectory, unexpected appearance of wildlife, poor choice of launch area)	4 (Damage to vehicle components, damage to wildlife)	4, Low	Launch in an open area with high visibility. Be aware of the surroundings when choosing a launch area and launching.	Perform visual sweep of launch field to ensure no wildlife is present. Gently encourage wildlife displacement if any is present: if cannot be displaced, relocate launch location.	4, Low

Wildlife Contact with Launch Pad	1 (Failure to monitor the launch pad, poor choice of launch area)	4 (Possible inability to launch the launch vehicle, unpredictable launch behavior or trajectory)	4, Low	Have at least one team member monitoring the launch pad at all times. 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.	Perform visual sweep of launch field to ensure no wildlife is present. Gently encourage wildlife displacement if any is present: if cannot be displaced, relocate launch location.	4, Low
Battery Leakage	3 (Absence of -or damage to - battery casing causing puncture or cracking)	4 (Possible toxic acid leak, heavy metal contamination)	12, 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.	Inspect battery casing prior to launch to ensure the battery is properly protected and unlikely to become punctured.	4, Low
Fire	5 (Exhaust caused by launch vehicle engine)	5 (Possible spread of wildfire, burns to personnel)	25, Very High	Ground will be cleared per NAR standard. Fire extinguishers will be on hand. Flame retardant tarp will be deployed to prevent catching of fire. Launch will not be performed on dry brush.	Inspection by safety officer will be performed to ensure compliance with NAR safety standard on minimum clear area. Launch site will be sprayed with water as necessary.	5, Low
Kinetic Damage to Buildings	2 (launch vehicle veers off trajectory causing landing in occupied area)	4 (Repairable destruction to building)	8, Medium	Choose launch site that is remote enough to make this risk negligible.	Ensure minimum distance from significant buildings/structures exceeds minimum personnel distance as established by NAR safety standard by a factor of at least 3.	4, Low
Kinetic Damage to Terrain	5 (launch vehicle has excessive landing speed)	1 (Creation of small ground divots, mild inconvenience to wildlife and	5, Low	Simulate landing conditions to ensure parachute generates sufficient drag to slow launch vehicle	Dual simulations will be performed to ensure proper parachute performance.	2, Minimal

		flora)		to acceptable parameters.		
Unstable Ground	2 (Poor choice of launch site, inclement weather creating mud or softening the ground)	3 (Personnel may slip or fall and damage equipment or themselves, launch pad may sink into the ground and cause an unexpected trajectory)	6, Low	A rigid system which can be used to support the launch pad, such as wooden planks (if needed to reduce their flammability, they may be wetted directly underneath the rocket), choice of a launch site which has rigid ground, observation of launch pad condition shortly before launch.	Use designated launch areas as designated to which must strictly follow this rule to be approved.	3, Low
Obstructions on Launch Field	4 (Rocky terrain, soft/uneven dirt, scrub, sand)	3 (Payload deployment failure)	12, Medium	Payload deployment will be tested on terrains including loose rocks, sand, and dirt ruts.	Testing will be documented by payload team lead (See PT_04.1).	3, Low

Table 7.6: Environmental Hazard Analysis

7.6. Project Risks Analysis

Hazard	Likelihood (Cause)	Severity (Effect)	Risk	Mitigation	Verification	Post Mitigation Risk
Improper Funding	3 (Lack of revenue)	5 (Inability to purchase parts)	15, High	Create and execute a detailed funding plan properly, minimize excessive spending by having multiple members check the necessity of purchases.	Have each team verify purchases with team lead to ensure the team is still within their given budget.	5, Low
Failure To Receive Parts	2 (Shipping delays, out of stock orders)	5 (Cannot construct and fly vehicle)	10, Medium	Order parts while in stock well in advance of needed date.	Order parts a month before needed. Acquire lead time from supplier for accountability.	5, Low
Damage to or Loss of Parts	2 (Failure during testing, improper part care during construction,	5 (Cannot construct or fly vehicle without spare parts)	10, Medium	Have extra parts on hand in case parts need to be replaced, follow all safety procedures for	Confirm a minimum number of parts needed so the team is able to	5, Low

	transportation, or launch)			transportation, launch, and construction.	obtain duplicates for certain parts. Assign responsibility for more important and expensive parts.	
Rushed Work	2 (Rapidly approaching deadlines, unreasonable schedule expectations)	4 (Threats of failure during testing or the final launch due to a lower quality of construction and less attention paid to test data)	8, Medium	Set deadlines which both keep the project moving at a reasonable pace and leave room for unforeseen circumstances.	Have team leads verify that projects are being completed before the deadline arrives.	4, Low
Major Testing Failure	2 (Improper construction of the launch vehicle, insufficient data used before creating the launch vehicle's design)	5 (Damage to vehicle parts, possible disqualification from the project due to a lack of subscale flight data, an increase in budget for buying new materials, delay in project completion)	10, Medium	Ensure parts used fall within specifications of required use. Take care to perform tests correctly.	Conduct proper tests to ensure that the designs are in fact reliable.	5, Low
Unavailable Test Launch Area	2 (Failure to locate a proper area to launch subscale launch vehicles for testing, failure to receive an FAA waiver for the test launch)	5 (Disqualification from the project due to a lack of subscale flight data)	10, Medium	Secure a reliable test launch area and FAA waiver well in advance of the dates on which test launch data is required.	Schedule a launch date a well in advance and set a deadline for when the FAA waiver is to be completed and submitted.	5, Low
Loss or Unavailability of Work Area	1 (Construction, building hazards, loss of lab privileges)	4 (Temporary inability to construct vehicle)	4, Low	Follow work area regulations and have secondary spaces available.	Inform members of proper work area etiquette to prevent loss of lab privileges. Regularly	4, Low

					confirm that the team has access to secondary locations if the need arises.	
Failure in Construction Equipment	1 (Improper long-term maintenance of construction equipment, improper use or storage of equipment)	3 (Possible long-term delay in construction)	3, Low	Ensure proper maintenance and use of construction equipment and have backup equipment which can be used in case of an equipment breakdown.	Inspect equipment before and after use to confirm the equipment is functioning properly.	3, Low
Insufficient Transportation	1 (Insufficient funding or space available to bring all project members to launch sites or workplace)	3 (Loss of labor force, team members lose knowledge of what is happening with the project, low attendance to the final launch)	3, Low	Organize and budget for transportation early and keep track of dates on which large amount of transportation are needed.	Organize transportation for at least a month in advance and make sure either enough drivers are secured or buses are rented.	3, Low
Inactivity / Low Availability of Personnel	2 (Members are unable or unwilling to work due to an increase in classwork or other mandatory activities)	5 (Low attendance, loss of team members, labor shortages, inability to construct vehicle)	10, Medium	Train all members to work in all areas necessary.	Utilization of work time table.	5, Low
Damage By Non-Team Members	1 (Accidental damage caused by other workspace users)	5 (Extensive repairs necessary, delay in construction)	10, Medium	Separate all components from other areas of the workspace as necessary.	Ensure only team members as known can have access to components.	5, Low
Damage During Transit	2 (Mishandling during transportation)	5 (Inability to fly launch vehicle)	10, Medium	Protect all launch vehicle components during transit.	Ensure launch vehicle safety secured by testing.	5, Low
Weather Delays	3 (Poor weather conditions during test launches, such	5 (Possible disqualification from the project due to a lack of	15, High	Have multiple dates available on which test launches can be conducted in case of adverse weather	Have backup date planned before.	5, Low

	as high wind speeds, ice and frost, or storms)	subscale flight data)		conditions.		
Payload Unable to Capture Sample	2 (Improper testing, poor design, design failure/ damage)	5 (Inability to complete mission)	10, Medium	Testing will be performed on sample collection system for a variety of potential sample analogs.	Testing will be documented by payload team lead (See PT_04.1)	5, Low

Table 7.7 Project Risks Analysis

8. Project Plan

8.1. Launch Vehicle Quick Reference

8.1.1. Lower Airframe

Sub-Group	Lower Airframe	**Weight**	10.5 lbm	
Responsible Subteam	Construction Team	**Manufacturing Process**	Filament Spun, Commercial CNC	
Sub-Components	Fins (x3), Airframe, Motor Tube, Centering Rings (x3), Rail Buttons (x2), Motor Retainer, Motor Tube, Parachute, Shock cord	**Vendor**	Wildman Rocketry	
Material	G12 FWFG, G10 FG, and 2 Part Epoxy-Resin, Aircraft Grade Aluminum	**Component Cost**	Airframe: $231.50 Centering Rings: $30.00 Motor Retainer: $51.00 Motor Tube: $41.02 Total: $353.52	1 3 ($10.00 ea) 1 1 1
General Dimensions	ID: 6 in OD: 6.17 in Length: 40 in			
Group Description	The purpose of the lower airframe is to house the fins, the motor and its mount, and the main recovery system.			

Table 8.1: Lower Airframe Quick Reference

Failure Modes and Effects Assembly Analysis					
Hazard	**Likelihood (Cause)**	**Severity (Effect)**	**Risk**	**Mitigation**	**Verification**
Material Buckling	2 (Damage to or cracking of airframe, unexpected pressure from rocket motor)	4 (Possible catastrophic loss of launch vehicle)	10, Medium	Visually inspect airframe for cracks or scratches prior to launch, ensure only approved motors are used in launch.	Inspection will be performed by construction team of all components prior to onsite launch vehicle construction, and safety team lead and project manager prior to installation on

					launch pad.
Damage to Fins	3 (Kinetic damage to fins, damage to fins in transit)	2 (Improper launch trajectory)	6, Low	Properly pack fins during transport to launch field, inspect fins before installation on airframe.	Construction team will inspect fins before installation, packing material will be added between and around fins before transit.
Loss of Fins	3 (Improper epoxy application)	2 (Improper launch trajectory, out of specification pre-launch stability)	6, Low	Visually and physically inspect epoxy, only use appropriately rated epoxies for launch vehicle applications.	Look and feel along all epoxy seams for gaps or cracks in epoxy, check performance ratings for epoxy prior to purchase.

Table 8.2: Lower Airframe FMEA

8.1.2. Avionics Bay

Sub-Group	Avionics Bay Group	**Weight**	6.34 lbm	
Responsible Subteam	Avionics Team	**Manufacturing Process**	Filament Spun, Commercial CNC, 3D Printing	
Sub-Components	3.7V LiPo Battery, 9V Battery, Altimeter Sled, Battery Guard, Bulkheads (x2), Camera, Camera Holder, Charge Wells (x4), Coupler, I-bolts (x2), RRC3+ Sport Altimeter, Camera Electronics Sled, E-matches (x4), Switch Holders (x4), Switches (x4), Telemetrum Altimeter, Terminal Blocks (x4), Threaded Rods (x2), Assorted Wiring, Screws, Nuts, and Washers	**(Major) Component Cost**	Telemetrum: $300.00 RRC3+ Sport: $85.00 Switch: $4.60 Secondary Payload System: $80.00 3.7V LiPo Battery: $11.50 9V Battery: $1.50 Charge Well: $16.00 E-match: $2.40 Terminal Block: $14.20 Threaded Rod: $24.00 All Fiberglass Components: $65.00 All 3D Printed Parts: $2.15 All Other Components: $25.00	1 1 4 ($1.15 each) 1 1 1 4 ($4.00 each) 4 ($0.60 each) 4 ($3.55 each) 2 ($12.00 each) 1 1 1

			Total: $631.35	
Material	G10 FG, 2 Part Epoxy-Resin, PLA, Aluminum, Plastic, etc.			
General Dimensions	ID: 5.775”	OD: 6”	Length: 14” (Coupler), 17.25” (Total)	
Group Description	The primary purpose of the avionics bay is to house the primary and redundant altimeter/ejection systems, provide an attachment point for the drogue and main parachutes, and house the secondary payload (camera) system.			

Table 8.3: Avionics Bay Quick Reference

Failure Modes and Effects Assembly Analysis					
Hazard	**Likelihood (Cause)**	**Severity (Effect)**	**Risk**	**Mitigation**	**Verification**
Cracking of Altimeter Sled	3 (Dropping of sled, over torquing of fixture nuts)	4 (Possible damage to electronics, inability to track altitude)	12, Medium	Visually inspect sled for cracks prior to and after installation on threaded rod, avoid over torquing of fixture nuts.	Inspection of sled will be performed during and after installation in launch vehicle
Insufficient Battery Charge	4 (Team neglects to charge or use fresh batteries)	4 (Failure of avionics components)	16, High	Charge all batteries the night before launch, check batteries for proper voltage prior to installation, ensure avionics are functioning prior to launch.	Avionics team lead will be responsible for batteries and battery installation.
Poor or Loose Lead Connections	3 (Pulled leads, lead corrosion)	4 (Failure of avionics components)	12, Medium	Inspect all leads before installation in avionics components (tin as necessary), pull test all leads before installation in launch vehicle.	Leads will be inspected and manually pull tested by avionics team lead before installation.

Table 8.4: Avionics Bay FMEA

8.1.3. Upper Airframe

Sub-Group	Upper Airframe Group	Weight	5.61 lbm	
Responsible Subteam	Construction Team	Manufacturing Process	Filament Spun	
Sub-Compo nents	Airframe, Drogue Parachute, Shock cord, Rail button	Vendor	Wildman Rocketry	
Material	G12 FWFG	Component Cost	Airframe: $185.00 Total: $185.00	1
General Dimensions	ID: 6 in OD: 6.17 in Length: 48 in			
Group Description	The purpose of the upper airframe is to house the drogue recovery system, as well as part of the payload bay.			

Table 8.5: Upper Airframe Quick Reference

Failure Modes and Effects Assembly Analysis					
Hazard	**Likelihood (Cause)**	**Severity (Effect)**	**Risk**	**Mitigation**	**Verification**
Material Buckling	1 (Damage to or cracking of airframe)	4 (Possible catastrophic loss of launch vehicle)	4, Low	Visually inspect airframe for cracks or scratches prior to launch, ensure only approved motors are used in launch.	Inspection will be performed by construction team of all components prior to onsite launch vehicle construction, and safety team lead and project manager prior to installation on launch pad.
Parachute Failure	3 (Inability to deploy, damage to parachute)	5 (Possible ballistic trajectory)	15, High	Visually inspect parachute for damage prior to installation, properly pack parachute.	Parachute will be inspected in full deployment on the ground for holes or rips, parachute will only be packed by qualified team members.

Table 8.6: Upper Airframe FMEA

8.1.4. Payload Bay (Including Nosecone)

Sub-Group	Payload Bay	**Weight**	Nosecone: 2.8 lbm UAV: 2.7 lbm Payload R&D System: 9.0 lbm Total: 14.5 lbm
Responsible Subteam	Payload Team	**Manufacturing Process**	Filament Spun, Commercial CNC, 3D Printing, Waterjet
Sub-Components	Nosecone, Payload coupler, Payload R&D system, UAV	**Major Vendors**	Wildman Rocketry, McMaster Carr, Digikey, Servocity, Adafruit, Hobbyking
Material	G12 FWFG, G10 FG, and 2 Part Epoxy-Resin, Aircraft Grade Aluminum, Nylon-6 (UAV), PLA (UAV and some R&D components)	**Component Cost**	Nosecone & Coupler: $129.00 Payload R&D system: $975.03 Payload UAV: $963.11: **Total:** $2067.14
General Dimensions	**Overall system:** ID: 6.00" OD: 6.15" Length: 49" **Nosecone:** ID: 6.00" OD: 6.15" Length: 36.0" **Payload Bay:** ID: XX" OD: 5.775" Length: 18" **UAV (closed configuration):** 13" x 3.43" x 4.60" **UAV (open configuration):** 13.78" x 3.43" x 4.60"		
Group Description	The payload bay is responsible for safely retaining the UAV payload throughout the flight. To do so, this system includes a sophisticated active retention and deployment system. This system also includes the launch vehicle's nosecone, forming the topmost component of the vehicle's airframe.		

Table 8.7: Payload Bay Quick Reference

Failure Modes and Effects Assembly Analysis					
Hazard	**Likelihood (Cause)**	**Severity (Effect)**	**Risk**	**Mitigation**	**Verification**
Material Buckling	1 (Damage to or cracking of nosecone or coupler)	4 (Possible catastrophic loss of launch vehicle)	4, Low	Visually inspect nosecone and couplers for cracks or scratches prior to launch, ensure only approved motors are used	Inspection will be performed by construction team of all components prior to onsite launch vehicle construction, and

				in launch.	safety team lead and project manager prior to installation on launch pad.
Obstructions on Launch Field	4 (Rocky terrain, soft/uneven dirt, scrub, sand)	3 (Payload deployment failure)	12, Medium	Payload deployment will be tested on terrains including loose rocks, sand, and dirt ruts.	Testing will be documented by payload team lead (See PT_04.1).
Loose Payload in Payload Bay	3 (Damage to payload retention system, improper payload installation)	4 (Improper orientation at payload deployment, possible mass shift affecting flight trajectory)	12, Medium	Payload retention system will be checked for cracks before sealing of launch vehicle, payload installation will be inspected as part of pre-launch procedure.	Inspection steps will be listed in pre-flight checklist.
Improper Payload Orientation	4 (Damage to orientation system, system unable to read payload orientation)	4 (Failure to perform payload mission)	16, High	Payload bay will be visually inspected before sealing of launch vehicle for damage to orientation motors and physical obstructions.	Inspection steps will be listed in pre-flight checklist.
Payload Rotor Arm Deployment Failure	2 (Damage to payload deployment springs, payload deployment obstruction)	4 (Failure to perform payload mission)	8, Medium	Payload spring systems will be inspected for damage and replaced as necessary.	Payload team lead will be responsible for final payload inspection prior to launch day.

Table 8.7: Payload Bay FMEA

8.2. Avionics Testing

8.2.1. Altimeter Continuity Test - A_01

Test ID: A_01 **Test Name:** Altimeter Continuity Test	**Objective and Tested Variables:** This test ensures that continuity can be achieved when an e-match is connected to the drogue and main outputs of both the Telemetrum and RRC3+ Sport altimeters in a variety of temperatures. The tested variable is	**Reason for Test:** This test validates the use of each altimeter within the avionics bay of the launch vehicle. Each altimeter will need to achieve continuity when connected in the final

Related Requirements: 3.12.2, T3.5 **Related Test IDs:** A_02	the number of beeps (or *dits* for the Telemetrum) that occur every five seconds after altimeter initialization.	system, and this test verifies the capability of the altimeters to do that at varying temperatures.
	Success Criteria: Each altimeter passes this test if it emits three beeps (or *dits* for the Telemetrum) every five seconds after the initialization routine for all three trials in both temperature extremes, indicating successful continuity for a dual deploy configuration.	
Methodology	1. A 9V battery and a switch were connected to the RRC3+ Sport altimeter. An e-match was also connected to each of the drogue and main outputs. 2. The altimeter was powered on and allowed to complete its initialization routine. 3. The number of continuity beeps that were subsequently emitted was then recorded for each of the three trials. 4. The same procedure was repeated with the Telemetrum altimeter (but with a 3.7V LiPo battery and listening for *dits* instead). 5. The entire test was conducted in both the early fall and in the winter in order to verify that continuity for a dual deploy configuration in both hot and cold environmental conditions can be achieved.	
Possible Impacts	Analyzing the results from this test will verify the plan for the launch vehicle avionics bay. For the RRC3+ Sport altimeter, 3 beeps every 5 seconds in each of the temperature climates will validate its inclusion in the avionics bay. For the Telemetrum altimeter, 3 *dits* every 5 seconds will also verify its inclusion. These results are critical to the design of the launch vehicle, as these altimeters will be used throughout the flight, especially at apogee and at the main deployment altitude.	
Results **Status:** *In Progress*	The first round of testing was conducted in the early fall to simulate warmer temperatures. Both the RRC3+ Sport and Telemetrum altimeters passed this round of testing. The second set of trials for this test will be conducted between January 10th and January 12th. This time period was chosen in order to test the continuity in colder temperatures. The data from the cold-weather testing will be analyzed the same way as the initial trials.	

8.2.2. Altimeter Ejection Vacuum Test - A_02

Test ID: A_02 **Test Name:** Altimeter Ejection Vacuum Test **Related Requirements:** 3.1.1, T3.4, T3.4.1, T3.4.2, T3.4.3 **Related Test IDs:** A_01	**Objective and Tested Variables:** This test ensures that both the Telemetrum and RRC3+ Sport altimeters ignite the drogue ejection charge at apogee (or one second after apogee for the RRC3+ Sport) and the main ejection charge at the correct descent altitude (800') (or 700' for the RRC3+ Sport) at pressures corresponding to these theoretical altitudes. The tested variable is the illumination (or lack thereof) of LEDs wired to the altimeters to signify ignition. **Success Criteria:** The Telemetrum altimeter passes this test if the magnitude of the difference between the apogee altitude and the altitude the drogue LED illuminates at is less than 500' and the altitude the main LED illuminates at is between 800 ± 50' for all three trials. The RRC3+ Sport altimeter passes this test if the drogue delay (the time between apogee and illumination of the drogue LED) is between 0.75 and 1.75 seconds (as it is programmed to be 1.00 second) and the main LED illuminates between 800 ± 50' for all three trials.	**Reason for Test:** This test justifies the design of the avionics bay within the launch vehicle and verifies that the altimeters will ignite the ejection charges at the correct times.
Methodology	1. A sheet of plexiglass was prepared by drilling one large hole into it. A wine stopper was placed into this hole. 2. A ring of plumbers' putty was placed around the rim of the bowl. 3. To test one altimeter, an LED was connected to each the drogue and main outputs, and this system (along with the Altimeter One) was placed in the glass bowl. 4. A wine bottle air remover pump was then used to remove air through the stopper. Once the process of removing air was halted at the expected apogee altitude (the digital display of the Altimeter One indicated when this was), the drogue LED was expected to illuminate (or one second after apogee for the RRC3+ Sport). 5. Finally, the stopper was very slightly lifted away from the bowl to slowly allow air back inside it, causing the altitude to decrease according to the Altimeter One. The main LED was expected to illuminate at pressures corresponding to an altitude of 800' (or 700' for the RRC3+ Sport). 6. The flight data was downloaded onto a laptop for analysis.	
Possible Impacts	The results of this test will verify the planned design of the avionics bay, specifically the pairing of the RRC3+ Sport and Telemetrum altimeters with ejection charges. Verifying this design will ensure the inclusion of ejection charges is safe, and that they will only ignite at the correct times in flight.	
Results **Status:** *Incomplete*	This test will be conducted on the 11th of January. This testing date ensures that the test will be completed with enough time to make any necessary changes before flight.	

8.2.3. Avionics Ejection Black Powder Test - A_03

Test ID: A_03 **Test Name:** Avionics Ejection Black Powder Test **Related Requirements:** 3.2, T3.3 **Related Test IDs:** N/A	**Objective and Tested Variables:** The objective of this test is to ensure that the black powder ejection system for the recovery components functions correctly. The tested variable is how much black powder should be used in each canister, measured in grams.	**Reason for Test:** This test justifies the use of black powder canisters within the avionics bay ejection system on the launch vehicle.
	Success Criteria: Each black powder canister passes this test if its ignition resulted in at least 6' of separation of the avionics bay from the corresponding airframe, as well as full ejection of the corresponding parachute, for at least one amount of black powder equal to or greater than 5g for the upper airframe and 2g for the lower airframe.	
Methodology	1. The black powder canister on the upper airframe side of the avionics bay was filled with 5g of black powder. 2. An e-match connected to a 10' extension wire was inserted into the canister. 3. The avionics bay was reconnected to the upper airframe using shear pins. 4. The person conducting the test stood 10' away from the system and connected a 9V battery to the extension wire. The ejection charges were then expected to ignite and result in separation of the two components. If they did indeed separate, the distance between them was measured in feet. 5. If the success criteria were not met, the procedure was repeated using increasing amounts of black powder (in 1g increments) until 6' of separation and full ejection of the parachute were achieved. This last amount of black powder was then recorded as the ideal amount of black powder. 6. The procedure was also repeated for the black powder canister on the lower airframe side of the avionics bay, with 2g of black powder.	
Possible Impacts	The results of this test will ensure the black powder canisters are safe to use within the avionics bay of the launch vehicle. Specifically, the separation distance of at least 6' will verify each individual black powder canister. The overall verification of this system will add confidence that the launch vehicle will safely descend after reaching apogee.	
Results **Status:** *Incomplete*	This test will be conducted on the 12th of January. This test date allows for enough time to conduct the test, analyze data, and make necessary changes before flight.	

8.2.4. Avionics Battery Drain Test - A_04

Test ID: A_04 **Test Name:** Avionics Battery Drain Test **Related Requirements:** 3.12.2, T3.2 **Related Test IDs:** N/A	**Objective and Tested Variables:** This test ensures that the altimeter batteries can last for the duration of the launch. The tested variable is each battery's electric power over time.	**Reason for Test:** This test will verify that the batteries deliver enough power to the altimeters throughout the duration of the launch.
	Success Criteria: Each battery passes this test if, connected to its corresponding altimeter, the battery is able to keep it powered on for an hour and a half as well as stay within 40 and 60 mAh of electric charge (measured every half hour).	
Methodology	1. One 9V battery was connected to the RRC3+ Sport altimeter. 2. The altimeter was powered on, and the system was left running for an hour and a half. 3. Every thirty minutes, the electric charge of the battery was recorded using a multimeter. 4. After an hour and a half, the altimeter was checked to see if it was still powered on. 5. The procedure was repeated with a 3.7V LiPo battery and the Telemetrum altimeter.	
Possible Impacts	Successful completion of this test will add to the team's confidence that the altimeters will be able to return critical altitude values and initiate important flight events.	
Results **Status:** *Incomplete*	This test will be conducted on the 11th of January. In the event of a test failure, the timing allows for time to test and install other batteries before flight.	

8.2.5. Parachute Drop Test - A_05

Test ID: A_05 **Test Name:** Parachute Drop Test **Related Requirements:** 3.3, T3.1 **Related Test IDs:** N/A	**Objective and Tested Variables:** This test ensures the drogue parachute opens within a consistent time frame after being ejected and the main parachute allows the largest section of the launch vehicle to land with a kinetic energy below the threshold specified in the competition requirements.	**Reason for Test:** This test will verify the inclusion of the designated drogue and main parachutes in the design of the launch vehicle. These parachutes will need to be reliable in order to ensure a safe decent of the launch vehicle after apogee.
	Success Criteria: The drogue parachute passes this test if the three recordings for the amount of time it took for it to come to a fully opened state are within one second of each other. The main parachute passes this test if, after being dropped from an already opened state, the calculated kinetic energy of the largest section of the launch vehicle is below 75 ft-lbf.	
Methodology	1. Fifty pounds of weights were attached to the drogue and main parachutes separately to provide a stand-in for the final launch vehicle, which is expected to weigh about 50lbm. 2. Through prior testing from last season, it was determined that approximately 40' was an ample distance for the drogue parachute to open from a correctly-folded state. Therefore, it was folded in the packing configuration and dropped from a height of approximately 40'. 3. The drogue parachute was dropped three times from this height; using a stopwatch, the amount of time it took to fully open was measured. 4. The main parachute was dropped three times from the same height in a similar fashion, except it was held open by two people prior to being dropped; using a stopwatch, the amount of time it took to reach the ground was measured. This time was then used to calculate the landing kinetic energy of the largest section of the launch vehicle.	
Possible Impacts	Verifying the drogue and main parachutes will ensure the launch vehicle has a safe and controlled descent after apogee.	
Results **Status:** *Incomplete*	This test will be conducted on the 12th of January. This allows for time to test and install other parachutes if necessary.	

8.3. Payload Testing

Test Phases

The payload test program will be divided into two phases: the Developmental Testing (DT) phase, and Operational Testing (OT) phase. The DT phase will primarily consist of component-level, subsystem, and individual system testing to verify and document performance characteristics. The OT phase will primarily consist of verifying mission capability. OT will ultimately determine the effectiveness and reliability of the UAS for its designed mission. A flowchart displaying the test program and its progress is shown in Figure 8.1 below. Italicized on the bottom of each block are the tests that must be completed prior to entry into the next block. Each of these tests are mapped to a single requirement and all of the tests can be found in Table 8.9.

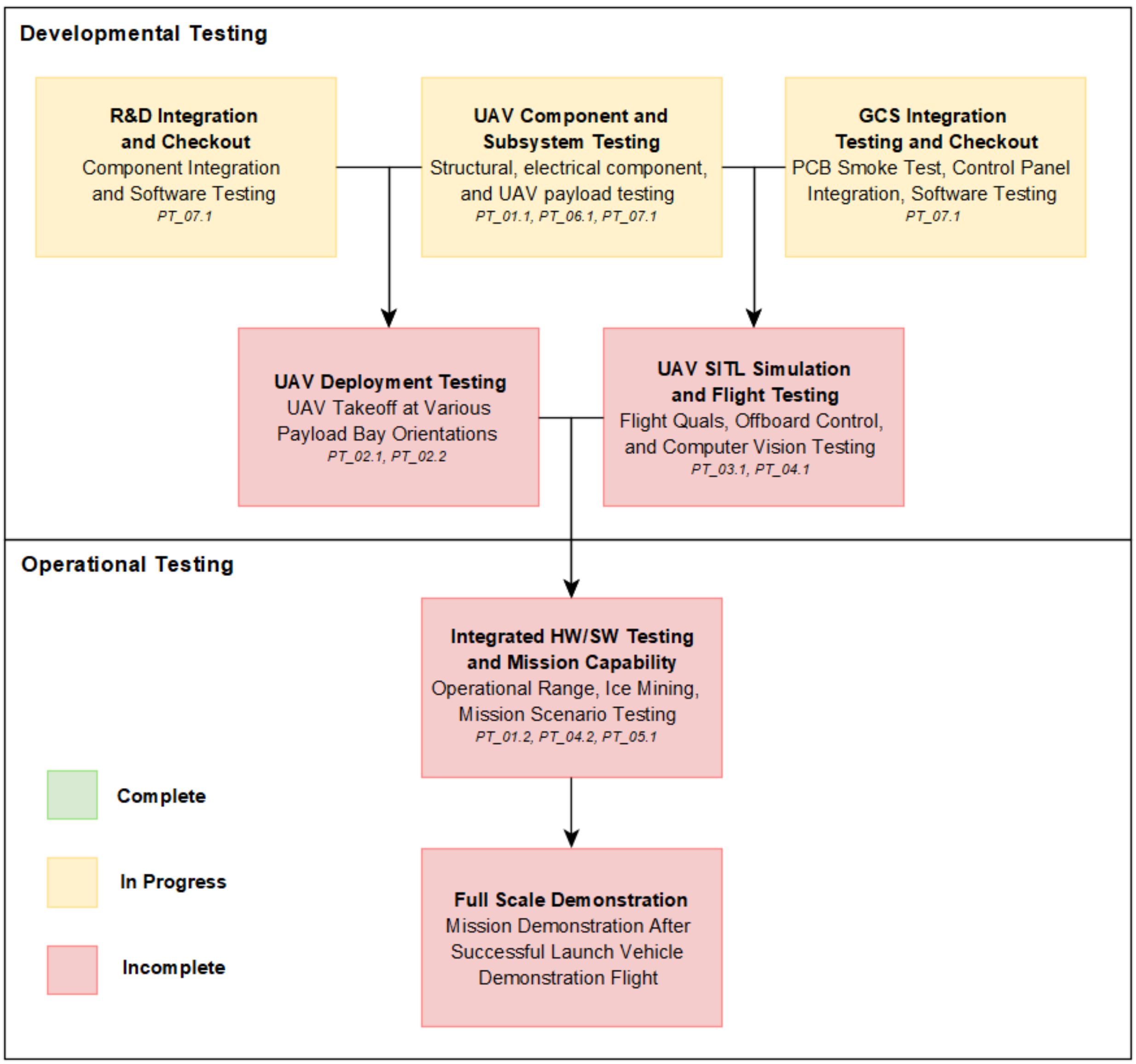


Figure 8.1: Payload Test Plan Flowchart

Tests and Test Conditions

Verification of certain requirements, both NASA and team-derived, necessitated multiple tests/test points. Each requirement is mapped to its own unique test identifier and each test identifier has a set of 1 or more tests. The payload test summary can be found below in Table 8.9 and test summary sheets can be found directly below the table.

Req. ID	Test ID	Test	SUT	DT/OT	Status
4.3.2	PT_01.1	IMPS Stand Test	UAV	DT	In Progress
	PT_01.2	Onboard Ice Mining Test	UAV	OT	Incomplete
PR_2.6	PT_02.1	Variable Launch Orientation Test	UAV/R&D	DT	Incomplete
	PT_02.2	Variable Sled Orientation Test	UAV/R&D	DT	Incomplete
PR_2.7	PT_03.1	Flight Controller Tuning	UAV	DT	Incomplete
PR_2.8	PT_04.1	SITL Ice Recovery Testing	UAV/GCS	DT	Incomplete
	PT_04.2	Recovery Area Identification Testing	UAV/GCS	OT	Incomplete
PR_2.9	PT_05.1	RF Comms Testing	All	DT	Incomplete
PR_2.10	PT_06.1	XWing Structural Testing	UAV	DT	In Progress
PR_2.11	PT_07.1	Battery Drain and Power Testing	All	DT	In Progress

Table 8.9: Payload Test Summary Matrix

8.3.1. IMPS Stand Testing - PT_01.1

Test ID: PT_01.1 **Test Name:** IMPS Stand Testing **Related Requirements:** 4.3.2 **Related Test IDs:** PT_01.2, PT_04.1, PT_04.2	**Objective and Tested Variables:** To confirm the ice mining and procurement system's ability to extract different shapes and sizes of simulated lunar ice material. The test variable is the volume of collected material.	**Reason for Test:** This test is integral to verifying that the IMPS is capable of meeting the mission requirement of collecting at least 10 mL of lunar ice. Additionally, since the exact material properties of the simulated lunar ice is unknown, analyzing the system's performance with different materials may inform small changes of the IMPS design moving forward.
	Success Criteria: The test is successful if the IMPS extracts 10+ mL of each sample of simulated lunar ice material.	
Methodology	1. Place simulated lunar ice material in a 1'x1' containment unit. Enough lunar ice material should be used such that a depth of at least 2" is created. 2. A test stand shall shall be developed to simulate how the IMPS mounts to the UAV. Place the test stand, with the IMPS attached, on top of the simulated lunar ice material. 3. Run the IMPS for 15 seconds. 4. Measure the total volume of material collected in each scoop component of the IMPS. 5. Repeat steps 1-4 for at least 3 different simulated lunar ice materials over varying shapes, sizes, and textures.	
Possible Impacts	Successful completion of this test indicates that the current design of the IMPS system is capable of collecting the proper amount of lunar ice material. Geometric design and control strategy changes could occur if performance of the system does not meet the given specifications.	
Results **Status:** *In Progress*	This test is not yet completed.	

8.3.2. On-Board Ice Mining Test - PT_01.2

Test ID: PT_01.2 **Test Name:** On-Board Ice Mining Test **Related Requirements:** 4.3.2 **Related Test IDs:** PT_01.2, PT_04.1, PT_04.2	**Objective and Tested Variables:** To confirm the ice mining and procurement system's ability to extract simulated lunar ice material under flight conditions. The test variable is the volume of collected material.	**Reason for Test:** This test is integral to verifying that the IMPS is capable of meeting the mission requirement of collecting at least 10 mL of lunar ice. Additionally, this test ensures that the IMPS properly integrates into the overall UAV system and is capable of containing sampled material while the vehicle flies away from the recovery site.
	Success Criteria: The test is successful if the IMPS extracts and contains 10+ mL of simulated lunar ice material.	
Methodology	1. Create a simulated lunar ice recovery area by placing a 3' diameter of simulated lunar ice material in the center of a 10'x10' tarp. The lunar ice material should have an average depth of 2". 2. Fly the UAV 100' above the simulated recovery area. 3. From 100' AGL, slowly land the vehicle on the simulated recovery area. 4. Upon landing, remotely trigger the IMPS to begin extracting simulated lunar ice material. 5. After 15 seconds, trigger the IMPS to stop lunar ice extraction. 6. Fly the UAV at least 10' linearly away from the recovery area and land the vehicle. 7. Measure the volume of lunar ice material collected by the IMPS.	
Possible Impacts	Successful completion of this test indicates that the current design of the IMPS system is capable of collecting the proper amount of lunar ice material. Changes to the interface between the ice mining system and the UAV could result from this test.	
Results **Status:** *Incomplete*	This test is not yet completed.	

8.3.3. Variable Launch Orientation Test - PT_02.1

<table>
<tr><td rowspan="2">Test ID: PT_02.1
Test Name: Variable Launch Orientation Test
Related Requirements: PR_2.6
Related Test IDs: PT_02.2</td><td>Objective and Tested Variables: Determine the maximum angle at which the R&D system can still successfully deploy the UAV. This test should push the R&D system to failure, thus finding the point at which UAV deployment should no longer be attempted. The test variable is the angle of the payload bay with respect to the ground.</td><td>Reason for Test: This test is necessary to gain a better understanding of the limits of the R&D system. This limit can then be compared to predicted landing orientations to determine compliance with mission requirements. This data can be compared with the data found in PT_02.2 to create a grid of potential UAV launch orientations.</td></tr>
<tr><td colspan="2">Success Criteria: This test is successful when an angle with respect to the ground in which UAV deployment is no longer possible is found.</td></tr>
<tr><td>Methodology</td><td colspan="2">1. Place the UAV inside the payload bay in the configuration in which it will land during the mission.
2. Prop the upper-airframe up off of the ground, such that the entire assembly with the payload bay enclosed makes an angle of 5° with the ground.
3. Remotely signal the R&D system to initiate the UAV deployment sequence. Take note of how the R&D system responds to its orientation with respect to the ground.
4. After the UAV deployment sequence is complete, initiate deployment of the UAV.
5. If the UAV successfully takes off in step 4, repeat steps 2-4 in 5° increments. Repeat until the UAV is no longer able to deploy.
6. Repeat steps 1-5 propping the nosecone up off of the ground instead of the upper-airframe.</td></tr>
<tr><td>Possible Impacts</td><td colspan="2">If the payload team decides that the angle found in this test is not large enough, design modifications to increase this maximum angle might be made.</td></tr>
<tr><td>Results
Status: Incomplete</td><td colspan="2">This test is not yet completed.</td></tr>
</table>

8.3.4. Variable Sled Orientation Test - PT_02.2

<table>
<tr><td rowspan="2">Test ID:
PT_02.2
Test Name:
Variable Sled Orientation Test
Related Requirements:
PR_2.6
Related Test IDs:
PT_02.1</td><td>Objective and Tested Variables: Determine the maximum angle with respect to the axis of the payload bay in which the UAV can successfully deploy from the sled. This test should verify how precise the gyroscopic reorientation system within the broader R&D system needs to be. The test variable is the payload sled angle.</td><td>Reason for Test: This test is necessary to gain a better understanding of the limits of the R&D system. Identifying how far off-axis the deployment sled can be, while still enabling a successful take-off, will provide information about the necessary precision of the R&D reorientation system. This data can be compared with the data found in PT_02.1 to create a grid of potential UAV launch orientations.</td></tr>
<tr><td colspan="2">Success Criteria: This test is successful when an angle with respect to the payload bay axis, in which UAV deployment is no longer possible, is found.</td></tr>
<tr><td>Methodology</td><td colspan="2">1. Place the UAV inside the payload bay in the configuration in which it will land during the mission.
2. Separate the nose cone and the upper-airframe, simulating the beginning of the UAV deployment process.
3. Manually rotate the sled on which the UAV sits 2° from the ideal take-off orientation.
4. Initiate deployment of the UAV. Take note of how the orientation of the sled affects the UAV’s takeoff.
5. If the UAV successfully takes off in step 4, repeat steps 2-4 in 2° increments. Repeat until the UAV is no longer able to deploy.</td></tr>
<tr><td>Possible Impacts</td><td colspan="2">If the payload team decides that the angle found in this test is not large enough, design modifications to increase this maximum angle might be made.</td></tr>
<tr><td>Results

Status: Incomplete</td><td colspan="2">This test is not yet completed.</td></tr>
</table>

8.3.5. Rate Controller Tuning - PT_03.1

<table>
<tr><td rowspan="2">Test ID:
PT_03.1
Test Name:
Rate Controller Tuning
Related Requirements:
PR_2.7
Related Test IDs:
N/A</td><td>Objective and Tested Variables: Tune the flight controller’s (angular) rate and attitude controller to improve flight qualities and reduce the effects of noise and disturbances. The test variables are roll rate, pitch rate, and yaw rate PID gains, attitude proportional gain, and vehicle response.</td><td>Reason for Test: This test ensures that the UAV is properly tuned to improve flight quality, increase efficiency, reduce vibrations that may affect onboard hardware, and reduce the likelihood of a crash. Failure to perform this test may lead to poor handling qualities, reduced flight time, and a reduction in the performance of other vehicle hardware.</td></tr>
<tr><td colspan="2">Success Criteria: The test is successful if changes to the PID gain parameters result in an improvement to the vehicle’s response to a setpoint angular rate (roll rate, pitch rate, or yaw rate) and increased maneuverability.</td></tr>
<tr><td>Methodology</td><td colspan="2">Testing of the vehicle’s angular velocity response will involve the use of pre-programmed mission files that will utilize the offboard control capability of the flight controller. Each of these pre-programmed mission files will focus on either roll, pitch, or yaw, and will supply a brief step function commanding a roll, pitch, or yaw rate shortly after takeoff. The UAV will then land be tuned automatically before being flown again until results are satisfactory. These pre-programmed missions will be tested in simulation before they are tested on the UAV.
1. Place the UAV in a flight ready configuration such that it is powered and ready to takeoff.
2. Establish and ensure a connection between the GCS and UAV.
3. Load and begin autotuning mission of either roll, pitch, or yaw.
4. After the UAV lands, adjust the PID parameters.
5. Repeat steps 1-4 until results are satisfactory.</td></tr>
<tr><td>Possible Impacts</td><td colspan="2">Successful completion of this test will allow for smoother, more efficient flight of the UAV.</td></tr>
<tr><td>Results
Status: Incomplete</td><td colspan="2">This test is not yet completed.</td></tr>
</table>

8.3.6. SITL Ice Recovery Testing - PT_04.1

<table>
<tr><td rowspan="2">Test ID:
PT_04.1
Test Name:
SITL Ice Recovery Testing
Related Requirements:
PR_2.8
Related Test IDs:
PT_04.2</td><td>Objective and Tested Variables: Verify the performance of the UAV's vision-guided descent algorithm in a simulated environment. The test variable is the UAV targeted landing accuracy and landing velocity.</td><td>Reason for Test: This test is essential for verifying the performance of the vision-guided descent algorithm. As this phase of the mission involves the landing of the UAV, successful completion of this event in a software environment is essential to mitigating the safety risks posed by testing on the flight vehicle. Successful completion of this test is required before performing PT_04.2 which involves repeating this procedure with the physical UAV.</td></tr>
<tr><td colspan="2">Success Criteria: This test is considered successful if the UAV lands softly within 1' of the center of the lunar ice recovery area. A "soft" landing is defined as a maximum landing velocity of 2 ft/s.</td></tr>
<tr><td>Methodology</td><td colspan="2">This test is completely performed in a software environment due to the associated risks with landing the UAV while its software is still in development. As such, this test will be performed many times to test new versions of the landing software. Each of the following steps should therefore be followed in the context of a software simulation.
1. Place a simulated lunar ice recovery area on the ground plane. This recovery area should be 10'x10' with a target circle of diameter 3' at the recovery area's center.
2. Position the UAV approximately 100' above the recovery area, offset 10' from the recovery area's center.
3. Run the UAV's vision-guided descent algorithm. Record the velocity of the UAV throughout its descent and take note of the quality of the descent. Is the control system tuned for accurate, efficient performance?
4. If the UAV's landing velocity and landing position meet the given success criteria, repeat the procedure at least 3 additional times, varying the initial lateral position offset from the recovery area up to 25'. Take note of how the vision-guided descent algorithm reacts to different amounts of initial offset.</td></tr>
<tr><td>Possible Impacts</td><td colspan="2">This test will be conducted many times as the landing algorithm is further developed. This test directly impacts the development of this algorithm and how it will be implemented on the UAV.</td></tr>
<tr><td>Results
Status: Incomplete</td><td colspan="2">This test is not yet completed.</td></tr>
</table>

8.3.7. Recovery Area Identification Testing - PT_04.2

<table>
<tr><td rowspan="2">Test ID:
PT_04.2
Test Name:
Recovery Area Identification Testing
Related Requirements:
PR_2.8
Related Test IDs:
PT_04.1</td><td>Objective and Tested Variables: Verify the performance of the UAV's vision-guided descent algorithm. The test variable is the UAV targeted landing accuracy and landing velocity.</td><td>Reason for Test: This test is essential for verifying the performance of the vision-guided descent algorithm.</td></tr>
<tr><td colspan="2">Success Criteria: This test is considered successful if the simulated UAV lands softly within 1' of the center of the lunar ice recovery area. A "soft" landing is defined as a maximum landing velocity of 2 ft/s.</td></tr>
<tr><td>Methodology</td><td colspan="2">This test closely mirrors PT_04.1, in which the vision-guided descent algorithm is tested in a software environment. This test should not be attempted until PT_04.1 has been successfully completed. This test should be completed in an open area where it is safe to fly a UAV to heights up to 100'.
1. Construct a lunar ice sample recovery area. This recovery area should be 10'x10' with a target circle of diameter 3' at the recovery area's center.
2. Fly the UAV approximately 100' above the recovery area, offset approximately 10' from the recovery area's center.
3. Run the UAV's vision-guided descent algorithm. Record the velocity of the UAV throughout its descent and take note of the quality of the descent. Is the control system tuned for accurate, efficient performance?
4. If the UAV's landing velocity and landing position meet the given success criteria, repeat the procedure at least 3 additional times, varying the initial lateral position offset from the recovery area up to 25'. Take note of how the vision-guided descent algorithm reacts to different amounts of initial offset.</td></tr>
<tr><td>Possible Impacts</td><td colspan="2">Results from this test directly inform the development of the landing algorithm and its implementation on the UAV. This test informs the tuning of the PID controller for guiding the vehicle's landing.</td></tr>
<tr><td>Results

Status: Incomplete</td><td colspan="2">This test is not yet completed.</td></tr>
</table>

8.3.8. RF Comms Test - PT_05.1

<table>
<tr><td rowspan="2">Test ID:
PT_05.1
Test Name:
RF Comms Test
Related Requirements:
PR_2.9
Related Test IDs:
N/A</td><td>Objective and Tested Variables: Verify the range of the wireless communication systems for sending data between the GCS and UAV. The test variable is the wireless data link range.</td><td>Reason for Test: This test ensures that proper communication between the GCS and the payload system can be achieved over any range that could reasonably be expected during the mission.</td></tr>
<tr><td colspan="2">Success Criteria: This test is successful if all data links demonstrate a maximum point-to-point range of at least 1 mile.</td></tr>
<tr><td>Methodology</td><td colspan="2">The UAS has three separate data links that each must complete this test. These data links include the 900 MHz XBee radios, the 915 MHz Holybro telemetry transmitters, and the 2.4 GHz RC transmitter. The following procedure shall be completed for each of these data links to ensure healthy wireless connections can be established. Note: The details of the procedure outlined below is slightly different for each data link. Consult their documentation for details specific to each product.
1. Establish the “local” end of the test setup. This includes the RF receiver that is stationary throughout the test. In the mission, this receiver is located in the GCS. The receiver should be placed 2-3’ above the ground, connected to a device that can record RSSI and packet loss. Power the receiver on.
2. Utilizing a GPS-enabled phone (or other such device), walk the “remote” RF receiver .25 miles away from the local receiver.
3. Power the remote receiver on. Measure the amount of time it takes for a connection to be established between the two devices.
4. Record the RSSI and packet loss reported by the local RF receiver.
5. Power the remote receiver off.
6. Repeat steps 2-5 in increments of .25 miles up to 1 mile.</td></tr>
<tr><td>Possible Impacts</td><td colspan="2">Successful completion of this test confirms that the RF systems of the payload have the range necessary to cover all potential distances seen on launch day. A failure of this test could lead to a component-level redesign of the system, such as adding a higher power antenna to increase range.</td></tr>
<tr><td>Results

Status: Incomplete</td><td colspan="2">This test is not yet completed.</td></tr>
</table>

8.3.9. X-Wing Structural Test - PT_06.1

<table>
<tr><td rowspan="2">Test ID:
PT_06.1
Test Name:
X-Wing Structural Test
Related Requirements:
PR_2.10
Related Test IDs:
N/A</td><td>Objective and Tested Variables: Verify that the UAV airframe structure can withstand reasonably expected flight loads. The test variable is the structural integrity of the UAV airframe.</td><td>Reason for Test: This test is necessary to validate the physical design of the UAV. The maximum theoretical force the UAV could experience in flight would be that which transpires by an assertion of 100% throttle by the FCC. This test ensures that the airframe structure is capable of handling this force and yields data that can be utilized in further FEA analysis.</td></tr>
<tr><td colspan="2">Success Criteria: This test is considered successful if the UAV withstands no structural damage during the test.</td></tr>
<tr><td>Methodology</td><td colspan="2">This test is designed to push the UAV to its maximum reasonably expected flight loads. As such, precautions must be taken to ensure the safety of those involved with the test. The test should be conducted in an open area and all test personnel must stand at least 15’ away from the UAV while the vehicle is powered on and armed. A load cell is used in this test to quantify the maximum force the UAV can exert. This data can be used in further FEA analysis of the UAV airframe.
1. Tether the UAV to the load cell. The tether should be about 2’ long and attached to the underside of the UAV, near the center of its yaw axis. The load cell should be firmly attached to a flat, rigid surface.
2. Check the UAV to ensure the airframe is in its nominal flight configuration and the battery is at 100% charge.
3. With an RC controller, bring the UAV to a hover, slowly letting the slack out of the tether.
4. Once the tether is taut, slowly push the throttle up to 100%. Hold the throttle here for 10 seconds before bringing the UAV back to a hover.
5. Land the UAV and disarm it.
6. Inspect the UAV for any signs of structural failure. Pay careful attention to the x-wing mechanism and the four armatures.
7. If no damage is found, repeat steps 2-6 two additional times.</td></tr>
<tr><td>Possible Impacts</td><td colspan="2">Successful completion of this test indicates that the UAV’s airframe can withstand reasonably expected flight loads. If this test fails, the UAV’s airframe may need to be reinforced for additional strength or the total amount of thrust produced by the motors may need to be limited in the UAV’s software.</td></tr>
<tr><td>Results
Status: In Progress</td><td colspan="2">This test is not yet completed.</td></tr>
</table>

8.3.10. Battery Drain and Power Testing - PT_07.1

Test ID: PT_07.1 **Test Name:** Battery Drain and Power Testing **Related Requirements:** PR_2.11 **Related Test IDs:** N/A	**Objective and Tested Variables:** Ensure the batteries used to power the GCS and R&D systems are capable of powering each system for a duration exceeding that of the entirety of the mission. The test variables are the GCS and R&D respective system on-times and battery cell voltages during power consumption.	**Reason for Test:** This test is necessary to ensure that all electrical systems on both the R&D and GCS systems have enough battery to operate throughout the entire mission timeline. This test will not only look at the energy capacity of the batteries, but also how the supplied voltage varies. The voltage difference between a fully charged and an empty LiPo battery is about 30% and it is important to make sure that all electronics and electrical actuators can operate at a reduced voltage.
	Success Criteria: This test is successful if the GCS and R&D electronics are both powered by battery for a period of time exceeding 4 hours.	
Methodology	The batteries will be tested both on their own and with their respective systems. Batteries will be drained from full to empty and their individual cell batteries will be monitored to study how supply voltage vary along the mission timeline. For the R&D system, the battery will supply idle power for 4 hours. After 4 hours, the R&D will initiate its high-power deployment until the battery has drained. The GCS employs its own battery monitoring system which will be used to measure power consumption as the battery is drained. 1. Connect the fully charged LiPo battery to its respective load or SUT and record its initial voltage values. 2. Drain the battery until the battery has reached 85% of its nominal voltage or until it has been under load for 4 hours and record cell voltages. 3. For R&D, enable deployment equipment and leave on until the battery has been completely drained.	
Possible Impacts	Successful completion of this test verifies that the GCS and R&D power systems can power their respective systems well beyond the expected duration of the mission. Failure of this test indicates the need for a redesign of the power system, likely a larger capacity battery.	
Results **Status:** *In Progress*	This test is not yet completed.	

8.4. Derived Requirements & Verification Plans

8.4.1. NASA-Derived R&VP

The following list of requirements are derived by the NASA Student Launch team and need to be completed and followed in order to ensure that the PSP-SL team is constantly meeting the safety standards of high powered flight and completing the goals set by the Student Launch team. The PSP-SL team has looked at all requirements and broken each into several sections to ensure that the team is meeting / exceeding each NASA derived requirement. The sections are as follows: General Requirements & Verification Plans (R&VP), Vehicle R&VP, Avionics & Recovery R&VP, Payload R&VP, and Safety R&VP. Each of the above sections will be broken into individual requirements and will be verified via one of 4 ways: Inspection, Demonstration, Test, and / or Analysis.

1. **Inspection:** Thorough examination of the system or process using physical manipulation, measurements, or observation via the five human senses.
2. **Demonstration:** Manipulation of a given system or process intended to verify that the results match the expected results.
3. **Test:** Verification of a given system or process through the use of a procedure, given relevant inputs, to ensure that the system or process meets and exceeds expected results specified in the requirements.
4. **Analysis:** Verification of a given system or process through utilizing calculations, academic theory, and system models.

Each of the following requirements, both team derived and NASA derived, has been given a unique ID number which references the requirements from the NASA Student Launch Handbook, written by the NASA Student Launch team.

Additionally, the PSP-SL team has generated a table format which organizes R&VP for each requirement. Each table lists requirement ID, the requirement's description, the team's verification plan for the requirement, additional comments for the requirement (if applicable), the ID for the test which verifies the requirement (if applicable), the status of the requirement (complete or incomplete), and one of four colors which indicates the type of verification. These colors are shown along the left side of each table, and multiple colors means multiple methods of verification. For requirements which are not applicable to the PSP-SL team, these colors were left as white, as was the coloration of the verification status box. A guide to which color corresponds to each verification method follows:

Inspection	Demonstration
Analysis	Test

Table 8.10: Verification Method Color Key

8.4.1.1. General R&VP

D	**Requirement ID:** 1.1 **Description:** Students on the team will do 100% of the project, including design, construction, written reports, presentations, and flight preparation with the exception of assembling the motors and handling black powder or any variant of ejection charges, or preparing and installing electric matches (to be done by the team's mentor). Teams will submit new work. Excessive use of past work will merit penalties.	**Verification Plan:** PSP-SL members will demonstrate the new work they have completed by submitting milestone documents and will demonstrate the understanding they have gained by doing the work themselves during PowerPoint presentations. **Comments:** Not Applicable (N/A)
	Status: *In Progress*	**Verification Test ID:** N/A

D	**Requirement ID:** 1.2 **Description:** The team will provide and maintain a project plan to include, but not limited to the following items: project milestones, budget and community support, checklists, personnel assignments, Science, Technology, Engineering and Math (STEM) engagement events, and risks and mitigations.	**Verification Plan:** Project plan completion will be demonstrated by turning in the milestone reports which contain it. **Comments:** All of this is covered in both section seven (7) and after all team & NASA derived requirements and verification
	Status: *In Progress*	**Verification Test ID:** N/A

D	**Requirement ID:** 1.3 **Description:** Foreign National (FN) team members must be identified by the Preliminary Design Review (PDR) and may or may not have access to certain activities during launch week due to security restrictions. In addition, FN's may be separated from their team during certain activities on site at Marshall Space Flight Center.	**Verification Plan:** Foreign national team members' contact information will be demonstrated when it is submitted alongside PDR documentation. **Comments:** All PSP-SL FN have been submitted to the board for approval.
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** 1.4 **Description:** The team must identify all team members attending launch week activities by the Critical Design Review (CDR). Team members will include: • Students actively engaged in the project throughout the entire year. • One mentor (see requirement 1.13).	**Verification Plan:** PSP-SL member contact information will be demonstrated when it is submitted alongside CDR documentation.

	• No more than two adult educators.	**Comments:** The team has submitted our launch week team member list concurrently with all CDR documents.
Status: *Complete*		**Verification Test ID:** N/A

D	**Requirement ID:** 1.5 **Description:** The team will engage a minimum of 200 participants in educational, hands-on Science, Technology, Engineering, and Mathematics (STEM) activities, as defined in the STEM Engagement Activity Report, by FRR. To satisfy this requirement, all events must occur between project acceptance and the FRR due date and the STEM Engagement Activity Report must be submitted via email within two weeks of the completion of the event.	**Verification Plan:** STEM Engagement Activity Reports will be completed and demonstrated to the NASA Student Launch team via email throughout the course of the project. Activity Reports will be submitted within a week of each event occurring so documentation can be written accurately.
		Comments: As of CDR the team has engaged with 1000+ students and is continually planning more events.
Status: *Complete*		**Verification Test ID:** N/A

D	**Requirement ID:** 1.6 **Description:** The team will establish a social media presence to inform the public about team activities.	**Verification Plan:** PSP-SL's social media information will be publicly available online and links to each social media outlet of the team will be provided to the NASA Student Launch team.
		Comments: The team has a social media presence on Twitter, Facebook, Instagram, and a personal website.
Status: *Complete*		**Verification Test ID:** N/A

D	**Requirement ID:** 1.7 **Description:** Teams will email all deliverables to the NASA project management team by the deadline specified in the handbook for each milestone. In the event that a deliverable is too large to attach to an email, inclusion of a link to download the file will be sufficient.	**Verification Plan:** All deliverables will be demonstrated to the NASA project management via email by the deadlines listed in the Project Plan section. All milestone deliverables will be completed at least a week in advance so they can be reviewed and submitted on time.
		Comments: The team ensures that all deliverables are submitted the day before the deadlines.

	Status: *In Progress*	**Verification Test ID:** N/A

D	**Requirement ID:** 1.8 **Description:** All deliverables must be in PDF format.	**Verification Plan:** The altitude will be determined through simulation with open launch vehicle.
		Comments: N/A
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** 1.9 **Description:** In every report, teams will provide a table of contents including major sections and their respective sub-sections.	**Verification Plan:** Tables of contents will be included at the beginning of each milestone report and will be included when the report is submitted to the NASA Student Launch Team.
		Comments: N/A
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** 1.10 **Description:** In every report, the team will include the page number at the bottom of the page.	**Verification Plan:** Each milestone report submitted to the NASA Student Launch team will have a page number visible at the bottom-right corner of each page.
		Comments: N/A
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** 1.11 **Description:** The team will provide any computer equipment necessary to perform a video teleconference with the review panel. This includes, but is not limited to, a computer system, video camera, speaker telephone, and a sufficient Internet connection. Cellular phones should be used for speakerphone capability only as a last resort.	**Verification Plan:** PSP-SL will demonstrate its capability to conduct video teleconferences by participating in all milestone presentation conferences with the NASA Student Launch team.
		Comments: N/A
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** 1.12 **Description:**	**Verification Plan:** PSP-SL will demonstrate its full scale vehicle's compatibility with the launch pads

	All teams will be required to use the launch pads provided by Student Launch's launch services provider. No custom pads will be permitted on the launch field. At launch, 8' 1010 rails and 12' 1515 rails will be provided. The launch rails will be canted 5 to 10 degrees away from the crowd on launch day. The exact cant will depend on launch day wind conditions.	provided by the launch services provider by launching its full scale vehicle demonstration flight on a 12' 1515 launch rail. **Comments:** The team has chosen to use a 12' 1515 launch rail.
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** 1.13 **Description:** Each team must identify a "mentor." A mentor is defined as an adult who is included as a team member, who will be supporting the team (or multiple teams) throughout the project year, and may or may not be affiliated with the school, institution, or organization. The mentor must maintain a current certification, and be in good standing, through the National Association of Rocketry (NAR) or Tripoli Rocketry Association (TRA) for the motor impulse of the launch vehicle and must have flown and successfully recovered (using electronic, staged recovery) a minimum of 2 flights in this or a higher impulse class, prior to PDR. The mentor is designated as the individual owner of the launch vehicle for liability purposes and must travel with the team to launch week. One travel stipend will be provided per mentor regardless of the number of teams he or she supports. The stipend will only be provided if the team passes FRR and the team and mentor attend launch week in April.	**Verification Plan:** Information on PSP-SL's mentor will be provided within project milestone documentation, which will be supplied to the NASA Student Launch team via email. This information will include the mentor's affiliation with NAR and TRA and the mentor's number of level 1, 2, and 3, high-power rocketry flights, as defined by NAR. **Comments:** N/A
	Status: *Complete*	**Verification Test ID:** N/A

8.4.1.2. Vehicle R&VP

A	**Requirement ID:** 2.1 **Description:** The vehicle will deliver the payload to an apogee altitude between 3,500' and 5,500' above ground level (AGL). Teams flying below 3,000' or above 6,000' on Launch Day will be disqualified and receive zero altitude points towards their overall project score.	**Verification Plan:** The altitude will be determined through simulation with open launch vehicle. **Comments:** This will also be verified through the use of RASAero and our avionics trajectory MATLAB code
	Status: *In Progress*	**Verification Test ID:** N/A

A	**Requirement ID:** 2.2 **Description:**	**Verification Plan:** The team will use open rocket calculations to predict the target altitude.

	Teams shall identify their target altitude goal at the PDR milestone. The declared target altitude will be used to determine the team's altitude score during Launch Week.	**Comments:** This will also be verified through the use of RASAero and our avionics trajectory MATLAB code
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 2.3 **Description:** The vehicle will carry one commercially available, barometric altimeter for recording the official altitude used in determining the Altitude Award winner. The Altitude Award will be given to the team with the smallest difference between their measured apogee and their official target altitude on launch day. This altimeter may also be used for deployment purposes (see Requirement 3.4)	**Verification Plan:** The existence of altimeter will be verified through inspection. **Comments:** A redundant commercially available altimeter will be used to verify launch vehicle apogee
	Status: *Complete*	**Verification Test ID:** N/A

T	**Requirement ID:** 2.4 **Description:** The launch vehicle will be designed to be recoverable and reusable. Reusable is defined as being able to launch again on the same day without repairs or modifications.	**Verification Plan:** The team will reuse the launch vehicle in subscale and full scale test flights. **Comments:** N/A
	Status: *In Progress*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.5 **Description:** The launch vehicle will have a maximum of four (4) independent sections. An independent section is defined as a section that is either tethered to the main vehicle or is recovered separately from the main vehicle using its own parachute.	**Verification Plan:** The team will demonstrate that there are no more than 4 independent sections. **Comments:** The teams current design consists of three (3) tethered sections.
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 2.5.1 **Description:** Coupler/airframe shoulders which are located at in-flight separation points will be at least 1 body diameter in length.	**Verification Plan:** The team will measure and make sure the coupler/airframe shoulders that are located at in-flight separation points are at least 1 body diameter in length. **Comments:** The avionics bay is our only section which is located on an in-flight separation point and the upper and lower airframe overlay with the coupler 6" (1 diameter)

Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 2.5.2 **Description:** Nosecone shoulders which are located at in-flight separation points will be at least ½ body diameter in length.	**Verification Plan:** The team will measure and make sure the nosecone shoulders that are located at in-flight separation points are at least 1 body diameter in length. **Comments:** Our current design does not have an in-flight separation point at the nosecone shoulder.
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.6 **Description:** The launch vehicle will be capable of being prepared for flight at the launch site within 2 hours of the time the Federal Aviation Administration flight waiver opens.	**Verification Plan:** The preparation time of the launch vehicle will be verified through testing at both full scale launch vehicle and payload demonstration flights. **Comments:** The team has made it a priority to prepare the night prior to ALL launch days.
	Status: *Incomplete*	**Verification Test ID:** N/A

T	**Requirement ID:** 2.7 **Description:** The launch vehicle and payload will be capable of remaining in launch-ready configuration on the pad for a minimum of 2 hours without losing the functionality of any critical on-board components, although the capability to withstand longer delays is highly encouraged.	**Verification Plan:** Ability of the vehicle and payload to remain in launch-ready configuration will be verified by ground testing. **Comments:** N/A
	Status: *Incomplete*	**Verification Test ID:** A_04 AND PT_07.1

D	**Requirement ID:** 2.8 **Description:** The launch vehicle will be capable of being launched by a standard 12-volt direct current firing system. The firing system will be provided by the NASA-designated launch services provider.	**Verification Plan:** The team will use a standard 12-volt direct current firing system during its full-scale test launches to ensure successful integration with such a system can be achieved. **Comments:** This will be tested at both the full scale and payload demonstration flights

Status: *Incomplete*	Verification Test ID: N/A

D	**Requirement ID:** 2.9 **Description:** The launch vehicle will require no external circuitry or special ground support equipment to initiate launch (other than what is provided by the launch services provider).	**Verification Plan:** Ability to launch without external circuitry or special ground support will be achieved by electronic testing and subscale flight(s). **Comments:** N/A
	Status: *Complete*	**Verification Test ID:** N/A

I D	**Requirement ID:** 2.10 **Description:** The launch vehicle will use a commercially available solid motor propulsion system using ammonium perchlorate composite propellant (APCP) which is approved and certified by the National Association of Rocketry (NAR), Tripoli RocketryAssociation (TRA), and/or the Canadian Association of Rocketry (CAR).	**Verification Plan:** Use of required solid motor propulsion system will be achieved by inspection and demonstration. **Comments:** The team has decided, as of CDR, to use the CTI L1115 4 grain SRM. This motor is compliant with NAR, TRA, and CAR. More discussion can be found in Section 5.3.
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.10.1 **Description:** Final motor choice will be declared by the Critical Design Review (CDR) milestone.	**Verification Plan:** The team will demonstrate that the team has the final motor choice by showing it in CDR. **Comments:** The team has decided, as of CDR, to use the CTI L1115 4 grain SRM. This motor is compliant with NAR, TRA, and CAR. More discussion can be found in Section 5.3.
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.10.2 **Description:** Any motor change after CDR must be approved by the NASA Range Safety Officer (RSO) and will only be approved if the change is for the sole purpose of increasing the safety margin. A penalty against the team's overall score will be incurred when a motor change is made after the CDR milestone, regardless of the reason.	**Verification Plan:** In the case the team must use another motor, the team will document the motor change after approval from the NASA RSO. **Comments:** The team currently has all three (3) motors on hand and does not anticipate any need to change.
	Status: *Incomplete*	**Verification Test ID:** N/A

I D	**Requirement ID:** 2.11 **Description:** The launch vehicle will be limited to a single stage.	**Verification Plan:** Limitation of a single stage in launch vehicle will be verified by inspection. **Comments:** N/A
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 2.12 **Description:** The total impulse provided by a College or University launch vehicle will not exceed 5,120 Newton-seconds (L-class). The total impulse provided by a High School or Middle School launch vehicle will not exceed 2,560 Newton-seconds (K-class).	**Verification Plan:** Total impulse will be verified by consulting specifications provided by the manufacturer of the solid rocket motor. **Comments:** The chosen motor, the CTI L1115 4 grain, meets this requirement with a total impulse of 5015 N-s.
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 2.13 **Description:** Pressure vessels on the vehicle will be approved by the RSO and will meet requirements 2.13.1, 2.13.2, and 2.13.3	**Verification Plan:** See respective verifications for 2.13.1, 2.13.2, and 2.13.3. The vehicle will be presented to the RSO for hands on inspection. **Comments:** N/A
	Status: *In Progress*	**Verification Test ID:** N/A

T A	**Requirement ID:** 2.13.1 **Description:** The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) will be 4:1 with supporting design documentation included in all milestone reviews.	**Verification Plan:** Use of minimum 4:1 safety factor will be verified through inspection (safety factor will appear in the calculation to determine max expected altitude. **Comments:** FEA and an avionics ejection test will be conducted to provide verification for this requirement.
	Status: *Incomplete*	**Verification Test ID:** A_03

	Requirement ID: 2.13.2 **Description:** Each pressure vessel will include a pressure relief valve that sees the full pressure of the tank and is capable of withstanding the maximum pressure and flow rate of the tank.	**Verification Plan:** N/A **Comments:** There is no pressure tank in the current design of the vehicle, therefore this requirement does not apply.

Status: N/A	**Verification Test ID:** N/A

	Requirement ID: 2.13.3 **Description:** The full pedigree of the tank will be described, including the application for which the tank was designed and the history of the tank. This will include the number of pressure cycles put on the tank, the dates of pressurization/depressurization, and the name of the person or entity administering each pressure event.	**Verification Plan:** N/A Comments: There is no pressure tank in the current design of the vehicle, therefore this requirement does not apply.
	Status: N/A	**Verification Test ID:** N/A

I	**Requirement ID:** 2.14 **Description:** The launch vehicle will have a minimum static stability margin of 2.0 at the point of rail exit. Rail exit is defined at the point where the forward rail button loses contact with the rail.	**Verification Plan:** The minimum static stability will be verified by measuring the axial distance between the center of gravity and the center of pressure of the vehicle. **Comments:** The team will also use verification off of the rail stability through the use of OpenRocket, RASAero, and an in-house stability code.
	Status: *In Progress*	**Verification Test ID:** N/A

A	**Requirement ID:** 2.15 **Description:** Any structural protuberance on the launch vehicle will be located aft of the burnout center of gravity.	**Verification Plan:** The team will verify this during launch vehicle design **Comments:** The team's launch vehicle does not have any structural protuberances.
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.16 **Description:** The launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit.	**Verification Plan:** The minimum velocity will be verified through OpenRocket simulations and the vehicle demonstration test flight. **Comments:** The team will use verification rail exit velocity through the use of OpenRocket, RASAero, and an in-house trajectory code.

Status: *Incomplete*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.17 **Description:** All teams will successfully launch and recover a subscale model of their launch vehicle prior to CDR. Subscales are not required to be high power launch vehicles.	**Verification Plan:** The team will demonstrate that the team has completed a launch successfully by showing proof in CDR. **Comments:** The team completed a successful subscale launch on 11/24/2019.
	Status: *Complete*	**Verification Test ID:** N/A

A	**Requirement ID:** 2.17.1 **Description:** The subscale model should resemble and perform as similarly as possible to the full-scale model, however, the full-scale will not be used as the subscale model.	**Verification Plan:** The team will scale the full size model down and use extra weight to resemble the full-scale model. **Comments:** The team completed a successful subscale launch on 11/24/2019. This vehicle closely resembled the teams 2020 full scale launch vehicle
	Status: Complete	**Verification Test ID:** N/A

I	**Requirement ID:** 2.17.2 **Description:** The subscale model will carry an altimeter capable of recording the model's apogee altitude.	**Verification Plan:** The team will inspect and make sure a capable altimeter is installed. **Comments:** The team used a MissileWorks RRC3+ Sport altimeter as a primary source for collecting telemetry. The team also used a redundant altimeter, JollyLogic AltimeterOne, to verify the apogee.
	Status: Complete	**Verification Test ID:** N/A

D	**Requirement ID:** 2.17.3 **Description:** The subscale launch vehicle must be a newly constructed launch vehicle, designed and built specifically for this year's project.	**Verification Plan:** The team will demonstrate and make sure it is newly built. **Comments:** The subscale launch vehicle resembles the 2020 PSP-SL full scale

		launch vehicle and was constructed in the Fall 2019.
Status: Complete		**Verification Test ID:** N/A

D	**Requirement ID:** 2.17.4 **Description:** Proof of a successful flight shall be supplied in the CDR report. Altimeter data output may be used to meet this requirement.	**Verification Plan:** The team will demonstrate that the team has made a successful flight. **Comments:** The team has attached our results from our subscale flight in Section 3.2.
Status: Complete		**Verification Test ID:** N/A

D	**Requirement ID:** 2.18.1 **Description:** Vehicle Demonstration Flight - All teams will successfully launch and recover their full-scale launch vehicle prior to FRR in its final flight configuration. The launch vehicle flown must be the same launch vehicle to be flown on launch day.	**Verification Plan:** The team will demonstrate and make sure the launch vehicle meets the requirements. **Comments:** Our launch vehicle is currently under construction and the team plans to be ready to launch anywhere from the last week in January through February.
Status: *Incomplete*		**Verification Test ID:** N/A

D	**Requirement ID:** 2.18.1.1 **Description:** The vehicle and recovery system will have functioned as designed.	**Verification Plan:** The team will verify the functions of vehicle and recovery system through sub scale and full scale test flight demonstration. **Comments:** The team will conduct the following tests in the coming weeks, prior to demonstration flights.
Status: *In Progress*		**Verification Test ID:** A_01, A_02, A_03, A_41, A_05

D	**Requirement ID:** 2.18.1.2 **Description:** The full-scale launch vehicle must be a newly constructed launch vehicle, designed and built specifically for this year's project.	**Verification Plan:** The team will demonstrate and make sure the launch vehicle is newly designed and built. **Comments:** The full scale launch vehicle is currently under construction and mirrors the team's 2020 design.

	Status: *In Progress*	**Verification Test ID:** N/A

	Requirement ID: 2.18.1.3 **Description:** The payload does not have to be flown during the full-scale Vehicle Demonstration Flight.	**Verification Plan:** N/A **Comments:** The team's current design requires the payload Retention & Deployment system, regardless if the drone is being flown.
	Status: N/A	**Verification Test ID:** N/A

I	**Requirement ID:** 2.18.1.3.1 **Description:** If the payload is not flown, mass simulators will be used to simulate the payload mass.	**Verification Plan:** The team will inspect to make sure the mass simulator is used, in the case that the payload is not configured properly to be inserted to the vehicle. **Comments:** The team's current design requires the payload Retention & Deployment system, regardless if the drone is being flown.
	Status: *Incomplete*	**Verification Test ID:** N/A

A	**Requirement ID:** 2.18.1.3.2 **Description:** The mass simulators will be located in the same approximate location on the launch vehicle as the missing payload mass.	**Verification Plan:** The team will analyze and verify that they are in the same approximate locations. **Comments:** The team will use the overall payload weight and system center of gravity as verification for the location for the mass simulator, in the case that mass simulators are used.
	Status: *Incomplete*	**Verification Test ID:** N/A

	Requirement ID: 2.18.1.4 **Description:** If the payload changes the external surfaces of the launch vehicle (such as with camera housings or external probes) or manages the total energy of the vehicle, those systems will be active during the full-scale Vehicle Demonstration Flight.	**Verification Plan:** The team will inspect and make sure the systems are active during demonstration flight. **Comments:** The team's 2020 design currently does not have any external surfaces.
	Status: N/A	**Verification Test ID:** N/A

I	**Requirement ID:** 2.18.1.5 **Description:** Teams shall fly the launch day motor for the Vehicle Demonstration Flight. The team may request a waiver for the use of an alternative motor in advance if the home launch field cannot support the full impulse of the launch day motor or in other extenuating circumstances (such as weather).	**Verification Plan:** The team will make sure the motor is the launch day motor by inspecting the motor. **Comments:** The team currently has all three (3) motors on hand and does not anticipate any need to change.
	Status: *Incomplete*	**Verification Test ID:** N/A

A	**Requirement ID:** 2.18.1.6 **Description:** The vehicle must be flown in its fully ballasted configuration during the full-scale test flight. Fully ballasted refers to the same amount of ballast that will be flown during the launch day flight. Additional ballast may not be added without a re-flight of the full-scale launch vehicle.	**Verification Plan:** The team will do calculations and analyze data and make sure the vehicle is flown in its fully ballasted configuration. **Comments:** As of CDR, the team does not anticipate being under the predicted mass and therefore will not fly with any ballast.
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 2.18.1.7 **Description:** After successfully completing the full-scale demonstration flight, the launch vehicle or any of its components will not be modified without the concurrence of the NASA Range Safety Officer (RSO).	**Verification Plan:** The team will inspect and make sure nothing is being modified regularly. **Comments:** The team is in progress of constructing and launching our full scale launch vehicle.
	Status: *In Progress*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.18.1.8 **Description:** Proof of a successful flight shall be supplied in the FRR report. Altimeter data output is required to meet this requirement.	**Verification Plan:** The team will demonstrate that a successful flight occurred in the FRR report. **Comments:** AltusMetrum Telemetrum, MissileWorks, and JollyLogic AltimeterOne will all be flown to ensure that an apogee is successfully collected.
	Status: *Incomplete*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.18.1.9 **Description:** Vehicle Demonstration flights must be completed by the FRR submission deadline. No exceptions will be made. If the Student Launch office determines that a Vehicle Demonstration Re-flight is necessary, then an extension may be granted. THIS EXTENSION IS ONLY VALID FOR RE-FLIGHTS, NOT FIRST TIME FLIGHTS. Teams completing a required re-flight must submit an FRR Addendum by the FRR Addendum deadline.	**Verification Plan:** The team will demonstrate that a successful flight has occurred. **Comments:** The team is on track to launch the launch vehicle by the end of January and February will be a backup incase of delays.
	Status: *Incomplete*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.18.2 **Description:** Payload Demonstration Flight - All teams will successfully launch and recover their full-scale launch vehicle containing the completed payload prior to the Payload Demonstration Flight deadline. The launch vehicle flown must be the same launch vehicle to be flown on launch day. The purpose of the Payload Demonstration Flight is to prove the launch vehicle's ability to safely retain the constructed payload during flight and to show that all aspects of the payload perform as designed. A successful flight is defined as a launch in which the launch vehicle experiences stable ascent and the payload is fully retained until it is deployed (if applicable) as designed.	**Verification Plan:** The team will demonstrate that the vehicle can be successfully flown with the completed payload. **Comments:** The team is currently planning on flying an inactive drone, but a fully integrated payload system during the vehicle demonstration flight. If the team is required to fly again, a third motor is already on hand. The retention and deployment system will be fully active for both demonstration flights. However the drone's software/hardware will still be in the process of being completed.
	Status: *Incomplete*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.18.2.1 **Description:** The payload must be fully retained until the intended point of deployment (if applicable), all retention mechanisms must function as designed, and the retention mechanism must not sustain damage requiring repair.	**Verification Plan:** The team will demonstrate this through a successful payload demonstration flight. **Comments:** The team is currently planning on flying an inactive drone, but a fully integrated payload system during the vehicle demonstration flight. If the team is required to fly again, a third motor is already on hand. The retention and deployment system will be fully active for both demonstration flights. However the drone's software/hardware will still be in the process of being completed.
	Status: *Incomplete*	**Verification Test ID:** N/A

I	**Requirement ID:** 2.18.2.2 **Description:** The payload flown must be the final, active version.	**Verification Plan:** The team will verify that the payload flown is the final version by inspecting the payload. **Comments:** The team is currently planning on flying an inactive drone, but a fully integrated payload system during the vehicle demonstration flight. If the team is required to fly again, a third motor is already on hand. The retention and deployment system will be fully active for both demonstration flights. However the drone's software/hardware will still be in the process of being completed.
	Status: *Incomplete*	**Verification Test ID:** N/A

	Requirement ID: 2.18.2.3 **Description:** If requirements 2.18.2.1 and 2.18.2.2 are met during the original Vehicle Demonstration Flight, occurring prior to the FRR deadline and the information is included in the FRR package, the additional flight and FRR Addendum are not required.	**Verification Plan:** N/A **Comments:** N/A
	Status: N/A	**Verification Test ID:** N/A

D	**Requirement ID:** 2.18.2.4 **Description:** Payload Demonstration Flights must be completed by the FRR Addendum deadline. NO EXTENSIONS WILL BE GRANTED.	**Verification Plan:** The team will show that the payload demonstration flight has been completed by showing proof in FRR. **Comments:** N/A
	Status: *Incomplete*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.19 **Description:** An FRR Addendum will be required for any team completing a Payload Demonstration Flight or NASA required Vehicle Demonstration Re-flight after the submission of the FRR Report.	**Verification Plan:** The team will submit an FRR Addendum for re-flight after submitting FRR report. **Comments:** N/A
	Status: *Incomplete*	**Verification Test ID:** N/A

	Requirement ID: 2.19.1 **Description:** Teams required to complete a Vehicle Demonstration Re-Flight and failing to submit the FRR Addendum by the deadline will not be permitted to fly the vehicle at launch week.	**Verification Plan:** N/A **Comments:** N/A
	Status: N/A	**Verification Test ID:** N/A

	Requirement ID: 2.19.2 **Description:** Teams who successfully complete a Vehicle Demonstration Flight but fail to qualify the payload by satisfactorily completing the Payload Demonstration Flight requirement will not be permitted to fly the payload at launch week.	**Verification Plan:** **Comments:** N/A
	Status: N/A	**Verification Test ID:** N/A

	Requirement ID: 2.19.3 **Description:** Teams who complete a Payload Demonstration Flight which is not fully successful may petition the NASA RSO for permission to fly the payload at launch week. Permission will not be granted if the RSO or the Review Panel have any safety concerns.	**Verification Plan:** N/A **Comments:** N/A
	Status: N/A	**Verification Test ID:** N/A

I	**Requirement ID:** 2.20 **Description:** The team's name and launch day contact information shall be in or on the launch vehicle airframe as well as in or on any section of the vehicle that separates during flight and is not tethered to the main airframe. This information shall be included in a manner that allows the information to be retrieved without the need to open or separate the vehicle.	**Verification Plan:** The team will inspect and make sure all required information exist on parts. **Comments:** The team plans on attaching a note on each section with the team contact information and name.
	Status: *Incomplete*	**Verification Test ID:** N/A

I	**Requirement ID:** 2.21 **Description:** All Lithium Polymer batteries will be sufficiently protected from impact with the ground and will be brightly colored, clearly marked as a fire hazard, and easily distinguishable from other payload hardware.	**Verification Plan:** The team will inspect the placement and visuals of lithium polymer batteries to ensure this requirement is met. **Comments:** The team will wrap all LiPo batteries in bright orange duck

		tape to ensure that the batteries are clearly marked.
	Status: *In Progress*	**Verification Test ID:** N/A

I D	**Requirement ID:** 2.22.1 **Description:** The launch vehicle will not utilize forward canards. Camera housings will be exempted, provided the team can show that the housing(s) causes minimal aerodynamic effect on the launch vehicle's stability.	**Verification Plan:** Absence of forward canards will be verified through inspection and demonstration. **Comments:** The team's 2020 design does not and will not use any canards or external camera housings.
	Status: *Complete*	**Verification Test ID:** N/A

I D	**Requirement ID:** 2.22.2 **Description:** The launch vehicle will not utilize forward firing motors.	**Verification Plan:** Absence of forward firing motors will be verified through inspection and demonstration. **Comments:** The team's 2020 design does not and will not use any forward firing motors.
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** 2.22.3 **Description:** The launch vehicle will not utilize motors that expel titanium sponges (Sparky, Skidmark, Metal-Storm, etc.)	**Verification Plan:** Absence of prohibited motors will be verified through demonstration at the vehicle demonstration flight. **Comments:** The team's 2020 design does not and will not use any motors that expel titanium sponges.
	Status: *Complete*	**Verification Test ID:** N/A

	Requirement ID: 2.22.4 **Description:** The launch vehicle will not utilize hybrid motors.	**Verification Plan:** Absence of hybrid motors will be verified through demonstration at the vehicle demonstration flight. **Comments:** The team's 2020 design does not and will not use any hybrid motors.
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 2.22.5 **Description:** The launch vehicle will not utilize a cluster of motors.	**Verification Plan:** Absence of a cluster of motors will be verified through inspection and demonstration.

D		**Comments:** The team's 2020 design does not and will not utilize a cluster of motors.
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 2.22.6 **Description:** The launch vehicle will not utilize friction fitting for motors.	**Verification Plan:** Absence of friction fitting will be verified through inspection. **Comments:** The team's 2020 design does not and will not use any friction fitting for motors..
	Status: *Complete*	**Verification Test ID:** N/A

A	**Requirement ID:** 2.22.7 **Description:** The launch vehicle will not exceed Mach 1 at any point during flight.	**Verification Plan:** The team will use calculation and open launch vehicle simulation to verify the max speed does not exceed Mach 1. **Comments:** This is verified through OpenRocket, RASAero, and in-house trajectory MATLAB code.
	Status: *Complete*	**Verification Test ID:** N/A

A	**Requirement ID:** 2.22.8 **Description:** Vehicle ballast will not exceed 10% of the total unballasted weight of the launch vehicle as it would sit on the pad (i.e. a launch vehicle with an unballasted weight of 40lbm. on the pad may contain a maximum of 4lbm. of ballast).	**Verification Plan:** The team will verify this using calculation. **Comments:** As of CDR, the team does not anticipate being under the predicted mass and therefore will not fly with any ballast.
	Status: *Complete*	**Verification Test ID:** N/A

A	**Requirement ID:** 2.22.9 **Description:** Transmissions from onboard transmitters will not exceed 250 mW of power (per transmitter).	**Verification Plan:** The team will calculate and make sure that the transmission will not exceed 250mW. **Comments:** The team does not utilize any transmitters that exceed 250 mW.
	Status: *Complete*	**Verification Test ID:** N/A

	Requirement	Verification
T A	**Requirement ID:** 2.22.10 **Description:** Transmitters will not create excessive interference. Teams will utilize unique frequencies, hand- shake/passcode systems, or other means to mitigate interference caused to or received from other teams.	**Verification Plan:** The team will test and calculate to make sure that the transmitter does not create excessive interference. **Comments**: The team will ensure that proper transmission procedures are used during any and all transmissions.
	Status: *In Progress*	**Verification Test ID:** N/A

	Requirement	Verification
I A	**Requirement ID:** 2.22.11 **Description:** Excessive and/or dense metal will not be utilized in the construction of the vehicle. Use of lightweight metal will be permitted but limited to the amount necessary to ensure structural integrity of the airframe under the expected operating stresses.	**Verification Plan:** The team will inspect and calculate to make sure the usage of metal is within standard. **Comments:** The team will ensure that proper transmission procedures are used during any and all transmissions.
	Status: *In Progress*	**Verification Test ID:** N/A

8.4.1.3. Avionics and Recovery R&VP

	Requirement	Verification
I	**Requirement ID:** 3.1 **Description:** The launch vehicle will stage the deployment of its recovery devices, where a drogue parachute is deployed at apogee, and a main parachute is deployed at a lower altitude. Tumble or streamer recovery from apogee to main parachute deployment is also permissible, provided that kinetic energy during drogue stage descent is reasonable, as deemed by the RSO.	**Verification Plan:** Having these components will be verified via inspection. **Comments:** The team will not be utilizing tumble or streamer recovery for the 2020 SL competition.
	Status: *Complete*	**Verification Test ID:** N/A

	Requirement	Verification
T	**Requirement ID:** 3.1.1 **Description:** The main parachute shall be deployed no lower than 500'.	**Verification Plan:** The parachute will be deployed at 800' to ensure that by the time it reaches 500', it is fully open. **Comments:** This will be completed after conducting test A_02 on the 11th of January.
	Status: *In Progress*	**Verification Test ID:** A_02

D	**Requirement ID:** 3.1.2 **Description:** The apogee event may contain a delay of no more than 2 seconds.	**Verification Plan:** This will be verified through subscale and full scale launches. **Comments:** This was completed on the 24th of November as part of the subscale launch. This test will be validated again via vehicle and payload demonstration flights.
	Status: Complete	**Verification Test ID:** N/A

D	**Requirement ID:** 3.1.3 **Description:** Motor ejection is not a permissible form of primary or secondary deployment.	**Verification Plan:** The PSP-SL team will not deploy using motor ejection at all, and this will be verified through subscale and full scale launches. **Comments:** This was completed during the subscale launch on the 24th of November
	Status: Complete	**Verification Test ID:** N/A

T	**Requirement ID:** 3.2 **Description:** Each team must perform a successful ground ejection test for both the drogue and main parachutes prior to the initial subscale and full-scale launches.	**Verification Plan:** At least 6' of separation of the avionics bay from the corresponding airframe, as well as full ejection of the corresponding parachute, must be achieved from each black powder charge. **Comments**: This will be completed after conducting test A_03 around the 12th of January.
	Status: *In Progress*	**Verification Test ID:** A_03

T	**Requirement ID:** 3.3 **Description:** Each independent section of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf at landing.	**Verification Plan:** During parachute drop testing, after the main parachute is dropped from an already opened state, the calculated kinetic energy of the largest section of the launch vehicle will be calculated. **Comments:** This will be completed after conducting test A_05 around the 12th of January.
	Status: *In Progress*	**Verification Test ID:** A_05

I	**Requirement ID:** 3.4 **Description:** The recovery system will contain redundant, commercially available altimeters. The term “altimeters” includes both simple altimeters and more sophisticated flight computers.	**Verification Plan:** Having these components will be verified via inspection. **Comments:** The team will use theAltusMetrum Telemetrum, MissileWorks, and JollyLogic AltimeterOne will all be flown to ensure that an apogee is successfully

		collected. The AltimeterOne will be outside the avionics bay and does not initiate ejection charge deployment.
Status: *Complete*		**Verification Test ID:** N/A

I	**Requirement ID:** 3.5 **Description:** Each altimeter will have a dedicated power supply, and all recovery electronics will be powered by commercially available batteries.	**Verification Plan:** Having these components will be verified via inspection. **Comments:** The two altimeter-battery systems will be completely separated as can be seen in Figure 4.4.
Status: *Complete*		**Verification Test ID:** N/A

D	**Requirement ID:** 3.6 **Description:** Each altimeter will be armed by a dedicated mechanical arming switch that is accessible from the exterior of the launch vehicle airframe when the launch vehicle is in the launch configuration on the launch pad.	**Verification Plan:** Accessibility will be verified through subscale and full scale launches. **Comments:** The team has designed a switch mount that a rocker switch sits in. This switch is accessible from the exterior of the rocket. Please reference Figure 4.3 and 4.12.
Status: *Complete*		**Verification Test ID:** N/A

D	**Requirement ID:** 3.7 **Description:** Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	**Verification Plan:** This will be verified through subscale and full scale launches. **Comments:** The rocker switch is mounted such that it cannot be disarmed from flight forces. This will be proven through all vehicle flights.
Status: *Complete*		**Verification Test ID:** N/A

I	**Requirement ID:** 3.8 **Description:** The recovery system electrical circuits will be completely independent of any payload electrical circuits.	**Verification Plan:** Independence will be verified via inspection. **Comments:** The altimeter-ejection systems will not be wired to the secondary payload (camera) system in the avionics bay in any way, nor to the primary payload system.
Status: *Complete*		**Verification Test ID:** N/A

I	**Requirement ID:** 3.9 **Description:** Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.	**Verification Plan:** The use of shear pins will be verified through inspection. **Comments:** The team's 2020 design utilizes shear pins for both the main and drogue parachute compartments.
Status: *Complete*		**Verification Test ID:** N/A

D	**Requirement ID:** 3.10 **Description:** The recovery area will be limited to 2,500 ft. radius from the launch pads.	**Verification Plan:** This will be verified through the full scale launch. **Comments:** This will be completed after demonstration at the full scale launch. This will also be verified via OpenRocket, RASAero, and in-house drift-trajectory MATLAB code.
Status: *Incomplete*		**Verification Test ID:** N/A

D	**Requirement ID:** 3.11 **Description:** Descent time will be limited to 90 seconds (apogee to touch down).	**Verification Plan:** This will be verified through subscale and full scale launches. **Comments:** This has also been verified via OpenRocket, RASAero, and in-house trajectory MATLAB code.
Status: *Complete*		**Verification Test ID:** N/A

D	**Requirement ID:** 3.12 **Description:** An electronic tracking device will be installed in the launch vehicle and will transmit the position of the tethered vehicle or any independent section to a ground receiver.	**Verification Plan:** This will be verified through both vehicle and payload demonstration flights **Comments:** The AltusMetrum Telemetrum is able to transmit telemetry and GPS coordinates to the team during flight.
Status: *Complete*		**Verification Test ID:** N/A

I	**Requirement ID:** 3.12.1 **Description:** Any launch vehicle section or payload component, which lands untethered to the launch vehicle, will contain an active electronic tracking device.	**Verification Plan:** Having these components will be verified via inspection. **Comments:** No launch vehicle section or payload component will land untethered to the launch vehicle.
Status: *Complete*		**Verification Test ID:** N/A

T	**Requirement ID:** 3.12.2 **Description:** The electronic tracking device(s) will be fully functional during the official flight on launch day.	**Verification Plan:** The functionality of the tracking devices will be verified through individually testing each major electronic system within the avionics bay. **Comments:** This will be completed after conducting test A_04 and the remainder of test A_01 around the 11th of January.
	Status: *In Progress*	**Verification Test ID:** A_01, A_04

D	**Requirement ID:** 3.13 **Description:** The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	**Verification Plan:** This will be verified via demonstration during the full scale launch. **Comments:** This will be completed after demonstration at the full scale launch.
	Status: *Incomplete*	**Verification Test ID:** N/A

I	**Requirement ID:** 3.13.1 **Description:** The recovery system altimeters will be physically located in a separate compartment within the vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.	**Verification Plan**: This will be verified via inspection. **Comments:** No other radio frequency transmitting device and/or magnetic wave producing device will be placed in the avionics bay with the altimeters.
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 3.13.2 **Description:** The recovery system electronics will be shielded from all onboard transmitting devices to avoid inadvertent excitation of the recovery system electronics.	**Verification Plan:** This will be verified via inspection. **Comments:** The team's 2020 design will include shielding between the recovery system electronics and the payload transmitters.
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 3.13.3 **Description:** The recovery system electronics will be shielded from all onboard devices which may generate magnetic waves (such as generators, solenoid valves, and Tesla coils) to avoid inadvertent	**Verification Plan:** This will be verified via inspection.

	excitation of the recovery system.	**Comments:** The team's 2020 design does not and will not include onboard devices which may generate magnetic waves.
Status: *Complete*		**Verification Test ID:** N/A

I	**Requirement ID:** 3.13.4 **Description:** The recovery system electronics will be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	**Verification Plan:** This will be verified via inspection. **Comments:** The team's 2020 design does not and will not include any other onboard devices which may adversely affect the proper operation of the recovery system electronics.
Status: *Complete*		**Verification Test ID:** N/A

8.4.1.4. Payload R&VP

D A	**Requirement ID:** 4.2 **Description:** The team will design a UAV payload that will safely be carried by a high powered rocket. The UAV will deploy from the launch vehicle and recover simulated lunar ice. The UAV will be designed to be safe and follow all rules and regulations.	**Verification Plan:** The UAV will undergo a series of developmental and operational testing and simulations that will validate the capability of the UAV to complete a successful ice sample recovery. These tests will validate that the UAV can functionally perform every mission function on its own as well as in sequence. In addition, all safety critical structural components used on the UAV and UAV retention system will be designed to have a safety factor of at least two using FEA. **Comments:** The UAV is currently being tested in simulation to verify its ice mining and recovery capability. Demonstration of a safe payload flight and ice sample recovery is set to be complete during the full-scale demonstration flight. This requirement will be verified upon successful demonstration in simulation and during the full-scale flight
Status: *In Progress*		**Verification Test ID:** N/A

D	**Requirement ID:** 4.3.1 **Description:** The launch vehicle will be launched from the NASA-designated launch area using the provided launch pad. All hardware utilized at the recovery must launch on the vehicle.	**Verification Plan:** The payload on the launch vehicle will be designed to contain all the hardware necessary for mission completion, All manipulation and control of the payload post recovery will be done through wireless commands issued from the GCS. **Comments:** The payload has been designed to contain all necessary hardware on the launch vehicle. Requirement will be verified through flight demonstration at competition.

Status: *Incomplete*	**Verification Test ID:** N/A

Method	Requirement	Verification
A T	**Requirement ID:** 4.3.2 **Description:** The team will be able to recover ice samples from five different recovery areas with each recovery site being 3' in diameter and contain sample material extending to at least 2" below the surface.	**Verification Plan:** The UAV and ice mining system will be tested in simulation and in a makeshift sampling area to confirm that the UAV will be able to detect, track, and extract an ice sample from a recovery area. **Comments:** Payload tests PT_01.1 and PT_01.2 are in the process of being complete. Requirement will be verified upon completion of these two tests showing successful ice sample recovery capability.
	Status: *In Progress*	**Verification Test ID:** PT_01.1, PT_01.2

Method	Requirement	Verification
D	**Requirement ID:** 4.3.3 **Description:** The recovered ice sample will be a minimum of 10 milliliters (mL).	**Verification Plan:** The UAV ice mining and procurement system will be tested to mine and contain at least 10mL of ice both on a test stand and during a mission. This capability will be demonstrated during both PT_01.1 and PT_02.1. **Comments:** The ice mining system has been designed to recover more than 10ml of ice. Requirement will be verified upon demonstration of recovery capability during PT_01.1 and PT_01.2, which are currently in progress.
	Status: *In Progress*	**Verification Test ID:** N/A

Method	Requirement	Verification
D	**Requirement ID:** 4.3.4 **Description:** Once the sample is recovered, it must be stored and transported at least 10' linearly from the recovery area.	**Verification Plan:** The UAV will be programmed to fly at least 10' linearly away from the recovery site at full ice capacity. **Comments:** Requirement will be verified during the full-scale demonstration flight.
	Status: *Incomplete*	**Verification Test ID:** N/A

Method	Requirement	Verification
I D	**Requirement ID:** 4.3.5 **Description:** The team must abide by all FAA and NAR rules and regulations.	**Verification Plan:** The operation of the UAV will abide by any and all rules and regulations from the FAA and NAR applicable to the operation of model aircraft. In addition, the operation of the UAV will follow any rules regulations issued by state and other local governments. **Comments:** All FAA and NAR rules and regulations have been reviewed and are being complied with. Checklists and testing

		procedures have been written using these rules and regulations as a guideline.
Status: *In Progress*		**Verification Test ID:** N/A

I	**Requirement ID:** 4.3.6 **Description:** Black Powder and/or similar energetics are only permitted for deployment of in-flight recovery systems. And ground systems must employ mechanical systems.	**Verification Plan:** The deployment used for the payload has been designed not incorporate the use of energetics, but will strictly employ the use of a mechanical systems.
		Comments: Requirement has been verified as the payload strictly makes use of a mechanical deployment system.
Status: *Complete*		**Verification Test ID:** N/A

I	**Requirement ID:** 4.3.7 **Description:** Any part of the payload or vehicle that is designed to be deployed, whether on the ground or in the air, must be fully retained until it is deployed as designed.	**Verification Plan:** The ground-deployed payload will be retained and locked in the vehicle until a signal is sent from the GCS.
		Comments: Requirement has been verified as the payload has been designed to be locked until a signal is sent from the GCS to initiate the payload deployment procedure.
Status: *Complete*		**Verification Test ID:** N/A

I	**Requirement ID:** 4.3.7.1 **Description:** A mechanical retention system will be designed to prohibit premature deployment.	**Verification Plan:** The retention and deployment system will strictly make use of a mechanical system to retain and deploy the UAV.
		Comments: Requirement has been verified as deployment can only be signalled by the GCS.
Status: *Complete*		**Verification Test ID:** N/A

D	**Requirement ID:** 4.3.7.2 **Description:** The retention system will be robust enough to successfully endure flight forces experienced during both typical and atypical flights.	**Verification Plan:** All structural components in the retention system will be designed with a factor of safety of at least 2 and functionality of the retention system will be demonstrated during the full-scale demonstration flight.
A		**Comments:** FEA simulations of all structural components have been completed showing a minimum factor of safety of 2. Requirement will be verified upon successful completion of the full-scale demonstration flight.

Status: *In Progress*	**Verification Test ID:** N/A

I	**Requirement ID:** 4.3.7.3 **Description:** The designed system will be fail-safe.	**Verification Plan:** The mechanical system employed to retain the payload in-flight has been designed to be fail-safe and will function regardless of power delivery. **Comments:** Requirement has been verified as the all mechanical retention systems make use of strictly fail-safe locking mechanisms.
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 4.3.7.4 **Description:** Exclusive use of shear pins will not meet requirement 4.3.7.	**Verification Plan:** Shear pins will not be used for the retention of the payload. **Comments:** Requirement has been verified as the payload does not make use of shear pins in it design.
	Status: *Complete*	**Verification Test ID:** N/A

	Requirement ID: 4.4.1 **Description:** Any experiment element that is jettisoned during the recovery phase will receive real-time RSO permission prior to initiating the jettison event.	**Verification Plan:** Shear pins will not be used for the retention of the payload. **Comments:** The UAV will not be jettisoned during the recovery phase and will not require real-time RSO permission.
	Status: N/A	**Verification Test ID:** N/A

	Requirement ID: 4.4.2 **Description:** The UAV, if designed to be deployed during descent, will be tethered to the vehicle with a remotely controlled release mechanism until the RSO has given permission to release the UAV.	**Verification Plan:** The UAV is not designed to be deployed during flight and therefore will not require a remotely controlled release mechanism that will be triggered during descent. **Comments:** The UAV will not be deployed during descent.
	Status: N/A	**Verification Test ID:** N/A

I	**Requirement ID:** 4.4.3 **Description:**	**Verification Plan:** The operation of the UAV shall follow all laws laid out by the FAA pertaining to the operation of rotorcraft and model aircraft.

	Teams flying UAVs will abide by all applicable FAA regulations, including the FAA's Special Rule for Model Aircraft.	**Comments:** FAA rules and regulations have been reviewed and are being complied with. This includes rules included in the FAA's Special Rule for Model Aircraft and the FAA Part 107.
	Status: *In Progress*	**Verification Test ID:** N/A

I	**Requirement ID:** 4.4.4 **Description:** Teams flying UAVs will abide by all applicable FAA regulations, including the FAA's Special Rule for Model Aircraft.	**Verification Plan:** The UAV will be registered with the FAA and its registration number is clearly marked on the outside of the vehicle. **Comments:** Requirement has been verified as the UAV has been registered with the FAA (Certificate Number FA3EXNERX4) and is clearly marked on the outside of the vehicle.
	Status: *Complete*	**Verification Test ID:** N/A

8.4.1.5. Safety R&VP

I	**Requirement ID:** 5.1 **Description:** Each team will use a launch and safety checklist. The final checklists will be included in the FRR report and used during the Launch Readiness Review (LRR) and any launch day operations.	**Verification Plan:** Design review documents will be inspected to contain a launch checklist encompassing at least Pre-Launch, Launch, and Post Launch operations. **Comments:** Checklists have been created to maintain continuity in launch procedures. Quality assurance stops insure the quality of work done in previous steps.
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 5.2 **Description:** Each team must identify a student safety officer who will be responsible for all items in section 5.3.	**Verification Plan:** A safety officer will be voted on before the competition season. **Comments:** The team safety officer for 2019-2020 is Noah Stover.
	Status: *Complete*	**Verification Test ID:** N/A

I	**Requirement ID:** 5.3.1 **Description:** Monitor team activities with an emphasis on safety during the design of vehicle and payload, the construction of vehicle and payload components, the assembly of vehicle and payload, the ground testing of vehicle and	**Verification Plan:** The safety officer will be held accountable by record of attendance by the heads of operations for construction and testing procedures, and the project manager for launch operations.

	payload, the subscale launch test(s), the full-scale launch test(s), the launch day, the recovery activities, and the STEM engagement activities.	**Comments:** Safety officer presence at launch day, construction days, and engagement activities encourages a focus on safe processes and operations.
Status: *In Progress*		**Verification Test ID:** N/A

I	**Requirement ID:** 5.3.2 **Description:** Implement procedures developed by the team for construction, assembly, launch, and recovery activities.	**Verification Plan:** Safety plans will be put in place for the operations of construction (machine operations, epoxy application, etc), launch (see 5.1), and vehicle recovery. **Comments:** Safety steps and procedures have been integrated in construction and launch day activities to ensure operational efficiency and team wellbeing,
Status: Complete		**Verification Test ID:** N/A

I	**Requirement ID:** 5.3.3 **Description:** Manage and maintain current revisions of the team's hazard analyses, failure modes analysis, procedures, and MSDS/chemical inventory data.	**Verification Plan:** Safety data will be reviewed before each milestone document submission and be updated accordingly. **Comments:** Analysis of project hazards allows for the mitigation of possible threats and the creation of a safe work environment.
Status: *In Progress*		**Verification Test ID:** N/A

I	**Requirement ID:** 5.3.4 **Description:** Assist in the writing and development of the team's hazard analyses, failure modes analysis, and procedures.	**Verification Plan:** Hazard analysis will be present in PDR document, and presence will be confirmed by Safety Officer and Project Manager **Comments:** Safety officer participation in the creation of hazard analysis encourages a focus on the comprehensive wellbeing of the launch vehicle, project, and team members.
Status: Complete		**Verification Test ID:** N/A

I	**Requirement ID:** 5.4 **Description:** During test flights, teams will abide by the rules and guidance of the local rocketry club's RSO. The allowance of certain vehicle configurations and/or payloads at the NASA Student Launch does not give explicit or implicit authority for	**Verification Plan:** Inform RSO and club president of launch in a reasonable time prior. Obtain permission to use payload in launch if payload is to be used. **Comments:** In order to organize launches we will contact necessary administrative heads to ensure our launch requirements are within the safety standards they hold for launches.

	teams to fly those vehicle configurations and/or payloads at other club launches. Teams should communicate their intentions to the local club's President or Prefect and RSO before attending any NAR or TRA launch.	
Status: *In Progress*		**Verification Test ID:** N/A

I	**Requirement ID:** 5.5 **Description:** Teams will abide by all rules set forth by the FAA.	**Verification Plan:** An extensive review of FAA guidelines and regulations and checked in order to make sure that all subteams follow and stay within the given regulations. **Comments:** We reviewed FAA guidelines in order to analyze possible risks and their mitigations. Before constructions or launches, we will go over guidelines again to make sure we are following the rules.
Status: *In Progress*		**Verification Test ID:** N/A

8.4.2. Team-Derived R&VP

8.4.2.1. Team-Derived General R&VP

D	**Requirement ID:** T1.1 **Description:** All milestone documents will be finished by the team at least one week ahead of required deadline; this ensures that the executive board may review and make edits prior to milestone submission.	**Verification Plan:** Subteam leads will ensure that each subteam finishes their respective section prior to this deadline. **Comments:** CDR was completed by the 3rd of January 2020 so that the project manager and assistant project manager could review, refine, and finalize.
Status: *In Progress*		**Verification Test ID:** N/A

I	**Requirement ID:** T1.2 **Description:** Each team member, regardless of position, will miss no more than five (5) meetings throughout the entirety of the competition; this ensures that team members stay actively engaged and are making meaningful contributions to the team's design.	**Verification Plan:** Team will take attendance prior to the start of each meeting **Comments:** As of CDR, the team has had to let several people go due to inconsistent attendance and missing more than the five (5) allowed meetings.

Status: *In Progress*	**Verification Test ID:** N/A

D	**Requirement ID:** T1.3 **Description:** STEM engagement activity reports will be submitted within a week of activity date so that documentation of even can be properly written.	**Verification Plan:** Social / Outreach team lead will bring activity reports to all outreach events to ensure that team members fill out reports at completion of event. **Comments:** The team has been working on refining the process for STEM engagement activity reports and is on track to meet this requirement.
	Status: *In Progress*	**Verification Test ID:** N/A

I	**Requirement ID:** T1.4 **Description:** In order to be eligible to attend launch week, each team member will be required to attend a minimum of three STEM educational outreach events.	**Verification Plan:** Attendance will be taken prior to the start of each team lead STEM outreach event. **Comments:** N/A
	Status: *In Progress*	**Verification Test ID:** N/A

8.4.2.2. Team-Derived Vehicle R&VP

D A	**Requirement ID:** T2.1 **Description:** The vehicle will carry the payload to an apogee altitude of 4325 +/- 100' AGL.	**Verification Plan:** Verification analysis will be performed in OpenRocket, secondary analysis will be performed in RASAero, tertiary analysis will be performed in avionics and recovery trajectory code, and final verification will be gathered from full scale launch altimeter data. **Comments:** N/A
	Status: *In Progress*	**Verification Test ID:** N/A

A	**Requirement ID:** T2.2 **Description:** The vehicle will be capable of carrying the payload to apogee with a 12.5 pound or less payload.	**Verification Plan:** 12.5 pounds will be included in the mass of simulations performed to account for the mass of the payload. Further verification will be performed from the gathering of altimeter data during final launch. **Comments:** N/A
	Status: *Complete*	**Verification Test ID:** N/A

D	**Requirement ID:** T2.3 **Description:** The flight path of the launch vehicle shall not differ from the vertical axis by an observable amount under any circumstances or launch conditions with the exception of the flight path at apogee.	**Verification Plan:** Visual inspection of all launch vehicle test flights during ascent will verify the vehicle's flight path does not vary from the vertical direction by unreasonable amounts. **Comments:** N/A
	Status: *Incomplete*	**Verification Test ID:** N/A

D	**Requirement ID:** T2.4 **Description:** The vehicle will be capable of successfully deploying the payload system's UAV after landing.	**Verification Plan:** This requirement will be verified by demonstrating payload deployment capabilities of the launch vehicle during the full scale flight or the payload demonstration flight. **Comments:** N/A
	Status: *In Progress*	**Verification Test ID:** N/A

D	**Requirement ID:** T2.5 **Description:** The full-scale launch vehicle will be successfully launched and recovered in a test flight before March 2, 2020, configured in the same configuration that will be used on the final launch day	**Verification Plan:** Full scale flight launch data will be recorded and included in milestone documentation to prove a full scale flight has occurred by the specified date and in the desired configuration. **Comments:** N/A
	Status: *Incomplete*	**Verification Test ID:** N/A

I	**Requirement ID:** T2.6 **Description:** If the launch day motor is not capable of being flown during the full-scale test flight, the replacement motor shall simulate as closely as possible the predicted maximum velocity and maximum acceleration of the launch day flight.	**Verification Plan:** The motor which is used to simulate the final motor to be used shall be the closest allowable motor impulse level and shall retain a size and mass which differs by no more than 10% than the motor to be used. **Comments:** N/A
	Status: *Incomplete*	**Verification Test ID:** N/A

D	**Requirement ID:** T2.7 **Description:** If the Student Launch office determines that a re-flight is necessary, then another flight of the full-scale vehicle will be conducted before March 20, 2020.	**Verification Plan:** Full scale re-flight launch data will be recorded and included in milestone documentation to prove a re-flight has occurred by the specified date. **Comments:** N/A
	Status: *Incomplete*	**Verification Test ID:** N/A

8.4.2.3. Team-Derived Avionics and Recovery R&VP

T A	**Requirement ID:** T3.1 **Description:** The main parachute will open with an altitude great enough for the heaviest section of the launch vehicle to land with a kinetic energy of less than 75 ft-lbf.	**Verification Plan:** Initial analysis was done using a coded simulation to select the optimal main parachute. Then, during the parachute drop test, after the main parachute is dropped from an already opened state, the kinetic energy of the largest section of the launch vehicle will be calculated. **Comments:** The analysis required was conducted through code when choosing parachutes before PDR. The verification will be completed after conducting test A_05 on the 12th of January.
	Status: *In Progress*	**Verification Test ID:** A_05

T	**Requirement ID:** T3.2 **Description:** Both of the batteries will consistently supply 40-60 mAh to their corresponding altimeter for a minimum of 1.5 hours.	**Verification Plan:** Each battery will be verified if, connected to its corresponding altimeter, the battery is able to keep it powered on for an hour and a half as well as stay within 40 and 60 mAh of electric charge (measured every half hour). **Comments:** This will be completed after conducting test A_04 on the 11th of January.
	Status: *Incomplete*	**Verification Test ID:** A_04

T A	**Requirement ID:** T3.3 **Description:** For ground testing, the black powder ejection system will create at least 6' of separation between the avionics bay and each airframe for at least one amount of black powder equal to or greater than 5 grams for the main ejection charge and 2 gram for the drogue ejection charge, as well as full ejection of each parachute.	**Verification Plan:** Ejection charge sizing analysis was performed using hand calculations to estimate a target amount of black powder to be used. Then the distance between the avionics bay and each airframe will be measured after test ejection, ultimately allowing the team to determine the most efficient amount of black powder to use. **Comments:** The required analysis has been completed and this requirement will be verified after conducting test A_03 on the 12th of January.

Status: *In Progress*	Verification Test ID: A_03

T	**Requirement ID:** T3.4 **Description:** Both altimeters must be able to consistently ignite both ejection charges at the appropriate times.	**Verification Plan:** Altimeter functionality will be verified through simulating a flight via a vacuum chamber. **Comments:** This will be completed after conducting test A_02 on the 11th of January.
	Status: *Incomplete*	**Verification Test ID:** A_02

T	**Requirement ID:** T3.4.1 **Description:** The primary drogue ejection charge will ignite within +/- 500' of apogee.	**Verification Plan:** Recorded drogue ignition altitude will be compared against apogee altitude. **Comments:** This will be completed after conducting test A_02 on the 11th of January.
	Status: *Incomplete*	**Verification Test ID:** A_02

T	**Requirement ID:** T3.4.2 **Description:** The redundant drogue ejection charge will have a drogue delay between 0.75 and 1.75 seconds.	**Verification Plan:** The duration of time between apogee and drogue ignition will be measured. **Comments:** This will be completed after conducting test A_02 on the 11th of January.
	Status: *Incomplete*	**Verification Test ID:** A_02

T	**Requirement ID:** T3.4.3 **Description:** The primary and redundant main ejection charges will ignite within +/- 50' of their programmed deployment altitudes (800' AGL for the primary altimeter and 700' AGL for the redundant altimeter).	**Verification Plan:** Recorded main ignition altitudes will be compared against programmed deployment altitudes for each altimeter. **Comments:** This will be completed after conducting test A_02 on the 11th of January.
	Status: *Incomplete*	**Verification Test ID:** A_02

T	**Requirement ID:** T3.5 **Description:** The altimeter firing sequence will be consistent across temperature extremes.	**Verification Plan:** Each altimeter will be verified if it emits three beeps every five seconds after the initialization routine in both temperature extremes, indicating successful continuity for a dual deploy configuration. **Comments:** This will be completed after finishing the procedure for test A_01 on the 11th of January.

Status: *In Progress*	Verification Test ID: A_01

8.4.2.4. Team-Derived Payload R&VP

I	**Requirement ID:** PR_2.1 **Description:** The vehicle shall not pose a significant safety risk to its surroundings during operation.	**Verification Plan:** Operation of the UAV will strictly adhere to predetermined flight paths. Any and all safety risks will be identified and addressed prior to flight, and proper documentation of emergency procedures will be available at hand. **Comments:** Each UAV flight is currently requiring the use of a flight test document that contains information relating to the safe operation of the UAV as well as all rules and regulations, both FAA and local, that must be complied with. This flight test document also contains a pre-flight checklist that ensures the operation of the UAV is safe and predictable.
	Status: *In Progress*	**Verification Test ID:** N/A

I	**Requirement ID:** PR_2.2 **Description:** Autonomous operation of the UAV shall require a designated Pilot-in-Command (PIC) monitoring UAV telemetry and status, and an observer that maintains visual line of sight with the UAV at all times.	**Verification Plan:** Every flight of the UAV will require a designated pilot-in-command (PIC) and observer, both with proper training and equipment. This information will be documented in a preflight checklist and a flight maintenance document. **Comments:** A PIC and observer are being designated and briefed prior to each flight.
	Status: *In Progress*	**Verification Test ID:** N/A

I D	**Requirement ID:** PR_2.3 **Description:** Control of the UAV with computer vision algorithms shall be minimally accompanied with real-time imaging data and a kill-switch.	**Verification Plan:** Testing of computer vision algorithms on a UAV in-flight will require imaging data to be transmitted and monitored by the PIC. A kill-switch must be easily accessible at all times by the PIC, and the PIC shall remain in constant contact with the observer. **Comments:** Flight testing of the UAV with offboard is not taking place until the GCS construction has been completed and its functionality verified.
	Status: *In Progress*	**Verification Test ID:** N/A

D	**Requirement ID:** PR_2.4 **Description:**	**Verification Plan:** Switches will be tested in simulation and using flight hardware (assuming the test will not risk the safety

	Switches on the GCS that alter the operation or flight mode of the UAV, that are mission critical, or are essential for the safe operation of the UAV will require checkout testing.	of the vehicle nor its surroundings) to ensure expected behavior is met.
		Comments: Demonstration of GCS switch functionality will be completed during PT_04.1 in simulation and PT_03.1/PT_04.2 with UAV hardware.
	Status: *Incomplete*	**Verification Test ID:** N/A

I	**Requirement ID:** PR_2.5 **Description:** The operation of the UAS shall adhere to Purdue's Operating Procedures For Use of UAS and Model Aircraft.	**Verification Plan:** The operation of the UAV shall follow all rules laid out by Purdue University pertaining to the operation of UAVs.
		Comments: All rules laid out by Purdue University regarding the operation of UAVs have been reviewed and are being complied with.
	Status: *In Progress*	**Verification Test ID:** N/A

T	**Requirement ID:** PR_2.6 **Description:** The UAV shall be able to deploy from different landing orientations and conditions.	**Verification Plan:** UAV deployment will be tested with varying landing orientations of the payload bay to identify worst-case deployment scenarios. Additionally, UAV deployment will be tested with various angles of the UAV sled to identify worst-case sled orientations that still allow for successful deployment of the UAV.
		Comments: UAV deployment capability will tested in tests PT_02.1 and PT_02.2. Requirement will be verified upon completion of these tests.
	Status: *Incomplete*	**Verification Test ID:** PT_02.1, PT_02.2

T	**Requirement ID:** PR_2.7 **Description:** The UAV shall have smooth, safe, and stable flight throughout the entirety of its mission.	**Verification Plan:** Low-level control loop gains will be tuned, utilizing standard PID tuning procedures, to ensure safe and stable flight of the UAV.
		Comments: Flight controller tuning will take place during test PT_03.1. Requirement will be verified upon completion of this test.
	Status: *Incomplete*	**Verification Test ID:** PT_03.1

T	**Requirement ID:** PR_2.8 **Description:**	**Verification Plan:** Recovery area identification performance will be tested with a sample data set of recovery area images.

	The UAV's computer-vision and imaging system shall identify ice mining recovery areas and facilitate guided-descent of the UAV at such locations.	Guided descent of the UAV will be tested in computer simulations as well as UAV test flights with sample recovery area test articles.
		Comments: The computer-vision software and navigational software will be tested in PT_04.1 and PT_04.2. Requirement will be verified upon completion of these tests.
Status: *Incomplete*		**Verification Test ID:** PT_04.1, PT_04.2

T	**Requirement ID:** PR_2.9 **Description:** Communication and data links within the payload system will maintain a strong connection throughout the course of the mission with a minimum operational range of 1 mile.	**Verification Plan:** Key metrics quantifying the performance of data links within the system will be analyzed under various simulated mission conditions. Examples of such metrics include packet loss and wireless signal strength.
		Comments: All payload wireless data links will undergo range testing in PT_05.1. Requirement will be verified upon completion of this test.
Status: *Incomplete*		**Verification Test ID:** PT_05.1

T	**Requirement ID:** PR_2.10 **Description:** The structure of the UAV shall be able to withstand in-flight forces that can be reasonably expected over the course of the mission.	**Verification Plan:** A load cell will be employed to quantify the maximum amount of force the UAV can exert. Finite element analysis will also be performed on the UAV to verify the structural integrity of the vehicle.
		Comments: The UAV will undergo in-flight structural load testing in PT_06.1. Requirement will be verified upon completion of this test.
Status: *In Progress*		**Verification Test ID:** PT_06.1

T	**Requirement ID:** PR_2.11 **Description:** Batteries for the R&D and GCS shall minimally allow system operation for at least 4 hours under normal operation. UAV batteries shall have at least 1.5 times the battery capacity necessary for the longest duration flight time expected.	**Verification Plan:** All batteries used on these systems will undergo drain testing to determine how long the battery is expected to last and how certain mission phases and system actions impact battery characteristics.
		Comments: GCS and R&D batteries will be tested in PT_07.1. Requirement will be verified upon completion of this test.
Status: *In Progress*		**Verification Test ID:** PT_07.1

8.4.2.5. Team-Derived Safety R&VP

I	**Requirement ID:** T5.1 **Description:** Every team member must have a Pocket Safety Document on their person for all launch day, construction, assembly, or test operation.	**Verification Plan:** Team members will be asked to display Pocket Safety Documents before operation occurs. **Comments:** Separate pocket safety documents have been written for the operations of testing, machining, construction, and launch day.
	Status: *In Progress*	**Verification Test ID:** N/A

I	**Requirement ID:** T5.2.1 **Description:** Team members must be briefed on machine operations and operational hazards prior to construction operations.	**Verification Plan:** All team members performing machining or other construction operations will be present for a pre-operation briefing presentation covering PPE, machine operation, and other safety hazards approved by the safety officer. **Comments:** Pocket Safety Documents (See T5.1) will contain a condensed version of pre-operation briefing for on-site reference.
	Status: Complete	**Verification Test ID:** N/A

I	**Requirement ID:** T5.2.2 **Description:** Team members must be briefed on testing operations and operational hazards prior to tests.	**Verification Plan:** All team members performing testing will be present for a pre-test briefing presentation covering safety procedure, material hazards, and necessary PPE approved by the safety officer. **Comments:** Pocket Safety Documents (See T5.1) will contain a condensed version of pre-test briefing for on-site reference.
	Status: *In Progress*	**Verification Test ID:** N/A

I	**Requirement ID:** T5.2.3 **Description:** Team members must be briefed on launch operations and procedures prior to any sub-scale or full-scale launch.	**Verification Plan:** All team members performing launch procedures will be present for a pre-launch briefing presentation covering PPE, launch hazards, and proper launch procedure. **Comments:** Pocket Safety Documents (See T5.1) will contain a condensed version of launch briefing for on-site reference.
	Status: *In Progress*	**Verification Test ID:** N/A

I	**Requirement ID:** T5.3 **Description:** All students working with powered machinery must demonstrate a clear and	**Verification Plan:** Before operation, students must present verification of completion of requisite safety briefings, courses, or other material provided by the machine's holding group (Purdue Bechtel Innovation Design Center, Zucrow Labs, Purdue Aerospace Science Laboratory, etc).

	comprehensive knowledge of the machine and its requisite safety standards per the standards of the machining location.	**Comments:** The Purdue Bechtel Innovation Design Center requires online quizzes going over machine safety before the location is accessible to the student. All students performing machining operations have demonstrated proficiency prior to un
Status: *Complete*		**Verification Test ID:** N/A

I	**Requirement ID:** T5.4 **Description:** First aid equipment will be available and accessible at all launch day, construction, assembly, or test operations.	**Verification Plan:** Safety officer will be responsible for the upkeep and presentation of first aid kit. Kit's presence will be logged by safety officer, and initialed and dated by one witnessing team member. **Comments:** First aid kit contains multi-sized bandages, antibiotics, anti-inflammatories, anti-itch cream, pain killers, gauze, and an instant ice pack.
Status: *In Progress*		**Verification Test ID:** N/A

I	**Requirement ID:** T5.5 **Description:** A clear walking path free of hazards or pitfalls must be marked for use by team members on the launch field from the launch pad to observation area	**Verification Plan:** Safety team members will perform sweep of surroundings before rocket setup. A clear path will be denoted by two rows of flags containing the cleared path. **Comments:** Flags will be counted upon placement and recounted upon removal to ensure all flags are collected and accounted for.
Status: *In Progress*		**Verification Test ID:** N/A

8.5. Budgeting

8.5.1. Line Item Budget

Subteam	Purchaser	Item	Merchant	Quantity	Unit Cost	Total Cost
Payload Retention and Deployment						
Payload	Josh	Leveling servo	Servo city	1	$32.90	$32.90
Payload	Josh	Leveling clamp	servocity	1	$6.99	$6.99
Payload	Josh	Servo-worm coupler	servocity	1	$6.99	$6.99
Payload	Josh	Power Transmission shaft	servocity	2	$1.09	$2.18
Payload	Josh	Linear motion shafts	McMaster-Carr	4	$28.80	$115.20
Payload	Josh	Retaining rings	McMaster-Carr	1	$8.69	$8.69
Payload	Josh	Bushings	McMaster-Carr	12	$0.86	$10.32
Payload	Josh	turntable	McMaster-Carr	2	$3.11	$6.22
Payload	Josh	RH Leadscrew	McMaster-Carr	1	$16.60	$16.60
Payload	Josh	LH Leadscrew	McMaster-Carr	1	$20.24	$20.24
Payload	Josh	Leadscrew Couplers	Amazon	1	$8.69	$8.69
Payload	Josh	Eye nut	McMaster-Carr	1	$14.35	$14.35

Payload	Josh	Standoffs	McMaster-Carr	6	$2.60	$15.60
Payload	Josh	Right handed nuts	McMaster-Carr	1	$31.42	$31.42
Payload	Josh	Left handed nut	McMaster-Carr	1	$31.42	$31.42
Payload	Josh	Dual shaft stepper	Amazon	1	$21.50	$21.50
Payload	Josh	Eye nut bolt	McMaster-Carr	1	$5.29	$5.29
Payload	Josh	Xbee RF Module Kit	Digikey	1	$99.00	$99.00
Payload	Josh	Teensy 4.0 Microcontroller	PJRC	2	$19.95	$39.90
Payload	Josh	LiPo Battery	Amazon	1	$21.00	$21.00
Payload	Josh	Gyroscope	Amazon	1	$4.99	$4.99
Payload	Josh	Worm Set	Servocity	1	$21.99	$21.99
Payload	Josh	Bulk plate stock	McMaster-Carr	2	$16.78	$33.56
Payload	Josh	Stepper encoder	Digikey	1	$23.63	$23.63
Payload	Josh	50 ¼”-20 screws	McMaster-Carr	1	$5.08	$5.08
Payload	Josh	100 4-40 screws	McMaster-Carr	1	$8.96	$8.96
Payload	Josh	100 6-32 screws	McMaster-Carr	1	$8.96	$8.96
Payload	Josh	100 6-32 nuts	McMaster-Carr	1	$3.01	$3.01
Payload	Josh	Retaining ring pliers	McMaster-Carr	1	$16.12	$16.12
Payload	Josh	Pcb standoffs	McMaster-Carr	4	$2.86	$11.44
Payload	Josh	Activation switch	Digikey	1	$1.13	$1.13
Payload	Josh	100 push on retaining rings	McMaster-Carr	1	$7.63	$7.63
Payload	Josh	Eye nut	McMaster-Carr	1	$11.32	$11.32
Payload	Josh	Allen key set	McMaster-Carr	2	$6.79	$13.58
Payload	Josh	100 6/32 socket head screw	McMaster-Carr	1	$6.08	$6.08
Payload	Hicham	TO-220 clip on heatsink	Adafruit	2	$1.25	$2.50
Payload	Hicham	UBEC 5V @3A	Adafruit	2	$9.95	$19.90
Payload	Hicham	JST- XHP connectors pack	Amazon	1	$8.99	$8.99
Payload	Hicham	L7805 Voltage Regulator -5V	Sparkfun	4	$0.95	$3.80
Payload	Josh	Bushings	McMaster-Carr	10	$0.86	$8.60
Payload	Josh	Ball Bearings	McMaster-Carr	2	$6.48	$12.96
Payload	Josh	100 4-40 Screws 1/2"	McMaster-Carr	1	$9.20	$9.20
Payload	Josh	100 1/4"-20 Nuts	McMaster-Carr	1	$3.90	$3.90
Payload	Josh	100 1/4"-20 Screws	McMaster-Carr	1	$4.38	$4.38
Payload	Josh	10 6-32 Screws	McMaster-Carr	1	$4.00	$4.00
Payload	Josh	Ball Bearings	McMaster-Carr	2	$12.34	$24.68
Payload	Josh	Stepper Motor Driver	Robotshop	1	$18.00	$18.00
Payload	Josh	Teensy 4.0 Replacement	PJRC	1	$19.95	$19.95
Payload	Josh	3.3V to 5V Level shifter	Amazon	1	$3.98	$3.98
Payload	Josh	10 Socket Nuts	McMaster-Carr	1	$11.27	$11.27
Payload	Josh	100 4-40 Button Heads	McMaster-Carr	1	$3.02	$3.02
Payload	Josh	Self Aligning Bushing	igus	4	$5.22	$20.88
Payload	Josh	Fixed Alignment Bushing	igus	4	$5.83	$23.32
Payload	Josh	25 Press Fit Nuts	McMaster-Carr	1	$8.76	$8.76
Payload	Josh	100 Thin Nylock Nuts	McMaster-Carr	1	$3.71	$3.71
Payload Electronics and Controls						
Payload	Hicham	Pixhawk 4 Kit	Sparkfun	1	$249.99	$249.99

Payload	Hicham	ReadyToSky 915MHz 500mw	Amazon	1	$23.99	$23.99
Payload	Hicham	FrSky R-XSR RC Receiver	Amazon	1	$26.99	$26.99
Payload	Hicham	Raspberry Pi Zero W	Adafruit	2	$5.00	$10.00
Payload	Hicham	Nylon 6 Standoffs	McMaster-Carr	8	$2.00	$16.00
Payload	Hicham	3600mAh 3S 30 Turnigy Battery	Hobbyking	2	$27.65	$55.30
Payload	-	Toggle Switch and Cover	Sparkfun	4	$2.95	$11.80
Payload	-	15.6" Display	Amazon	1	$61.00	$61.00
Payload	-	15.6" Display Controller	Amazon	1	$26.68	$26.68
Payload	-	5mm LED Holder	Sparkfun	20	$0.50	$10.00
Payload	-	5mm LED Pack	Amazon	1	$10.99	$10.99
Payload	-	Momentary Push Button Switch x5	Amazon	2	$11.69	$23.38
Payload	-	LED Push Button Switch x5	Amazon	1	$11.97	$11.97
Payload	-	3Pin Toggle Switch x10	Amazon	1	$7.99	$7.99
Payload	-	Rocker Switch x15	Amazon	1	$6.99	$6.99
Payload	Hicham	Raspberry Pi 4 2GB	Adafruit	1	$45.00	$45.00
Payload	-	Taranis Q X7	Banggood	1	$97.99	$97.99
Payload	-	3.2" LCD Display	Banggood	1	$6.99	$6.99
Payload	Hicham	Raspberry Pi Zero Camera	Amazon	1	$24.83	$24.83
Payload Ice Mining						
Payload	Hicham	DC 6V Gear Motor High Torque	Amazon	2	$10.99	$21.98
Payload	Hicham	Low-Strength Steel Nylon-Insert Locknut: Zinc-Plated, 4-40 Thread Size	McMaster-Carr	1	$2.79	$2.79
Payload	Hicham	Set Screw Shaft Collar: for 1/4" Diameter, Black-Oxide 1215 Carbon Steel	McMaster-Carr	2	$1.15	$2.30
Payload	Hicham	12” D-Profile Rotary Shaft	McMaster-Carr	1	$9.40	$9.40
Payload	Hicham	torsion spring: 270 Degree Angle, Right-Hand Wound, 0.171" OD	McMaster-Carr	2	$5.22	$10.44
Payload	Hicham	Nylon Threaded Rod: 4-40 Thread Size, 2’ Long, Black	McMaster-Carr	1	$7.15	$7.15
Payload Airframe						
Payload	Hicham	Turnigy Multistar 21A ESC	Hobbyking	6	$9.81	$58.86
Payload	Hicham	Luminier 6.7x3x3 Folding Prop, 4 Pack	Amazon	2	$11.99	$23.98
Payload	Hicham	Turnigy Aerodrive SK3 1740Kv	HobbyKing	5	$18.03	$90.15
Payload	-	Nylon Sheet				$0.00
Payload	Hicham	Torsion Spring 5 Pack (180 Degree Right-Hand Wound, 0.556" OD)	McMaster-Carr	1	$1.00	$1.00
Payload	Hicham	Female Threaded Hex Standoff (18-8 Nylon, 1/4" Hex, 1" Long, 4-40 Thread)	McMaster-Carr	8	$2.23	$17.84
Payload	Hicham	18-8 Stainless Steel Hex Head Screws 100 Pack (6-32 Thread Size, 1/2" Long)	McMaster-Carr	1	$9.58	$9.58
Payload	Luke/Josh	316 Stainless Steel Washer 25	Menards	2	$0.49	$0.98

		Pack (Oversized, Number 12 Screw Size, 0.25" ID, 1" OD)				
Payload	Luke/Josh	Zinc Yellow-Chromate Plated Hex Head Screw 25 Pack (Grade 8 Steel, 1/4"-20 Thread Size, 1-1/2" Long, Fully Threaded)	Menards	2	$0.89	$1.78
Payload	Luke/Josh	Extreme-Strength Steel Extra-Wide Thin Hex Nut 2 Pack (Grade 2H, Zinc-Plated, 1/4"-20 Thread Size)	Menards	2	$0.49	$0.98
Payload Secondary Camera						
Payload	-	ELP 5MP Camera	Not Purchased Yet	1	$48.00	$48.00
Payload	-	Pimironi LiPo Sheet	Not Purchased Yet	1	$9.95	$9.95
Payload	-	LiPo Battery 3.7v 2000mAh	Not Purchased Yet	1	$12.95	$12.95
Payload	-	RPi Zero	Not Purchased Yet	1	$10.00	$10.00
Construction						
Construction	Luke	G12 Fiberglass Coupler Tubes (6" Dia. x 27" Len)	Wildman Rocketry	27	$4.89	$132.03
Construction	Luke	G12 Fiberglass Airframe (6" Dia x 60" Len)	Wildman Rocketry	1	$231.50	$231.50
Construction	Luke	G12 Fiberglass Airframe (6" Dia x 48" Len)	Wildman Rocketry	2	$185.00	$370.00
Construction	Luke	G12 Fiberglass 5:1 Von Karman Nosecone w/ Metal Tip (6" Dia.)	Wildman Rocketry	1	$129.00	$129.00
Construction	Luke	G10 Fiberglass Airframe Bulkplate (6" Dia. x 0125")	Wildman Rocketry	6	$9.00	$54.00
Construction	Luke	G10 Fiberglass Coupler Bulkplate (5.775" Dia. x 0125")	Wildman Rocketry	6	$9.00	$54.00
Construction	Luke	G12 Fiberglass Coupler Tubes (6" Dia. x 3" Len)	Wildman Rocketry	3	$4.89	$14.67
Construction	Luke	G12 Fiberglass Switch Band (6" Dia. x 3" Len)	Wildman Rocketry	2	$10.00	$20.00
Construction	Luke	Cesaroni (CTI) L1115 Classic High Power Rocket Motor	Wildman Rocketry	1	$292.99	$292.99
Construction	Luke	75mm Motor Retainer (need to check connection (flange))	Wildman Rocketry	1	$51.00	$51.00
Construction	Luke	G12 Fiberglass Centering Rings (6" OD x 75mm ID)	Wildman Rocketry	3	$10.00	$30.00
Construction	Luke	G12 Fiberglass Motor Tube (75mm Dia.)	Wildman Rocketry	1	$41.02	$41.02
Construction	Luke	Cesaroni (CTI) L1115 Classic High Power Rocket Motor	Wildman Rocketry	2	$292.99	$585.98
Construction	Luke	Structural Adhesive, Epoxy, Loctite 1C-Lv, 1.69 oz. Cartridge	McMaster-Carr	10	$17.78	$177.80
Construction	Luke	5.9" Long Taper Tip Nozzle with Bayonet Connection for Two-Part Cartridge	McMaster-Carr	40	$1.30	$52.00

Construction	Luke	Dispensing Gun for Two-Part Cartridge	McMaster-Carr	1	$23.76	$23.76
Avionics						
Avionics	Katelin	9V Battery Connectors	Amazon	1	$3.40	$3.40
Avionics/ Construction	Katelin	Latex Gloves	Amazon	1	$10.65	$10.65
Avionics	Katelin	Hex Wrench Set	Amazon	1	$16.26	$16.26
Avionics	Katelin	Terminal Blocks	Apogee Rockets	2	$5.20	$10.40
Avionics	Katelin	Altimeter Mounting Posts	Apogee Rockets	4	$5.48	$21.92
Avionics	Katelin	3.7V 900 mAh LiPo Battery	Apogee Rockets	1	$13.20	$13.20
Avionics	Katelin	FFFFG Black Powder (1 lb)	Graf & Sons	4	$48.33	$193.32
Avionics	Katelin	1' E-Matches (80 ct)	Electric Match	1	$53.98	$53.98
Avionics	Katelin	3' E-Matches (10 ct)	Electric Match	1	$25.48	$25.48
Avionics	Katelin	Red 28 Gauge Stranded Wire (90 ft)	Pololu Robotics & Electronics	1	$7.46	$7.46
Avionics	Katelin	Black 28 Gauge Stranded Wire (90 ft)	Pololu Robotics & Electronics	1	$7.47	$7.47
Avionics	Katelin	Rocker Switches	Digikey	4	$6.14	$9.59
Avionics	Katelin	USB A to USB Micro B Cable	Amazon	1	$2.68	$2.68
Avionics	Katelin	Terminal Blocks for Telemetrum	Digikey	2	$10.72	$15.44
Avionics	Katelin	Terminal Blocks	Apogee Rockets	2	$9.36	$12.91
Avionics	Katelin	Rocker Switches	Digikey	3	$7.20	$9.50

Table 8.11: Line Item Budget

8.5.2. Funding Plan

8.5.2.1. Sources of Funding

The team has a total budget of $11,050 and has fundraised $8,451 of this budget through donations, grants, engineering department, and sales of team spirit items. Currently, 44.2% of the current total has been raised through donations, 18.9% through the Aerojet Rocketdyne grant, 35.5% through collectively Purdue's Mechanical Engineering Department and the Electrical and Computer Engineering Department, and 1.4% through the sale of team apparel. There is $2,599 left to raise, and the team plans to do so through the Purdue Aeronautical and Astronautical Engineering Department travel grant, a $1600 grant, and the sale of team apparel.

8.5.2.2. Allocation of Funds

Section	Estimated Cost	Money Spent	Future Expenses	Section	Estimated Cost	Money Spent	Future Expenses
Construction	$3000.00	$2670.95	$200.00	**Outreach**	$100.00	$64.58	$2500.00
Avionics	$900.00	$921.62	$20.00	**Safety**	$250.00	$220.00	$75.00
Payload	$2000.00	$1574.89	$390.88	**Outreach**	$100.00	$64.58	$40.00
Branding	$550.00	$330.00	$600.00	**Total**	$11050.00	$5628.61	$3825.88

Table 8.12: Allocation of Funds

8.6. Completed STEM Engagement Events Since October

Event	Date of Event	Direct Interactions	Indirect Interactions
Purdue Space Day	11/9/2019	898	225
Activity 1: Water Rockets	11/9/2019	326	108
Activity 2: Moon Buggies	11/9/2019	120	60
Activity 3: Mars Rover	11/9/2019	240	51
Activity 4: Satellite Launch	11/9/2019	31	0
Activity 5: Group Leaders	11/9/2019	61	10
Activity 6: Apollo 13	11/9/2019	120	26
PESC IDEAS	11/16/2019	43	12
College Mentors for Kids	11/18/2019	28	0
College Mentors for Kids	11/20/2019	44	0
Boy Scouts- Space Exploration Merit Badge	12/14/2019	23	8
CDR Total		1036	245

Table 8.13: STEM Engagement Events from PDR to CDR

8.6.1. Purdue Space Day - 11/9/2019

8.6.1.1. Description

Purdue Space Day (PSD) is an annual event put on by students at Purdue University. There are three age groups: 3rd and 4th, 5th and 6th, and 7th and 8th. Each age group goes through 3 activities through the day and then they have the opportunity to meet an astronaut. The team had 20 members participate in this event. The majority of them participated in one activity that the team conducted, but others participated in alternative activities or guided the groups to and from activities. The activity that the team led was water rockets with 5th and 6th graders. The team also had team members involved with moon buggies with 3rd and 4th graders, Mars rovers with 7th and 8th graders, a satellite launch with 5th and 6th graders, and an Apollo 13 mission simulation with 5th and 6th graders. We also had two members guide groups of 5th and 6th grade children through all of the activities.

8.6.1.2. Activity Educational Goal

8.6.1.2.1. Activity 1: Water Rockets

This activity had children design and build a water rocket in groups. The goal of this activity was to help the children learn the science behind rocketry by challenging them to create the highest-flying rocket that landed safely. Through testing they were able to figure out how much water and how much pressure was required to produce the best results.

8.6.1.2.2. *Activity 2: Moon Buggies*

This activity had children build a moon buggy that was both light and efficient. The goal of this activity was to teach the children the best way to work together to solve a problem and how weight plays an important role in all engineering decisions - especially ones that deal with space travel.

8.6.1.2.3. *Activity 3: Mars Rover*

This activity had children plan and operate a wireless small-scale Mars rover. The goal of this activity was to expose the students to the development of technologies and techniques that are involved in executing a successful extraterrestrial rover mission.

8.6.1.2.4. *Activity 4: Satellite Launch*

This activity had children create a balloon-powered launch vehicle that was able to carry ping pong balls and/or cotton balls as a payload to the second floor of the building. The goal of this activity was to teach the students about energy and how different weights would affect how their vehicle would fly. They needed to figure out the best way to configure their payload to create a successful balloon vehicle.

8.6.1.2.5. *Activity 5: Group Leaders*

The goal of the group leaders was to transport and supervise groups of children and to help them through the various activities for the age group they were a part of.

8.6.1.2.6. *Activity 6: Apollo 13*

This activity had children channel fog through tubes to help the students understand how mission control helped the Apollo 13 crew. The goal of this activity was for the students to see the role that Mission Control plays in every space exploration mission and allow them to see space exploration from a different perspective.

8.6.1.3. Outcome

Through this event, the PSP-SL team was able to reach 898 students and expand their knowledge in space and engineering related activities. They were also able to learn the importance of teamwork within engineering and how it contributes to a better end result.

8.6.2. PESC IDEAS - 11/16/2019

8.6.2.1. Description

This was an event which had the kids rotate through a variety of engineering stations to learn about different types of engineering and do fun activities at each station. The team helped with the stomp rockets station.

8.6.2.2. Activity Educational Goal

The goal of this activity was to teach the students how rockets work and why it is important to have fins and stability within a rocket design.

8.6.2.3. Outcome

The students were able to learn the basic design of rockets and the theory behind how stomping on the stands launches them into the air. They also learned the importance of launch angles for their rockets to go the highest and farthest that they could go.

8.6.3. College Mentors for Kids - 11/18/2019

8.6.3.1. Description

College Mentors for Kids is a program that Purdue University offers which brings in local students once a week and pairs them with Purdue students. A College Mentors for Kids event was held in which a PSP-SL volunteer helped the Purdue Society of Women Engineers (SWE) make LEGO engine kits for kids that included a custom 3D printed part.

8.6.3.2. Activity Educational Goal

The goal of this activity was to teach the students hands on building skills as well as inform them about different types of engineering that they could do in the future.

8.6.3.3. Outcome

The kids were able to learn about the different parts of an engine and the different types of engineering so they will be more informed when they are thinking about future majors or careers.

8.6.4. College Mentors for Kids - 11/20/2019

8.6.4.1. Description

College Mentors for Kids is a program that Purdue University offers which brings in local students once a week and pairs them with a Purdue student. The team led the activity with groups of 1st and 2nd graders. The activity was to create balloon powered cars.

8.6.4.2. Activity Educational Goal

The goal of this activity was for the students to learn about energy and how it works in cars. The students were tasked to make their car go as far as it could go, so they had to consider ways in which the balloon's energy could be lost inefficiently.

8.6.4.3. Outcome

The students were able to learn all of the best ways to make their car the most efficient that it could be. They learned what energy was and how the balloon created the movement in the car. The students were able to determine the optimal amount of air required to have their car go the farthest or the fastest.

8.6.5. Boy Scouts Space Exploration Merit Badge-12/14/2019

8.6.5.1. Description

The team was able to help a Boy Scout troop obtain their Space Exploration Merit Badge. The team helped the scouts build model rockets and learn about the International Space Station.

8.6.5.2. Activity Educational Goal

The primary goal of this activity was to teach the scouts about the different parts of a rocket and help them safely build their own. Another goal of the activity was to teach them all of the components of the International Space Station and how it is a large collaboration between many different countries.

8.6.5.3. Outcome

The Boy Scouts were able to obtain their Space Exploration Merit Badge and learn about model rocketry and the International Space Station. They were able to make the most efficient rockets that they could based off what they had learned about the parts of a rocket and how they worked.

8.7. Plans for Future STEM Engagement

In the future, the team plans on working further with College Mentors for Kids. They conduct their events on campus, which increases team member participation. In addition, the team will be reaching out to more Boy Scout and Girl Scout troops to see if there will be any more opportunities to do activities with them. The team will also be working with the Purdue Space Day Ambassadors to have members of the team volunteer to help out at their events.

8.8. Project Timeline

The timeline the PSP-SL team will be following is shown below. The timeline outlines the following events: deadlines, launch opportunities, meetings or teleconferences with NASA officials, general team meetings, and miscellaneous events.

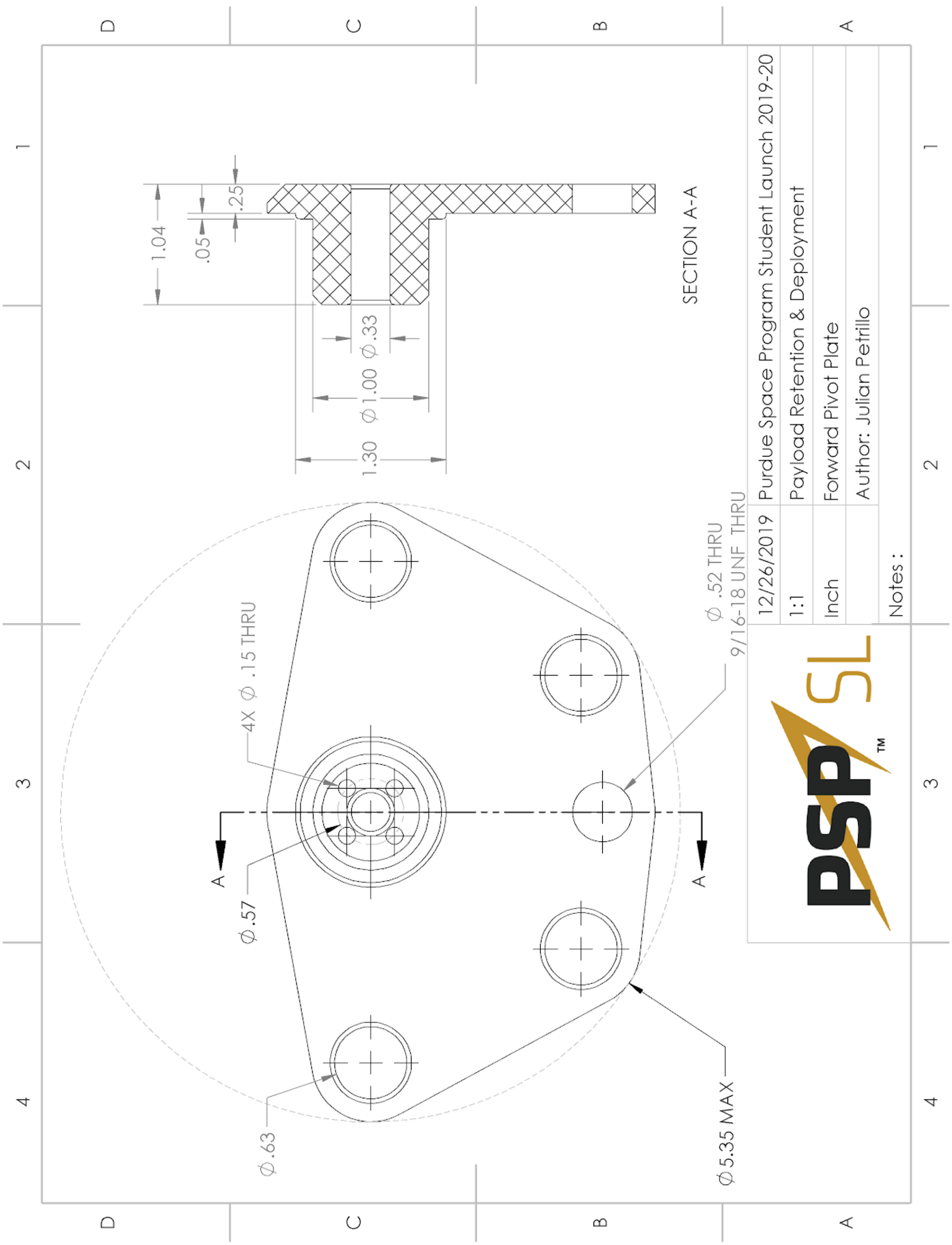
Date	Event	Date	Event
08/22/2019	NASA Releases 2020 Student Launch Handbook	01/13/2020	CDR video teleconferences start
09/01/2019	Purdue SL general meeting	01/11/2020-01/12/2020	Tentative Indiana Rocketry Launch
09/02/2019	LABOR DAY	01/19/2020	Purdue SL general meeting
09/07/2019-09/08/2019	Indiana Rocketry Launch	01/20/2020	MARTIN LUTHER KING JR. DAY
09/08/2019	Purdue SL general meeting	01/22/2020	CDR video teleconferences end
09/15/2019	Purdue SL general meeting	01/26/2020	Purdue SL general meeting
09/18/2019	Proposal due to project office by 3PM CDT	01/31/2020	FRR Q&A
09/22/2019	Purdue SL general meeting	02/02/2020	Purdue SL general meeting
09/29/2019	Purdue SL general meeting	02/09/2020	Purdue SL general meeting
10/03/2019	Awarded proposals announced	02/08/2020-02/09/2020	Tentative Indiana Rocketry Launch
10/06/2019	Purdue SL general meeting	02/16/2020	Purdue SL general meeting
10/07/2019-10/08/2019	OCTOBER BREAK	02/23/2020	Purdue SL general meeting
10/09/2019	Kickoff, PDR Q&A	03/01/2020	Purdue SL general meeting
10/12/2019-10/13/2019	Indiana Rocketry Launch @ Tab, Indiana	03/01/2020	Final day for full scale launch/Vehicle Demonstration Flight
10/13/2019	Purdue SL general meeting	03/02/2020	Vehicle Demonstration Flight data reported to NASA
10/20/2019	Purdue SL general meeting	03/02/2020	FRR report, slides, and flysheet posted

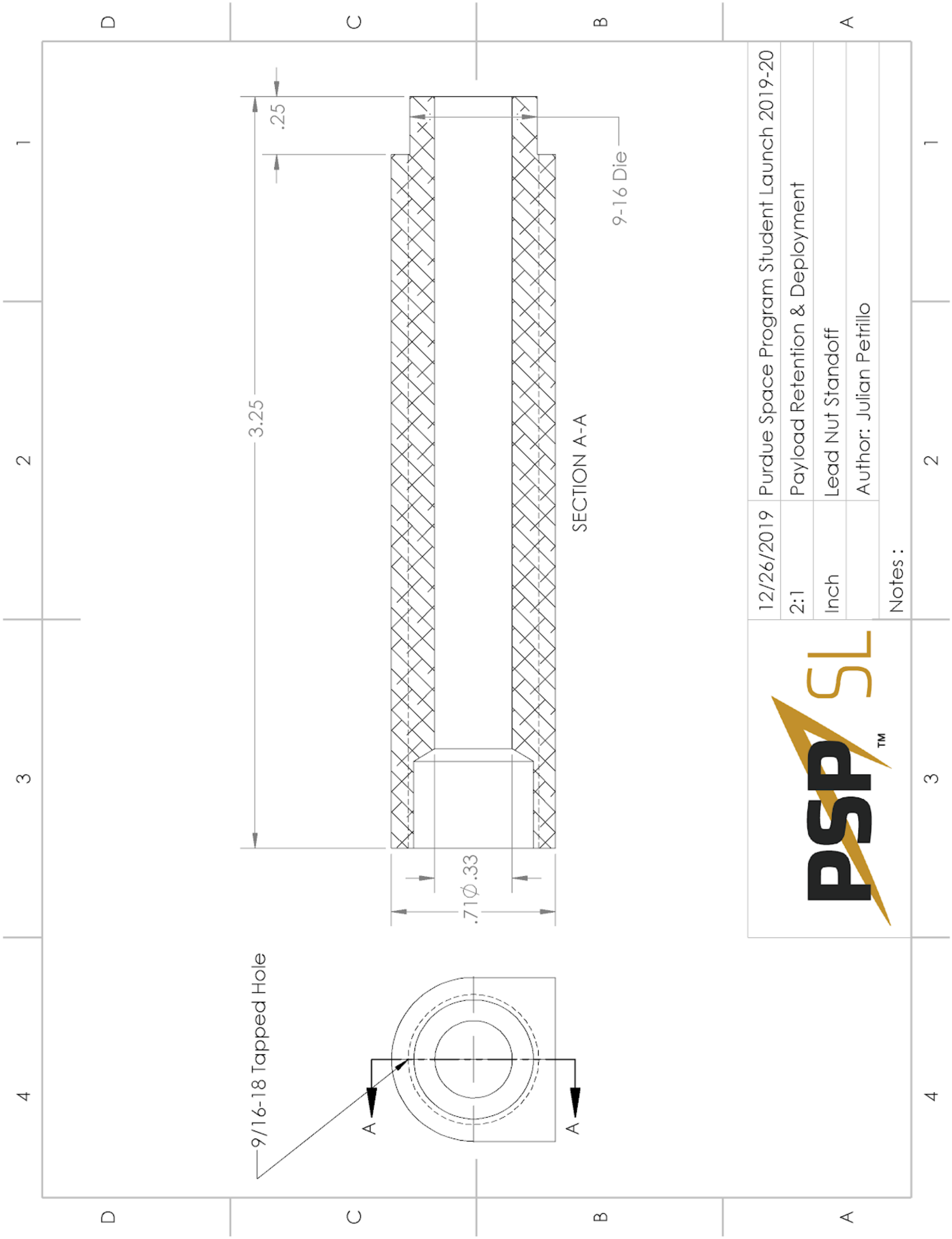
			online by 8AM CDT
10/25/2019	Web presence established, URLs sent to project office by 8AM CDT	03/06/2020	FRR video teleconferences start
10/27/2019	Purdue SL general meeting	03/08/2020	Purdue SL general meeting
11/01/2019	PDR report, slides, and flysheet posted online by 8AM CDT	03/14/2020-03/15/2020	Tentative Indiana Rocketry Launch
11/03/2019	Purdue SL general meeting	03/15/2020	Possible Purdue SL general meeting
11/04/2019	PDR video teleconferences start	03/16/2020-03/21/2020	SPRING BREAK
11/07/2019-11/10/2019	SEDS SpaceVision in Tempe, Arizona	03/19/2020	FRR video teleconferences end
11/09/2019-11/10/2019	Tentative Indiana Rocketry Launch	03/22/2020	Purdue SL general meeting
11/10/2019	Purdue SL general meeting	03/23/2020	Payload Demo Flight/Vehicle Demonstration Re-flight deadlines
11/15/2019-11/17/2019	Midwest Power Launch @ Princeton, Illinois	03/23/2020	FRR Addendum submitted to NASA by 8:00 AM CDT (if needed)
11/17/2019	Purdue SL general meeting	03/26/2020	Launch Week Q&A
11/20/2019	PDR video teleconferences end	03/29/2020	Purdue SL general meeting
11/24/2019	PSP-SL Subscale Launch	04/01/2020	Travel to Huntsville, Alabama
11/25/2019	CDR Q&A	04/01/2020	OPTIONAL – LRR for teams arriving early
11/27/2019-11/30/2019	THANKSGIVING BREAK	04/02/2020	Official launch week kickoff and activities
12/01/2019	Purdue SL general meeting	04/02/2020	LRR (If not done on 04/01)
12/07/2019-12/08/2019	Tentative Indiana Rocketry Launch	04/03/2020	Launch week activities
12/08/2019	Purdue SL general meeting	04/04/2020	Launch day
12/14/2019-01/13/2020	WINTER BREAK	04/04/2020	Awards Ceremony
01/10/2020	Final day for subscale launch	04/05/2020	Backup launch day
01/10/2020	Final motor choice made for launch	04/05/2020	Possible Purdue SL general meeting
01/12/2020	Possible Purdue SL general meeting	04/12/2020	Purdue SL general meeting
01/10/2020	CDR report, slides, and flysheet posted online by 8AM CDT	04/19/2020	Purdue SL general meeting
01/12/2020	Purdue SL general meeting	04/27/2020	PLAR posted online by 8AM CDT

Table 8.14: Project Timeline

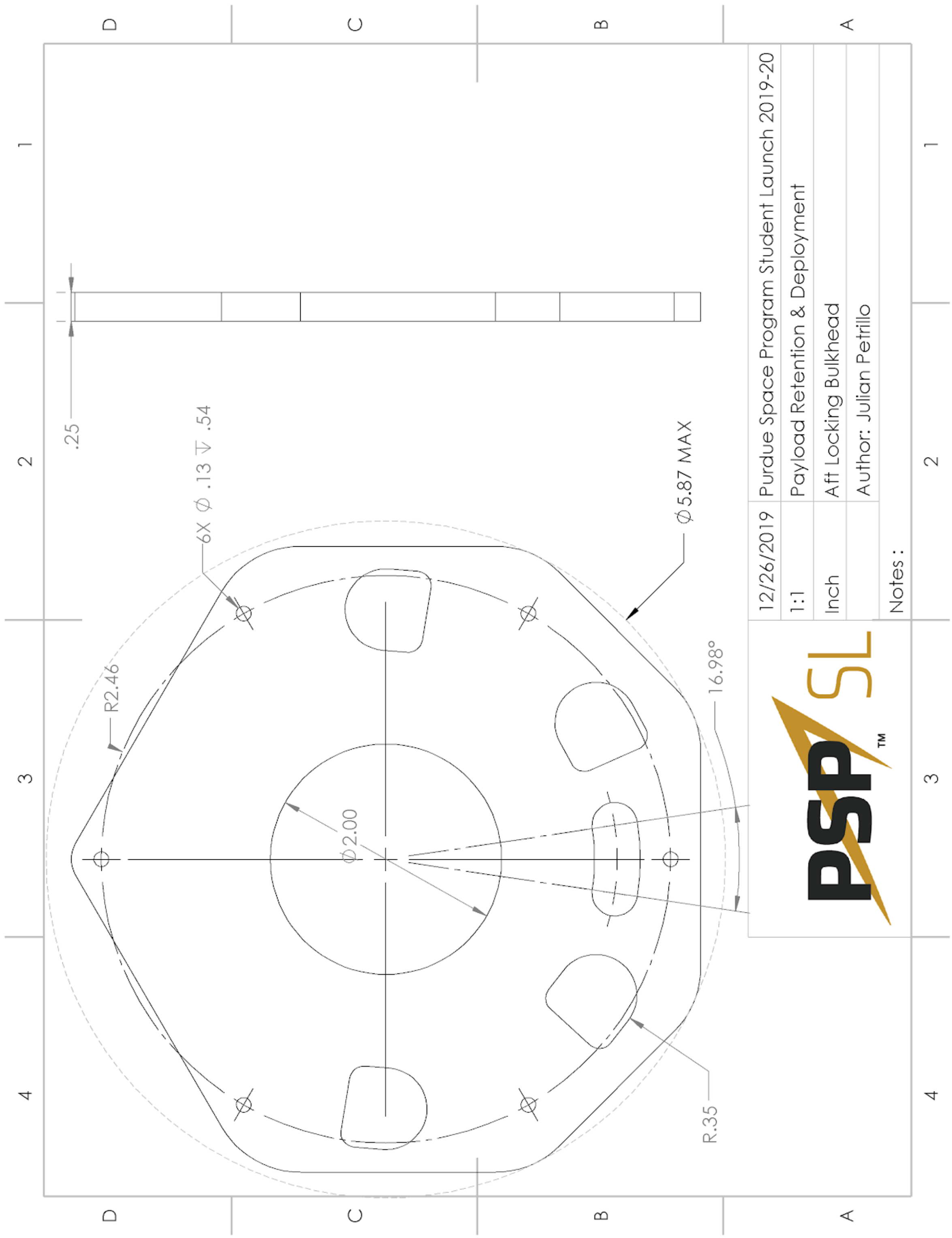
9. Appendix A

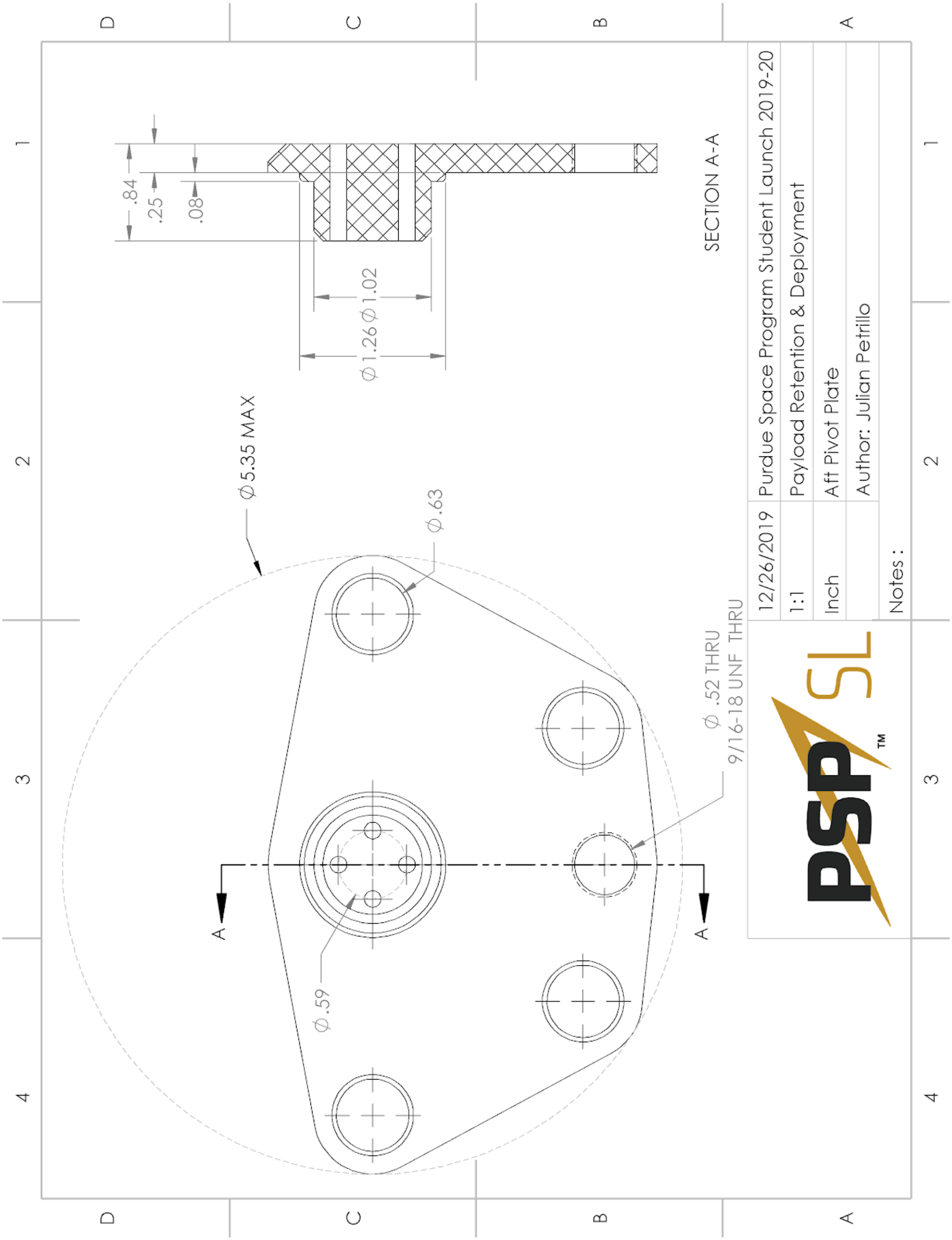
9.1. Additional Dimensional Drawings - Payload System





.25
3.25
9-16 Die
SECTION A-A
.71
Ø.33
9/16-18 Tapped Hole
A
A
Purdue Space Program Student Launch 2019-20
Payload Retention & Deployment
Lead Nut Standoff
Author: Julian Petrillo
12/26/2019
2:1
Inch
Notes :
PSP SL


12/26/2019
Purdue Space Program Student Launch 2019-20
1:1
Payload Retention & Deployment
Inch
Aft Locking Bulkhead
Author: Julian Petrillo
Notes :
.25
6X ⌀ .13 ↧ .54
⌀5.87 MAX
R2.46
⌀2.00
16.98°
R.35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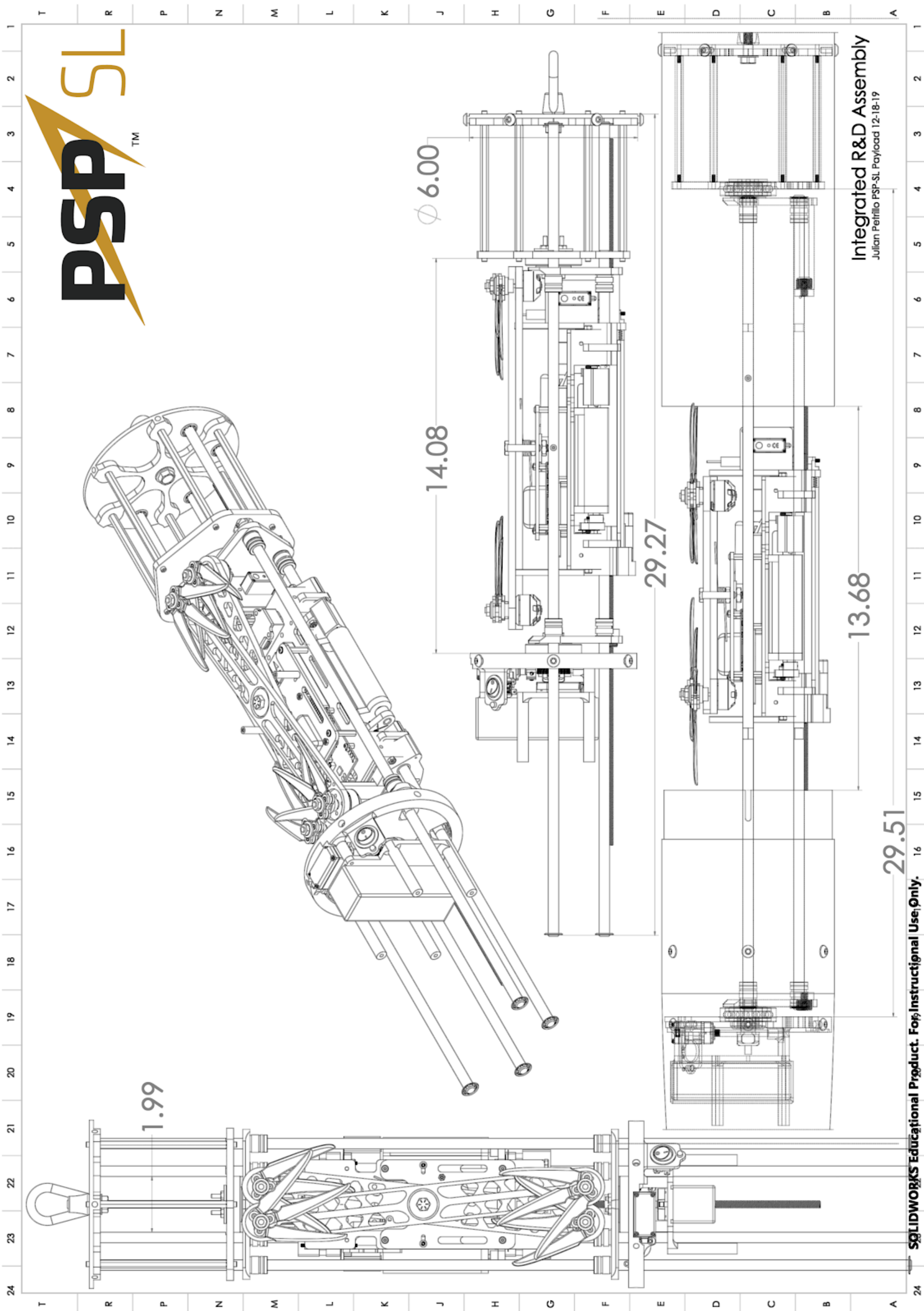
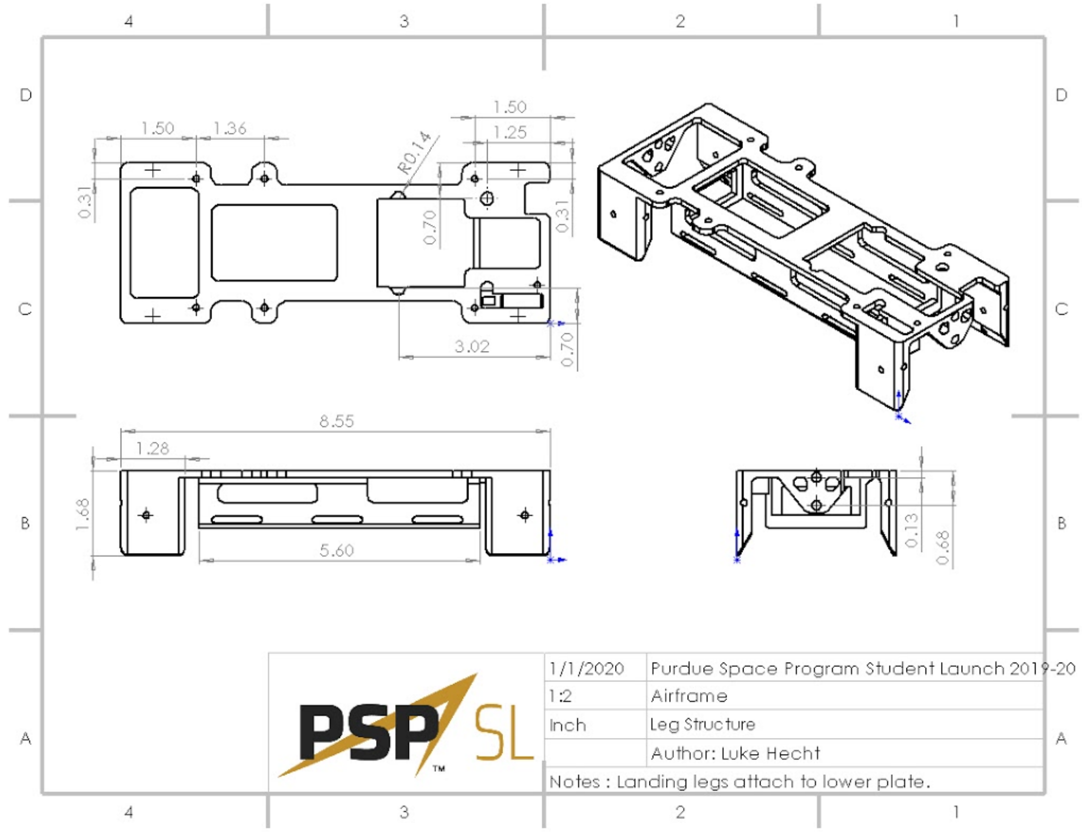
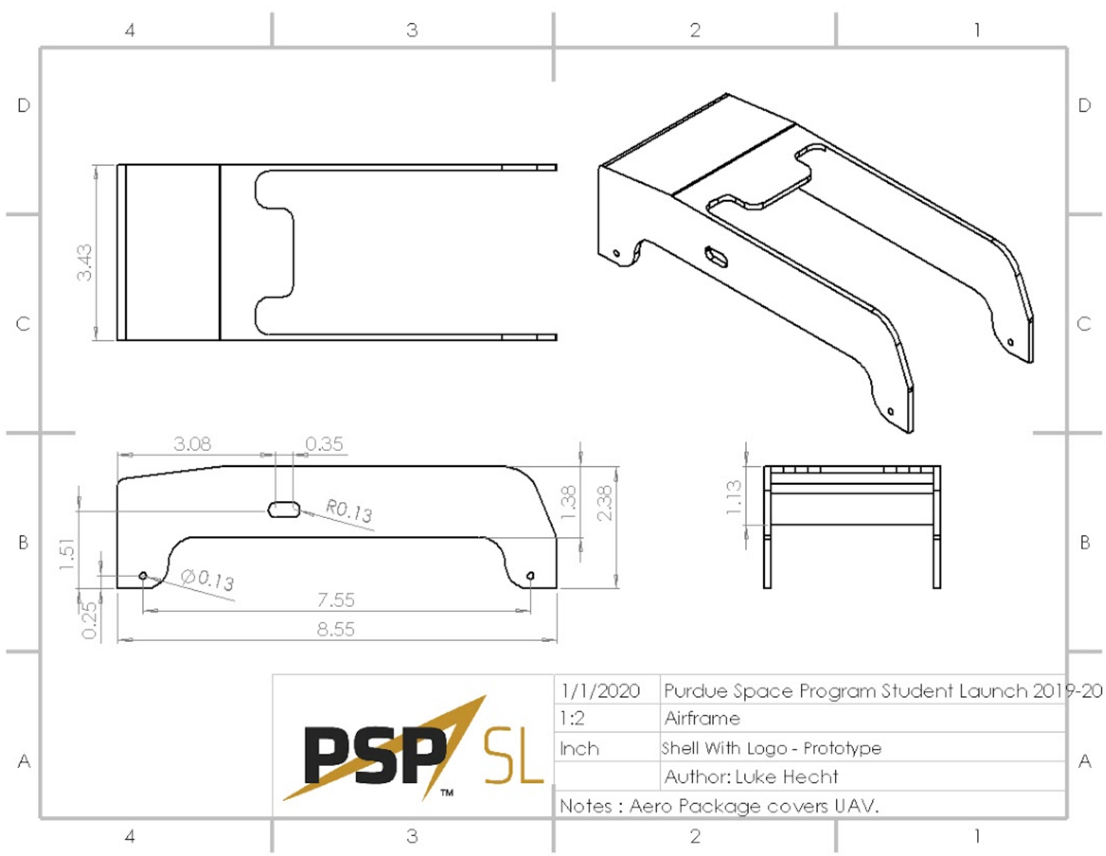
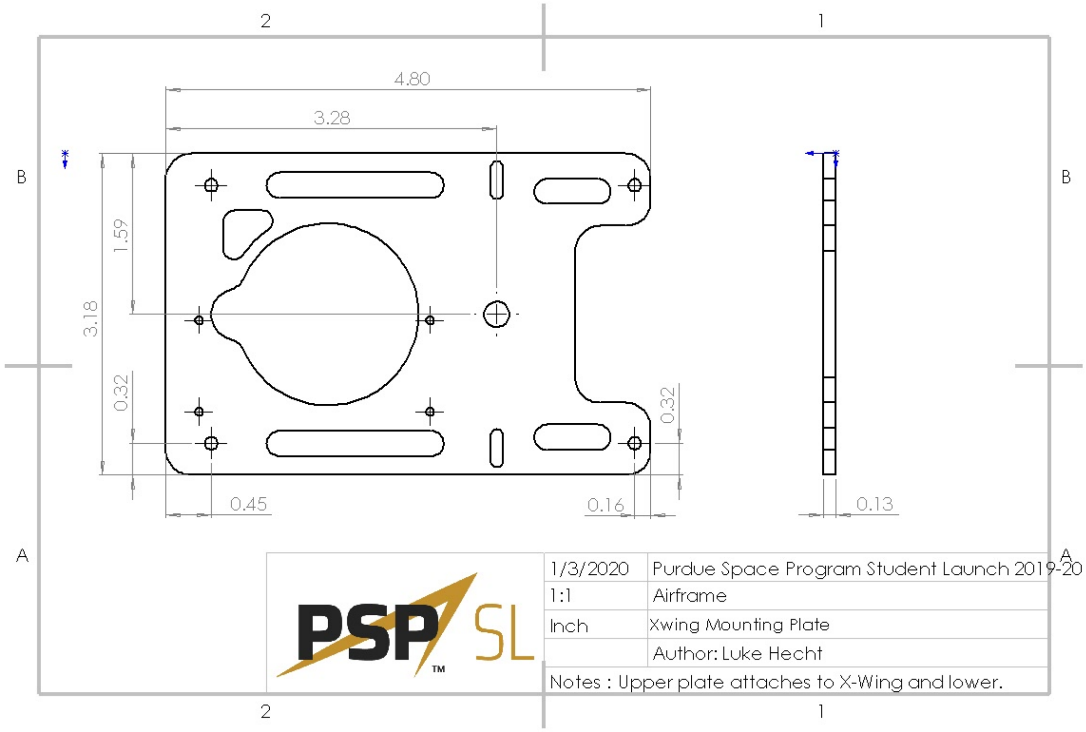
.84
.25
.08
Ø1.26
Ø1.02
Ø5.35 MAX
Ø.63
Ø.52 THRU
9/16-18 UNF THRU
Ø.59
A
A
Purdue Space Program Student Launch 2019-20
Payload Retention & Deployment
Aft Pivot Plate
Author: Julian Petrillo
12/26/2019
1:1
Inch
Notes :

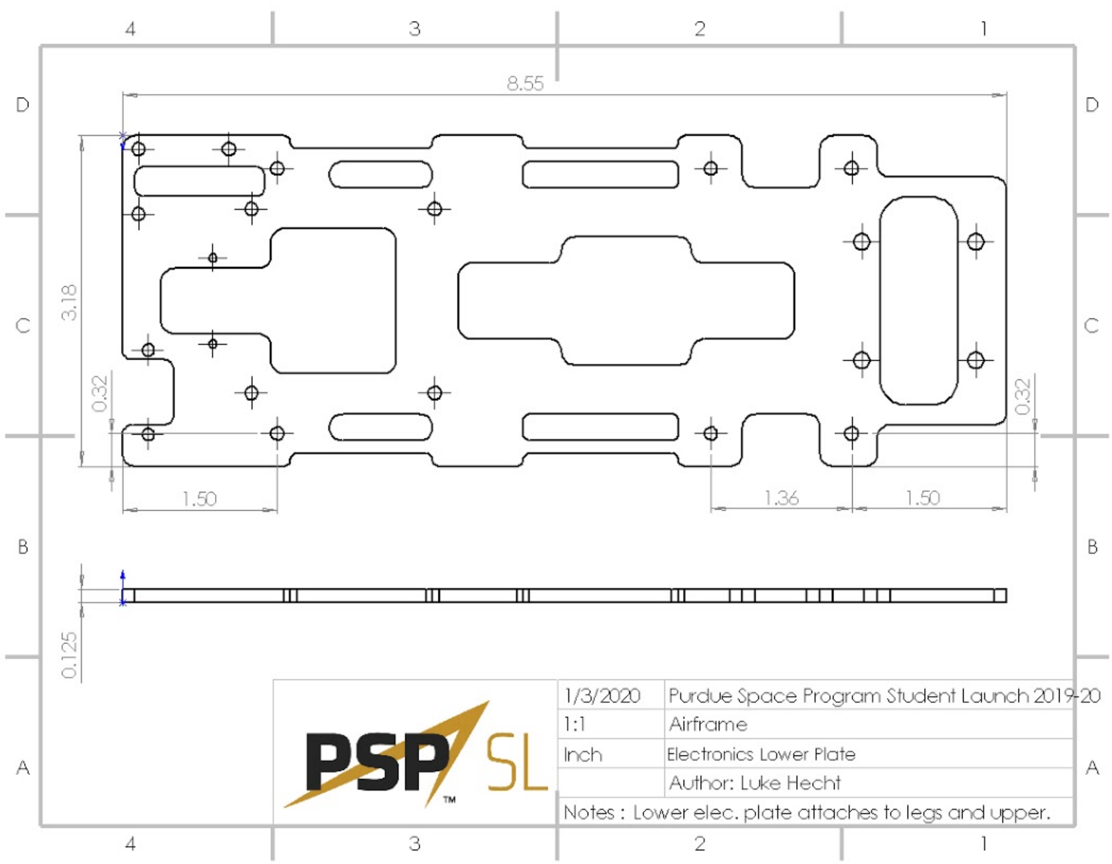

Integrated R&D Assembly
Julian Petrillo PSP-SL Payload 12-18-19
⌀6.00
14.08
29.27
13.68
29.51
1.99
SOLIDWORKS Educational Product. For Instructional Use Only.


1.50
1.36
1.50
1.25
R0.14
0.31
0.70
0.31
3.02
0.70
8.55
1.28
1.68
5.60
0.13
0.68
1/1/2020
Purdue Space Program Student Launch 2019-20
1:2
Airframe
Inch
Leg Structure
Author: Luke Hecht
Notes : Landing legs attach to lower plate.


3.43
3.08
0.35
R0.13
1.38
2.38
1.13
1.51
Ø0.13
0.25
7.55
8.55
1/1/2020
Purdue Space Program Student Launch 2019-20
1:2
Airframe
Inch
Shell With Logo - Prototype
Author: Luke Hecht
Notes : Aero Package covers UAV.



4.80
3.28
1.59
3.18
0.32
0.32
0.45
0.16
0.13
1/3/2020
Purdue Space Program Student Launch 2019-20
1:1
Airframe
Inch
Xwing Mounting Plate
Author: Luke Hecht
Notes : Upper plate attaches to X-Wing and lower.



8.55
3.18
0.32
0.32
1.50
1.36
1.50
0.125
1/3/2020
Purdue Space Program Student Launch 2019-20
1:1
Airframe
Inch
Electronics Lower Plate
Author: Luke Hecht
Notes : Lower elec. plate attaches to legs and upper.

10. Appendix B

10.1. NAR High Power Rocket Safety Code

- Certification. I will only fly high power rockets or possess high power rocket motors that are within the scope of my user certification and required licensing.
- Materials. I will use only lightweight materials such as paper, wood, rubber, plastic, fiberglass, or when necessary ductile metal, for the construction of my rocket.
- Motors. I will use only certified, commercially made rocket motors, and will not tamper with these motors or use them for any purposes except those recommended by the manufacturer. I will not allow smoking, open flames, nor heat sources within 25' of these motors.
- Ignition System. I will launch my rockets with an electrical launch system, and with electrical motor igniters that are installed in the motor only after my rocket is at the launch pad or in a designated prepping area. My launch system will have a safety interlock that is in series with the launch switch that is not installed until my rocket is ready for launch, and will use a launch switch that returns to the "off" position when released. The function of onboard energetics and firing circuits will be inhibited except when my rocket is in the launching position.
- Misfires. If my rocket does not launch when I press the button of my electrical launch system, I will remove the launcher's safety interlock or disconnect its battery, and will wait 60 seconds after the last launch attempt before allowing anyone to approach the rocket.
- Launch Safety. I will use a 5-second countdown before launch. I will ensure that a means is available to warn participants and spectators in the event of a problem. I will ensure that no person is closer to the launch pad than allowed by the accompanying Minimum Distance Table. When arming onboard energetics and firing circuits I will ensure that no person is at the pad except safety personnel and those required for arming and disarming operations. I will check the stability of my rocket before flight and will not fly it if it cannot be determined to be stable. When conducting a simultaneous launch of more than one high power rocket I will observe the additional requirements of NFPA 1127.
- Launcher. I will launch my rocket from a stable device that provides rigid guidance until the rocket has attained a speed that ensures a stable flight, and that is pointed to within 20 degrees of vertical. If the wind speed exceeds 5 miles per hour I will use a launcher length that permits the rocket to attain a safe velocity before separation from the launcher. I will use a blast deflector to prevent the motor's exhaust from hitting the ground. I will ensure that dry grass is cleared around each launch pad in accordance with the accompanying Minimum Distance table, and will increase this distance by a factor of 1.5 and clear that area of all combustible material if the rocket motor being launched uses titanium sponge in the propellant.
- Size. My rocket will not contain any combination of motors that total more than 40,960 N-sec (9208 pound-seconds) of total impulse. My rocket will not weigh more at liftoff than one-third

of the certified average thrust of the high power rocket motor(s) intended to be ignited at launch.

- Flight Safety. I will not launch my rocket at targets, into clouds, near airplanes, nor on trajectories that take it directly over the heads of spectators or beyond the boundaries of the launch site, and will not put any flammable or explosive payload in my rocket. I will not launch my rockets if wind speeds exceed 20 miles per hour. I will comply with Federal Aviation Administration airspace regulations when flying, and will ensure that my rocket will not exceed any applicable altitude limit in effect at that launch site.
- Launch Site. I will launch my rocket outdoors, in an open area where trees, power lines, occupied buildings, and persons not involved in the launch do not present a hazard, and that is at least as large on its smallest dimension as one-half of the maximum altitude to which rockets are allowed to be flown at that site or 1500’, whichever is greater, or 1000’ for rockets with a combined total impulse of less than 160 N-sec, a total liftoff weight of less than 1500 grams, and a maximum expected altitude of less than 610 meters (2000’).
- Launcher Location. My launcher will be 1500’ from any occupied building or from any public highway on which traffic flow exceeds 10 vehicles per hour, not including traffic flow related to the launch. It will also be no closer than the appropriate Minimum Personnel Distance from the accompanying table from any boundary of the launch site.
- Recovery System. I will use a recovery system such as a parachute in my rocket so that all parts of my rocket return safely and undamaged and can be flown again, and I will use only flame-resistant or fireproof recovery system wadding in my rocket.
- Recovery Safety. I will not attempt to recover my rocket from power lines, tall trees, or other dangerous places, fly it under conditions where it is likely to recover in spectator areas or outside the launch site, nor attempt to catch it as it approaches the ground.

10.2. NAR Minimum Distance Table

Installed Total Impulse (Newton-Seconds)	Equivalent High Power Motor Type	Minimum Diameter of Cleared Area (ft.)	Minimum Personnel Distance (ft.)	Minimum Personnel Distance (Complex Rocket) (ft.)
0 — 320.00	H or smaller	50	100	200
320.01 — 640.00	I	50	100	200
640.01 — 1,280.00	J	50	100	200
1,280.01 — 2,560.00	K	75	200	300
2,560.01 — 5,120.00	L	100	300	500
5,120.01 — 10,240.00	M	125	500	1000
10,240.01 — 20,480.00	N	125	1000	1500
20,480.01 — 40,960.00	O	125	1500	2000

Note: A Complex rocket is one that is multi-staged or that is propelled by two or more rocket motors

11. Appendix C

11.1. Pre-Launch Packing Lists

Items to be packed before leaving for launch site

Rocket body components

- ❏ Nose cone
- ❏ Upper airframe
- ❏ Lower airframe
- ❏ Avionics sled
- ❏ Payload retention system
- ❏ Rocket Motors
- ❏ Rocket Motor tubes

Payload

- ❏ General
  - ❏ Laptop with payload software suite
  - ❏ Laptop chargers
  - ❏ Multimeter
  - ❏ Jumper wires
  - ❏ Electronic tweezers
  - ❏ Wire strippers
  - ❏ Soldering iron
  - ❏ Solder
  - ❏ Solder wick
  - ❏ Solder helping hands
  - ❏ Stranded core wire
  - ❏ Solid core wire
  - ❏ LiPo charger
  - ❏ Phillips screwdriver
  - ❏ Small flathead screwdriver
  - ❏ Breadboard
  - ❏ Heat shrink kit
  - ❏ Electrical tape
  - ❏ Resistor kit
  - ❏ IC kit (diodes, transistors, etc.)
  - ❏ GCS Assembly
  - ❏ GCS batteries
  - ❏ GCS power adapter
  - ❏ Spare GCS buttons
  - ❏ Spare "missile" switches
  - ❏ Spare LCD display
  - ❏ Handheld GPS locator
  - ❏ 9V batteries
  - ❏ Pliers
  - ❏ Allen Key Set
  - ❏ RC Transmitter
  - ❏ RC Transmitter Battery
  - ❏ FAA Registration
  - ❏ Spare 32GB FAT32 SD Cards
  - ❏ Spare Propellers CW/CCW
  - ❏ Spare Propeller Hubs
  - ❏ Zipties
  - ❏ Double Sided Foam Tape
  - ❏ JST XH Connecters
  - ❏ Wire Crimper
  - ❏ Micro USB Cables
  - ❏ Loctite
  - ❏ Nuts/Bolts M3, 4-40
- ❏ UAV
  - ❏ UAV leg assembly
  - ❏ UAV electronics stack assembly
  - ❏ UAV X-Wing assembly
  - ❏ Airframe shell
  - ❏ Battery housing
  - ❏ Propellers
  - ❏ Propeller locking nuts
  - ❏ Brushless DC motors
  - ❏ Ice mining assembly
  - ❏ Ice mining DC motor
  - ❏ Pixhawk 4
  - ❏ Pixhawk PDU
  - ❏ Raspberry Pi Zero
  - ❏ Raspberry Pi camera
  - ❏ Raspi camera ribbon cable
  - ❏ LiDAR unit
  - ❏ GPS module
  - ❏ 915 MHz radio
  - ❏ Pixhawk wire kit
  - ❏ ESC
  - ❏ Spare bullet connectors

- ❏ LiPo battery
- ❏ Taranis Q X7

❏ Retention

- ❏ Upper airframe
- ❏ Nosecone
- ❏ Half Coupler w/ slot
- ❏ ¼-20 x ⅝” fasteners
- ❏ ¼-20 x ⅜” fasteners
- ❏ 4-40 x ½” fasteners
- ❏ 4-40 x ⅝”
- ❏ ¼-20 Nut
- ❏ 6-32 Nut
- ❏ 6-32 x ½”
- ❏ 6-32 x 1⅜”
- ❏ 4-40 x ¼”
- ❏ Long-body 4-40 insert nuts
- ❏ Flat-body ¼-20 press fit nuts
- ❏ Rotation Lock, L/R
- ❏ Rocket Stand
- ❏ Push-on retaining rings
- ❏ Lead nuts (left and right)
- ❏ 2" OD ball bearing
- ❏ Fixed alignment sleeve bearing
- ❏ Self-aligning sleeve bearing
- ❏ Notched ⅜” shafts
- ❏ 3D printed rotary nut plate w/ pass through
- ❏ HSR-2648CR Continuous Servo
- ❏ Actobotics shaft-servo coupler
- ❏ ¼” D shaft
- ❏ Worm gear assembly
- ❏ Rocker switch
- ❏ LED
- ❏ 3D printed nose cone interface bulkplate
- ❏ 3D printed battery housing
- ❏ 2200 mAh LiPo battery
- ❏ 6mm shaft encoder
- ❏ 3D printed bearing clamp collar
- ❏ 3D printed sled-encoder adapter
- ❏ Acrylic stationary retention plate
- ❏ 3D printed rotary nut plate w/o pass through
- ❏ 3D printed nut extender
- ❏ Aluminum bulkplate
- ❏ Long bulkplate standoff
- ❏ Eye nut
- ❏ M8 black-oxide bolt
- ❏ 3D printed bearing clamp plate
- ❏ Stepper motor
- ❏ Stepper motor driver
- ❏ R&D electronics PCB

Construction

- ❏ Drill
- ❏ Drill battery (charged and packed or charging)
- ❏ Drill bits (5/64”, 3/16”, 1/8”)
- ❏ Sandpaper
- ❏ Tape (duct tape and painter’s tape)
- ❏ Pliers (needle nose and regular)
- ❏ Screwdrivers (flathead and Phillips)
- ❏ Wrench set
- ❏ Allen key set
- ❏ 5-minute epoxy
- ❏ Acetone
- ❏ Paper towels
- ❏ Shear pins
- ❏ Extra eye bolts
- ❏ Extra nuts/washers
- ❏ Extra motor retainer
- ❏ Extra shock cord
- ❏ Extra rail button
- ❏ Threadlock
- ❏ Rotary tool
- ❏ Charged rotary tool battery

Safety team box

- ❏ Face masks
- ❏ Eye protection
- ❏ Ear protection
- ❏ Nitrile gloves
- ❏ Pocket rocket documents
- ❏ Marking flags
- ❏ First aid kit

Team toolbox

Avionics

- ❏ Assembled avionics bay
- ❏ Coupler

- Switch band
- (3) switch holders
- (3) rocker switches
- Camera mount
- Camera
- (6) 440 shear pins
- (2) bulkheads
  - (4) terminal blocks
  - (4) charge wells
  - (2) I-bolts
  - (8) 440 screws
  - (8) 440 hex nuts
  - (2) ¼-20 hex nuts
  - (2) ¼” washers
- (2) threaded rods
- (16) ¼-20 hex nuts
- (4) ¼” washers
- Avionics sled assembly
  - Altimeter sled 3D printed part
  - Battery guard 3D printed part
  - Telemetrum altimeter
  - RRC3+ Sport altimeter
  - 900 mAh 3.7V LiPo battery
  - 9V alkaline battery
  - 9V battery connector
  - (8) altimeter mounting posts
  - (8) 440 nylon screws
- (6) switch wires
- (4) e-matches
- (4) nitrile glove fingers
- 30g black powder
- Dog barf
- Aluminum foil
- Tape (masking or duck)

- (2) ¼” SS quick links
- 20’ long, ½” tubular nylon shock cord
- 40’ long, ½” tubular nylon shock cord
- Fruity Chutes Classic Elliptical drogue parachute
- Skyangle Cert-3 XXL main parachute
- (2) Nomex blankets
- Epoxy
- 9V batteries
- 9V battery connectors
- 900 mAh LiPo batteries
- Altimeter mounting posts/screws
- Telemetrum terminal blocks
- Switches
- 28 gauge red wire
- 28 gauge black wire
- Terminal blocks
- Black powder canisters
- ¼-20 hex nuts
- ¼” washers
- 440 screws
- 440 hex nuts
- 440 shear pins
- E-matches
- Long e-matches
- Nitrile gloves
- Black powder
- Dog barf
- Aluminum foil
- Tape (masking or duck)
- ¼” SS quick links
- Epoxy
- Skyangle Cert-3 XL main parachute
- Skyangle Cert-3 Drogue drogue parachute
- TeleDongle
- Yagi 3 Arrow Antenna
- SMA to BNC adapter
- Screwdriver set
- Hex wrench set
- Masking tape
- Duck tape
- Scissors
- Soldering iron
- Soldering material
- RRC3+ Sport manual**** (critical)
- Telemetrum manual**** (critical)
- Micro USB → USB cable (for Telemetrum)
- 5-pin joiner → USB interface (for RRC3+ Sport)
- Mini USB → USB cable (for RRC3+ Sport)
- Laptop with AltOS (for Telemetrum) and mDACS (for RRC3+ Sport) installed