The Zenith Program

Flight Readiness Review

FAMU-FSU College of Engineering

2525 Pottsdamer Street

Tallahassee, FL 32310

3/7/2023

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1 Summary

1.1 Team Summary

1.1.1 Team Information

1.1.1.1 Team Name

This team has dedicated itself to laying the groundwork for continued yearly participation in NASA Student Launch and expansion into experimental liquid-fueled engine development by the parent AIAA chapter. To that end, the team has deemed itself the first year of FAMU-FSU AIAA's rocket development program, called the Zenith Program.

1.1.1.1 Mailing Address

Mail to: FAMU-FSU AIAA

2525 Pottsdamer Street, Suite B111

Tallahassee, FL 32310

1.1.2 Mentor Information

1.1.2.1 Mr. Tom McKeown

- **Title:** Board Member, Spaceport Rocketry Association (NAR #342 / TRA #73)
- **Email:** mckeownt@ix.netcom.com
- **Phone:** 321-266-1928
- **NAR Flyer Number:** 57205
- **TRA Flyer Number:** 01922
- **NAR/TRA Certification Level:** Level 2

1.1.3 Huntsville Travel Plans

The team intends to attend launch week at NASA MSFC in Huntsville. An alternate launch site has been selected in Palm Bay, FL at the SRA Rocket Ranch should the team not be able to travel to Huntsville.

1.1.4 FRR Completion Time

The team spent approximately 200 hours working on the FRR document.

1.1.5 STEM Engagement Summary

The Team had to engage a minimum of 250 participants through the means of Direct education engagement and have a STEM Engagement activity report submitted by Flight Readiness Review. As far as engagement is concerned the team was able to engage with Sail High School and School of Arts and Science located in Tallahassee. This brought us to a minimum of 280 participants engaged. Sail High School STEM Engagement was done on the 28th of February and March 2nd, this consisted of a set of presentations to four of their engineering and physical science classes with 25-30 students in each class and started discussions about the main aspects of our senior design project and how the simple subjects that are being taught them apply to projects of a collegiate level. As well as give general rundown of the High School Division offered by NASA Student Launch and how many schools are competing as of now. The School of Arts and Sciences STEM Engagement took place on the 23rd and 24th of February which was done due to the large summary of participants, this consisted of 2 groups of 45 on each day. This event involved a small lecture/presentation on Aerodynamics that would eventually transition to a hands-on activity that would better explain the lecture to kids in ways they would understand.

Table 1-1. STEM Event Summary

Event/Group	Participant No.	Engagement Type	Description
Sail High School-Class 1	50	Direct Engagement	Presentation
Sail High School-Class 2	50	Direct Engagement	Presentation
School of Arts and Sciences	180	Direct Engagement	Aero Lecture/Hands on Activity
Present Total:	280	**-**	**-**

1.2 Launch Vehicle Summary

1.2.1 Target Altitude
The target altitude for the Zenith 1 is 4600 ft AGL.

1.2.2 Final Motor Selection
The final motor selected is the Aerotech L850W.

Table 1-1. Aerotech L850W Specifications

Motor Parameter	Value
Average Thrust	850 N
Initial Thrust	1,001 N
Maximum Thrust	1,866 N
Total Impulse	3,646 N-s
Burn Time	4.4 seconds

1.2.3 Vehicle Sections
The flight vehicle design is 97.75 inches in length with a body tube diameter of 6.12 inches and a total weight of 38.68 lbs. The static stability margin of the vehicle is 3.43 calibers and the max velocity the vehicle reaches is 545 ft/s. Section 3.1 expands further into the design of the vehicle, its subsections, and their weight distribution.

1.2.4 Vehicle Mass

Table 1-2. Vehicle Masses

Vehicle State	Mass (lbm)	Mass (g)
Dry Mass	34.12	15273
Wet Mass	38.68	17520

Component	Size (inches)	Weight (lbs.)
Upper Payload Bay	35	12.718
Nose Cone	20	5.297
Avionics Bay	12	4.822
Fin Can w/out propellants	40.05	14.341

1.2.5 Recovery System

The recovery system consists of two completely independent Entacore AIM 3 altimeters. At apogee, a 24” high strength elliptical parachute will deploy. At around 550 feet above ground an 84” toroidal shaped parachute will deploy. The system will use CO2 ejection charges to lower the risk of damaging any of the components in each payload bay, which is more likely to occur with the use of traditional black powder charges. The recovery system and all of its relevant modules are discussed further in Section 3.4.

Table 1-3. Recovery System Components

Component	Part Selected
Primary Flight Computer	AIM4 USB
Backup Altimeter	AIM4 USB
GPS Tracker/Locator 1	EggFinder RX
GPS Tracker/Locator 2	Apple AirTag
Ejection Charges	Tinder Rocketry Raptor CO2 Ejection Charge
Drogue Parachute	Fruity Chutes Classic Elliptical 24”
Main Parachute	Fruity Chutes Iris Ultra Standard 84”

1.2.6 Rail Size

The launch vehicle will utilize 1515 rails.

1.3 Payload Summary

The Payload has Summary has stayed the same since the critical design review. The payload is a static 3D printed camera housing, which will contain an Arduino Mega microcontroller, ArduCam mini camera, and stepper motor to drive the rotation of the camera. The camera housing has been designed as a pyramid, with a wide flat base to prevent tipping or rolling as it sets down on the ground under parachute. The microcontroller and camera are mounted to a 3D printed turret, driven by the stepper motor, with the camera protruding from the top of the pyramid into a protective Lexan plexiglass sides of a 3D printed frame. The Payload will be attached to the shock cord via four eye bolts and will be pulled out by the shock cord during upper payload bay separation.

2 Modifications since CDR

2.1 Modified vehicle criteria

Description of Change	Reason for the Change
Aft removable centering ring holds aft rail button	Prevents any movement from the centering ring and assists in fixing the rail button to the vehicle
Fin structure has been changed to eliminate the canted profile. The canted profile has been substituted with a spin tab position at the aft end of the fin and will be printed as one component with the fin (discussed in flight analysis).	The canted profile required the use of a base plate. The base plate resulted in deformation issues and fitment issues with the airframe.
Tail cone has been eliminated. Thrust plate remains in the design.	Tail Cone design was detrimental to vehicle performance and personnel safety (discussed in flight analysis).

2.2 Changes made to payload criteria

Description of Change	Reason for the Change
For the Payload Housing increased the material around the eyebolts to increase strength and decrease the likely hood of breaking	Had an oversite that after printing the Interior Electronics Frame couldn't side into the top.
Changed dimensions of Camera Housing.	Designed the Camera Housing before receiving the part.
Changed design to allow for the Arduino Mega to be put inside the Interior Electronics Frame	Had an oversite in the ability to put the Arduino Mega controller into the Frame after it was printed.
Changed the design of the Lexan top to be more like a picture frame. There is now a 3D printed Frame and have four panes of Lexan epoxied to the inside.	During manufacturing there was trouble molding it to form the correct shape.

2.3 Changes made to project plan

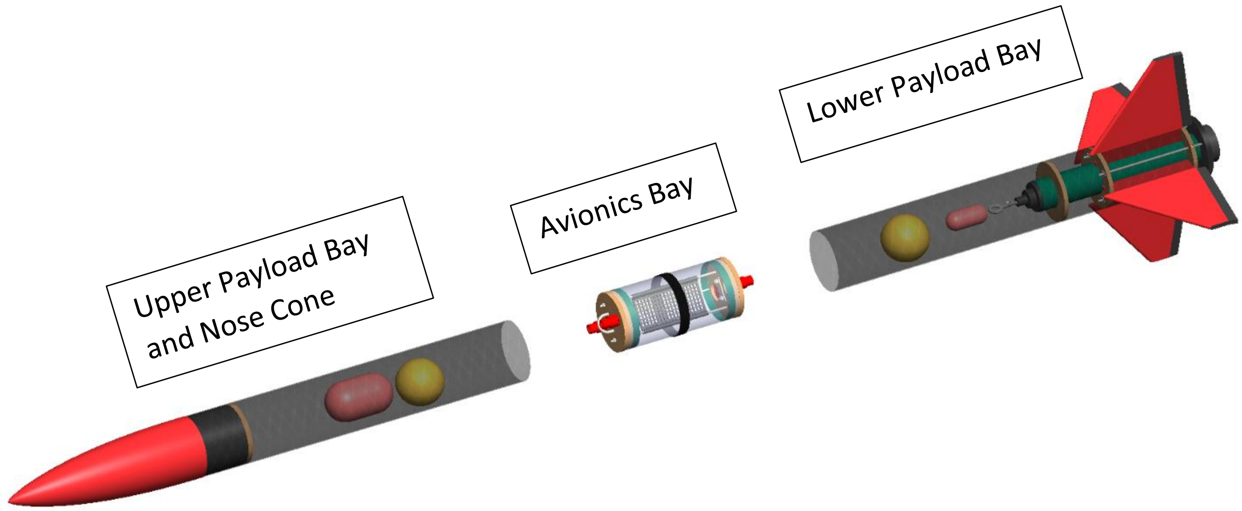
There are no notable changes to the project plan since last reported on. Manufacturing, testing, and test flight timelines are holding firm, STEM engagement was executed as expected, and budgeting and actual expenditures remain reasonably close.

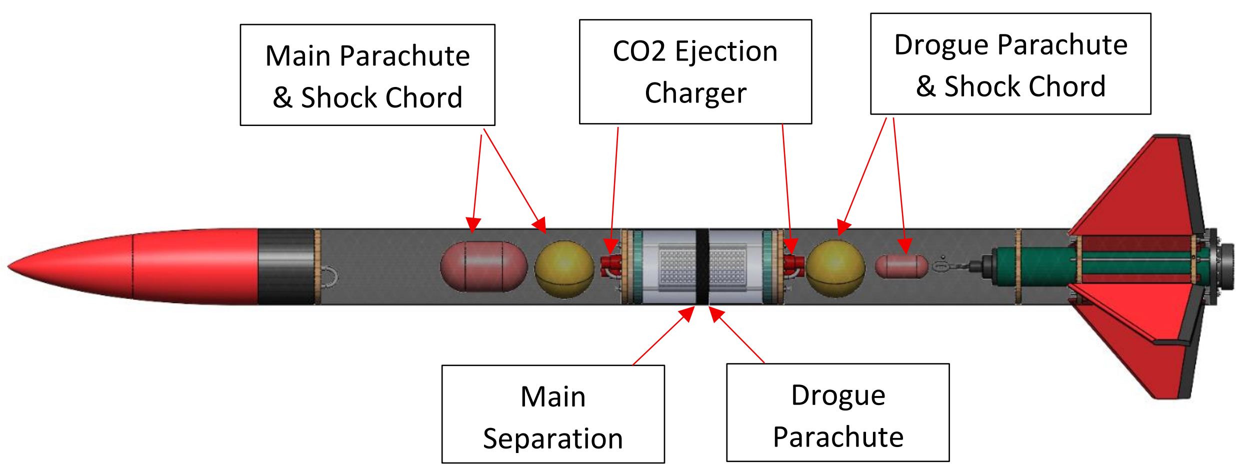
3 Vehicle Criteria

3.1 Design and Construction of Vehicle

3.1.1 Launch Vehicle Design

The launch vehicle will be 98 inches long with a loaded weight of 38.68 pounds and an unloaded weight of 34.12 pounds. Discussed in later sections, changes made to the launch vehicle include fin design and nose cone coupling method. The vehicle will separate between the lower payload bay and the avionics bay at apogee (drogue separation), and between the upper payload bay and avionics bay (main parachute separation). See figure 3.3 for visual representation.


Upper Payload Bay and Nose Cone
Avionics Bay
Lower Payload Bay


Main Parachute & Shock Chord
CO2 Ejection Charger
Drogue Parachute & Shock Chord
Main Separation
Drogue Parachute

3.1.1.1 Nosecone Configuration

The final design for the nosecone configuration has not changed and will be an LD-Haack Series with threaded shoulder attachment.

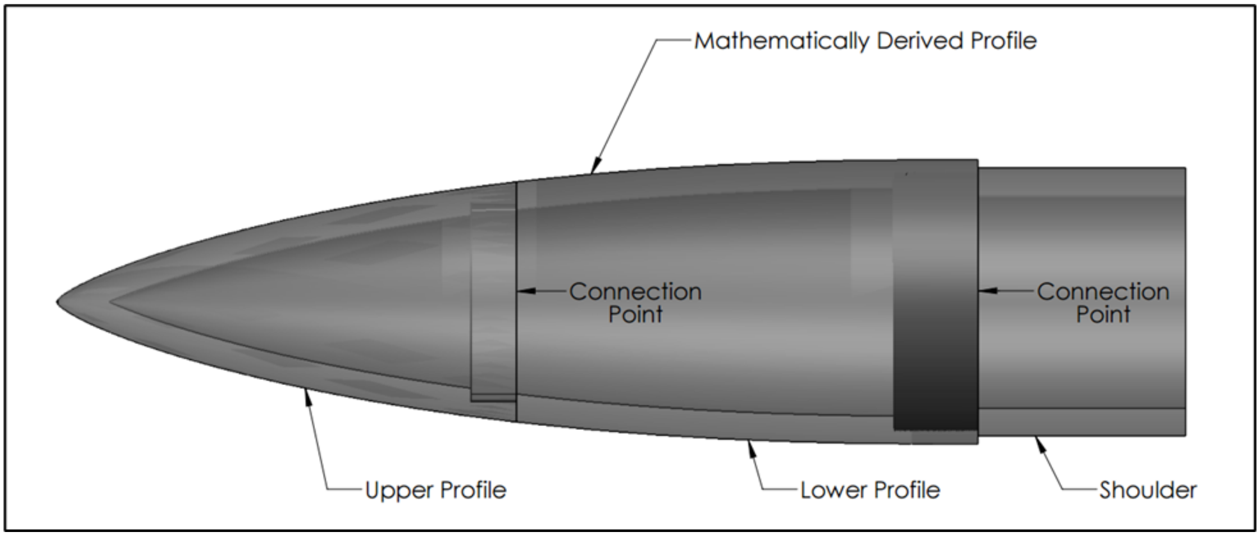


Figure 3-1. LD-Haack Series Nosecone Assembly

The nosecone configuration is separated into three sections: upper profile, lower profile, and shoulder. The configuration of the nosecone was ultimately separated into three sections for ease of manufacturing. Discussed further below in section 5.5.2, the upper profile and lower profile separated due to the coupler connector being too thin to withstand the ground hit impact. The solution to this issue will be to increase the thickness of the coupling connector between the upper and lower profile. In the previous design presented in CDR, the coupling piece between the upper and lower nose cone profiles was one sided and 1 inch long. The updated design shown in the engineering drawings below shows the coupler is now 0.8 inches thick and 2 inches long with 1 inch of the coupler in the lower section and 1 inch in the upper section. 3D printing specifications and notes have also been included in the drawings included in the drawings.

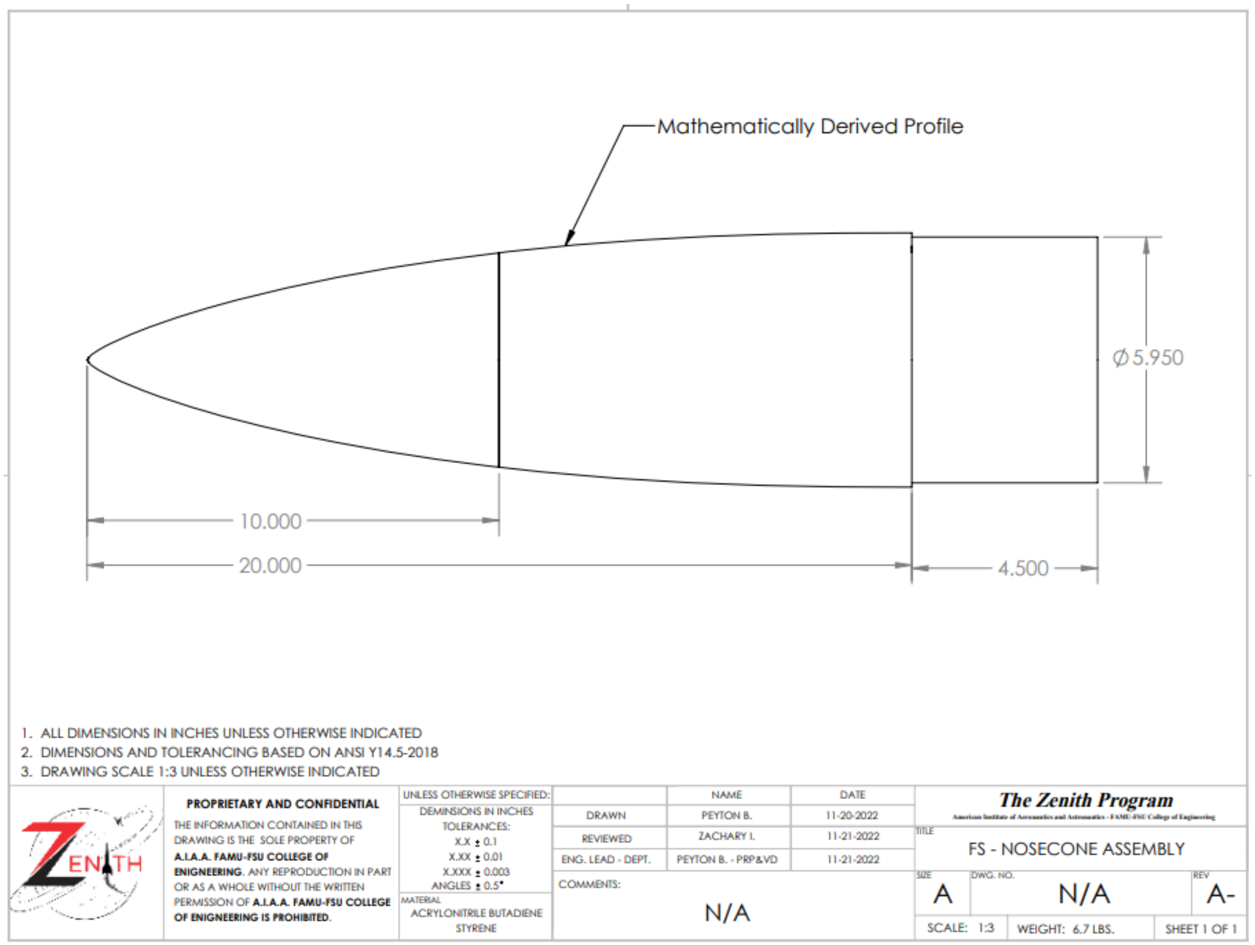
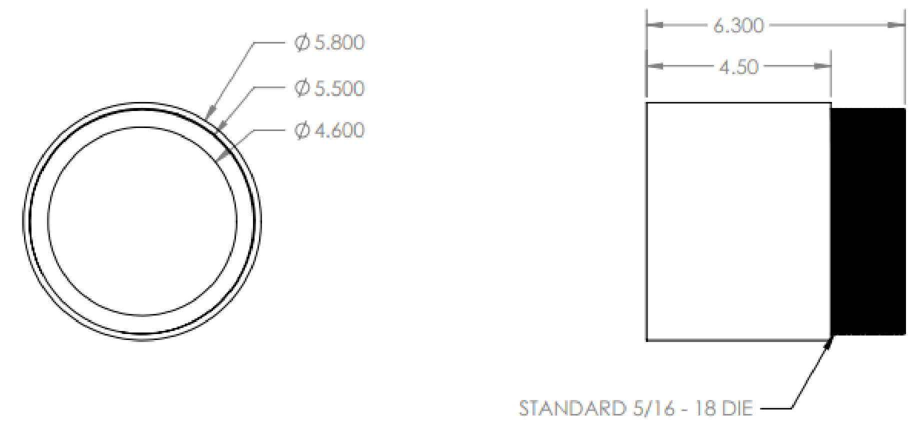


Figure 3-2. LD-Haack Nosecone Assembly CAD Drawing



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2. DIMENSIONS AND TOLERANCING BASED ON ANSI Y14.5-2018
3. DRAWING SCALE 1:3 UNLESS OTHERWISE INDICATED

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INFILL PATTERN: TRIAGULAR
NOZZLE TEMPERATURE: 235°C
BUILD PLATE TEMPERATURE: 110°C
PRINT SPEED: 60 mm/s
SUPPORT STRUCTURE: TREE AT 45° OVERHANG
SUPPORT PATTERN: ZIGZAG
SUPPORT DENSITY: 10%
BUILD PLATE ADHESION TYPE: BRIM

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TOLERANCES:
X.X ± 0.1
X.XX ± 0.01
X.XXX ± 0.003
ANGLES ± 0.5°

MATERIAL
ACRYLONITRILE BUTADIENE STYRENE

	NAME	DATE
DRAWN	PEYTON B.	11-20-2022
REVIEWED	ZACHARY L	11-21-2022
ENG. LEAD - DEPT.	PEYTON B. - PRP&VD	11-21-2022

COMMENTS:
- COMPONENT TO BE PRINTED USING ACRYLONITRILE BUTADIENE FILAMENT **ONLY**

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American Institute of Aeronautics and Astronautics - FAMU-FSU College of Engineering

TITLE: FS - NOSECONE - SHOULDER

SIZE	DWG. NO.	REV
A	N/A	A-

SCALE: 1:3 | WEIGHT: 1.7 LBS. | SHEET 1 OF 1

Figure 3-3. Nosecone Shoulder CAD Drawing

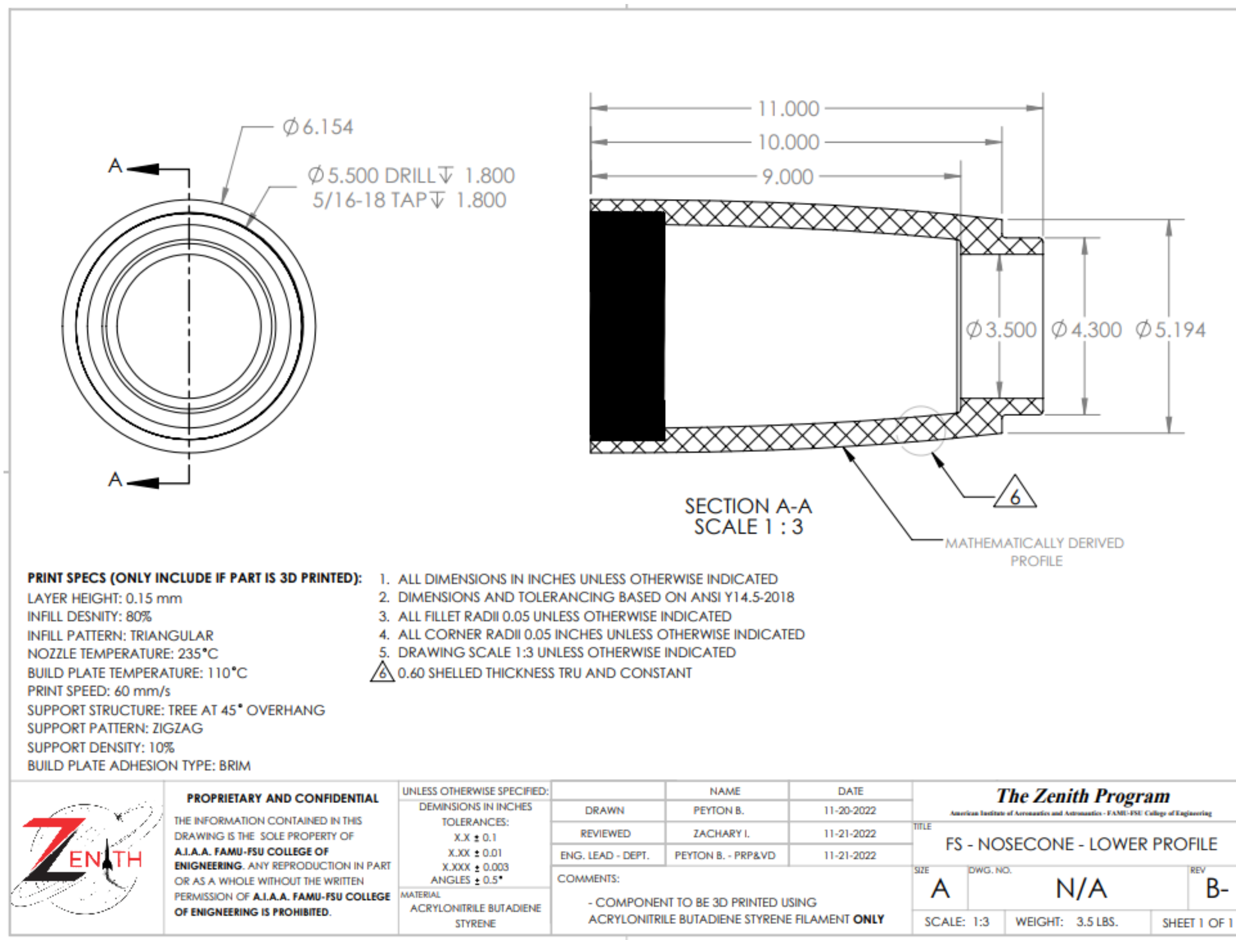


Figure 3-4. Nosecone Lower Profile CAD Drawing

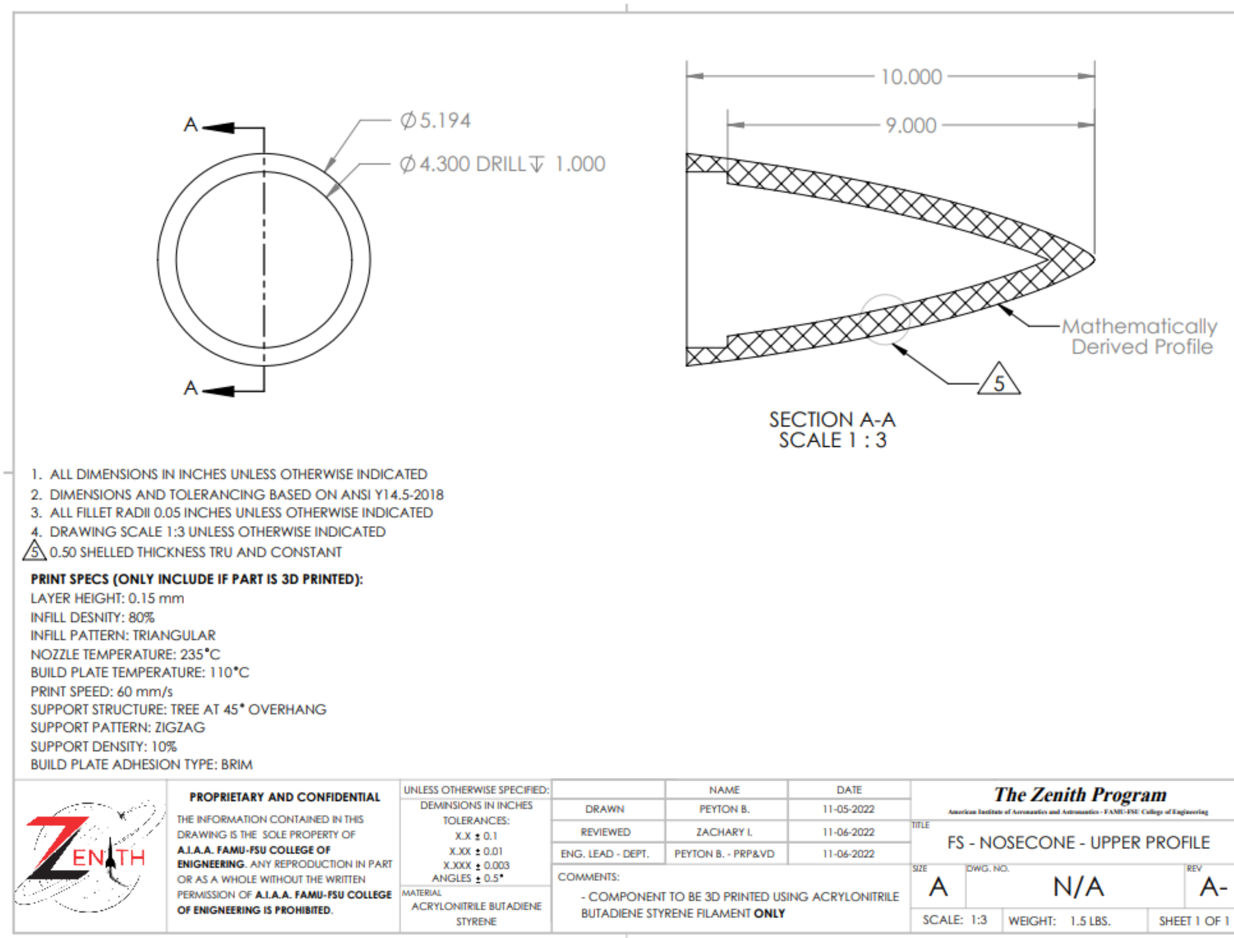


Figure 3-5. Nosecone Upper Profile CAD Drawing

3.1.1.2 <u>Upper Payload Bay</u>

The upper payload bay design has remained the same since CDR and consists of a 35-inch-long blue tube airframe with a forward bulkhead that sits flush against the nosecone shoulder. The manufactured dimensions are shown below along with their manufactured weights. The weights have altered slightly, each component was weighed post-manufacturing for accurate simulations and accurate documentation.

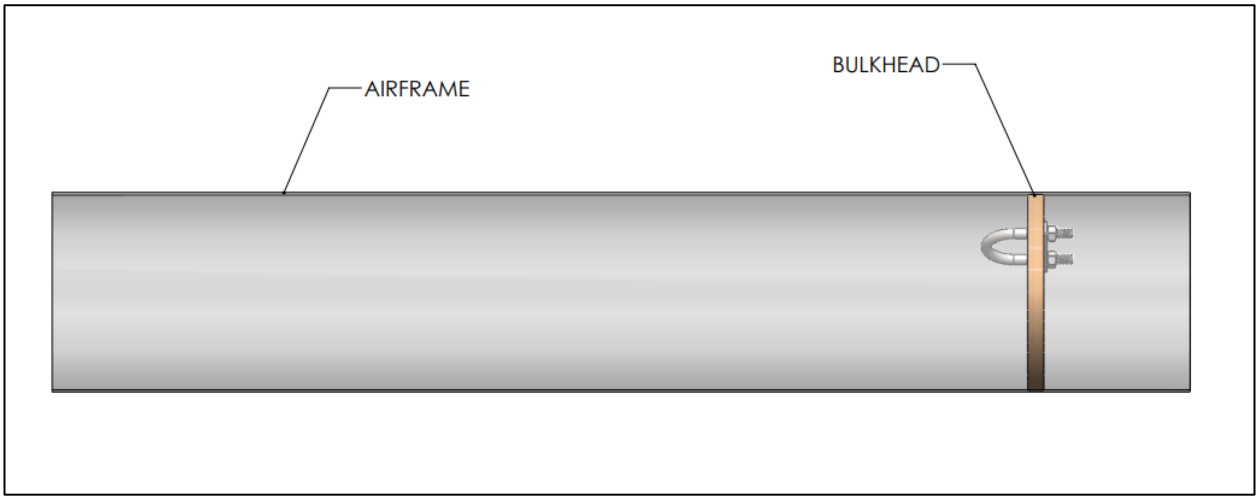


Figure 3-6. Upper Payload Bay Assembly

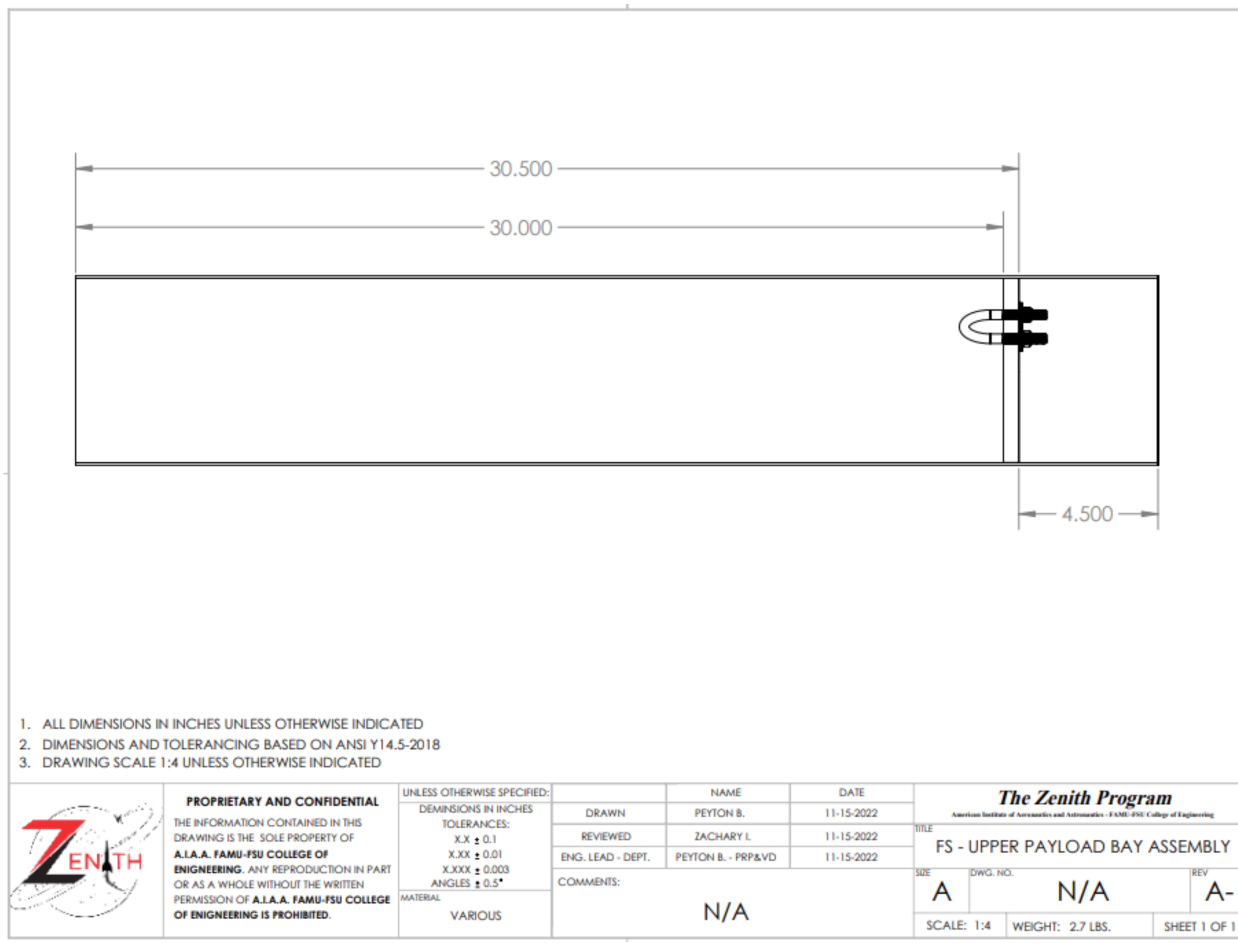


Figure 3-7. Upper Payload Bay Assembly CAD Drawing

(a) Upper Payload Bay Airframe

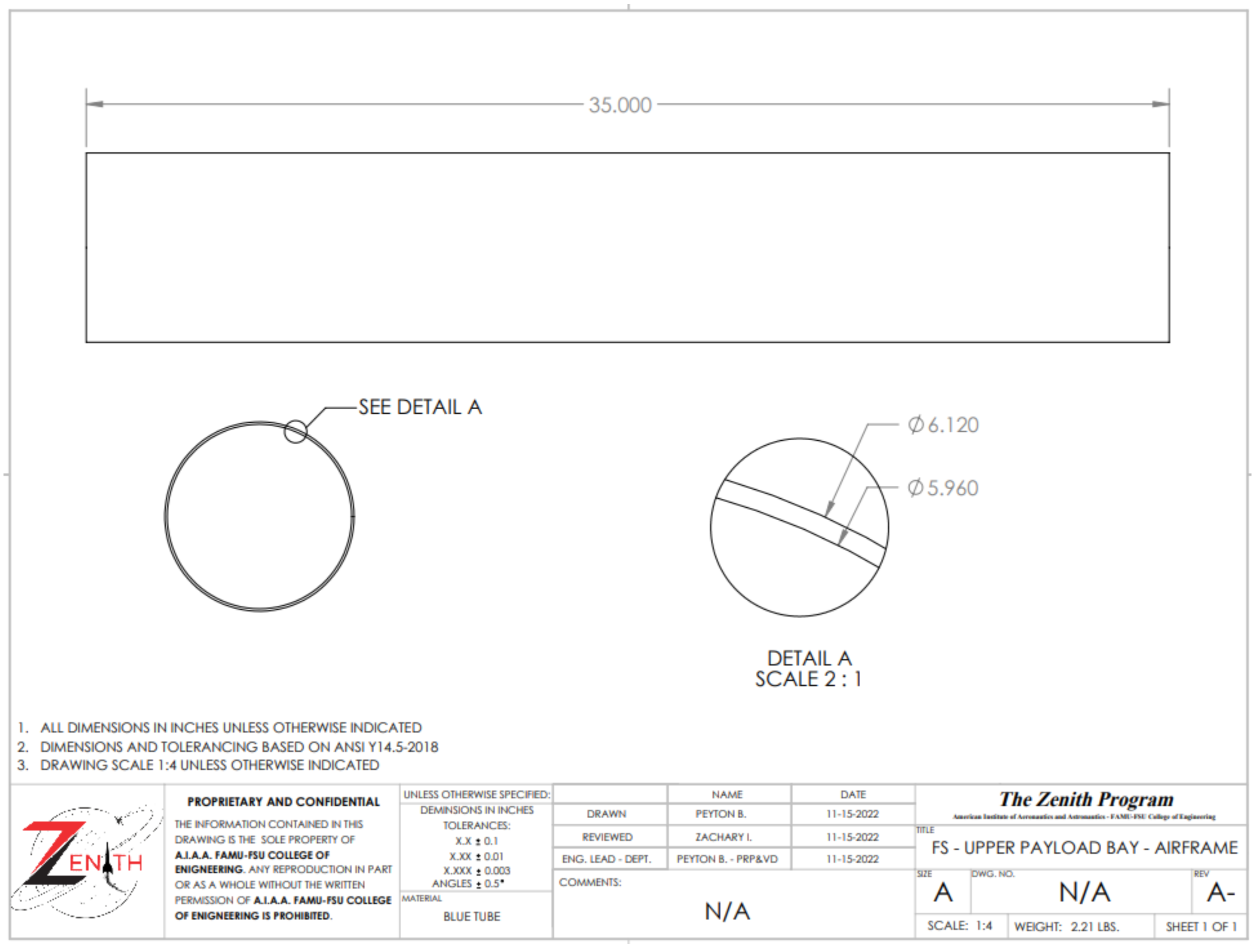


Figure 3-8. Upper Payload Bay Airframe CAD Drawing

(b) Nose Cone Bulkhead

All design aspects of the nosecone bulkhead have remained the same since CDR with a 5/16” U-Bolt and fastener plate configuration attached to the bulkhead for main parachute shock chord connection purposes. The ½” bulkhead is fabricated from Baltic Birch Plywood. Shown below are dimensioned CAD drawings of the nosecone bulkhead design and each of its components. Weights have been updated for accurate post-manufacturing weights.

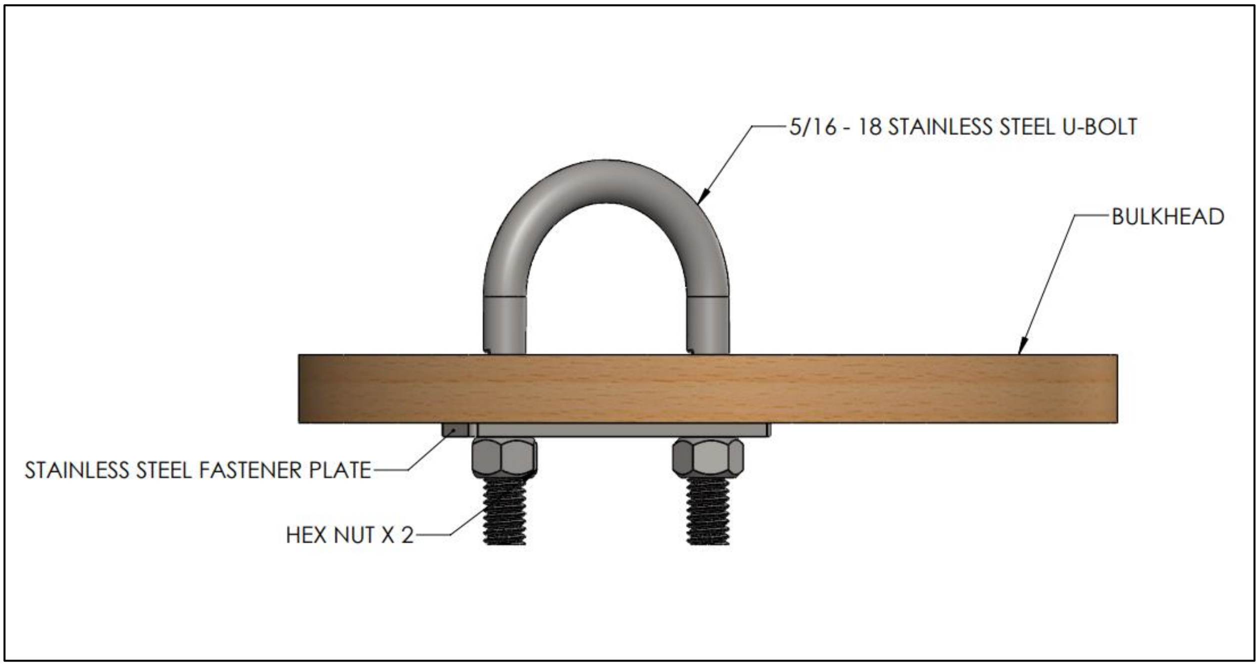


Figure 3-9. Nosecone Bulkhead Assembly

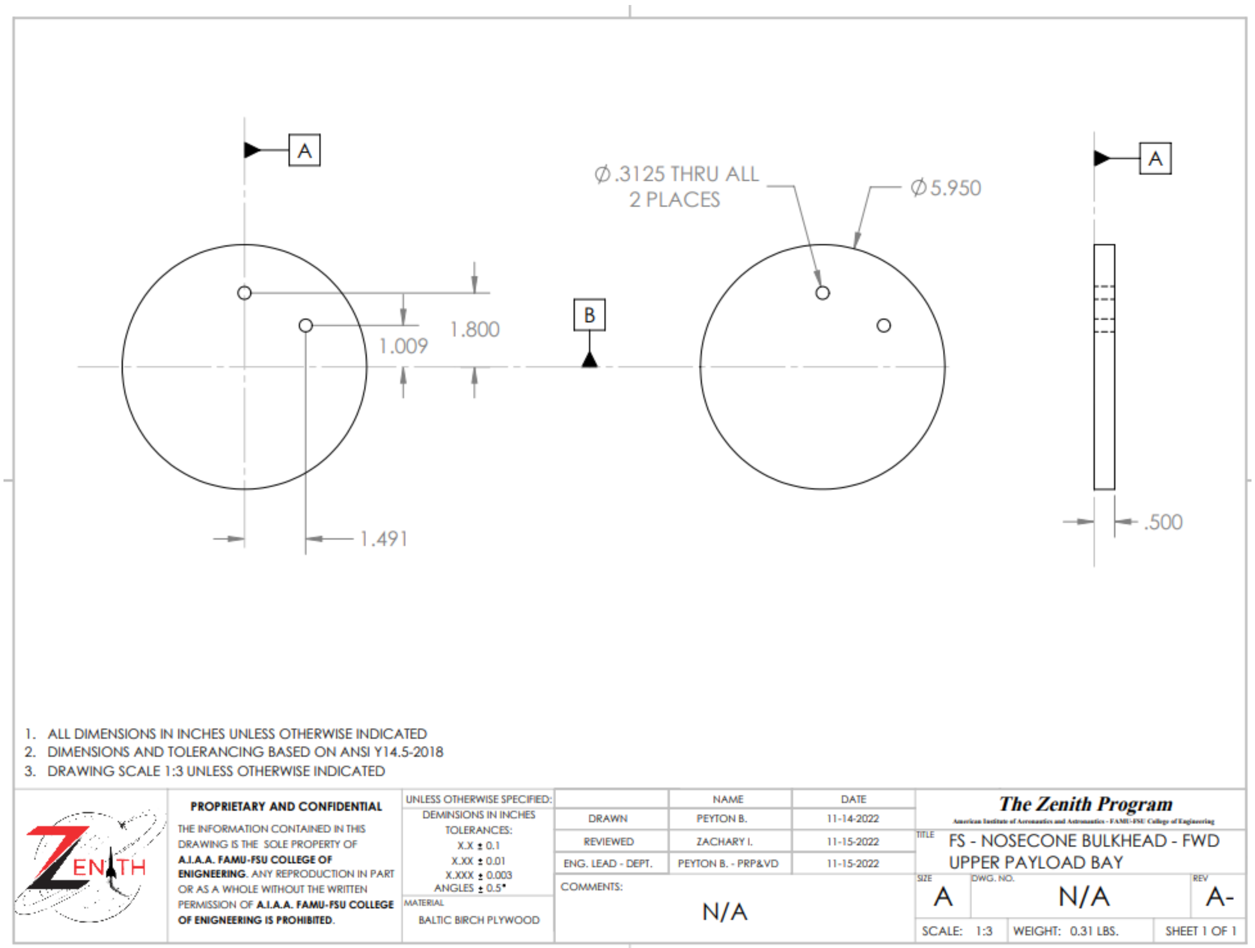


Figure 3-10. Nosecone Bulkhead CAD Drawing

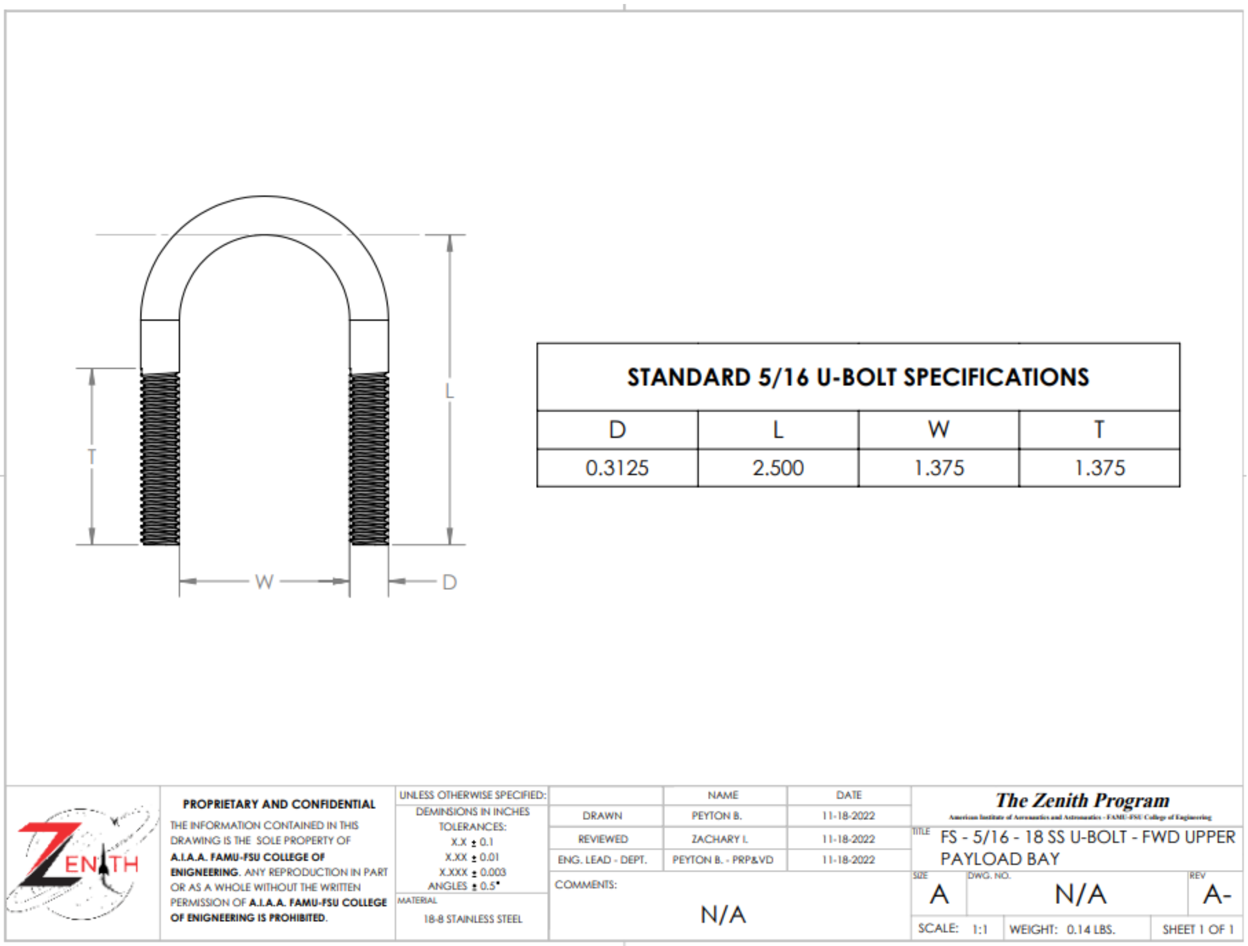
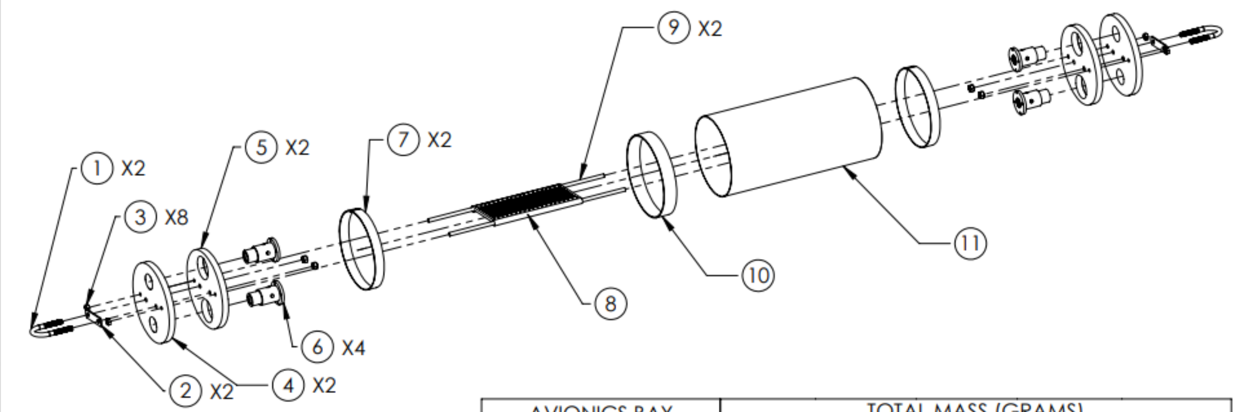


Figure 3-11. U-Bolt CAD Drawing

3.1.1.3 <u>Avionics Bay</u>

The avionics bay design and layout is relatively unchanged since CDR. The changes made were the positioning of the holes in the bulkhead that accommodate the threaded rods. The bulkheads were commercially ordered and came with the hole locations shown in figures 3-17 and 3-18. The avionics coupler is 12 inches in length and made from blue tube with a 1-inch-long outer airframe ring hugging the external surface of the coupler to provide a flush connection point between the coupler and the vehicle's airframe. The avionics bay as a whole is expected to weigh 4.83 lbs.



AVIONICS BAY ASSEMBLY	TOTAL MASS (GRAMS)				
	ATMPT 1	ATMPT 2	ATMPT 3	ATMPT 4	ATMPT 5
~~SIMULATED/DRAFTED~~	~~2056~~	~~N/A~~	~~N/A~~	~~N/A~~	~~N/A~~
PRE-MNFG	N/A	N/A	N/A	N/A	N/A
POST-MNFG	2187	N/A	N/A	N/A	N/A

PART #	PART NAME	QUANTITY	PART #	PART NAME	QUANTITY
1	5/16" U-BOLT	2	7	INNER AVIONICS BAY RING	2
2	U-BOLT PLATE	2	8	AVIONICS WIRING SLED	1
3	1/4" HEX NUT	8	9	THREADED ROD	2
4	OUTER AVIONICS BAY BULKHEAD	2	10	AIRFRAME SPACER	1
5	INNER AVIONICS BAY BULKHEAD	2	11	AVIONICS COUPLER	1
6	CO2 EJECTION CHARGES	4			

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MATERIAL: VARIOUS

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DRAWN	PEYTON B.	02/25/2023
REVIEWED	ZACHARY I.	02/25/2023
ENG. LEAD - DEPT.	PEYTON B. - P&VD	02/25/2023

COMMENTS:

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TITLE: FS - AVIONICS BAY ASSEMBLY

SIZE: A DWG. NO.: N/A REV: A-

SCALE: 1:9 WEIGHT: 4.83 LBS. SHEET 1 OF 1

Figure 3-12. AV Bay Transparent Assembly

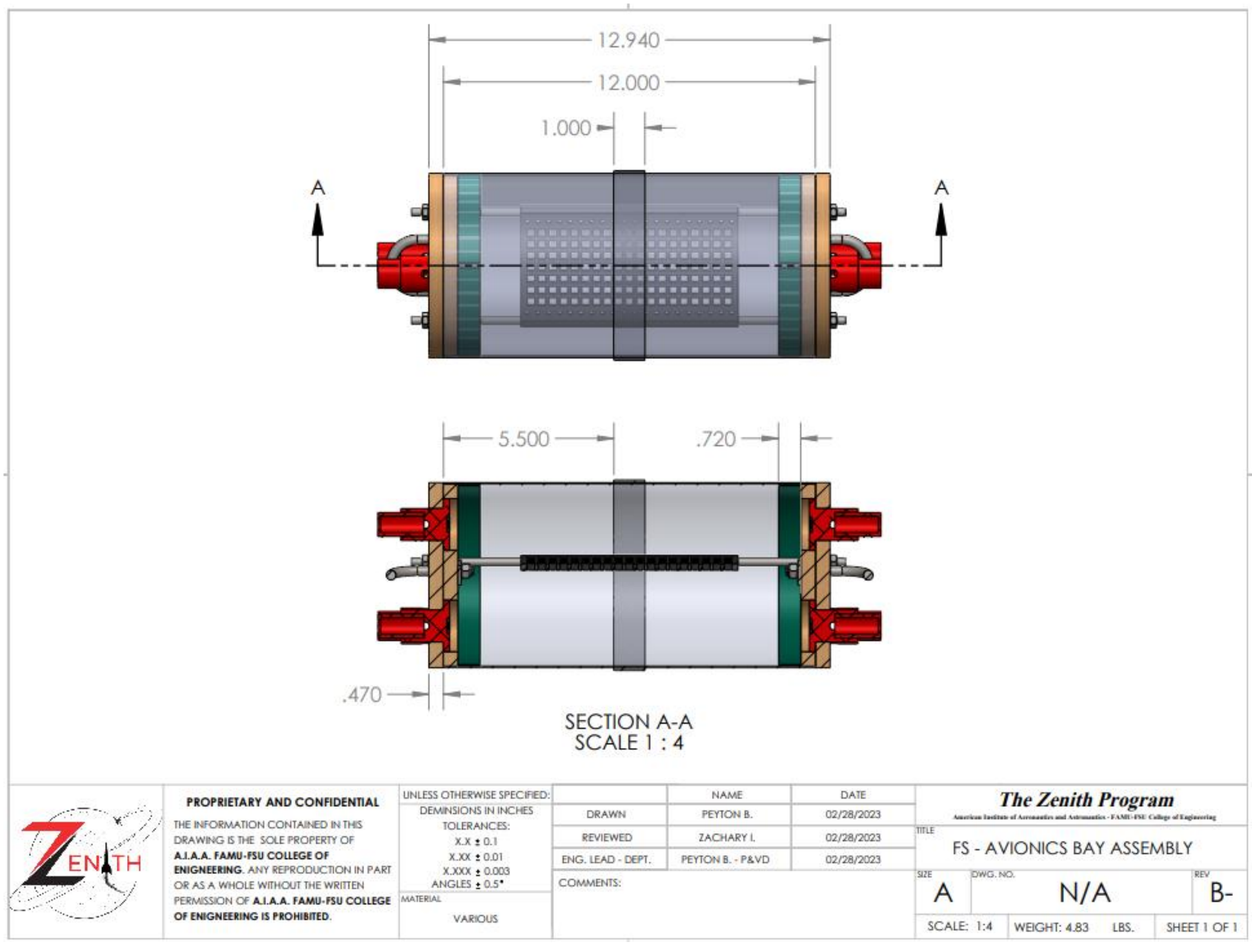


Figure 3-13. AV Bay Solid CAD Drawing

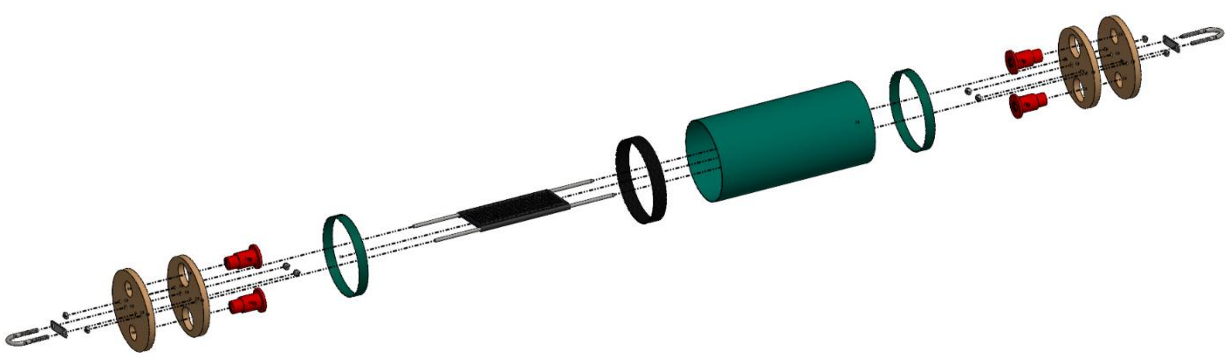


Figure 3-14. AV Bay Exploded Assembly

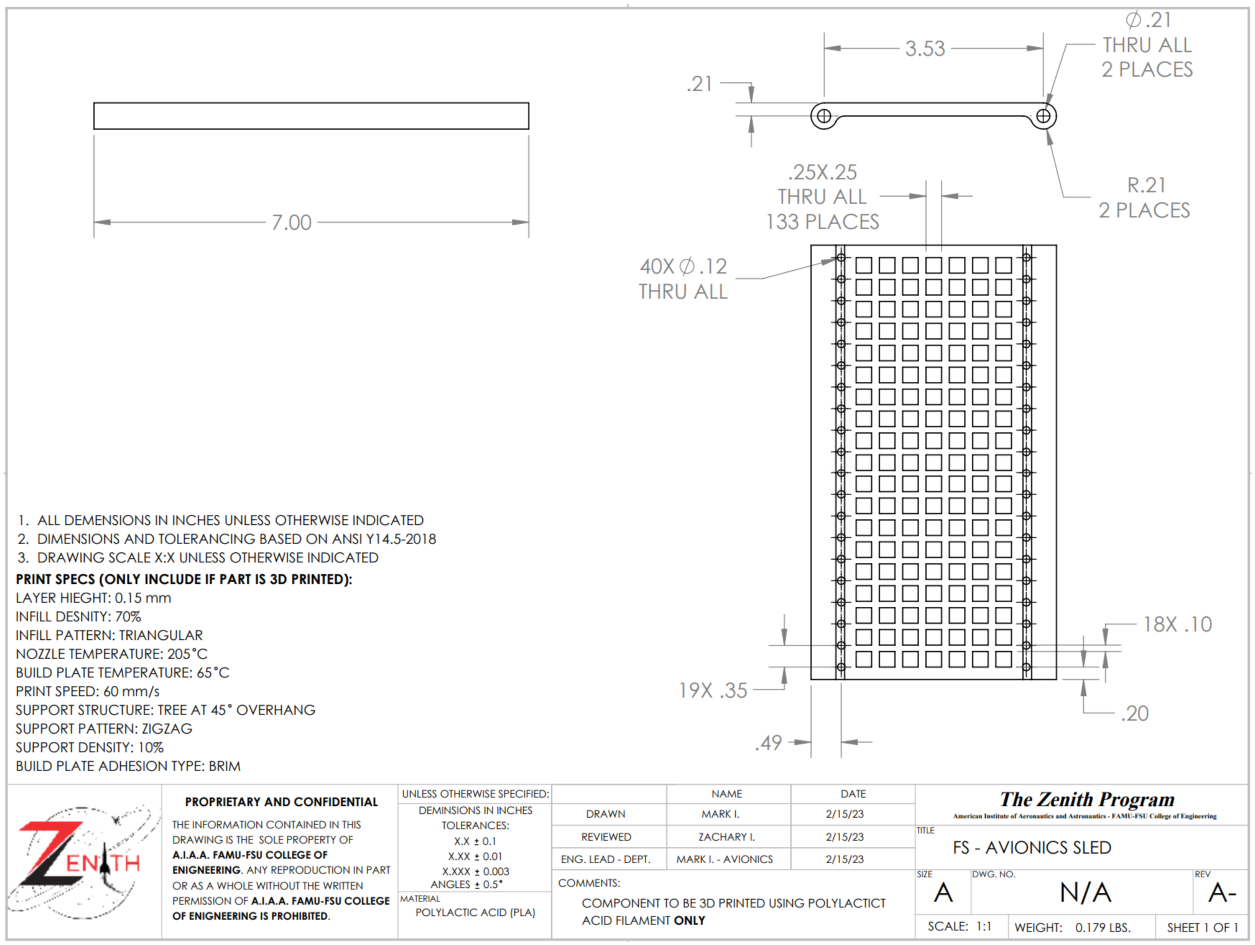


Figure 3-15 Avionics Sled Engineering Drawing

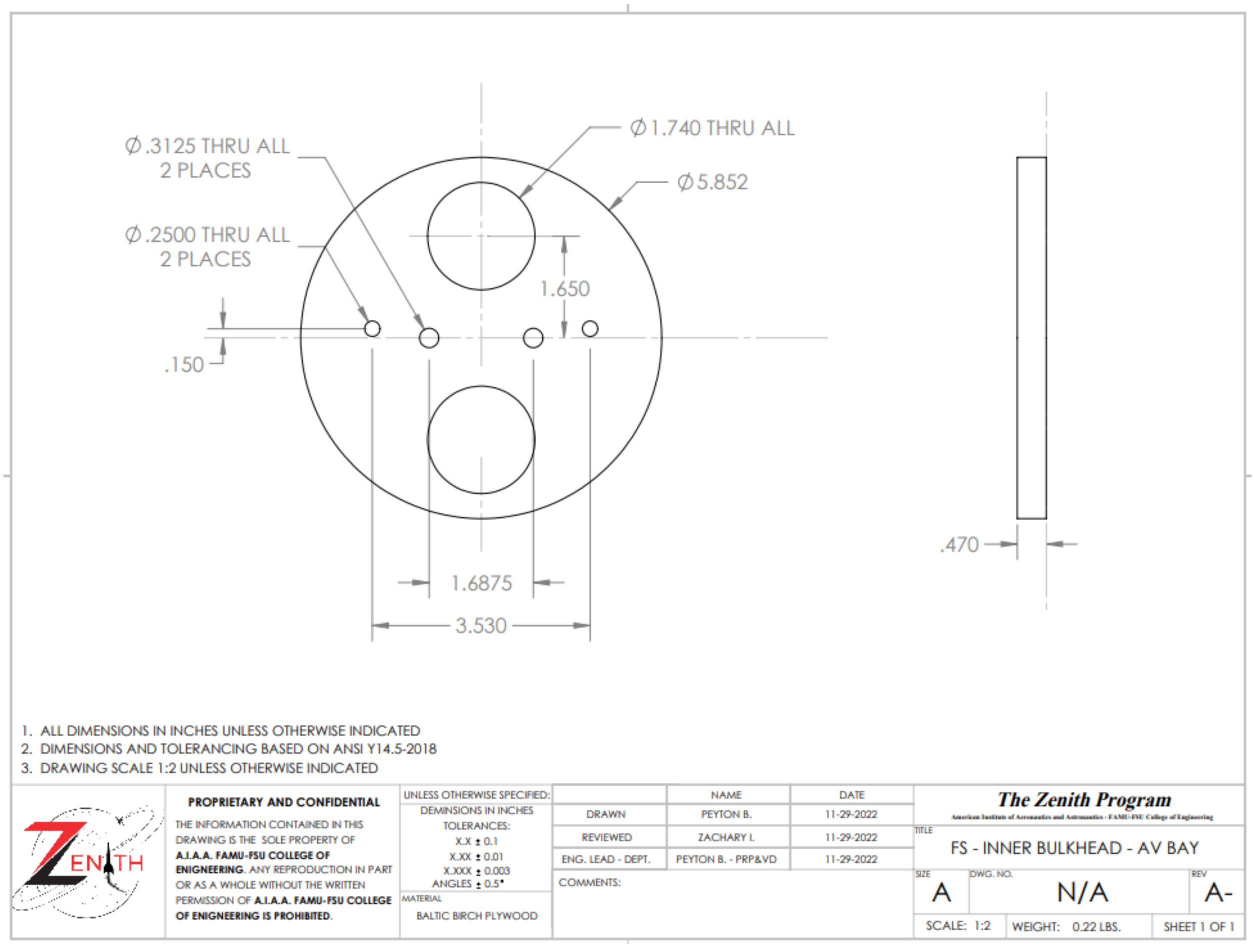


Figure 3-16. AV Bay Outer Bulkhead CAD Drawing

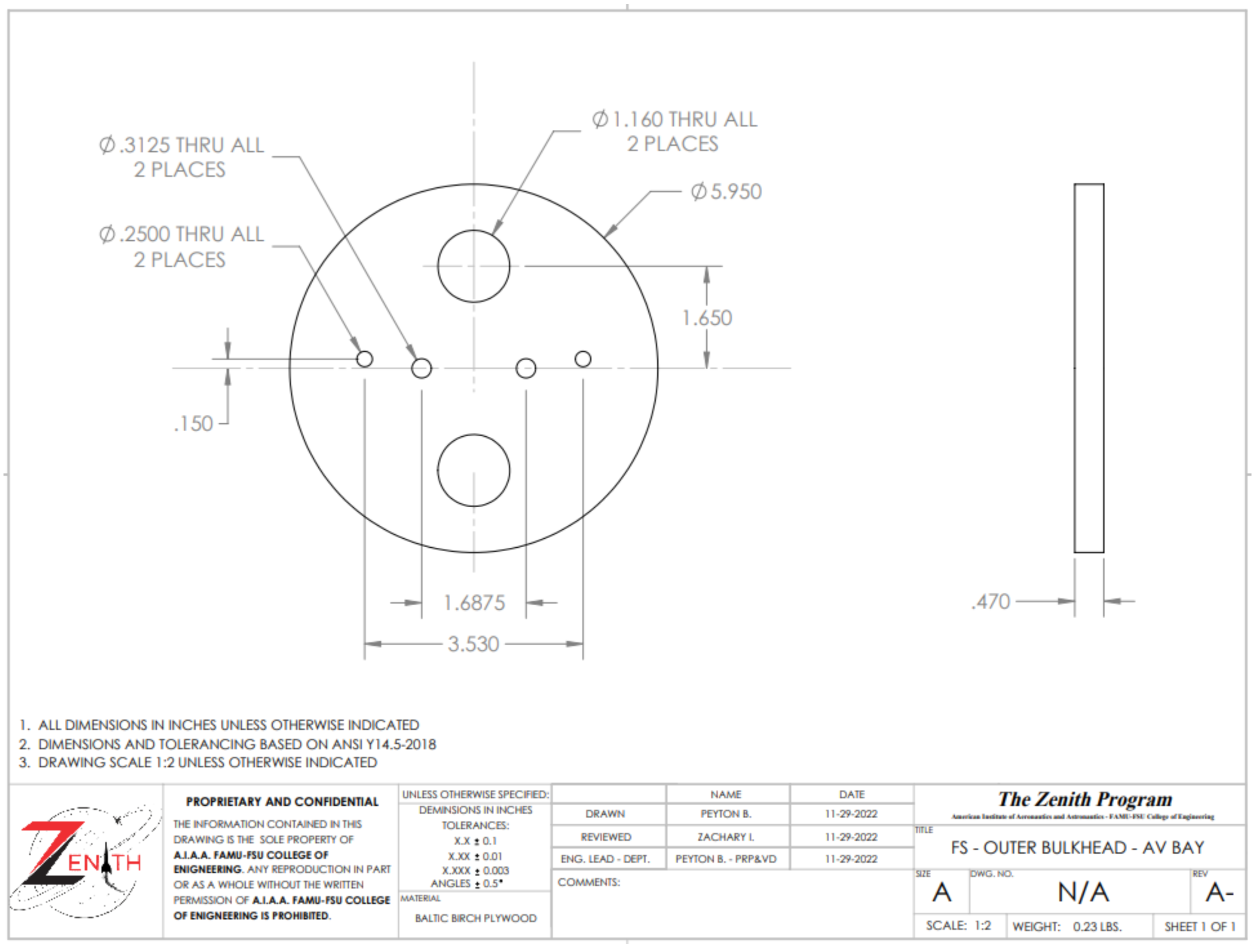


Figure 3-17. AV Bay Inner Bulkhead CAD Drawing

3.1.1.4 Fin Can/Lower Payload Bay

The lower payload bay design has a few minor design changes in regard to fin design and centering ring design. These changes are discussed below. Aside from fin and centering ring design, the rest of the components remain the same from CDR. The lower payload bay as a whole is 43.50 inches in length.

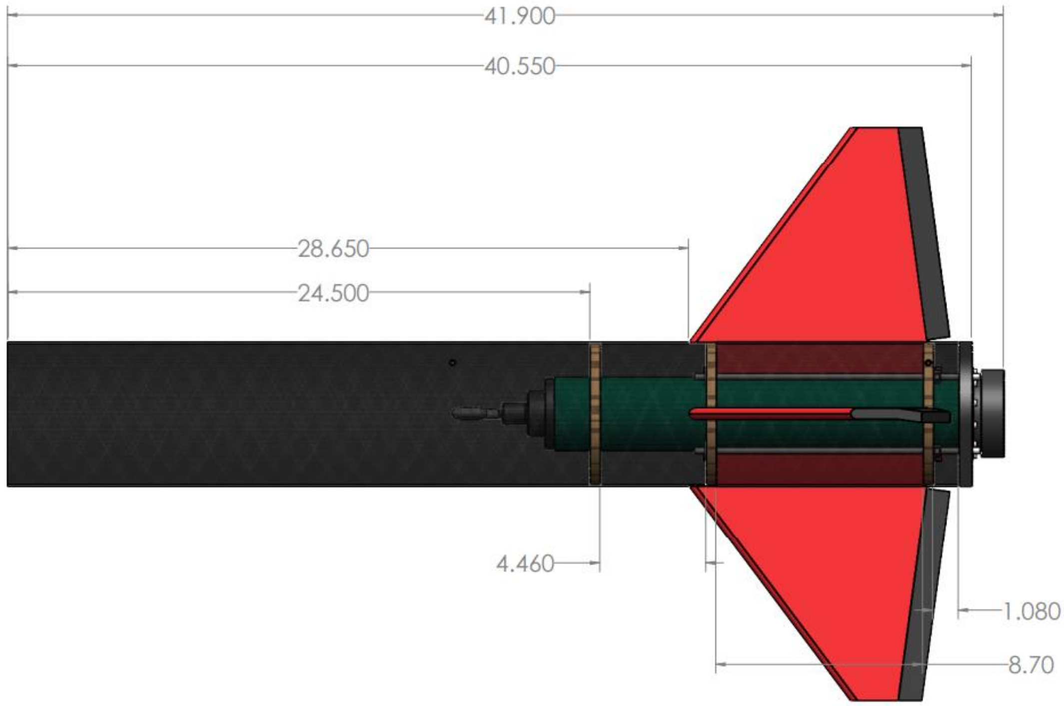


Figure 3-18. Lower Payload Bay/Fin Can Transparent Assembly

(a) Updated Airframe

The fin cut-outs for the lower payload bay airframe have been changed to accommodate the new fin design. The airframe design presented in CDR had fin slots cut at the aft end of the vehicle that were 13 inches in length and were cut to the end of the airframe. The new and updated fin design is design for the fins to be inserted vertically in and out of the vehicle, similar to traditional fin designs. Previous fin designs were modeled to have a base plate with a surface curve following the outer airframe diameter and were printed from PETG. The curved plate in the original design of the fins would frequently deform, regardless of the material used for printing, and often did not properly align with the airframe. The base plate was implemented to allow the fins to be canted off vertical to induce spinning stabilization. Due to structural printing deformation and fitment, the fin design was modified to a vertical stature with spin tabs located the end of the fins to replicate the canted feature in the last design. The new configuration is shown below along with an engineering drawing.

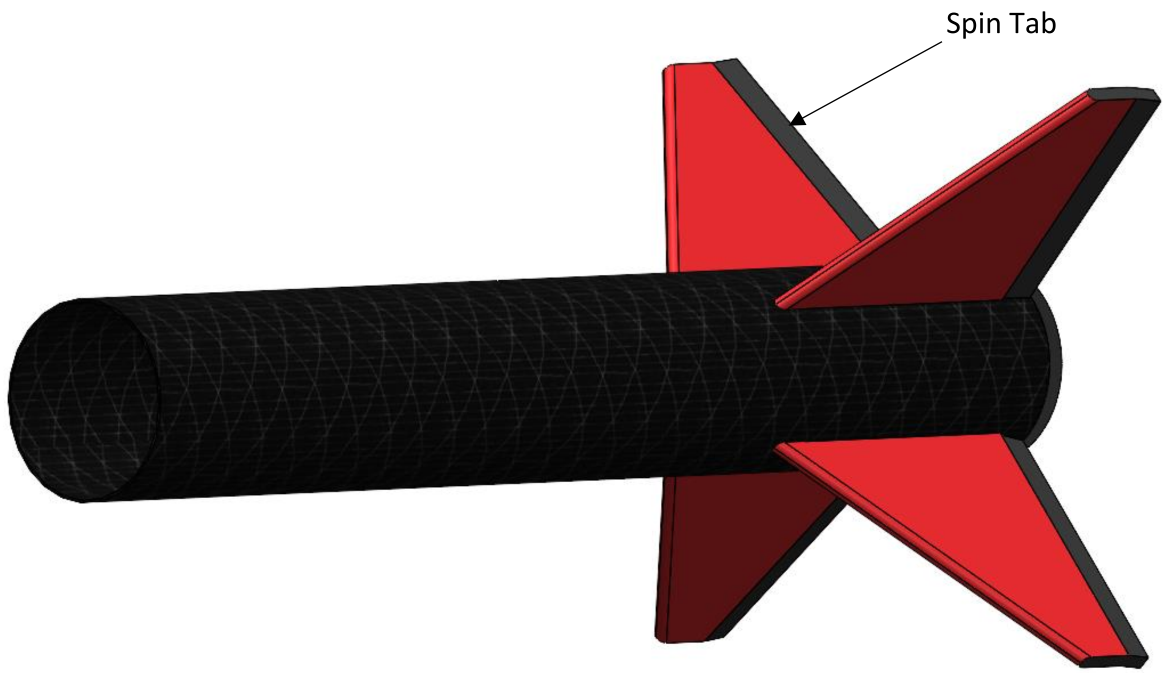
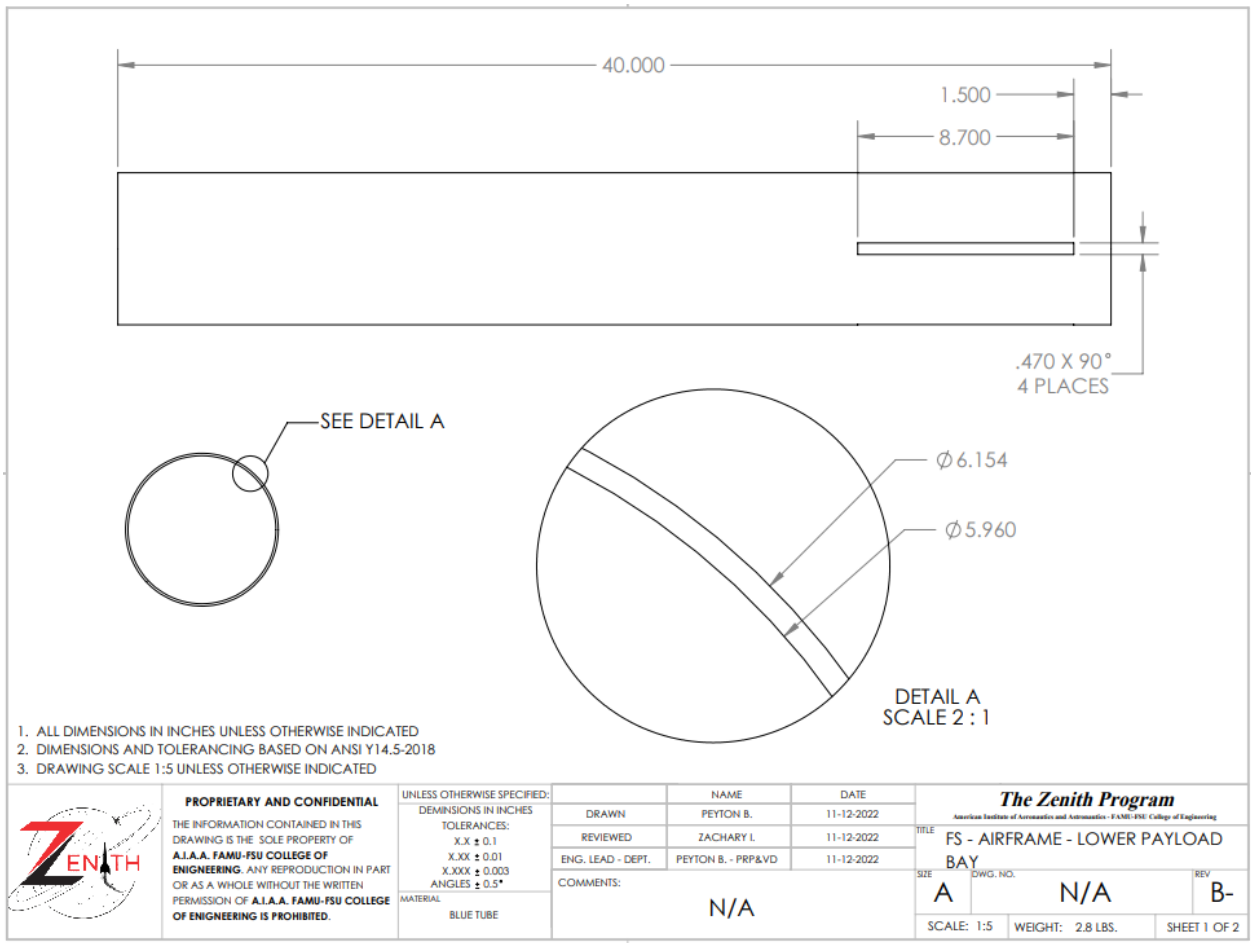


Figure 3-19. Lower Payload Bay Airframe and Fin Configuration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8.700
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Ø5.960
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UNLESS OTHERWISE SPECIFIED:
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X.XX ± 0.01
X.XXX ± 0.003
ANGLES ± 0.5°
MATERIAL
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NAME
DATE
DRAWN
PEYTON B.
11-12-2022
REVIEWED
ZACHARY I.
11-12-2022
ENG. LEAD - DEPT.
PEYTON B. - PRP&VD
11-12-2022
COMMENTS:
N/A
The Zenith Program
TITLE
FS - AIRFRAME - LOWER PAYLOAD BAY
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B-
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SHEET 1 OF 2

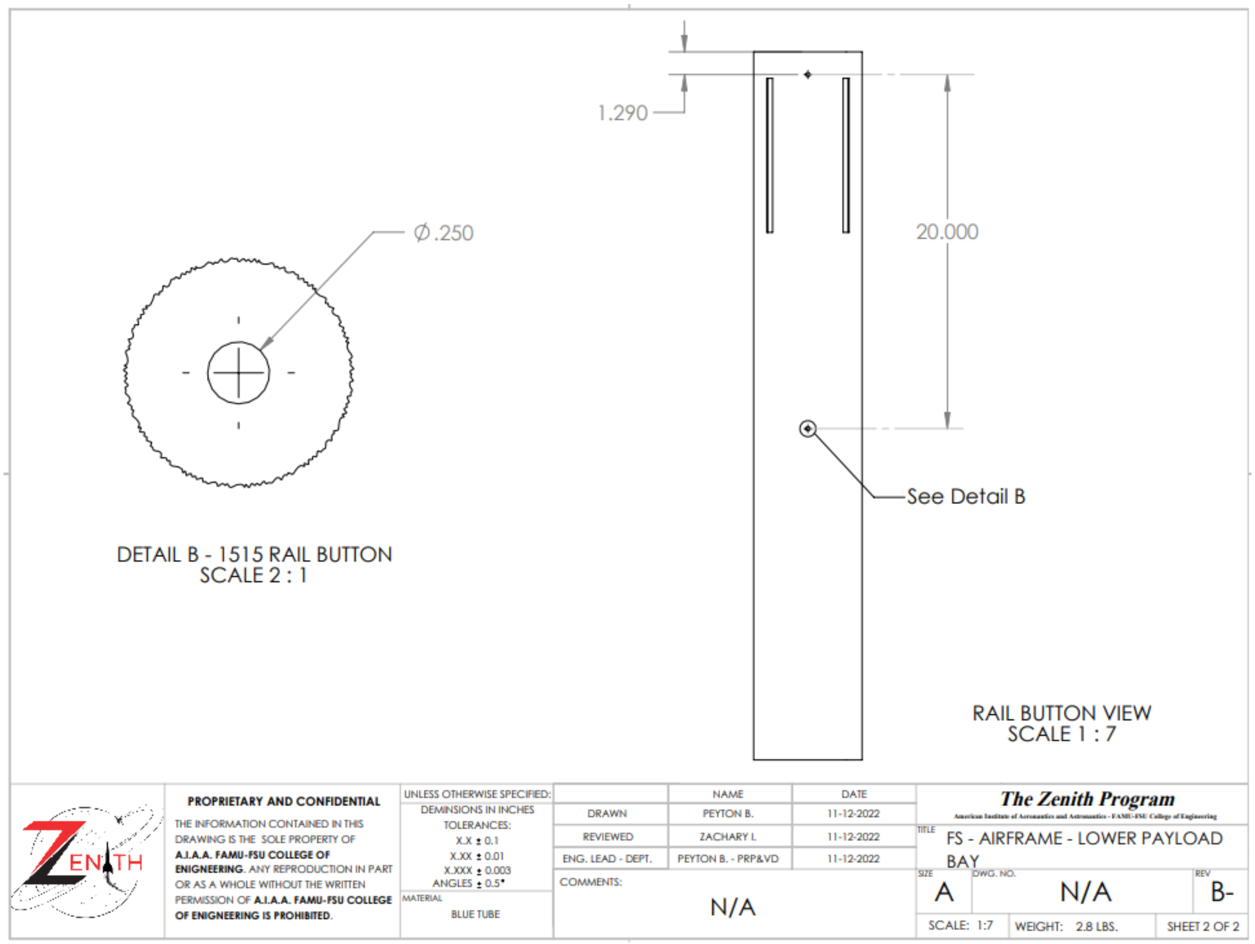


Figure 3-20. Lower Payload Bay Airframe w/ 1515 Rail Buttonhole

(b) Updated Thrust Structure

The thrust structure presented in CDR is relatively similar to the updated design, but with a few small changes.

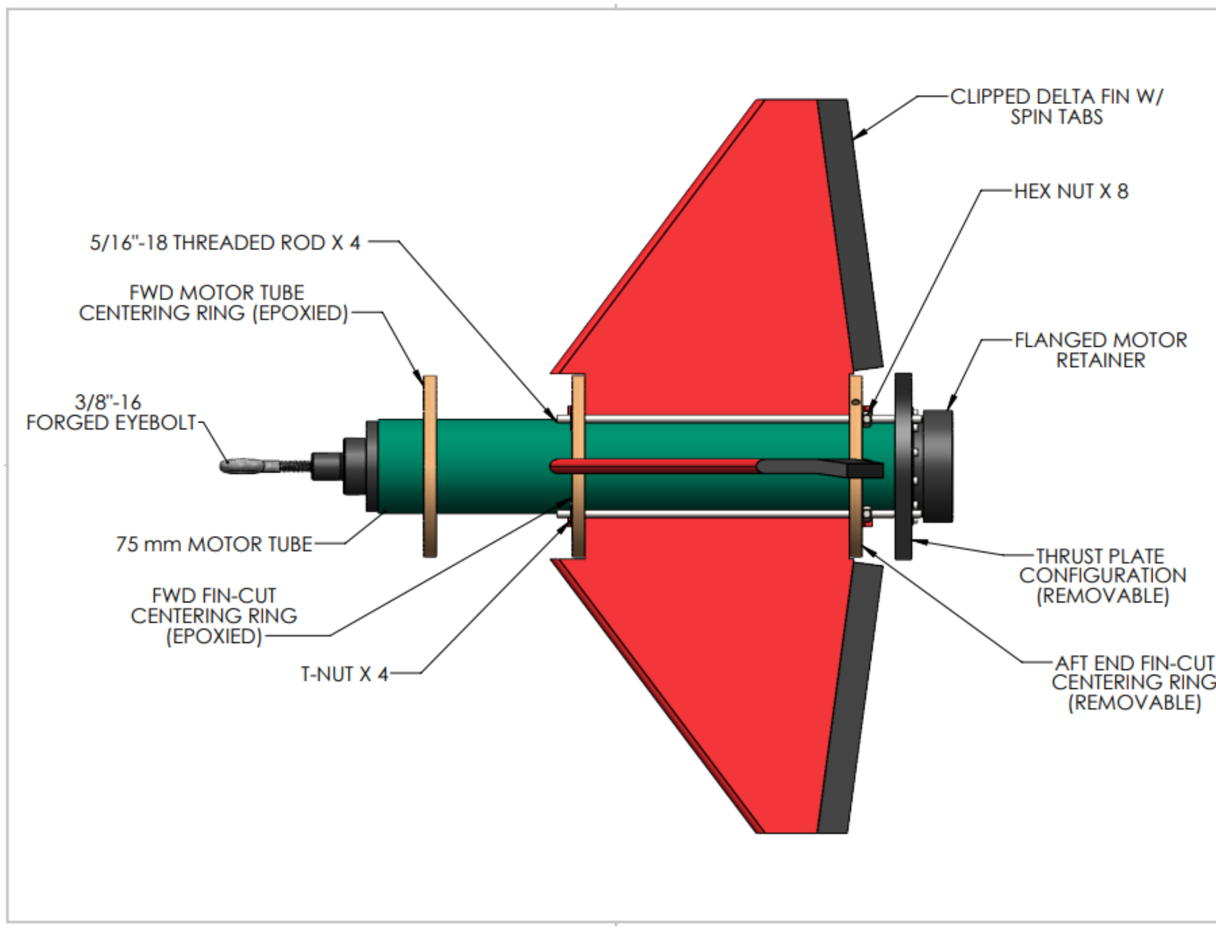


Figure 3-21. Labeled Fin Can Assembly

The fin tabs have been designed differently to fit into slots that were cut in the centering rings. The fins vertically slide into the slots and the centering rings, held in by the threaded rods, sandwich the fin tabs between the two centering rings to keep the fins fixed in place. The fin tabs extend to the motor tube for extra structural stabilization and to transfer thrust loads to the centering rings.

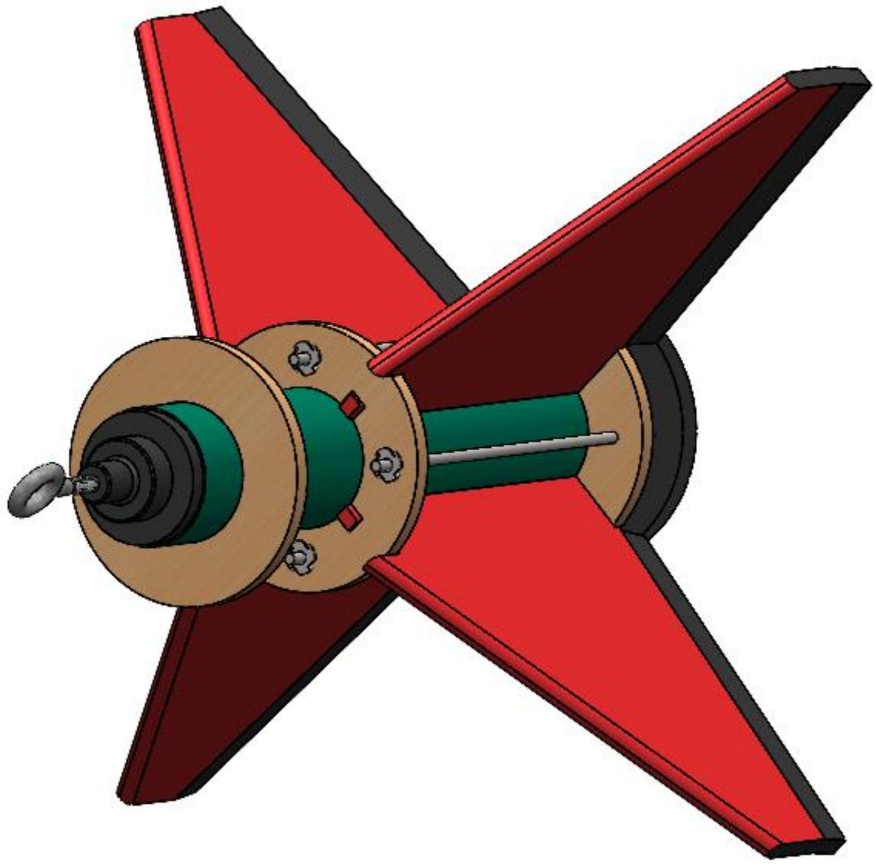


Figure 3-22. Fin Can Assembly

The fin assembly shown above is the updated thrust structure configuration. The fins will be 3D printed using ABS filament and have an estimated weight of 1.4 pounds each (670 grams). The spin tabs are vertically offset 8 degrees and are implemented to cause the vehicle to spin and positively affect the vehicle’s stability. Shown below are dimensioned CAD drawings of the clipped delta fins.

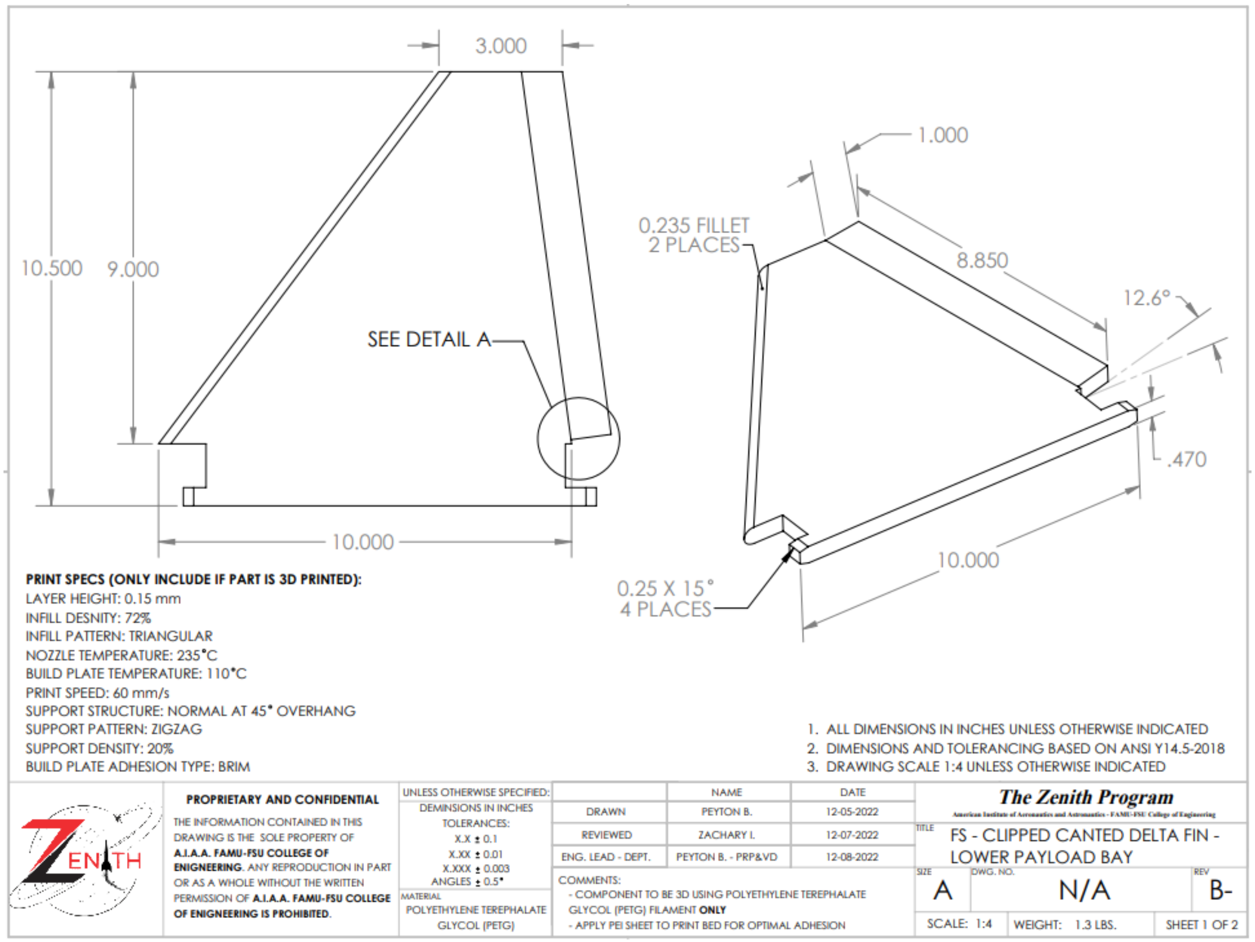


Figure 3-23. Canted Clipped Delta Fin CAD Drawing Sheet 1 of 2

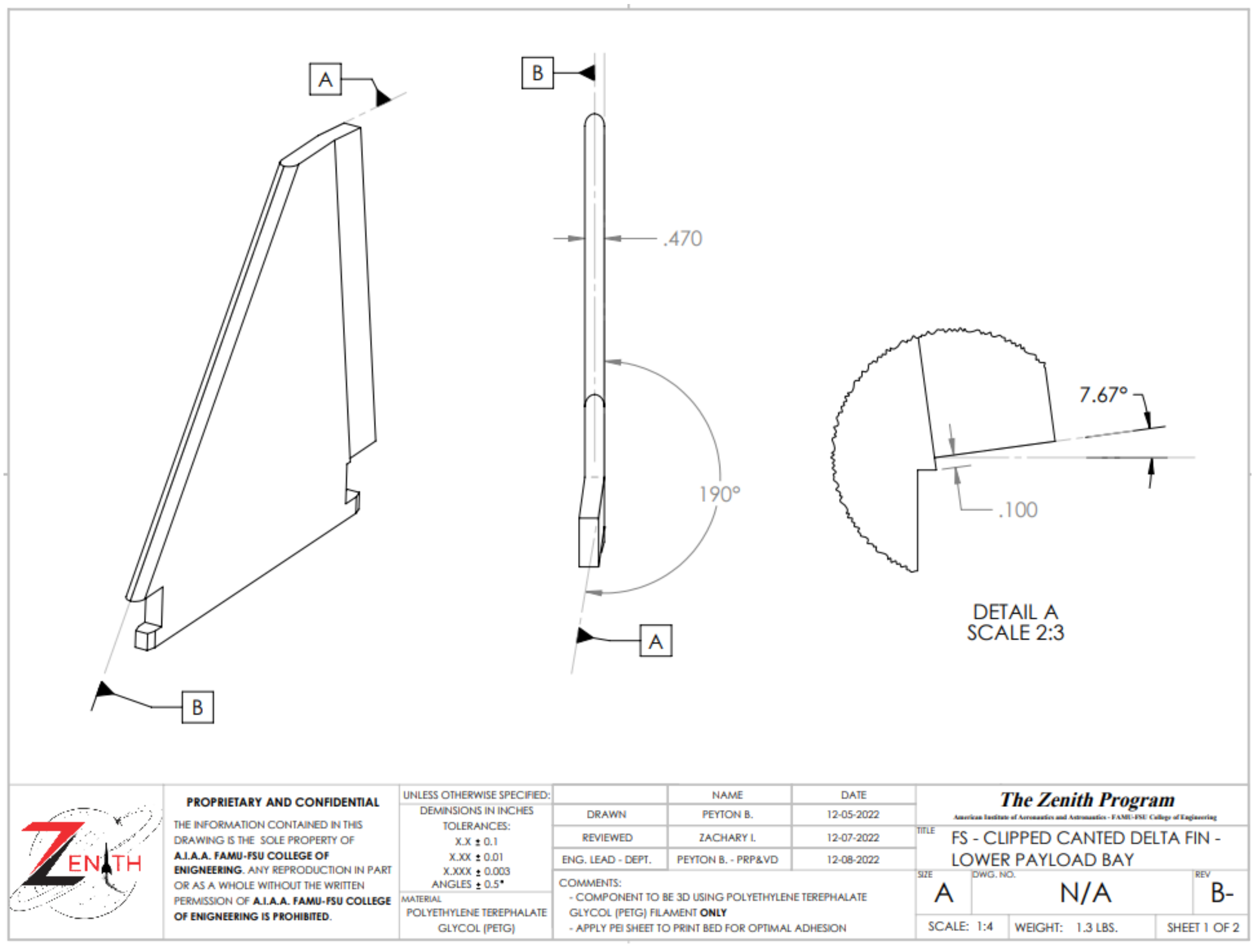


Figure 3-24. Canted Clipped Delta Fin CAD Drawing Sheet 2 of 2

As mentioned in both PDR and CDR, a leading concern with the vehicle's fins was the fins reaching the fin fluttering speed, which could lead to the fins shearing during flight. The new fin design has the same dimension as the design shown in CDR, aside from the base plate. The fin flutter speed shown below was calculated in MATLAB and gives the fin structure a nice factor of safety range.

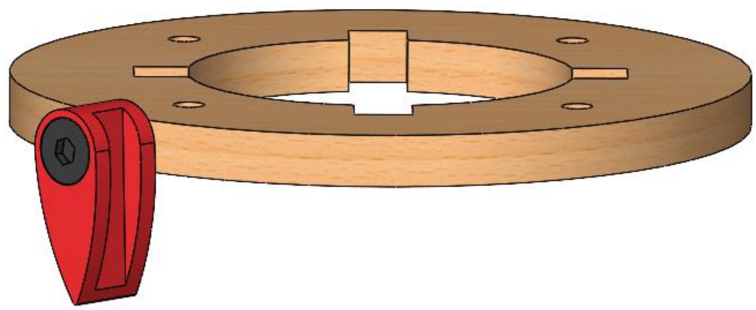
Table 3-1. Fin Flutter Speed Parameters

Fin Flutter Speed			
Parameter	**Symbol**	**Value**	**Unit**
Speed of Sound	a	1098.9	ft/s
Shear Modulus	G	151920	lb/in^2
Aspect Ratio	AR	1.0714	
Pressure	P	12.44	lb/in^2
Taper Ratio	l	0.2727	
Fin Thickness	t	0.47	inches
Root Chord	c	11	inches
Fin Flutter Speed	V_f	1446.3	ft/s

Table 3-2. Fin Flutter Speed Results

Max Vehicle Speed:	545 ft/s
Fin Flutter Speed:	1446.3 ft/s
Percent Flutter Speed Achieved:	38%
Factor of Safety:	2.65

The centering rings used for the thrust structure have kept the same general design since CDR. The cut-outs in the centering rings have been modified to accommodate the new fin design with skinnier tabs. The aft end centering ring has also been modified to allow the 1515 rail button to be screwed into for final assembly. The rail button placement not only fixes the aft end rail button to the vehicle, but also prevents the removable centering ring from rotating during flight. Show below are updated engineering drawings for the centering rings and an image of how the aft end rail button is fixed to the centering ring.



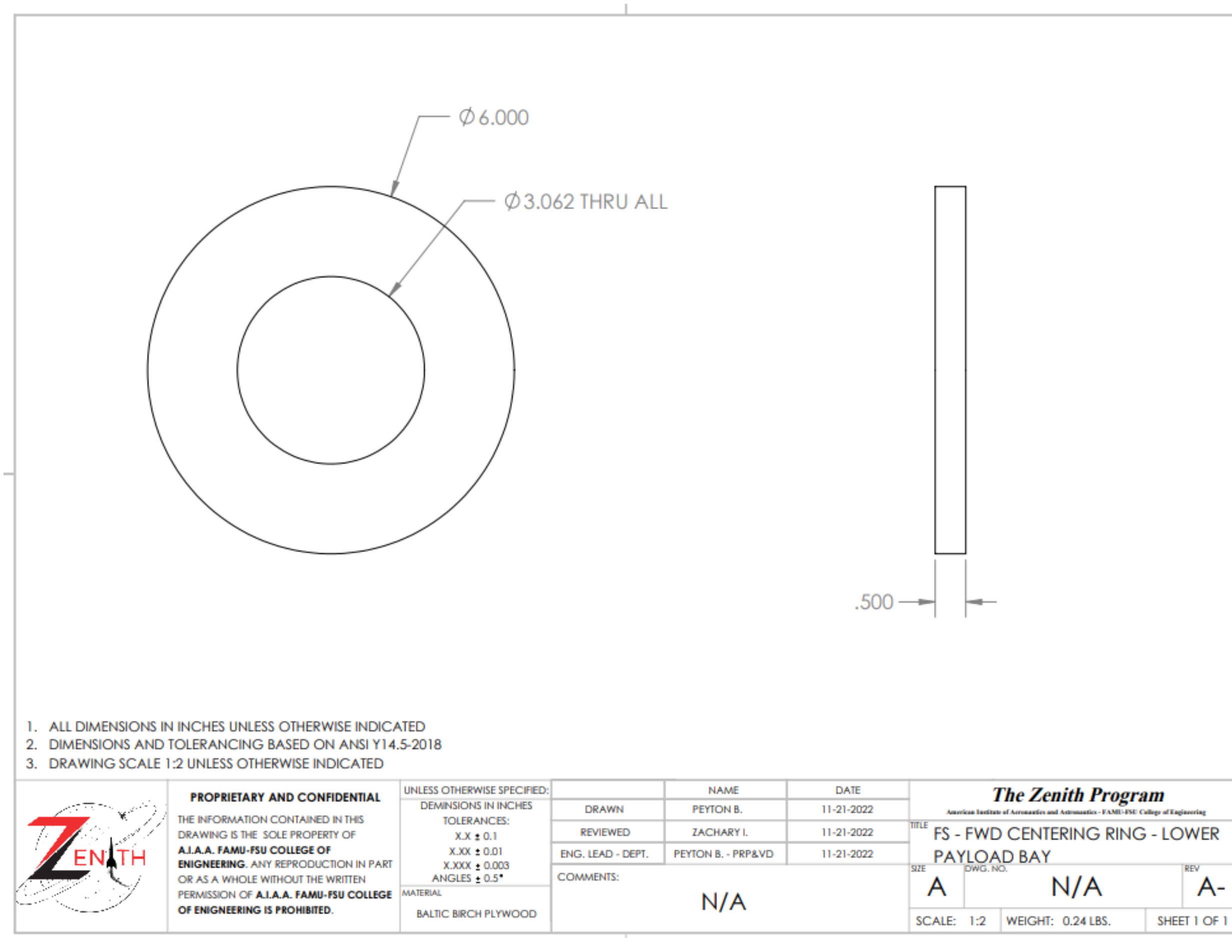


Figure 3-25. Forward Centering Ring CAD Drawing

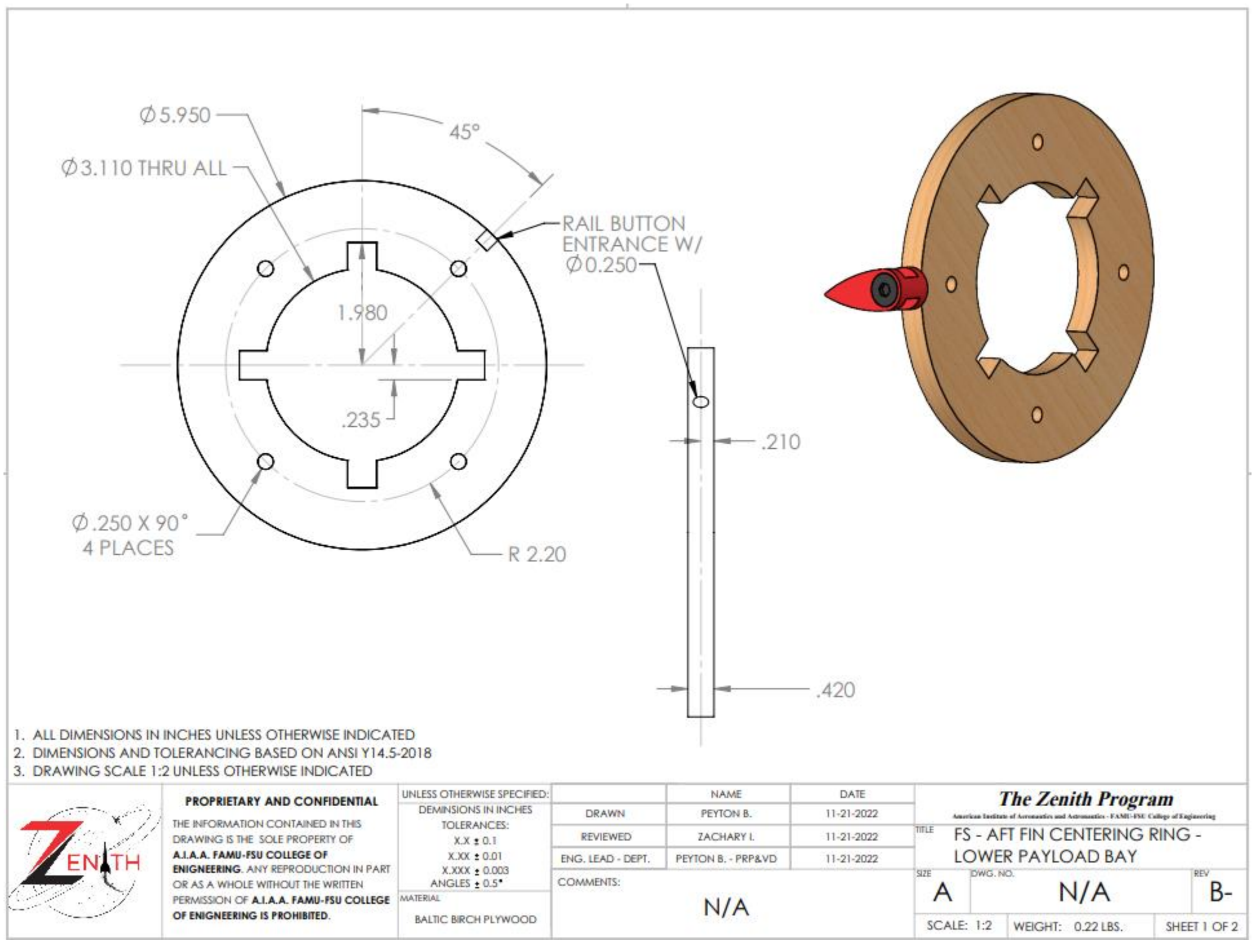


Figure 3-26. Aft Fin Centering Ring CAD Drawing

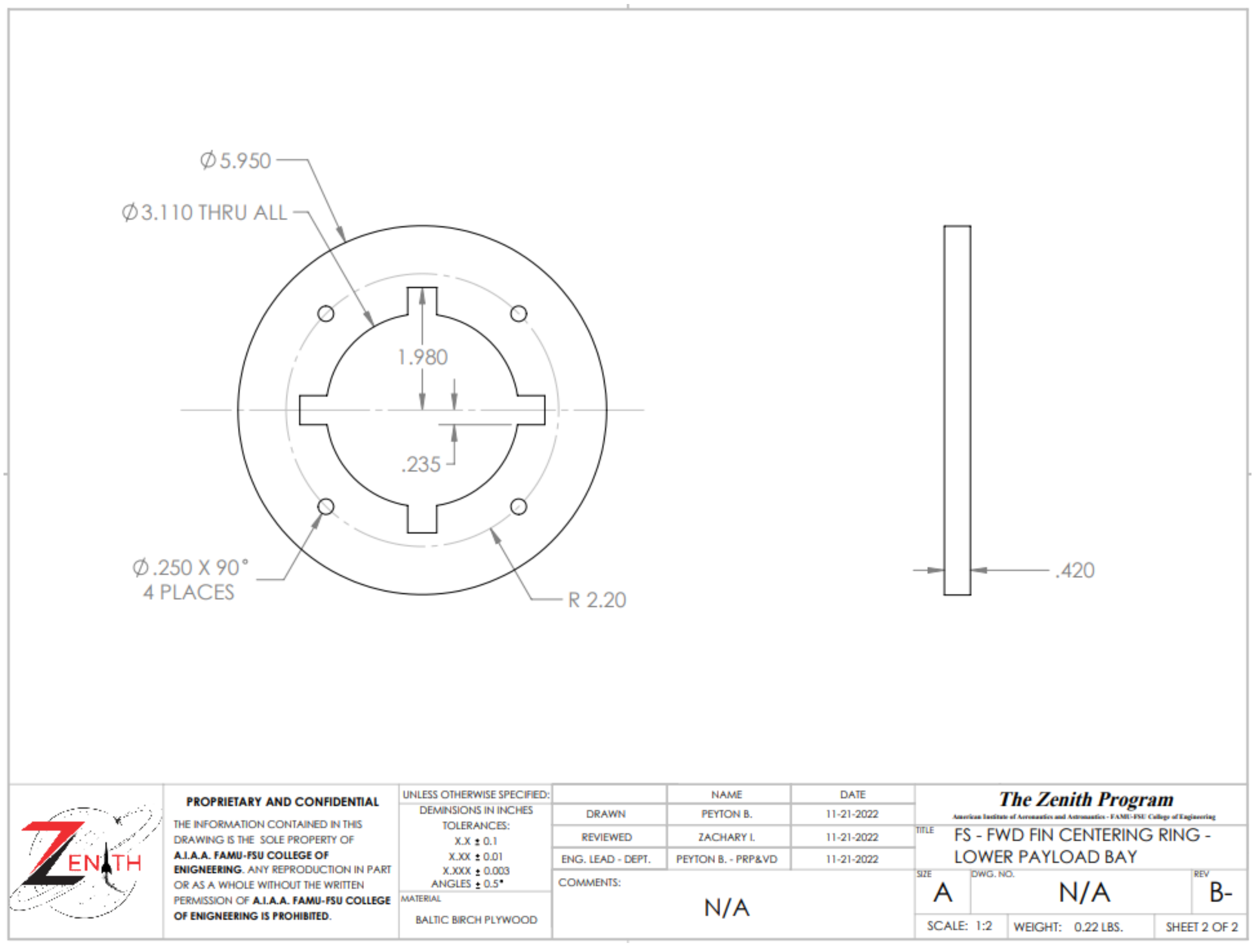


Figure 3-27. Forward Fin Centering Ring CAD Drawing

The thrust plate and tail cone configuration has changed since CDR, with the tail cone being removed from the design and the component having a total weight of 0.41 pounds. Shown below are images of how the tail cone is assembled to the thrust structure along with engineering drawings of the component. Mentioned in section #, the tail cone detached from the launch vehicle during the full-scale test launch. To avoid any further issue regarding the tail cone, it has been eliminated from the design as a whole.

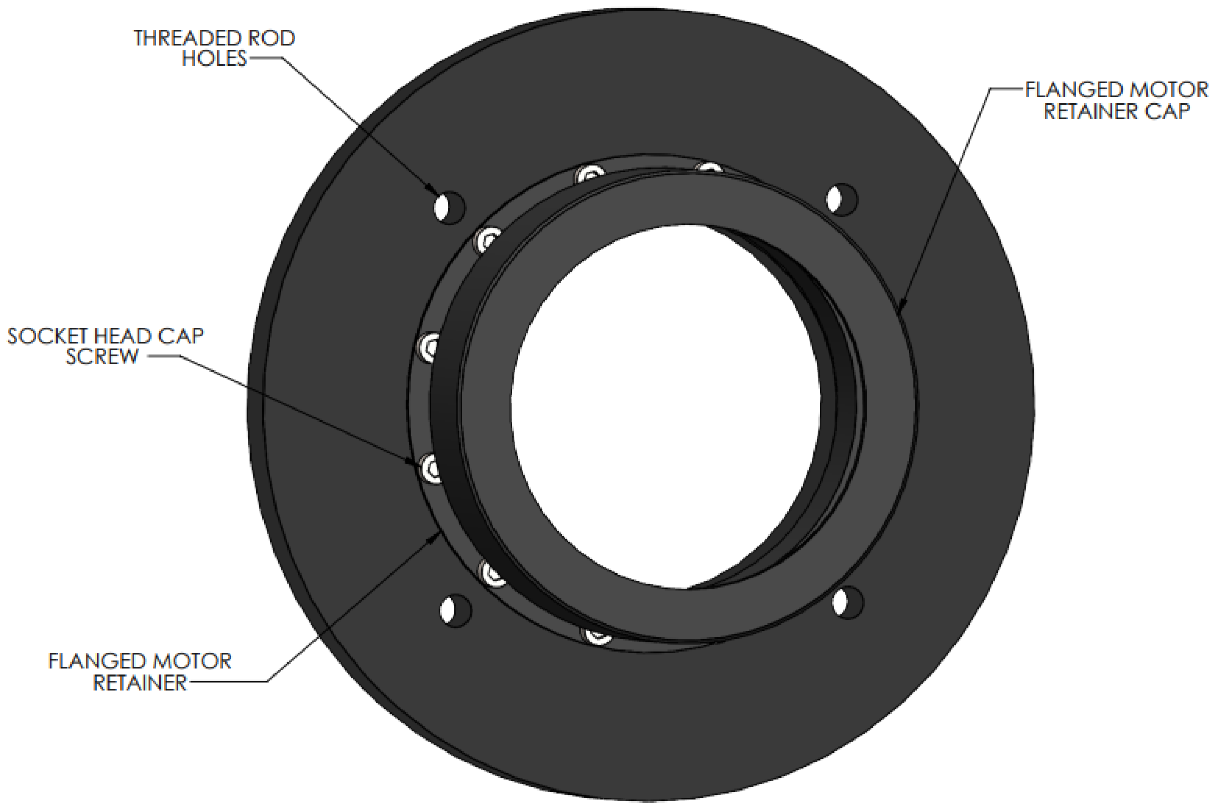


Figure 3-28. Thrust Plate CAD Model

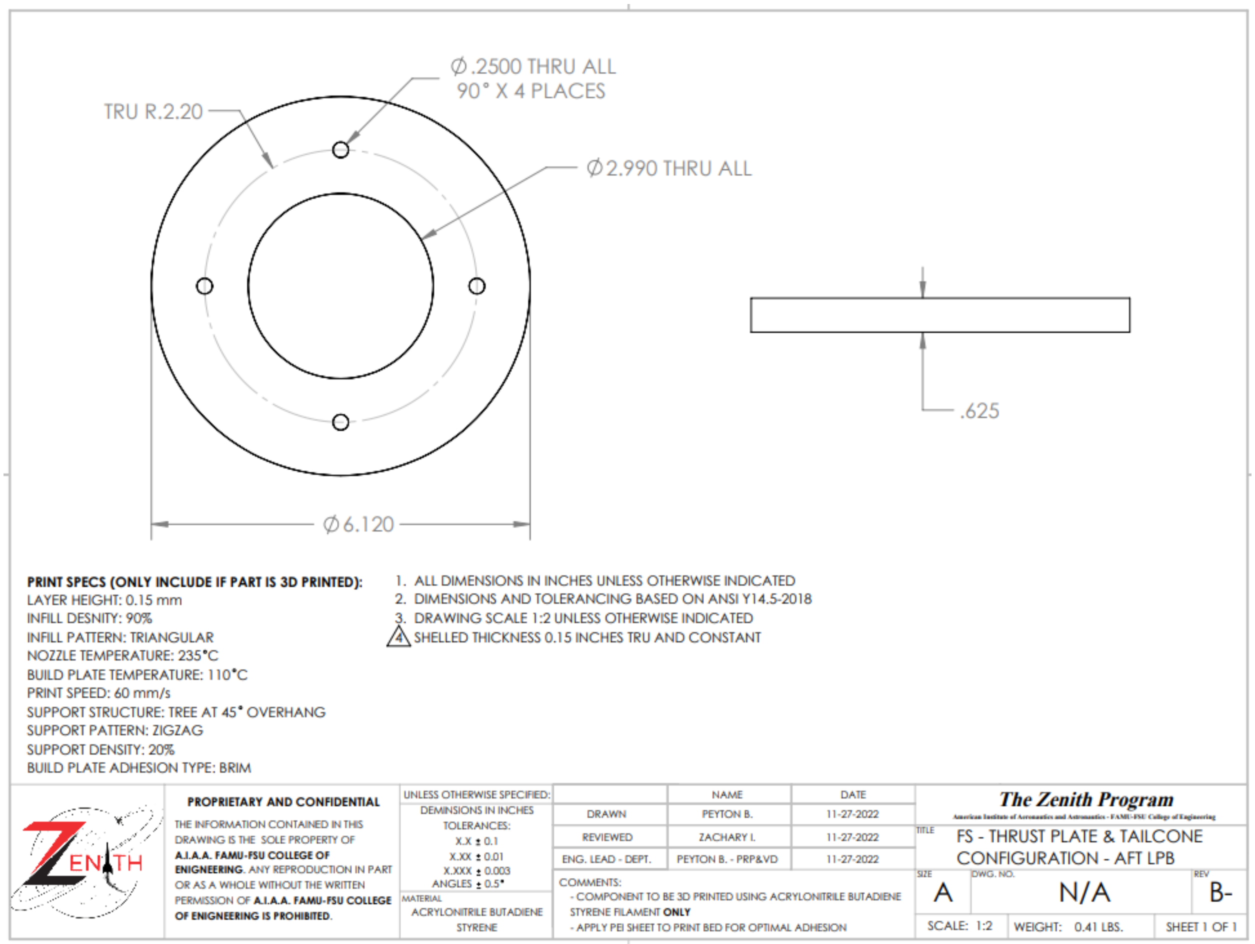


Figure 3-29. Tail Cone CAD Drawing

3.1.1.5 <u>Vehicle Component Weights</u>

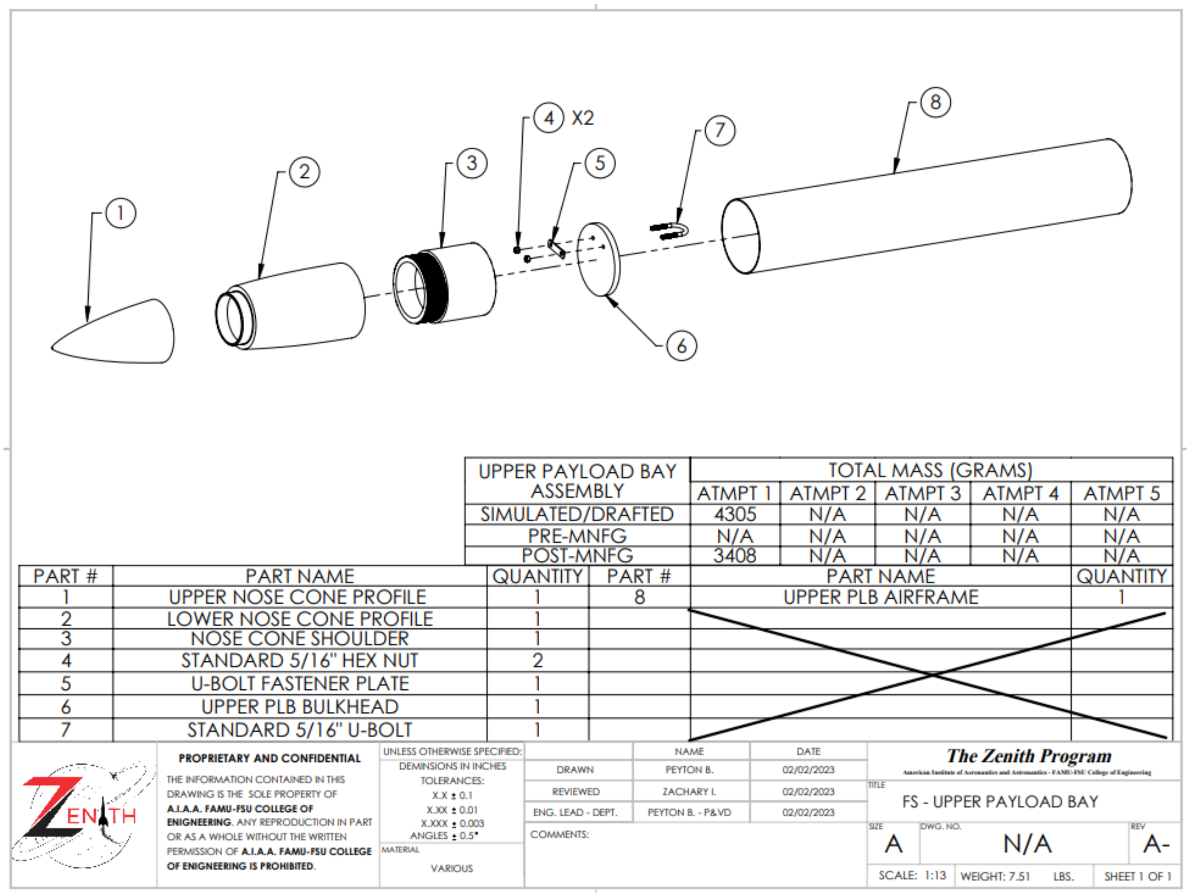
Table 3-3. Vehicle Weight Components

Component	Weight (lbs.)
Upper Payload Bay	12.718
Nose Cone	5.297
Nose Cone Bulkhead	0.319
U-Bolt with Hex Nuts/Washers/Fastener Plate	0.137
Quick Link	0.079
Airframe	2.2156
Main Parachute	1.188
Shock Chord	0.172
Payload	3.307

Avionics Bay	4.822
Airframe Coupler	0.069
Avionics Sled/2xThreaded rods	1.788
Inner Bulkhead x 2	0.503
Outer Bulkhead x 2	0.534
U-Bolt with Hex Nuts/Washers/Fastener Plate	0.53
Quick Link	0.079
CO2 Charges x 4 Loaded	0.811
Electronics	0.508
Fin Can	14.341
Airframe	2.764
Motor Tube	0.388
Flanged Motor Retention System	0.306
Motor Tube Centering Ring	0.238
Fin-Cut Centering Rings x 2	0.476
PETG Fin x 4	5.194
Shock Chord	0.172
Drogue Parachute	0.137
Eyebolt	0.149
1515 Rail Button x 2	0.0423
Threaded Rod x 4	0.434
Thrust Plate	0.410
Motor Case (w/out propellants)	3.631

3.1.2 Manufacturing and Assembly for Tested Vehicle

3.1.2.1 Upper Payload Bay and Nose Cone Configuration



UPPER PAYLOAD BAY ASSEMBLY	TOTAL MASS (GRAMS)				
	ATMPT 1	ATMPT 2	ATMPT 3	ATMPT 4	ATMPT 5
SIMULATED/DRAFTED	4305	N/A	N/A	N/A	N/A
PRE-MNFG	N/A	N/A	N/A	N/A	N/A
POST-MNFG	3408	N/A	N/A	N/A	N/A

PART #	PART NAME	QUANTITY	PART #	PART NAME	QUANTITY
1	UPPER NOSE CONE PROFILE	1	8	UPPER PLB AIRFRAME	1
2	LOWER NOSE CONE PROFILE	1			
3	NOSE CONE SHOULDER	1			
4	STANDARD 5/16" HEX NUT	2			
5	U-BOLT FASTENER PLATE	1			
6	UPPER PLB BULKHEAD	1			
7	STANDARD 5/16" U-BOLT	1			

Figure 3-30. UPB and Nose Cone Configuration

3.1.2.1.1 Nose Cone

The nose cone will be manufactured into three sections due to limited build volume on our 3D printer. The nose cone is 20 inches in length with a 4.5-inch shoulder coupler.

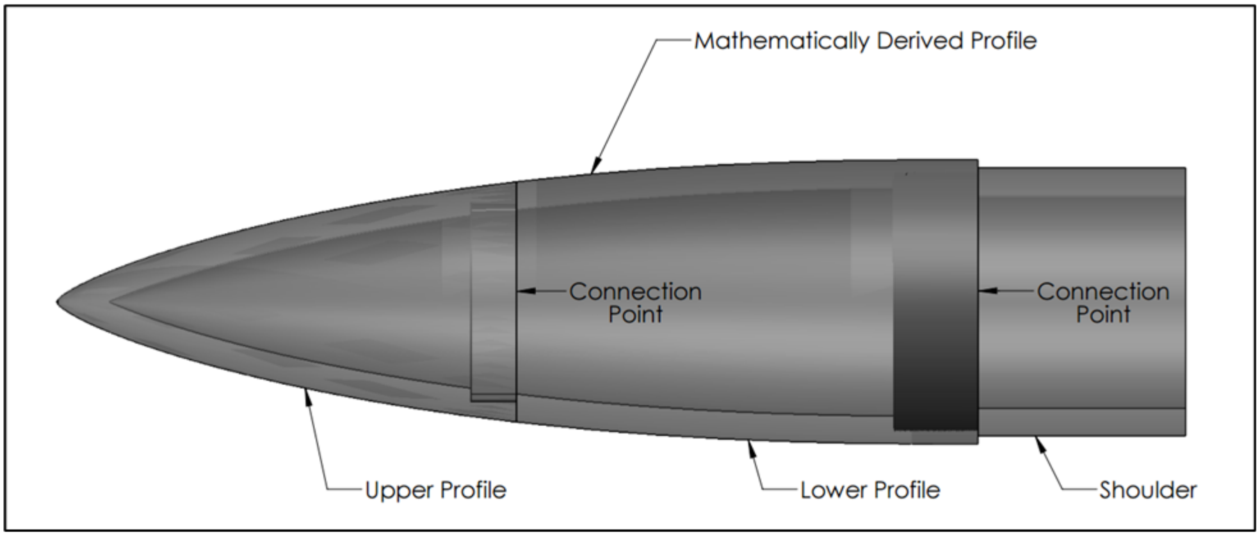


Figure 3-31. Nose Cone Configuration

The nosecone configuration is separated into three sections: upper profile, lower profile, and shoulder. The configuration of the nosecone was ultimately separated into three sections for ease of manufacturing. The upper profile is connected to the lower profile by a cylindrical shoulder. The shoulder, receiving end, and connection faces between the two sections will be epoxied together. The lower profile is connected to the shoulder coupler by a threaded cylindrical shoulder and the two sections (Lower Profile and Shoulder) will also be epoxied together. 3D printing specifications and notes have been included in the drawings included in the drawings. While printing, the specifications on the drawings will be implemented into the 3D printing software. For this component, Ultimaker Cura was used as the cross-functional 3D printing software.

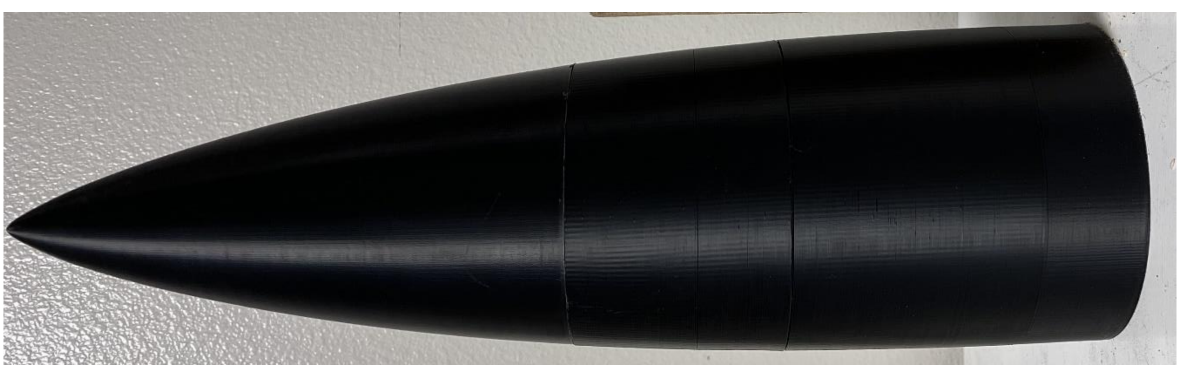


Figure 3-32. 3D Printed Nose Cone

The printing orientation shown below for each component is in accordance with the printing specifications on the engineering drawings. The print orientation of the object affects the placement and amount of structure was used to support over-hanging sections of the component. The printing support is not viewable in the images below due to an internal over-hand support structure.

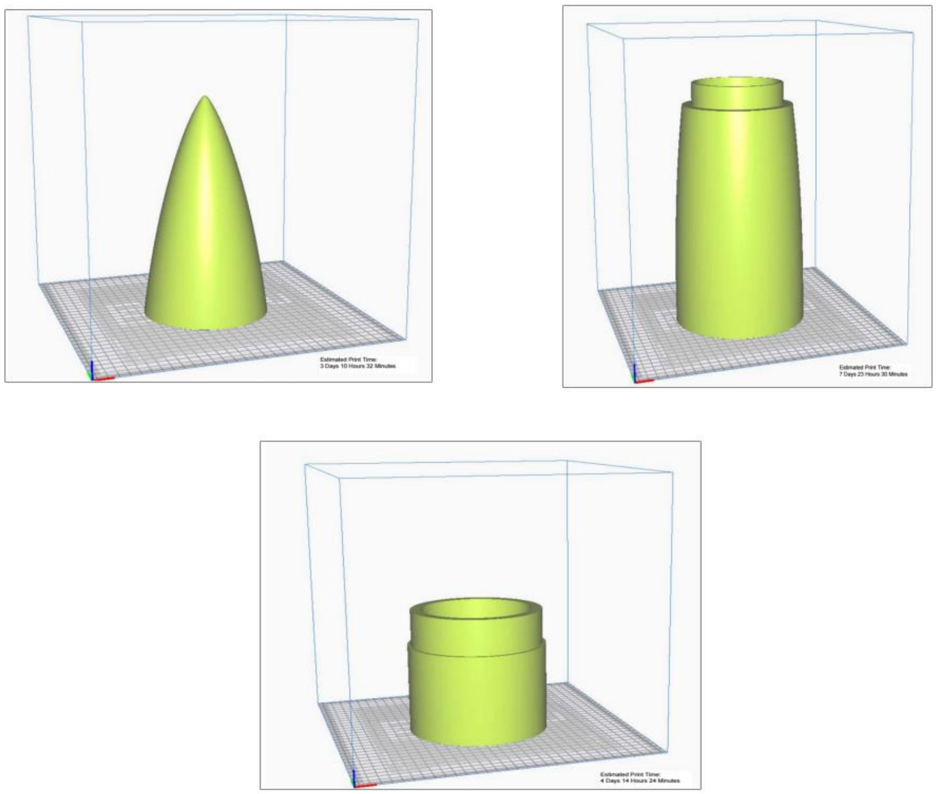


Figure 3-33. Nose Cone Sections Printing Orientation

Table 3-4. 3D Printing Specifications

Ultimaker Cura Printing Specifications	
Material	ABS
Layer Height	0.15 mm
Infill Density	85%
Infill Pattern	Triangular
Nozzle Temperature	235°C
Build Plate Temperature	110°C
Print Speed	60 mm/s
Support Structure	Tree Structure at 45° Overhang
Support Pattern	Zigzag
Support Density	10%

Build Plate Adhesion Type	10 mm thick Brim

3.1.2.1.2 Upper Payload Bay Airframe

The upper payload bay consists of a 35-inch-long blue tube airframe with a forward bulkhead that sits flush against the nosecone shoulder.

Figure 3-34. Upper Payload Bay Airframe

The bulkhead and U-bolt configuration is located at the front of the airframe, this allows the nose cone shoulder coupler to sit flush against the bulkhead and also ensures the main parachute shock chord will have a connection point to the vehicle (via U-Bolt). The upper payload bay airframe is cut from one of the two 48-inch airframes using a Dremel and cutting disk. The holes of the nose cone bulkhead are radially off set and were made using a power drill and a 5/16” bit. All residual wood splitting surrounding the holes was sanded down for a smooth surface.

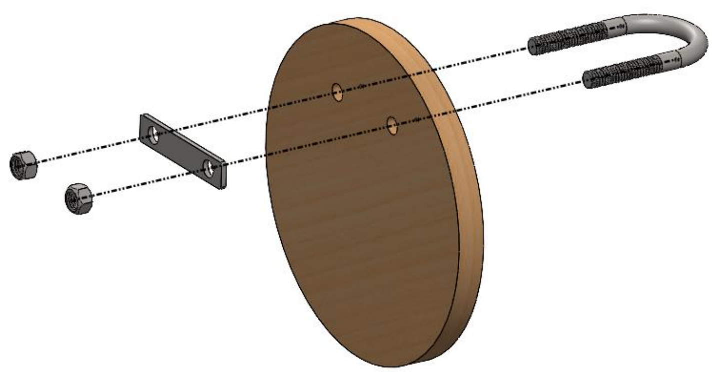


Figure 3-35. Bulkhead U-bolt Interface

The shoulder was epoxied into the airframe until the lip was pressed firmly against the top of the airframe tube. The epoxy was left to cure for 1 hour and then sandpaper was used to remove excess epoxy below the nosecone shoulder inside the airframe.

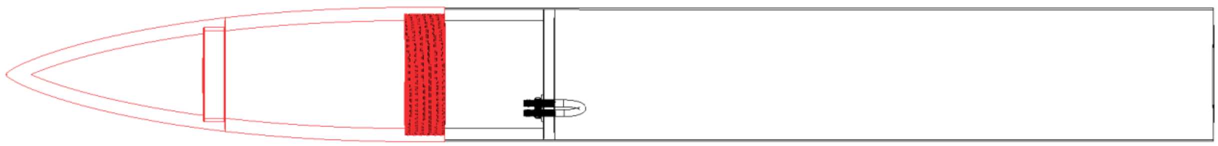
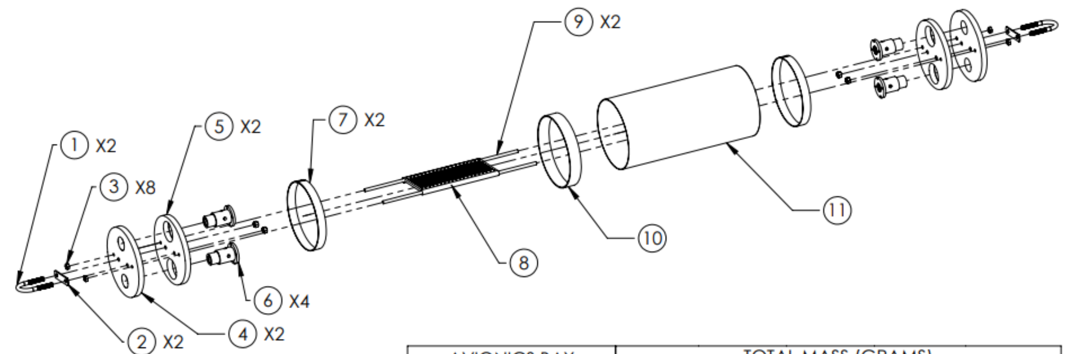


Figure 3-36. Nose Cone to UPB Interface

After the nosecone epoxy has set and the residual epoxy has been sanded down, a 1" ring of epoxy was applied directly below the nosecone shoulder and a layer was applied around the foreword bulkhead surface. The bulkhead was the oriented such that the U-bolt is facing away from the nosecone and then pushed through the airframe until it contacted the nosecone shoulder. A ring of instant-cure gap filler (purple super glue) was then applied at the bulkhead to airframe interface to seal the gap.

Figure 3-37. Mostly Assembled Vehicle

3.1.2.2 Avionics Bay



AVIONICS BAY ASSEMBLY	TOTAL MASS (GRAMS)				
	ATMPT 1	ATMPT 2	ATMPT 3	ATMPT 4	ATMPT 5
SIMULATED/DRAFTED	2056	N/A	N/A	N/A	N/A
PRE-MNFG	N/A	N/A	N/A	N/A	N/A
POST-MNFG	2187	N/A	N/A	N/A	N/A

PART #	PART NAME	QUANTITY	PART #	PART NAME	QUANTITY
1	5/16" U-BOLT	2	7	INNER AVIONICS BAY RING	2
2	U-BOLT PLATE	2	8	AVIONICS WIRING SLED	1
3	1/4" HEX NUT	8	9	THREADED ROD	2
4	OUTER AVIONICS BAY BULKHEAD	2	10	AIRFRAME SPACER	1
5	INNER AVIONICS BAY BULKHEAD	2	11	AVIONICS COUPLER	1
6	CO2 EJECTION CHARGES	4			

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X.XXX ± 0.003
ANGLES ± 0.5°
MATERIAL
VARIOUS

	NAME	DATE
DRAWN	PEYTON B.	02/02/2023
REVIEWED	ZACHARY I.	02/02/2023
ENG. LEAD - DEPT.	PEYTON B. - P&VD	02/02/2023

COMMENTS:

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TITLE
FS - AVIONICS BAY ASSEMBLY
SIZE A DWG. NO. N/A REV A-
SCALE: 1:9 WEIGHT: 4.53 LBS. SHEET 1 OF 1

Figure 3-38. Exploded View of Avionics Bay

3.1.2.2.1 Avionics Sled

Using a CAD model, the avionics sled was be printed from PLA filament, as this component takes no load during flight. The sled model includes a flat grid to zip-tie avionics components on to, supported by two longitudinal tubes through which the avionics bay threaded rods were ran. The print orientation is crucial for this component.

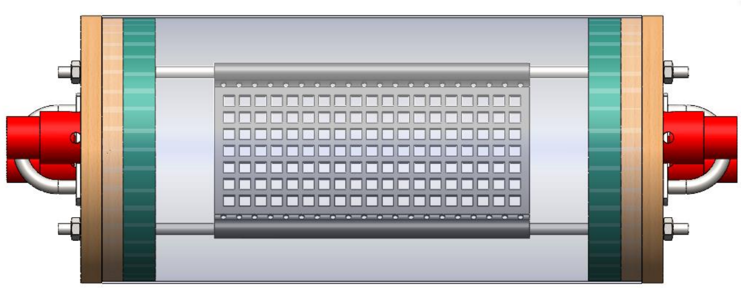


Figure 3-39. CAD Model of Avionics Bay

The avionics sled was printed vertically such that no supports were generated within the longitudinal tubes. Supports must be generated in the voids of the gridded plate in order to execute this print orientation. Residual material within the longitudinal tubes may prevent the threaded rods from passing through, and cannot easily be removed, thus the requirement for a vertical print orientation.

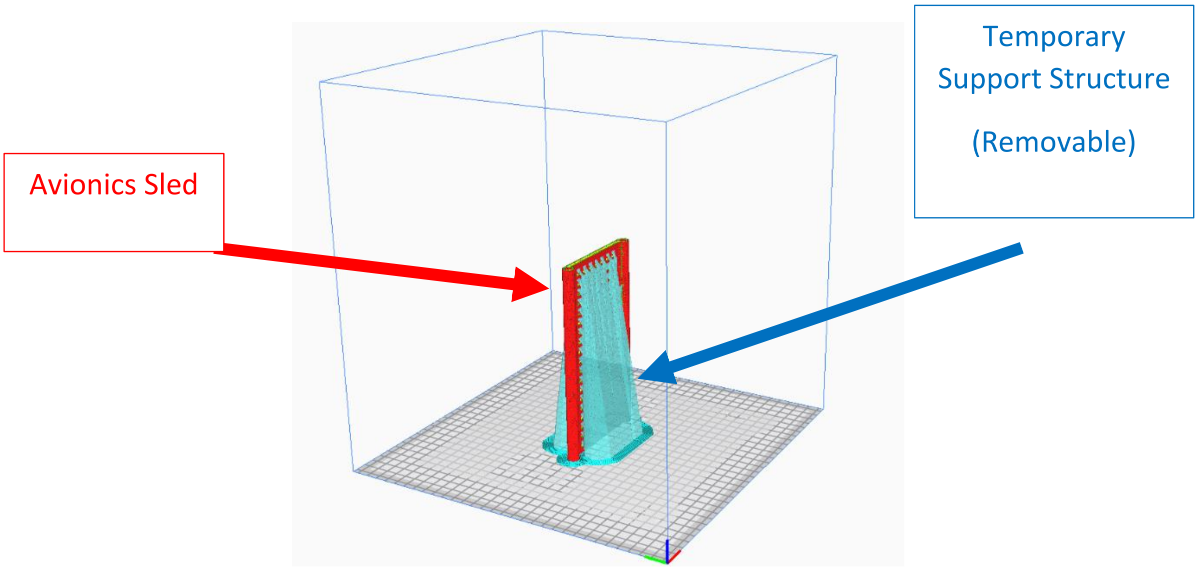


Figure 3-40. Printing Configuration of Avionics Sled

3.1.2.2.2 Avionics Placement and Wiring

The 3.4V LiPo batteries ordered for the program came with only change cables pre-installed. To make the batteries useful, leads were installed to the terminals. The outer protective casing (grey plastic material) was peeled away to expose the battery terminals. The red charging lead runs to the positive (+) terminal and the black to the negative (-) terminal. Two jumper wires, one red and one black, and a wire stripper was used to remove one end of the wire such that 1/8" of bare copper wire is unsheathed. A soldering iron and electrically conductive non-lead solder was substituted to attach the red jumper wire to the (+) terminal and the black jumper to the (-) terminal, ensuring that the charging leads were not melted off in the process. The kill switch was placed between one of the leads and the flight computer by stripping the end of the lead and the switch lead. The two exposed copper wires were twisted together and wrapped firmly with electrical tape. The two altimeter circuits are independent and run down opposite sides of the avionics sled. The GPS circuit is centrally located and does not include a kill switch, as it does not command pyrotechnics or energetics.

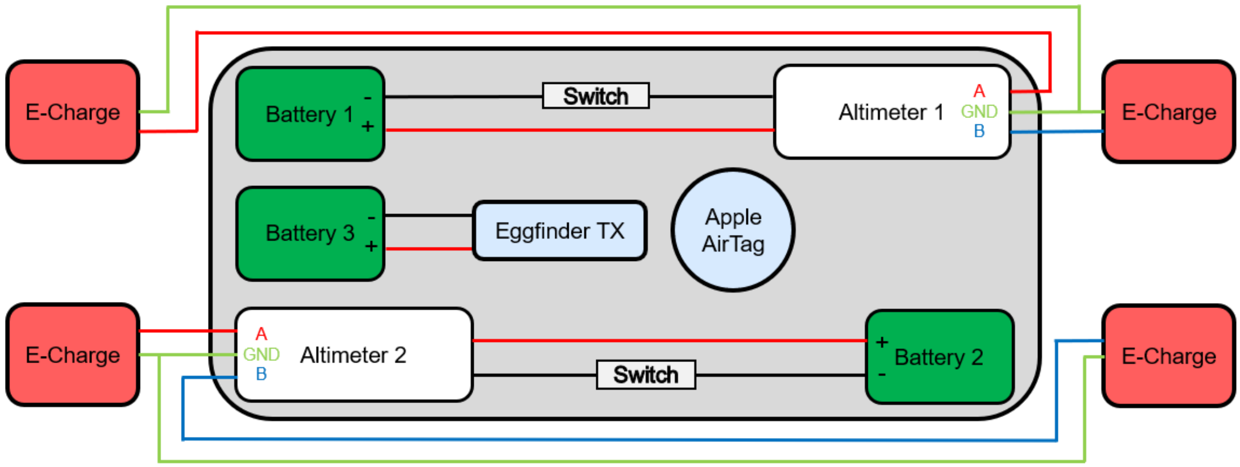
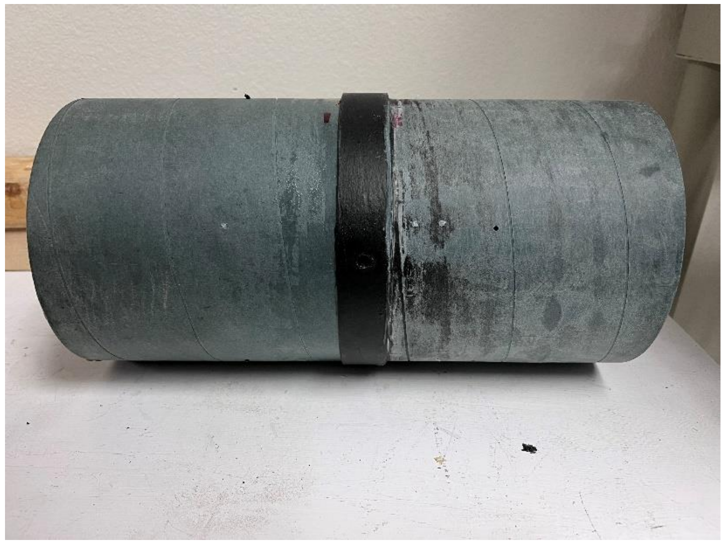


Figure 3-41. Avionics Wiring Diagram

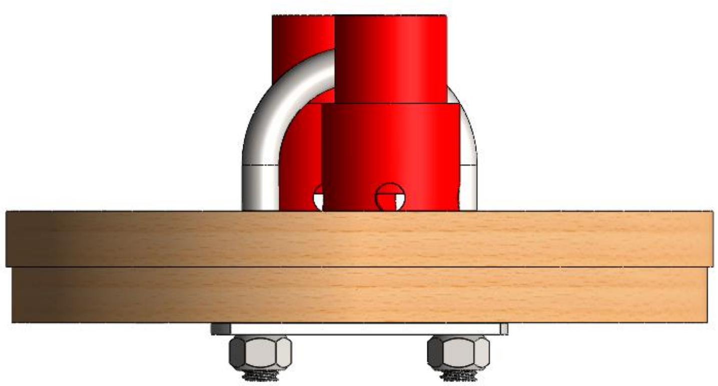
The wiring diagram excludes the GPS circuit as it is fully self-contained and inert. Each flight computer is wired to a drogue chute ejection charge (A line) and a main chute ejection charge (B line), to be fired at different points in the flight. The A and B lines are both (+) channels that share a common (-) channel, shown in green and labeled GND. The altimeters are activated by deactivating the kill switch on the (-) terminal of their respective batteries (refer to section 3.2.1).

3.1.2.2.3 Avionics Bay Assembly

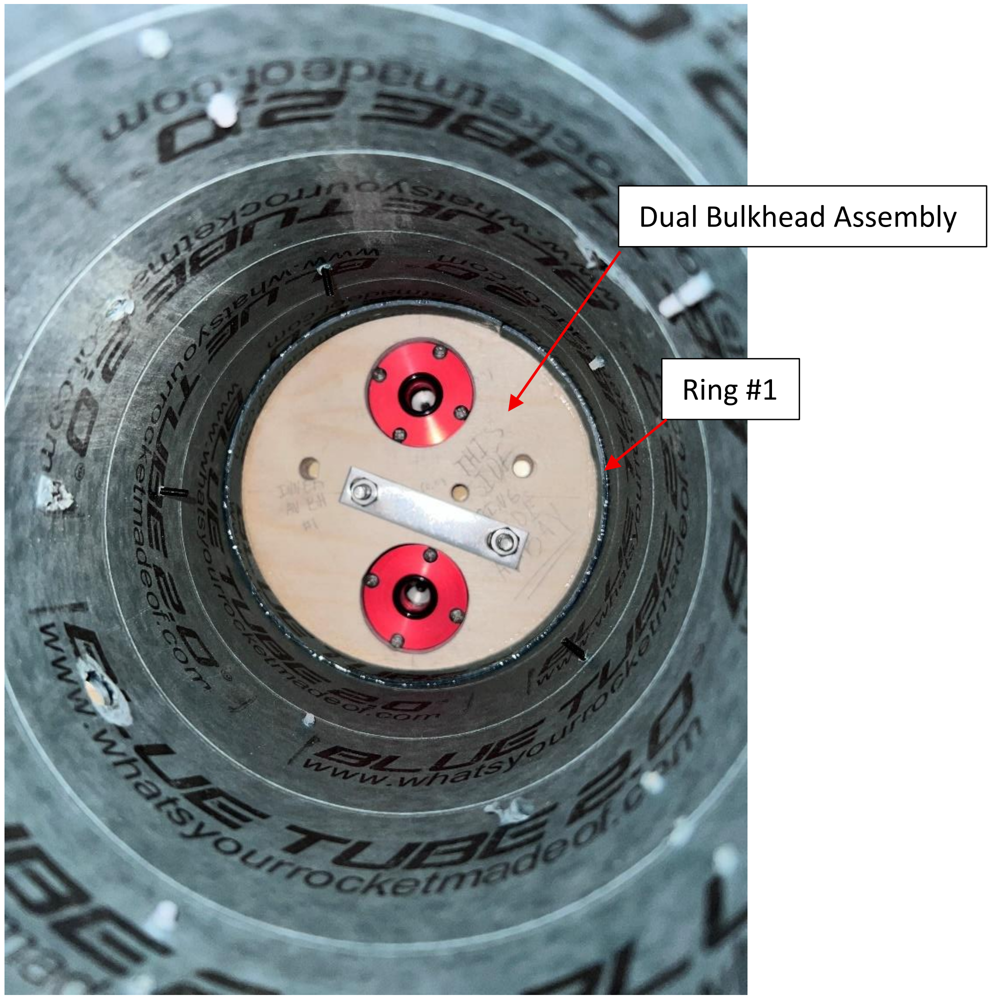
The airframe spacer is a 1” wide ring which sits at the center of the exterior of the avionics bay, creating a gap between the upper payload bay and lower payload bay airframe sections. The length of the avionics bay (without bulkheads) was measured and confirmed to 12” in length. A mark was made at the center point 6” from the end of the coupler. A mark was then made 0.5” from the central marking to create a 1” inch outline of where the ring should be placed. A thin layer of epoxy was then applied covering the measured 1”. The ring was then slid over the coupler to line up with the initial markings and left to set for 1 hour.



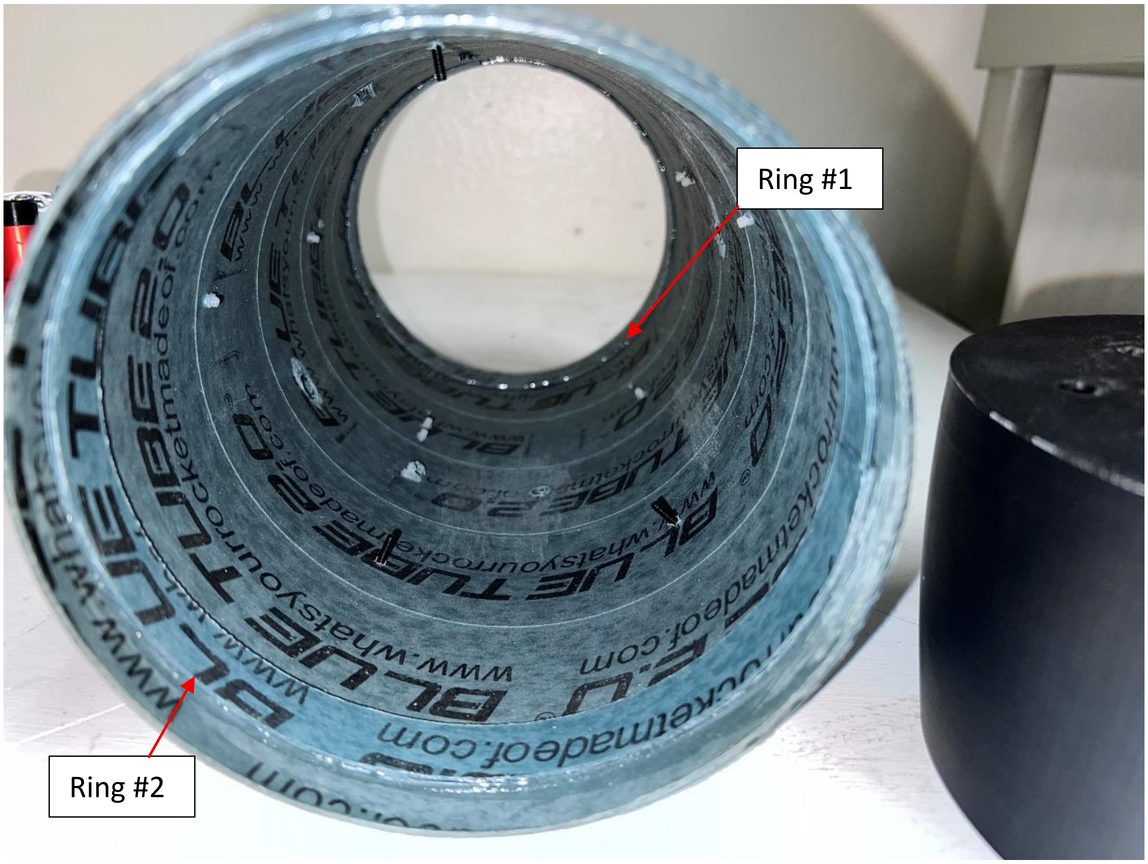
All Bulkhead modifications for the avionics bay were made by tracing 1:1 engineering drawings on the bulkhead and slowly scoring the wood in the traced areas. For future manufacturing, CAD models of both the inner and outer bulkheads will be used to program a CNC router in our in-house manufacturing shop. The scoring was a supplemental method used in the absence of the router for the CNC wood router. After bulkheads were manufacture, the airframe bulkheads were stacked on top of the coupler bulkhead to align the pre-drilled holes for the U-bolt. The bulkheads were the clamped firmly together and firmly affixed to a worktable. A power drill was used to enlarge the pre-drilled U-bolt holes at the center of the bulkhead from 1/4” to 5/16” for ease of assembly. The threaded ends of the U-bolt were passed through their respective holes and the fastener plate was positioned appropriately on the U-bolt legs. A wrench was then used to tighten the hex nuts of the U-bolts. The two flanged bases of the CO2 ejection system were fed through the holes cut in the bulkhead and screwed into place using hex head screws. The flange sits flush against the inner bulkhead and cylindrical tower extends through the outer bulkhead. This process was repeated until dual bulkhead assemblies were created.



The spacing rings are rings of blue tube approximately 0.7” in width which are epoxied inside the avionics bay for the inner bulkhead to rest against. A dual bulkhead assembly was placed at one end of the coupler. The inner bulkhead sits inside the tube and the outer bulkhead acts as a stopper, sitting flush against the outer edge of the bay. From the open end of the bay, a sharpie was used to mark out the location of the base of the inner bulkhead. The bulkheads were then removed, and a light ring of epoxy was applied to the marked area The ring was then inserted into the coupler and the dual bulkhead assembly was insert behind the ring to ensure the ring was position correctly.



After the epoxy was set, sandpaper was used to remove excess epoxy between spacing ring and edge of tube to allow for clean seating of inner bulkhead. This process was repeated on both sides of the avionics coupler.



The avionics sled and electronics were loaded inside the coupler and the dual bulkhead assemblies were attached on each side of the coupler. Electrical wires were routed through holes in the bulkhead assembly. The table below summarizes the holes that must be drilled in the avionics bay for various purposes:

Hole Size (Drill Bit)	Quantity	Location	Purpose
1/4"	1 per bulkhead	No exact requirement. Midway between ejection charges optimal.	Pass wires from flight computers though bulkheads to connect to e-matches
1/4”	3	Equally spaced around airframe spacer ring	Vent holes for flight computers to read pressure
3/8”	2	Adjacent on airframe spacer ring	Mount avionics kill switches internal. Pass push button through holes

3.1.2.3 Lower Payload Bay and Fin Can

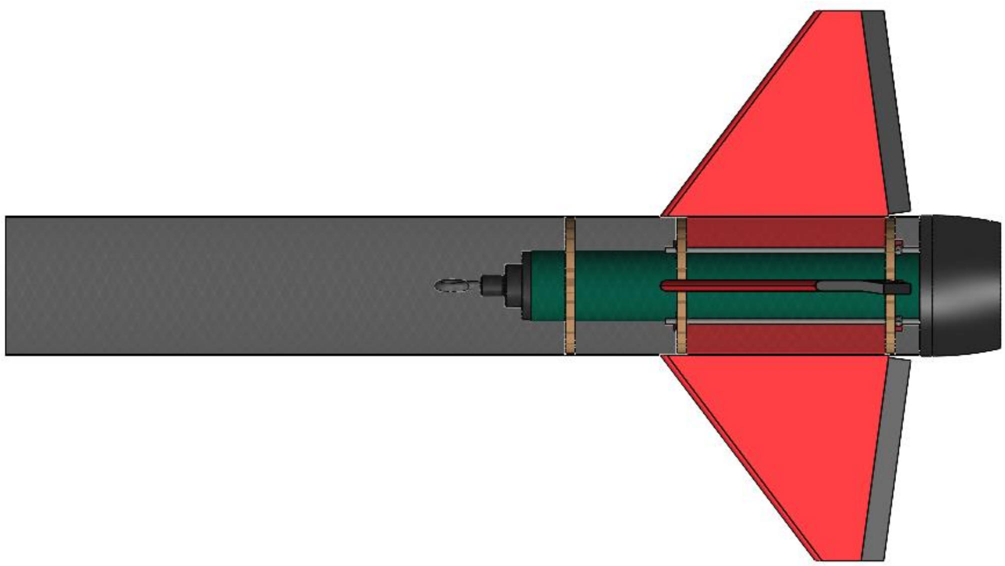
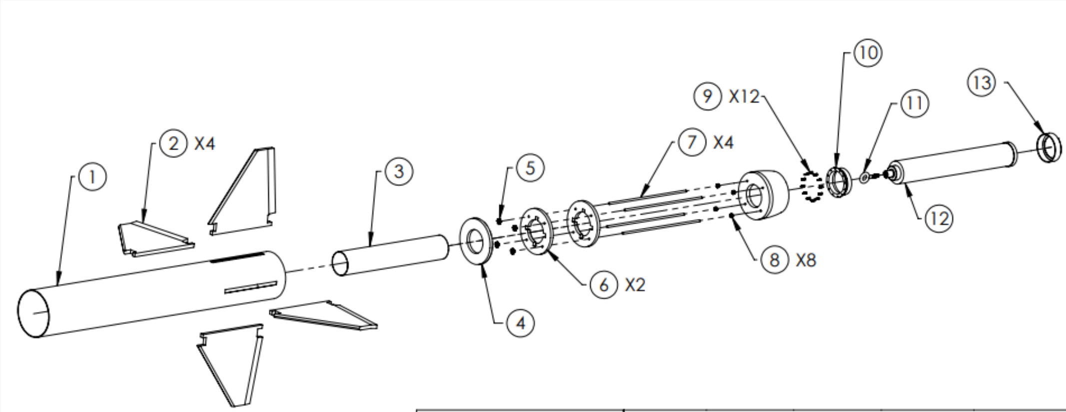


Figure 3-42. CAD Model of LPB and Fin Can



LOWER PAYLOAD BAY ASSEMBLY	TOTAL MASS (GRAMS)				
	ATMPT 1	ATMPT 2	ATMPT 3	ATMPT 4	ATMPT 5
SIMULATED/DRAFTED	4706	N/A	N/A	N/A	N/A
PRE-MNFG	N/A	N/A	N/A	N/A	N/A
POST-MNFG	4722	N/A	N/A	N/A	N/A

PART #	PART NAME	QUANTITY	PART #	PART NAME	QUANTITY
1	LOWER PAYLOAD BAY AIRFRAME	1	8	1/4" HEX NUTS	8
2	CLIPPED DELTA FIN	4	9	FLANGED MOTOR RETAINER SCREWS	12
3	MOTOR TUBE	1	10	FLANGED MOTOR RETAINER	1
4	FWD MOTOR TUBE CENTERING RING	1	11	5/16" FORGED EYE BOLT	1
5	1/4" T-NUT	4	12	PRO 75/3840 MOTOR CASE	1
6	FIN-CUT MOTOR TUBE CENTERING RING	2	13	FLANGED MOTOR RETAINER CAP	1
7	THREADED ROD	4	14		

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X.XX ± 0.01
X.XXX ± 0.003
ANGLES ± 0.5°
MATERIAL
VARIOUS

	NAME	DATE
DRAWN	PEYTON B.	02/02/2023
REVIEWED	ZACHARY I.	02/02/2023
ENG. LEAD - DEPT.	PEYTON B. - P&VD	02/02/2023

COMMENTS:

The Zenith Program
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TITLE: FS - LOWER PAYLOAD BAY
SIZE: A DWG. NO.: N/A REV: A-
SCALE: 1:13 WEIGHT: 10.4 LBS. SHEET 1 OF 1

Figure 3-43. Exploded View of LPB and Fin Can

3.1.2.3.1 Lower Payload Bay Airframe

The lower payload bay for the original design was cut using a Dremel for specific dimensions noted in the engineering drawing and then sanded down to fit a 3D printed fin plate.

Figure 3-44. Old Fin Slots on Airframe

The new airframe design shown below will be cut using the same method.

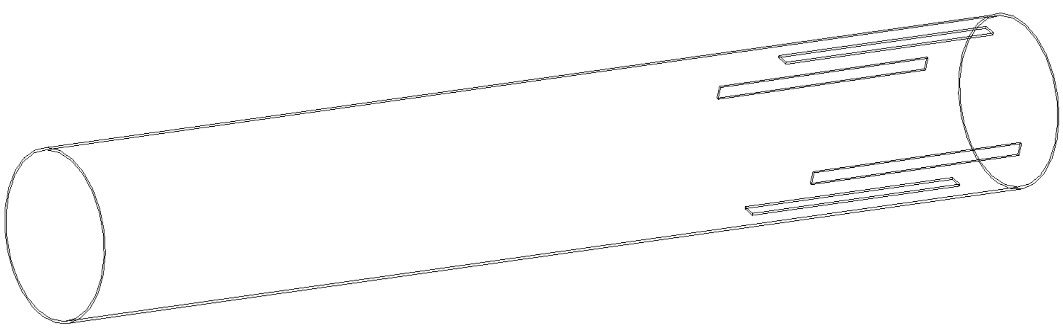


Figure 3-45. New Fin Slots on Airframe

3.1.2.3.2 Fin Manufacturing

The fins are 3D printed using PETG filament. Previous fin designs were modeled to have a base plate with a surface curve following the outer airframe diameter and were printed from ABS.

[Troubleshooting] ABS exhibits drastic deformation which ruined several prints. The increased layer adhesion in PETG solved this problem.

[Troubleshooting] The curved plate in the original design of the fins would frequently deform, and often did not properly align with the airframe. The new fin design will implement fins in which the entire fin passes directly through the airframe. Modeling clay will be used to create a smooth fin root fillet at the junction of fin and airframe.

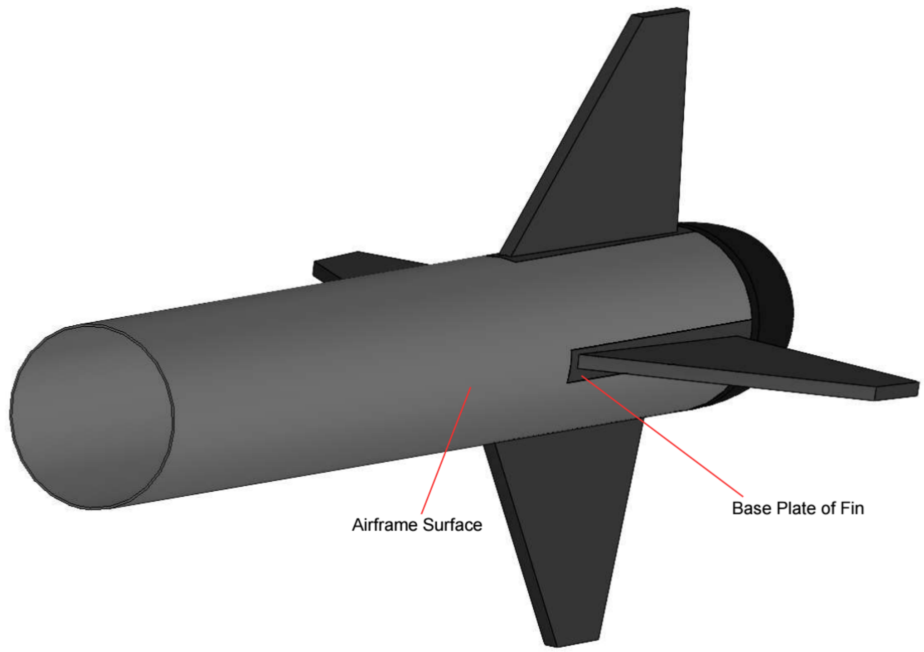


Figure 3-46. New Fin Can Design

The base plate was implemented to allow the fins to be canted off vertical to induce spinning stabilization. Due to structural printing deformation and fitment, the fin design was modified to a vertical stature with spin tabs located the end of the fins to replicate the canted feature in the last design.

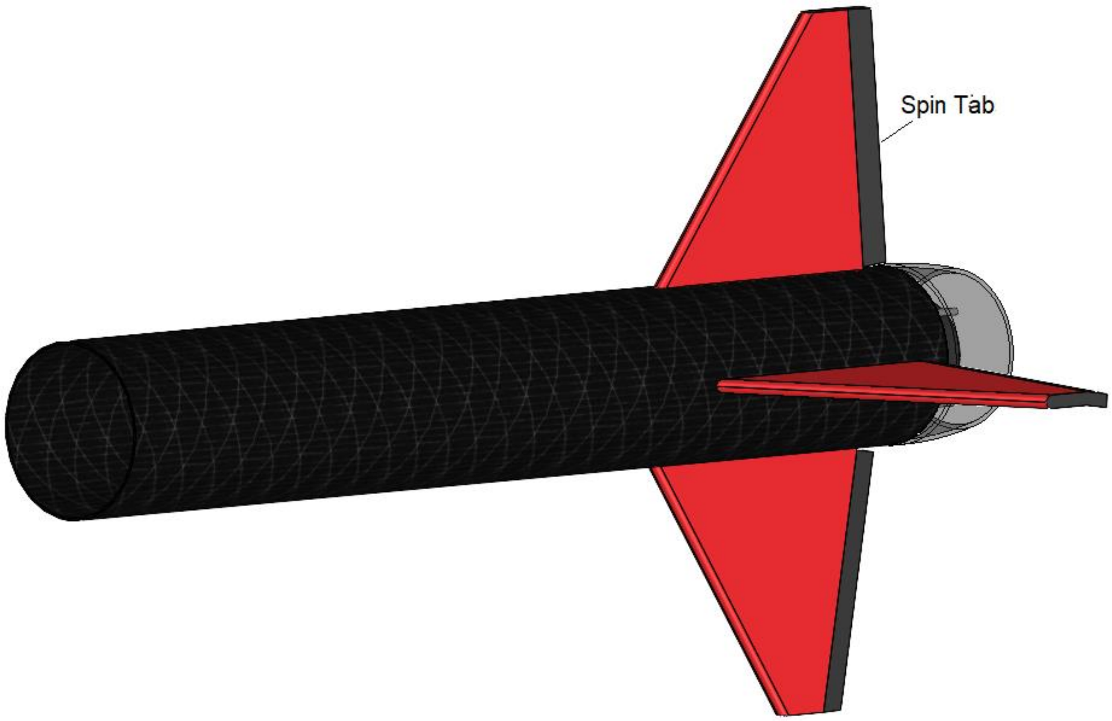


Figure 3-47. CAD Model of New Fin Can

The fins were printed vertically and supports were generated in the over-hanging section of the fin. All residual material was gently removed to ensure the fin tabs slid smoothly into the airframe and fin-cut centering rings. The following print orientation and printing specifications were executed.

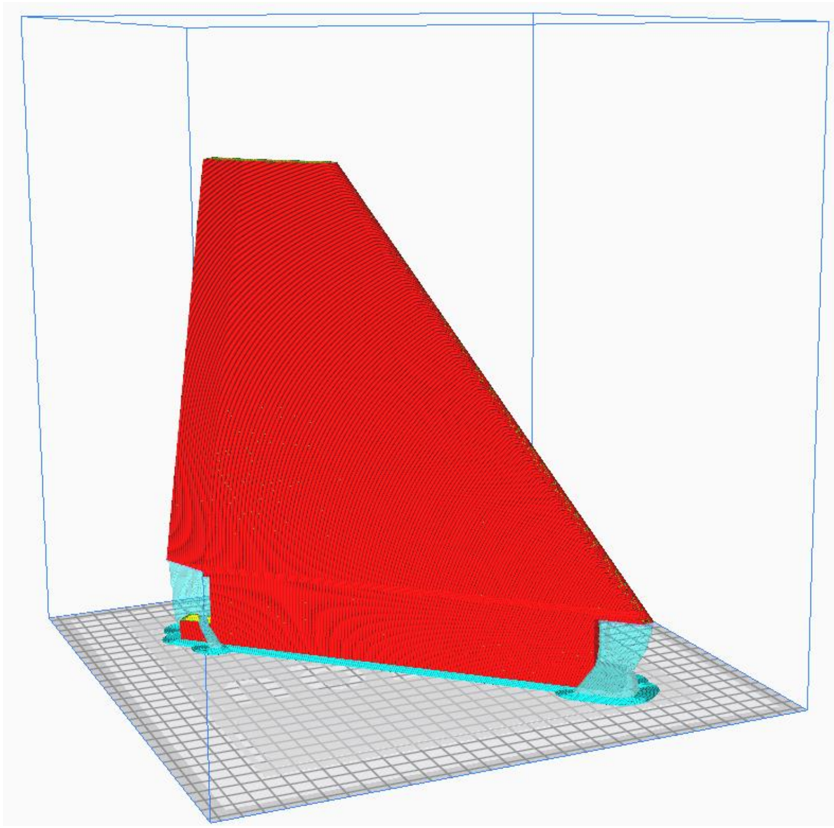


Figure 3-48. New Fin Design Print Orientation

Table 3-5. Fin 3D Printing Specifications

Ultimaker Cura Printing Specifications	
Material	PETG
Layer Height	0.15 mm
Infill Density	72%
Infill Pattern	Triangular
Nozzle Temperature	250°C
Build Plate Temperature	85°C
Print Speed	100 mm/s
Support Structure	Tree Structure at 45° Overhang
Support Pattern	Zigzag
Support Density	10%
Build Plate Adhesion Type	10 mm thick Brim

3.1.2.3.3 Centering Ring Manufacturing

The fin can is comprised of 3 centering rings. The forward motor tube centering ring is shipped with a hole cut through the middle to accommodate the outer diameter of the motor tube. The following two centering rings are shipped the same way with additional cutouts along the inner diameter of the rings. The slots in the centering rings prevent the fins from rotating and sandwich the fins together in place. This not only prevents the fins from moving, but also ensures the fin tabs are in constant contact with the motor tube surface, which helps translate thrust to the airframe. Manufacturing methods specified in section 3.1.2.2.3 were used to manufacture the fin-cut centering rings per the engineering drawings.

3.1.2.3.4 Tail Cone and Thrust Plate Configuration Manufacturing

The thrust plate and tail cone were printed as one component from ABS. As all printers have a printing tolerance, the tail cone and thrust plate configuration was printed with a smaller center hole need to allow the team to sand out the hole for a flush fit with the motor case. To account for ABS deformation, the thrust plate was printed 1/8” longer than need so the team could sand the plate down to a flush surface. The following print orientation and printing specifications were executed.

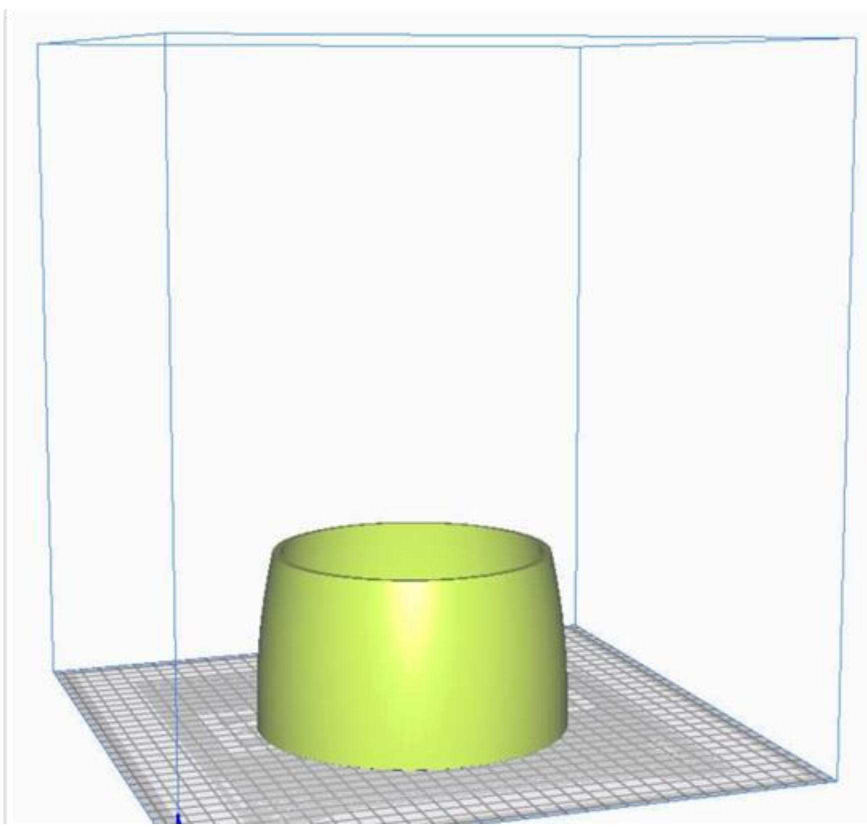


Figure 3-49. Tail Cone 3D Printing Orientation

Table 3-6. Tail Cone Printing Specifications

Ultimaker Cura Printing Specifications	
Material	ABS
Layer Height	0.15 mm
Infill Density	90%
Infill Pattern	Triangular
Nozzle Temperature	235°C
Build Plate Temperature	110°C
Print Speed	60 mm/s
Support Structure	Tree Structure at 45° Overhang
Support Pattern	Zigzag
Support Density	N/A
Build Plate Adhesion Type	10 mm thick Brim

Figure 3-50. Tail Cone

3.1.2.3.5 Motor Retainer Installation

The motor retainer installation is comprised of the flanged motor retainer, threaded adapters, and socket head cap screws.

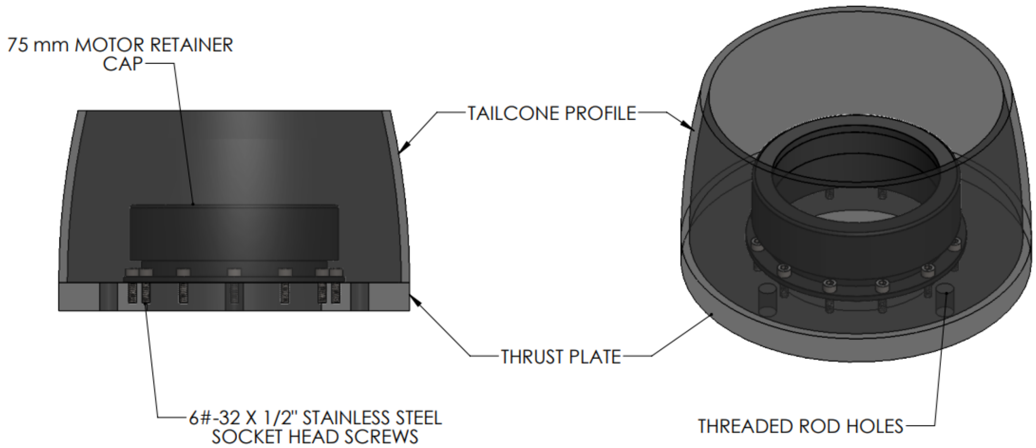


Figure 3-51. Labeled Tail Cone

After sanding, the diameter of the hole in the thrust plate is slightly larger than the PRO 75 motor case diameter to ensure the motor can slide in and out of the thrust structure. To ensure the threaded adapters are positioned correctly around the hole in the thrust plate, the motor case was fed through the thrust plate to center the flanged motor retainer and mark the socket head cap screw holes in the motor retainer lip. Power tools were used to drive the threaded adapters into the thrust plate. The holes in the retainer lip should align with each threaded adapter and the socket head cap screws should have no issue being screwed in and out. A layer of fiberglass was epoxied to the inner surface of the tail cone by epoxying stacked fiber glass sheets together and then epoxying the sheets inside of the tail cone.

Figure 3-52. Mounted Tail Cone

3.1.2.3.6 Forward and Central Ring Installation

The motor tube is fabricated from the same material as the airframe and was delivered longer than needed. The same cutting methods mentioned in section 3.1.2.2.3 were used to cut the motor tube to the desired length of 17 inches. A ring of epoxy was then spread 1 inch down from the top of the motor tube and the forward motor tube centering ring was slide on the motor tube and set in place to dry. The middle centering ring must be fixed in place accurately for the fins to fit in the airframe correctly. The ring was epoxied 11 inches up from the after end of the airframe. After the forward motor tube centering ring and the forward fin-cut centering ring were epoxied to the motor tube and the epoxy was set, the structure was epoxied into the airframe. A generous ring of epoxy was applied to both centering rings after inserted in the airframe. The motor tube lays flush against the surface of the thrust plate, so the bottom end of the motor tube was aligned with the aft end surface of the airframe before epoxying the structure into the airframe. Once the epoxy is set, from the top of the lower payload bay, apply instant-cure gap filler at the interface of the forward bulkhead and the airframe to fill any voids left by the epoxy. Flip the airframe and do the same at the interface of the middle centering ring.

3.1.2.3.7 Fin and Tail Cone Installation

The forward tab of each fin was clipped into the middle centering ring via the cutouts in the airframe. From the base of the vehicle, the aft end centering ring (removable centering ring) is pushed against the back end of the fin tab. The threaded rods are fed through their respective holes and tightened via hex nuts. Around 1.5” of threaded rod was left protruding from the base of the vehicle. This was to allow the tail cone and thrust plate configuration to be placed on the threaded rods. 1/4" hex nuts were used to tighten the thrust plate against the airframe surface.

Figure 3-53. Assembled Fin Can

3.2 Recovery Subsystem

The recovery system will ensure safety of the launch vehicle and all of its components upon descent. There are two recovery events, drogue parachute deployment and main parachute deployment. The two recovery events will be controlled by two completely independent Entacore AIM 3 altimeters. Each altimeter draws power from separate 3.7 V rechargeable Lithium-Ion batteries and are both armed with push-button kill switches for powering off the altimeters during pre-flight construction of the vehicle. The ejection charges will both be CO2 gas charges that are manufactured by Tinder Rocketry. These ejection charges were not only chosen to limit the extremity of the pyro events, but also to lower the safety concerns of using black powder that could severely damage components in either payload bay. The CO2 charge still requires the use of black powder, but only a small amount of it is housed in a sealed charge cup. Approximately 0.2 grams of 4F black powder is used to propel a puncture piston that penetrates the CO2 cartridge and releases gas into the payload bays. The black powder is loaded into a charge cup that has an e-match in it with the wire running through the bottom of the cup. The charge cup sits at the open-end of a pyro housing unit that holds it, along with the puncture piston and spring that is placed on top of the cup. The CO2 charges are designed to have the gas cartridge placed in the avionics bay and thread through a piece bolted to the outer bulkhead of the bay. On the other end of this bulkhead mount is where the pyro housing component is threaded onto it. The end of the bulkhead mount piece interfaced with the pyro housing has four equally distant vent holes to allow the CO2 gas to escape into the payload bay. The figure below shows the interface between the pyro housing, the bulkhead mount piece, and the outer bulkhead of the avionics bay.

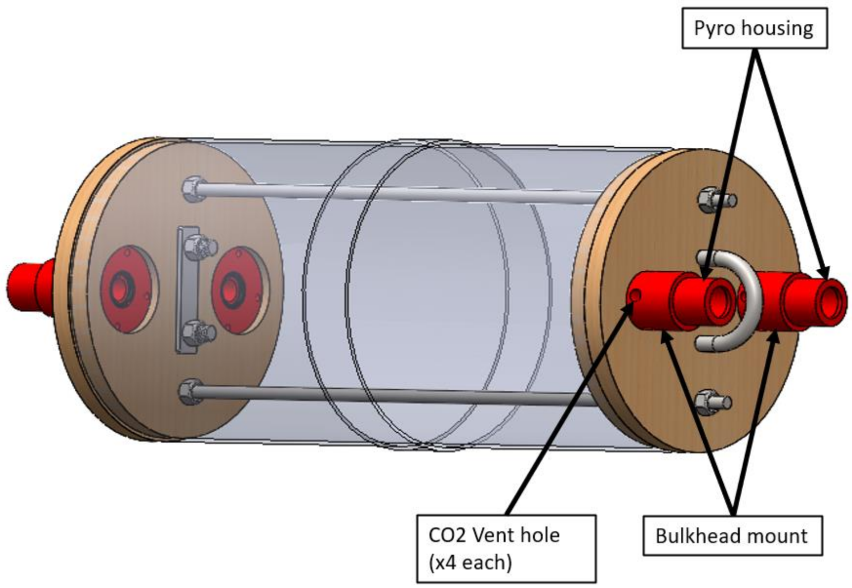


Figure 3-54. CO2 Ejection System and Bulkhead Interface

At apogee, the primary altimeter will send a current through the e-match that will fire the first ejection charge. The CO2 gas will rapidly pressurize the lower payload bay and shear the #4-40 nylon pins that connects it to the AV bay, deploying the drogue parachute. The secondary altimeter will fire the second charge on a time delay to ensure proper separation of the vehicle sections and also to not over pressurize the vehicle. Deployment of the main parachute is similar, with the only difference being that the payload will also be tethered to the recovery harness. The primary ejection charge for the main parachute is set to deploy at 550 ft AGL, and the redundant secondary charge will fire two seconds afterwards. Each parachute was bought off-shelf from a company called Fruity Chutes. A 24” classic elliptical parachute will be used for the drogue parachute and an 84” iris ultra-standard will be used as the main parachute. Both parachutes are tethered to 16 ft of 3000 lb rated 9/16-inch webbed Nylon, and 7 ft of 1500 lb rated tubular Kevlar shock cords that will be joined together to make up a total of 23 ft of shock cord. All connection points on the recovery harness are accomplished with the use of 1/4-inch quick links, except for the connection point to the eyebolt on the motor tube which requires the use of a 3/8-inch quick link. The two figures below illustrates the length and connection points on each recovery harness.

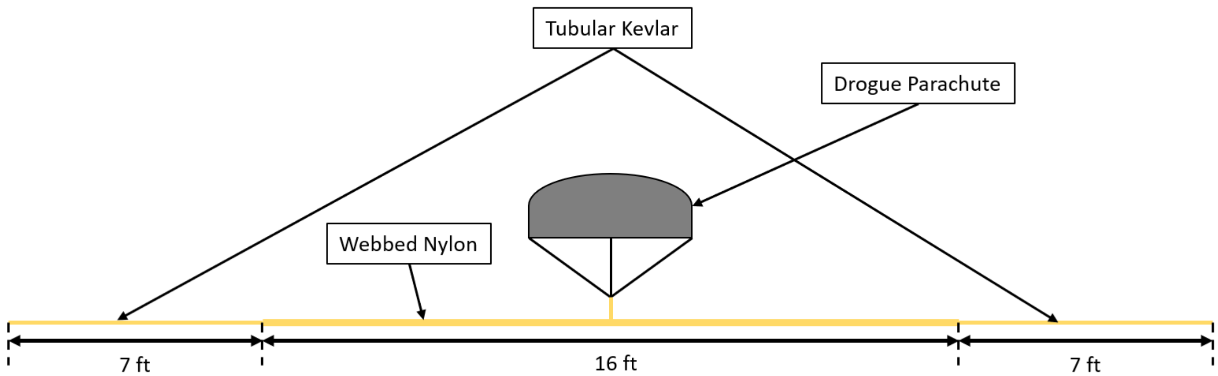


Figure 3-55. Drogue Parachute Recovery Harness Diagram

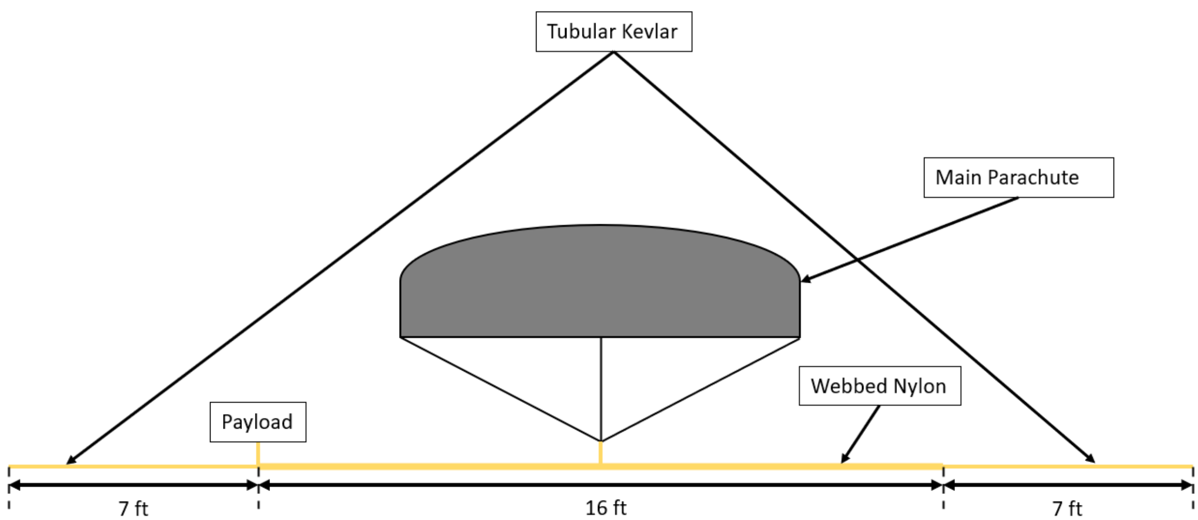


Figure 3-56. Main Parachute and Payload Recovery Harness Diagram

The vehicle should take approximately 80 seconds to recover, about 50 seconds under the drogue parachute and 30 seconds under the main parachute. The final descent rate of the vehicle under the main parachute is about 18.6 ft/s. Further details on descent speeds and times are discussed in section 3.3.2.

3.2.1 Altimeters

As mentioned previously the flight events will be monitored and controlled by two Entacore AIM 3 altimeters. Each altimeter will have its own set of batteries, wires, and push-button kill switches for redundancy. The push-button switches will intersect each wire between the negative terminals of the battery and altimeter. The AIM is equipped with two pyro channels, A and B, channel A for apogee and channel B for main. Channel A of each altimeter will send 6 amps of current to the CO2 ejection charges in the lower payload bay when the vehicle has reached apogee. Channel B of each will send the same amount of current to the charges in the upper payload bay at 550 ft AGL. For each ejection channel wire there is a ground wire paired to it, to allow current to flow through and fire-off the charges. The important characteristics to note of the AIM 3 is that it has a small form factor, it can store up to 30 min of flight data, it reads in data up to a max of 38,615 ft, and each comes with two pyro channels for ejection charges. Another good thing about the AIM 3 is that the software used to configure it is free and fairly simple to use. The following table lists the characteristics of said altimeter:

Table 3-7. AIM 3 Altimeter Specifications

Entacore AIM USB 3.0	
Form factor	70mm x 25mm x 15mm
Sensors	Pressure and temperature
Voltage input range (V)	3.7 - 12
Max Altitude (ft)	38,615
Ejection channels	2 (apogee and main)
Tracked data	Output status / continuity / battery voltage / temp. / time / altitude / velocity
Storage capacity	30 min of flight time
Sampling frequency	10 Hz
Other	Time delay options for pyro events / free software to alter settings / altitude beeps

3.2.2 GPS Trackers

There will be two separate GPS tracking devices used to locate the vehicle once it has fully recovered. The primary GPS module is an Apple AirTag device that will be placed at the center of the avionics sled. AirTag devices are equipped with Apple's U1 chip for ultra-wideband precision tracking that operates at a frequency of 6.24 GHz. The vehicle's location can be monitored in real time with the AirTag through the "Find My" app on any iPhone that is Bluetooth paired to the device. The AirTag also comes with a built-in CR2032 Lithium battery that lasts over a year before needing a replacement. It also comes with a built-in speaker for sonic locating. The secondary GPS unit is the Eggfinder TX transmitter and will act as a redundant tracking device in the case that the primary GPS unit fails. The Eggfinder TX requires a separate Eggfinder RX receiver module that hooks up to a laptop to track the vehicle's location. The receiver connects to a laptop with a USB cable and will be configured with either Google Earth or MapSphere to track the vehicle in real time. The Eggfinder TX operates at a frequency of 900 MHz and requires 100 mW to power. The Eggfinder will be powered by 3.7 V rechargeable lithium-ion battery. The electromagnetic fields generated by the GPS transmitters should not interfere with the altimeters and their functionality. The figures below are images of each GPS unit.

Figure 3-57. Apple AirTag GPS Tracking Device

Figure 3-58. Eggfinder GPS Tracking Device

3.3 Mission Performance Predictions

3.3.1 Target Altitude

As mentioned in both PDR and CDR, the Zenith program's declared altitude is 4,600 feet. This altitude was chosen based on multiple flight simulations at 10-15 MPH wind speeds at the coordinates of the launch competition.

3.3.2 Flight Profile and Stability Margin

After implementing accurate vehicle weights, multiple flight simulations were ran on a 12-foot launch rail with a 5-degree launch angle under windspeed conditions ranging from 5 to 15 MPH to obtain an updated flight profile.

3.3.2.1 Flight Profile

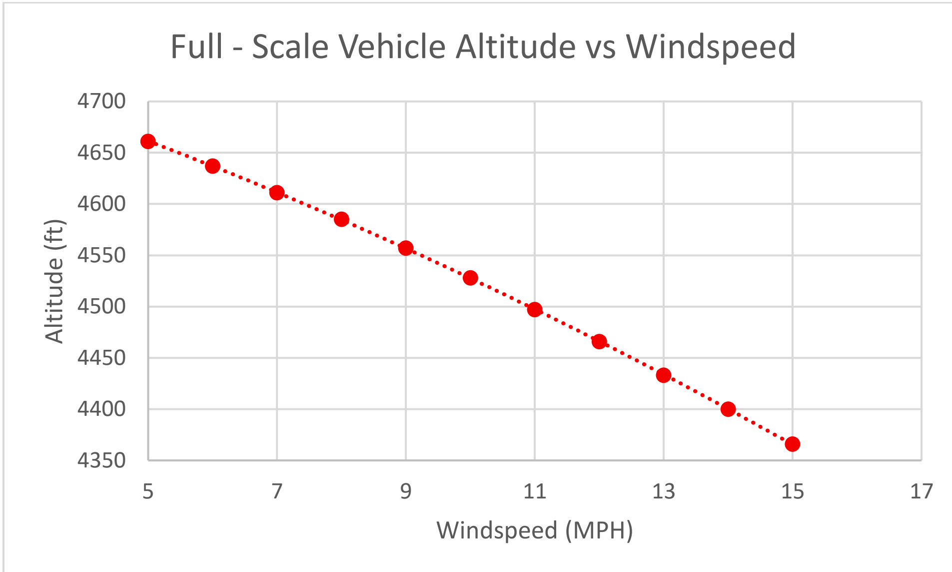


Figure 3-59. Altitude vs Wind Speed

The image above shows the maximum altitude of the vehicle decreases with increasing windspeed. As the vehicle's weight and component configuration deviates from the design used to predict the vehicle's apogee in PDR, the team expects the vehicle's competition apogee to be slightly under the declared altitude depending on the wind conditions. If the wind conditions stay within 5-11 mph, the vehicle will be able to reach within 100 feet of the proposed altitude. Shown below is the vertical/total velocity and acceleration versus time, along with the thrust curve simulated in OpenRocket Simulation software.

Full Scale Vehicle Vel vs Time

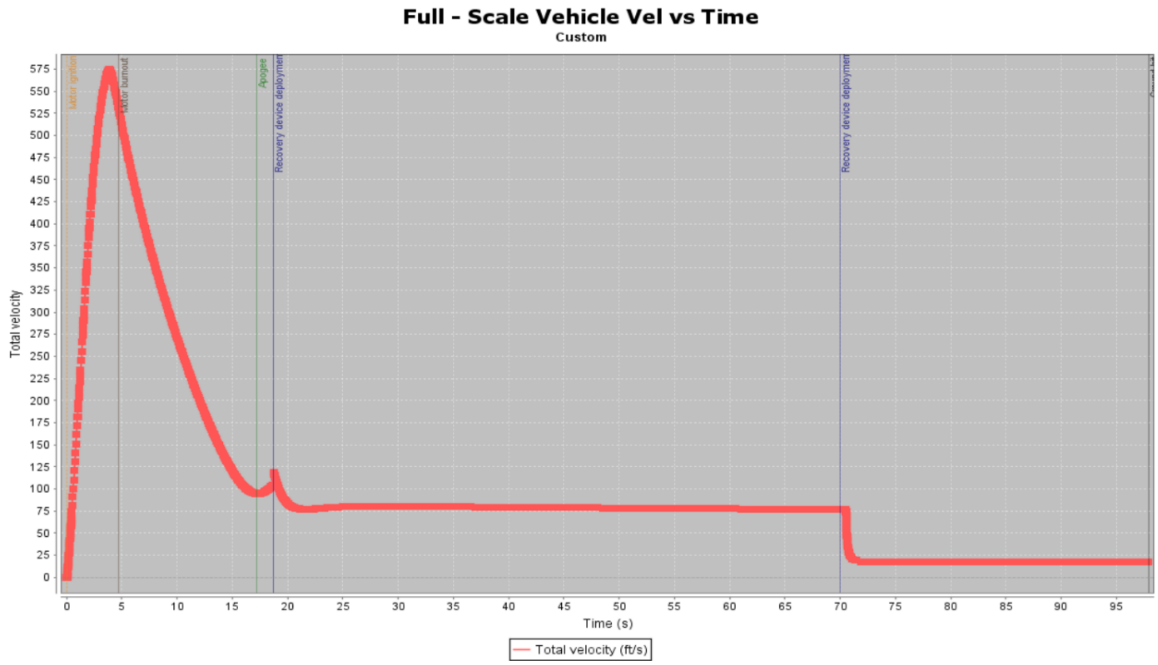


Figure 3-60. Velocity vs Time

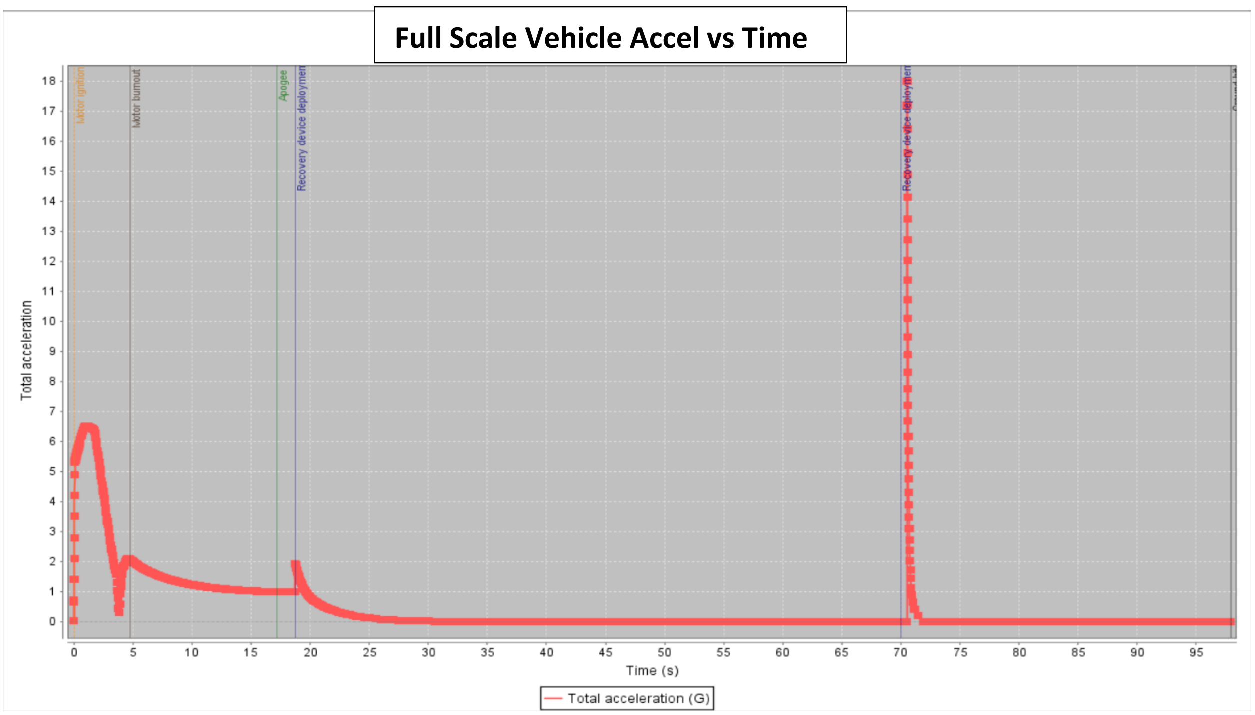


Figure 3-61. Acceleration vs Time

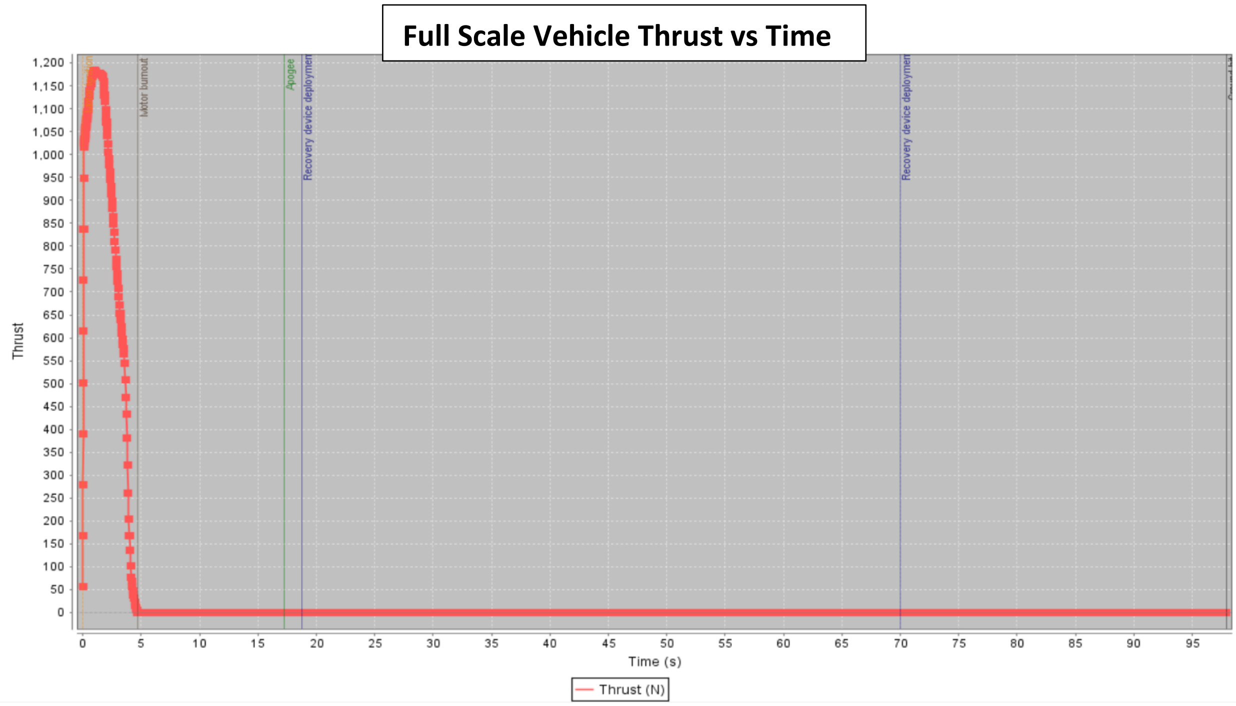


Figure 3-62. Thrust vs Time

3.3.2.2 Apogee Study

Mentioned in CDR, a hand calculated study of the vehicle's apogee was done to compare to the open rocket simulation. The theoretical apogee of the rocket is calculated by first calculating the altitude that the rocket achieves under powered ascent (the solid motor is producing thrust), followed by calculating the altitude that the rocket achieves after powered ascent while in the "coasting phase." The powered ascent altitude formula can be derived from Newtons 2nd law.

$$F = m * a \qquad (1)$$

Where F is Force, m is mass, and a is acceleration. Using definition of acceleration:

$$F = m\,\frac{dv}{dt} \qquad (2)$$

The entire force on the rocket can be written as:

$$F = T - W_{avg} - F_{drag} \qquad (3)$$

Where T is the trust force. F_{drag} is the drag force acting on the rocket. W_{avg} is the average weight force during powered ascent. For a more accurate calculation, the weight force can be integrated over the duration of the motor burn, however because the mass of the vehicle (17.368 kg) is large compared to the propellant mass (2.095 kg) and the burn time is relatively low (4.4 s), the method for determining the average mass in equation (4) is sufficient.

$$m_{avg} = m_{takeoff} - 0.5 * m_{propellant} \qquad (4)$$

and

$$W_{avg} = g * m_{avg} \qquad (5)$$

Where $m_{takeoff}$ is the mass of the vehicle at takeoff, $m_{propellant}$ is the mass of the propellant, and g is the gravitational constant of Earth. The gravitational acceleration, g, is assumed to be constant because the vehicle's target altitude is less than 5,000 feet. The drag force model used in this study is:

$$F_{drag} = k * v^2 \qquad (6)$$

Where v is the velocity of the vehicle and

$$k = 0.5 * \rho * C_d * A \qquad (7)$$

A is the greatest cross-sectional area of the vehicle and C_d is the coefficient of drag of the vehicle. For this calculation, a standard value was chosen for a vehicle of the same size. ρ is the density of the air. Again, because the rocket only travels to an altitude under 5,000 feet, we will assume this value to be constant. We can now plug equations (2), (5), and (6) into equation (3).

$$m \frac{dv}{dt} = T - g * m_{avg} - k * v^2 \qquad (8)$$

Equation 8 can be rearranged, and integrated to eventually obtain:

$$v = \sqrt{\frac{T - g * m_{avg}}{k}} * \frac{1 - e^{\frac{-2*t*k*\sqrt{\frac{T-g*m_{avg}}{k}}}{m_{avg}}}}{1 + e^{\frac{-2*t*k*\sqrt{\frac{T-g*m_{avg}}{k}}}{m_{avg}}}} \qquad (9)$$

Where T is the burn time and v is the velocity of the vehicle after the burn is complete. The height of the vehicle after the burn can now be written. To find the equation for height of the vehicle, equation (2) can be re-written as:

$$m * \frac{dv}{dt} = m * \frac{dv}{dh} * \frac{dh}{dt} = m * v * \frac{dv}{dh} \qquad (10)$$

By substituting equation (10) into equation (8), then rearranging and integrating, the following expression can be created:

$$h_{burnout} = \frac{m_{avg}}{2*k} * \ln\left(\frac{T - m_{avg}*g}{T - m_{avg}*g - k*v^2}\right) \quad (11)$$

Where $h_{burnout}$ is the height of the vehicle after the powered ascent. Now the height of the vehicle after its coasting portion of flight can be found. For these calculations, the mass of the vehicle without the propellant can be denoted as $m_{vehicle} = m_{takeoff} - m_{propellant}$. Using similar reasoning as above, with the exception of removing the impulse from the equation of motion, the following equation can be obtained:

$$h_{coasting} = \frac{m_{vehicle}}{2*k} * \ln\left(\frac{m_{avg}*g + k*v^2}{m_{avg}*g}\right) \quad (12)$$

Now that the equations for the altitude of both portions of the flight, they can be added together to find the total altitude of the vehicle that accounts for drag and changing mass.

$$h_{apogee} = h_{burnout} + h_{coasting} \quad (13)$$

Using the above equations, the theoretical apogee of the rocket was found to be 4,725 feet. The purpose of this study was to hand-calculate the maximum altitude of the Zenith 1 launch vehicle and compare this value to the altitude reached in open rocket under the same windspeed conditions. The graph below represents the altitude flight profile of the vehicle simulated in OpenRocket at 0 mph wind speed conditions.

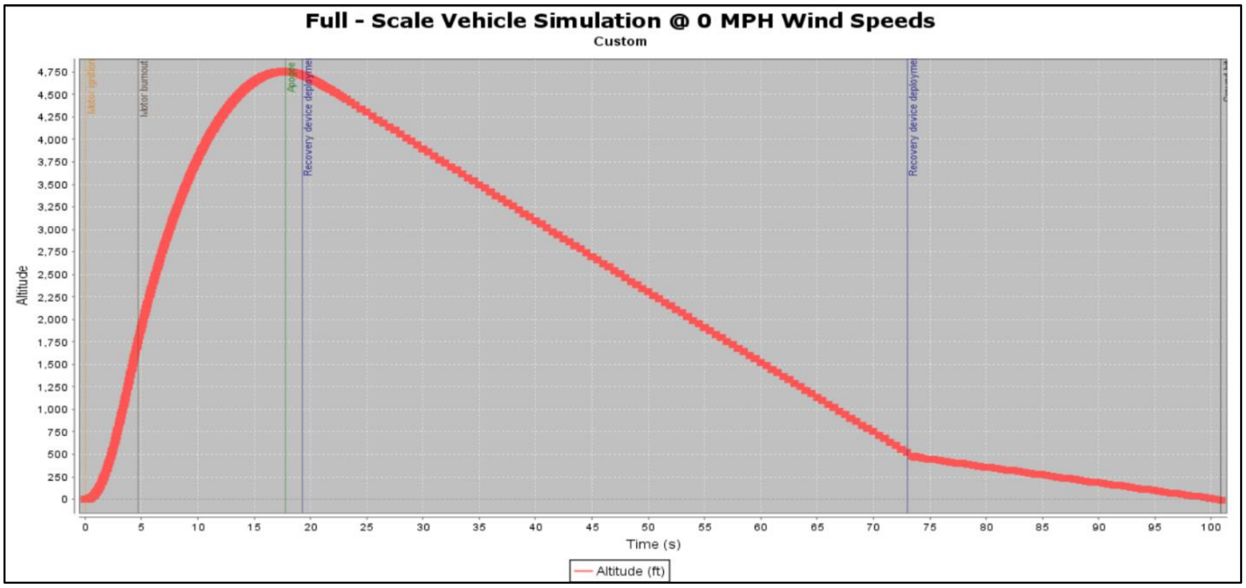


Figure 3-63. Altitude Profile at 0 MPH Wind

As seen in the image above, the launch vehicle reaches a maximum altitude of 4,750 feet, 25 feet higher than the altitude calculated by hand. The deviation between the two values could be explained by just the difference between programs. Although both programs calculate the altitude mathematically, OpenRocket has the upper-hand advantage as it was solely created to calculate and analyze a rocket's flight profile. In any case, both calculated values are below the declared altitude of the team. The vehicle was purposely designed to fly beyond the declared altitude under the specified conditions in the numerical solutions above. Considering the average daily windspeed in Huntsville, AL is 10-12 mph, the vehicle was designed under the strong assumption that there will be wind speeds on the day of the launch competition. This allows for the vehicle to over shoot under perfect conditions, in hopes of shooting spot on without perfect conditions. Please refer to section 3.3.2.1 where the altitude versus windspeed conditions from 5-10 mph are presented in a graph to track the change in apogee depending on the launch day wind speed conditions.

3.3.2.3 <u>Stability Margin</u>

Designing a vehicle around its stability margin is crucial as it plays a vital role in optimizing its flight performance and determining its overall success. While OpenRocket simulations may not precisely determine the stability margin of a real-time launch vehicle on the actual launch day, it provides an accurate measurement. The graph below displays the stability margin profile of the vehicle under the maximum permissible wind speed conditions during launch day.

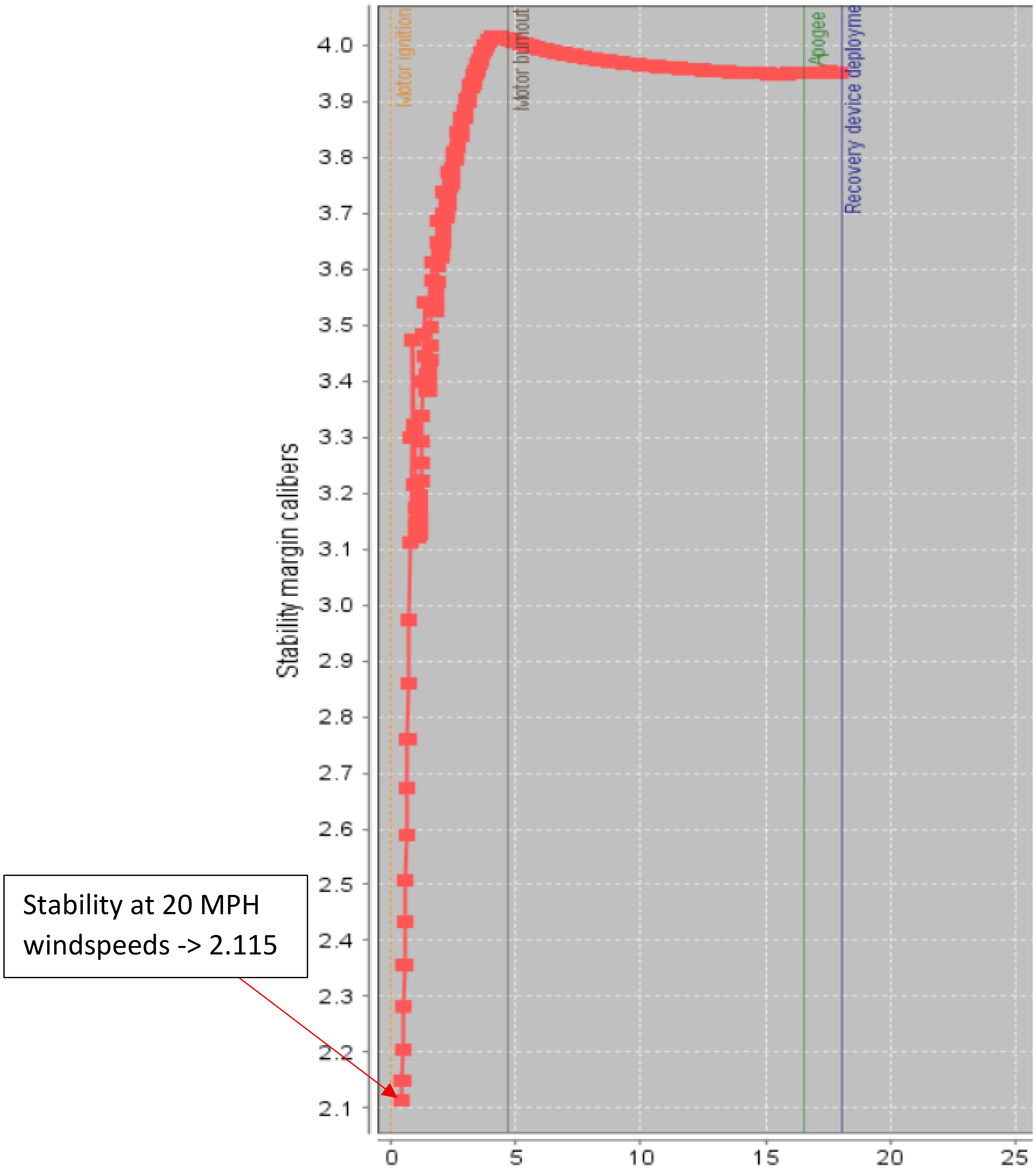


Figure 3-64. Stability Margin vs Time (at 20 MPH Wind)

The above figure indicates that the vehicle has a static stability margin of 2.115 calibers while at rest under the conditions of 20 mph wind speeds. The static stability margin displayed meets NASA's minimum requirement, ensuring that it also meets the requirement for wind speeds below 20 MPH. To double-check this, the table below illustrates the vehicle's stability margin at wind speeds of 0, 5, 10, 15, and 20 MPH, as per the geographical location and conditions of the launch competition.

Table 3-8. Stability Margin vs Wind Speed (at 5 degree cant)

5° Launch Angle	
Wind Speed (mph)	Static Stability Margin
0	3.408
5	2.950
10	2.595
15	2.323
20	2.115

Another method of calculating the static stability margin is the Barrowman's method. The Barrowman's method was written into a MATLAB script and used to calculate the vehicle's static stability margin at 0 MPH wind speeds. The following equation was used to calculate the static stability margin:

$$Stability\ Margin_{\ Static} = \frac{X_{CP} - X_{CG}}{D}$$

Where X_{CG} is the center of gravity and X_{CP} is the center of pressure, both measured from the tip of the nosecone. The center of gravity is given as 58.578 inches and the center of pressure can be found by

$$X_{CP} = \frac{C_N X_N + C_F X_F}{C_N + C_F}$$

The arm length of the fins, X_F can be found using the following equation:

$$X_F = X_B + \frac{X_R C_R + 2C_T}{3(C_R + C_T)} + \frac{1}{6}\left(C_R + C_R - \frac{C_R C_T}{C_R + C_T}\right)$$

Where X_B is the distance from the tip of the nosecone to the fin root chord leading edge, C_R is the length of the fin root chord, and C_T is the length of the fin tip chord.

The fin coefficient shown in equation, C_F, is represented as

$$C_F = \left(1 + \frac{R}{S+R}\right)\left(\frac{4N\left(\frac{S}{2R}\right)^2}{1 + \sqrt{1 + \left(\frac{2L_F}{C_R + C_T}\right)^2}}\right)$$

Where N is the number of fins and L_F is the length of the fin mid-chord line. L_F can be calculated using the fin semispan, S, and the following equation:

$$L_F = \sqrt{S^2 + \left(0.5C_T - 0.5C_R + \frac{S}{tan\theta}\right)}$$

The variables in the equations above and their calculated values are shown below in table.

Table 3-9. Stability Parameters

Parameter	Symbol	Value	Unit
Fin Semispan	S	9.00	Inches
Fin Tip Chord	C_T	3.00	Inches
Fin Root Chord	C_R	11.00	Inches
Fin Sweep Angle	θ	37	Degrees
Fin Mid-Chord Line	L_F	8.14	Inches
Radius of Airframe	R	3.00	Inches
Number of Fins	N	4	N/A
Nose Tip to Fin Root Chord Leading Edge	X_B	83.50	Inches
Fin Root Leading Edge to Fin Tip Leading Edge	X_R	6.77	Inches
	X_F	88.18	Inches
Arm Length of Nosecone	X_N	9.32	Inches
Nose Cone Length	L_N	20.00	Inches
Nose Cone Term	C_N	2.00	N/A
Fin term	C_F	16.95	N/A
Center of Pressure	X_{CP}	81.421	Inches
Center of Gravity	X_{CG}	60.657	Inches
Diameter of Airframe	D	6.154	Inches
Static Stability Margin	$Stability\ Margin_{\ Static}$	3.46	Calibers

Comparing the stability margin values calculated from OpenRocket and the MATLAB Code

Table 3-10. Stability Results

Method	Stability Margin (Calibers)
OpenRocket Simulation Software	3.37
Barrowman's Method	3.36

Any deviation between the two values is more than likely due to rounding the pre-defined MATLAB values in the coded script.

3.3.3 Descent Time, Wind Drift, and Max Landing Kinetic Energy Calculations

The descent time is determined by the descent velocity under each parachute and the altitudes at which they are deployed. The drift can be calculated, but a few assumptions must be made. The first assumption is that the launch vehicle reaches apogee directly above the launch pad. The second is that at apogee and main deployment altitude, the terminal velocities are reached instantaneously. Lastly, the wind speeds are applied uniformly on the vehicle, and it moves in one direction. Although these assumptions make the calculations not entirely accurate, it still gives a good understanding of how severe wind conditions can affect the vehicle's descent. To determine the total descent time of the vehicle the equation below was used:

$$t = \frac{h_a - h_m}{v_d} + \frac{h_m}{v_m}$$

where h_a is the apogee altitude, h_m is the main parachute deployment altitude, v_d is the descent velocity underneath the drogue parachute, and v_m is the descent velocity underneath the main parachute. For the declared altitude of 4600 ft, the calculated descent time is 80.7 seconds.

Table 3-11: Parachutes and Descent Rates

Parachute	**Drag Coefficient**	**Projected Area (ft^2)**	**Descent Speed (ft/s)**	**Descent Time (s)**	**Max Wind Drift at 20 MPH (ft)**
Fruity Chute 24" Classic Elliptical	1.5	3.02	76.90	52.66	1544.83
Fruity Chute 84" Iris Ultra Standard	2.2	37.29	18.06	30.46	893.42

Table 3-12: Descent Time and Drift

Wind Speed (mph)	Apogee (ft)	Descent Time (s)	Wind Drift (ft)
0	4600	83.12	0.0
5	4600	83.12	609.5
10	4600	83.12	1219.1
15	4600	83.12	1828.6
20	4600	83.12	2438.2

The kinetic energy of each section of the vehicle during descent is governed by the mass and velocity of each. The equation used to determine the kinetic energy of each section is stated below:

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

where m is the mass, and v is the velocity of the section. The maximum kinetic energy set by the requirement in the Student Launch Handbook is 75 ft-lb, which can be used to derive the maximum velocity for each descending section. The maximum velocity of each descending section is shown in the table below.

Table 3-13: Maximum Kinetic Energy

Section	mass (g)	mass (lbm)	mass (slug)	Descent Velocity (ft/s)	Kinetic Energy (ft-lb)
Nosecone + UPB	4025	8.8736	0.2758	18.591	47.662
Nosecone + UPB + payload	5525	12.1805	0.3786	18.591	65.424
Payload	1500	3.3069	0.1028	18.591	17.762
AV bay	2187	4.8215	0.1499	18.591	25.897
LPB + Fin can	2763	6.0914	0.1893	18.591	32.718

An important note about the table above is that the second row represents a failure mode, in which the payload does not exit the upper bay upon main deployment. In the unlikely event that this occurs, the landing kinetic energy of that section is still under the 75 ft-lb maximum requirement.

4 Payload Design

4.1 Changes since CDR

4.1.1 Interior electronics frame

Had an oversite in the design of being able to put the Arduino Mega controller into the interior electronics frame after the printing process was complete. To resolve this issue some of the material on one side will be removed allowing the Arduino Mega controller to slide into the print. Additionally extra material will be added back to keep its original strength. Below is the old and new design of the interior electronics frame.

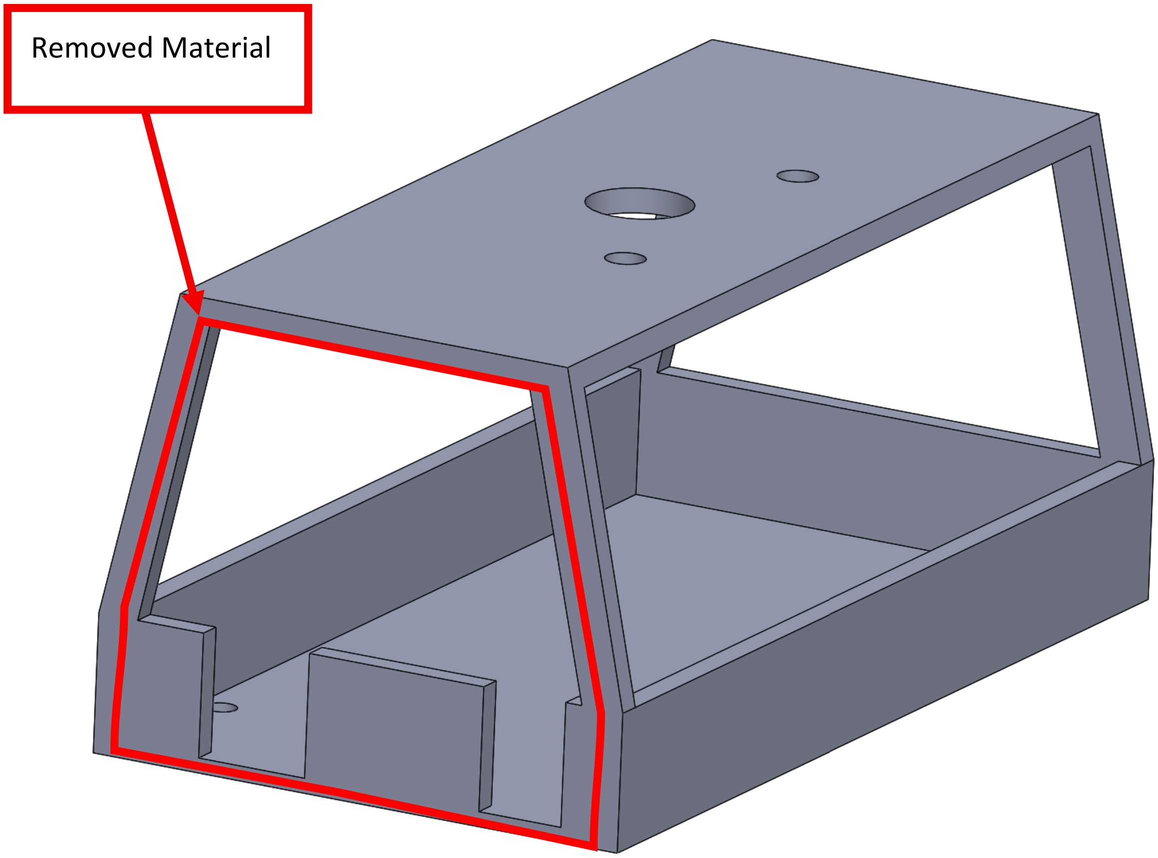


Figure 4-1. Previous Interior Electronics Frame

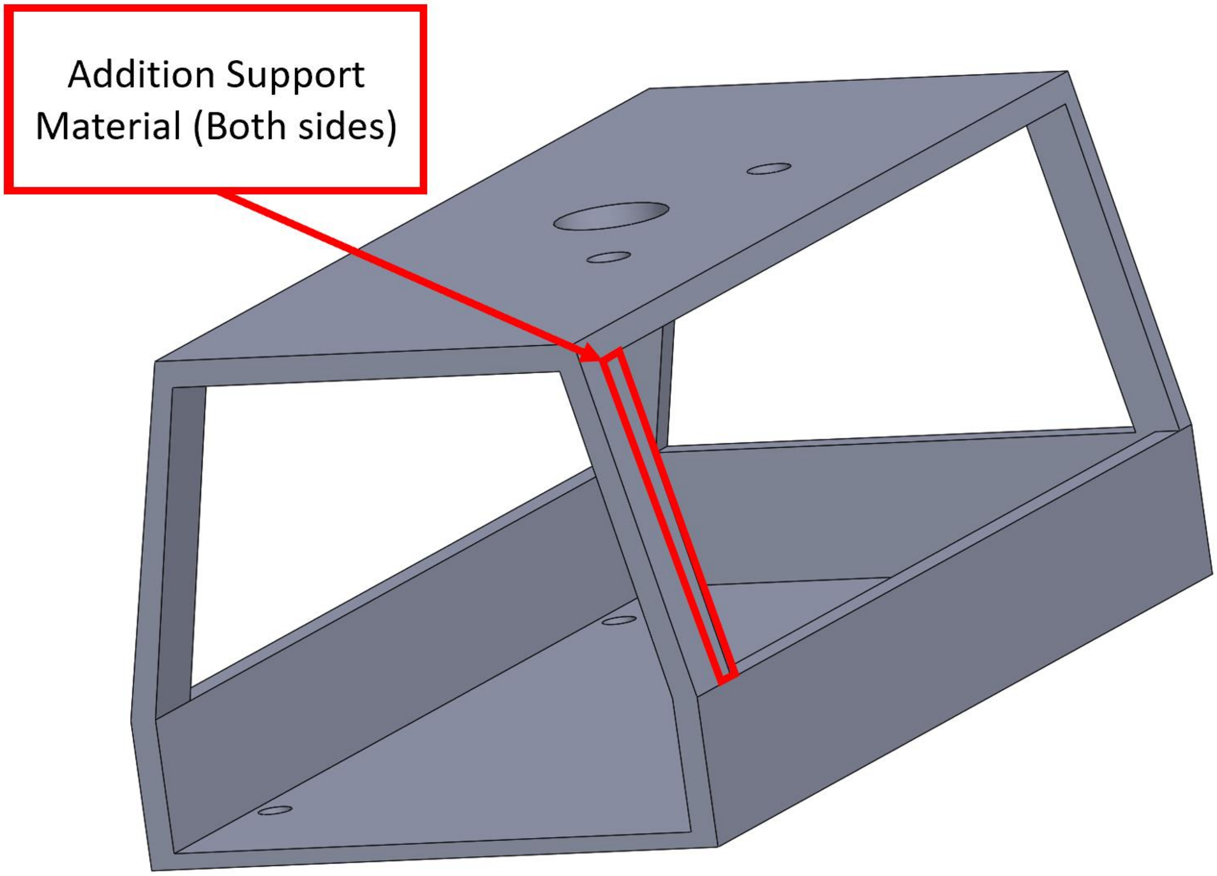


Figure 4-2. New Interior Electronics Frame

4.1.2 Camera Housing

The camera housing design was created before receiving the camera that is being utilized for the payload. So, the exact dimensions of the camera online were not the same as the camera that was received. The camera housing design will change in order to better suit the shape of the camera.

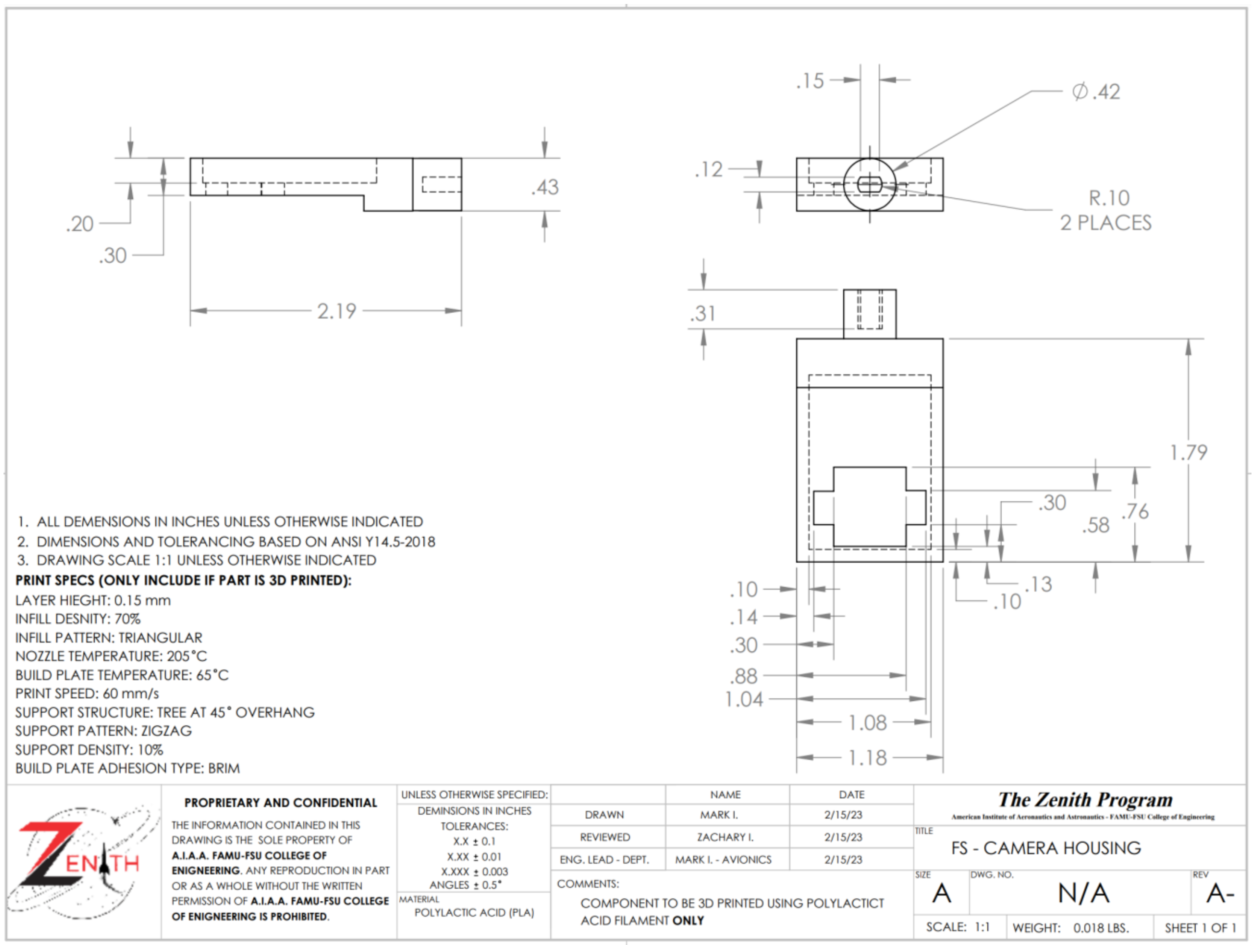


Figure 4-3. New Camera Housing Design Engineering Drawing

4.1.3 Payload Housing

After printing the original design, the interior electronics frame was unable to slide into it due to the top opening not being wide enough. The design has been widened to allow for the previous oversite. Also add additional material around the eyebolt locations for additional strength.

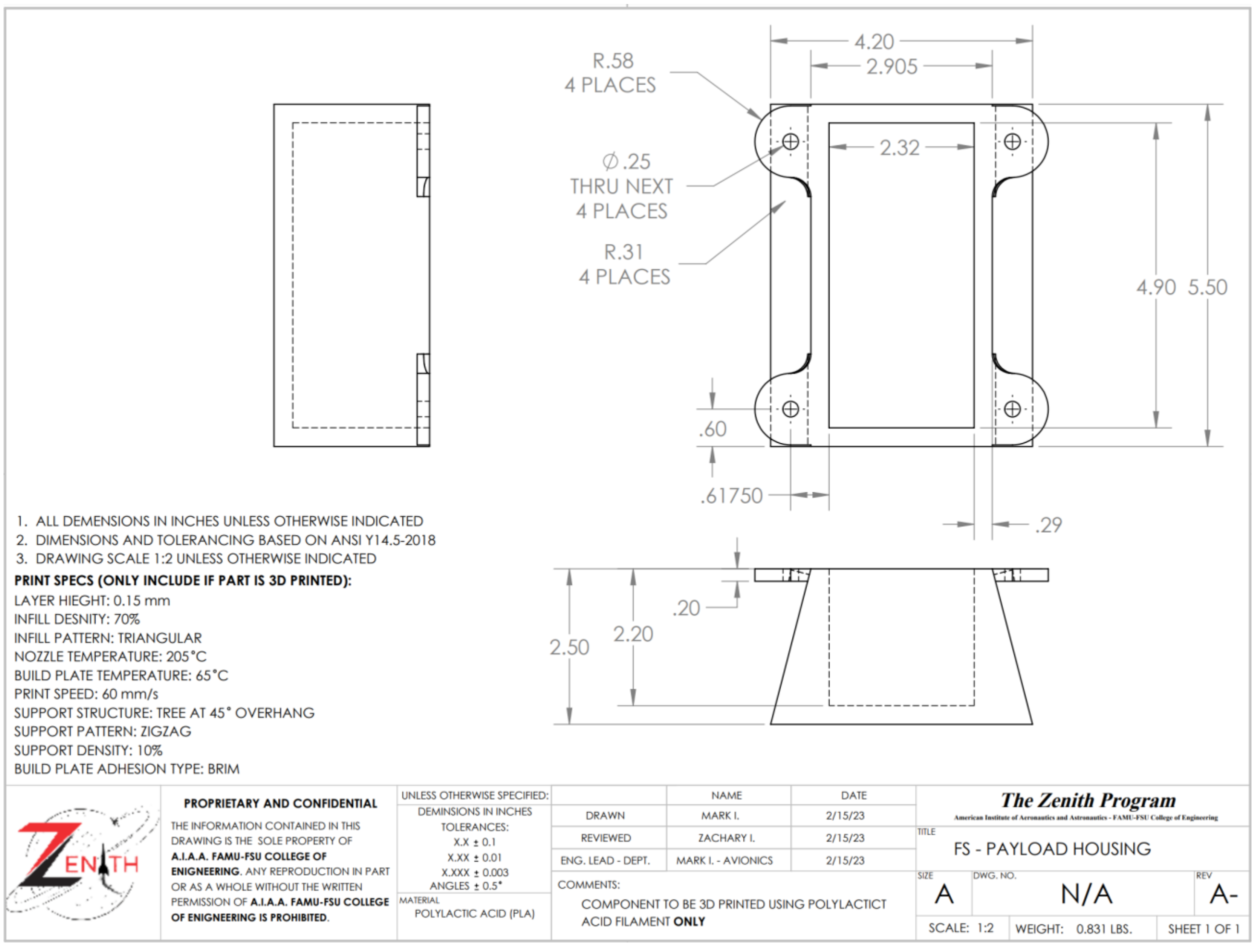


Figure 4-4. New Payload Housing Design Engineering Drawing

4.1.4 Lexan Cover

The changes made to the payload and camera housing make the need for the Lexan cover to change. Additionally, molding the Lexan was causing difficulty during manufacturing. Due to this difficulty in manufacturing the decision was made to make the cover more like a frame and have 4 different Lexan cutouts epoxied into the frame. These changes are shown in the design below.

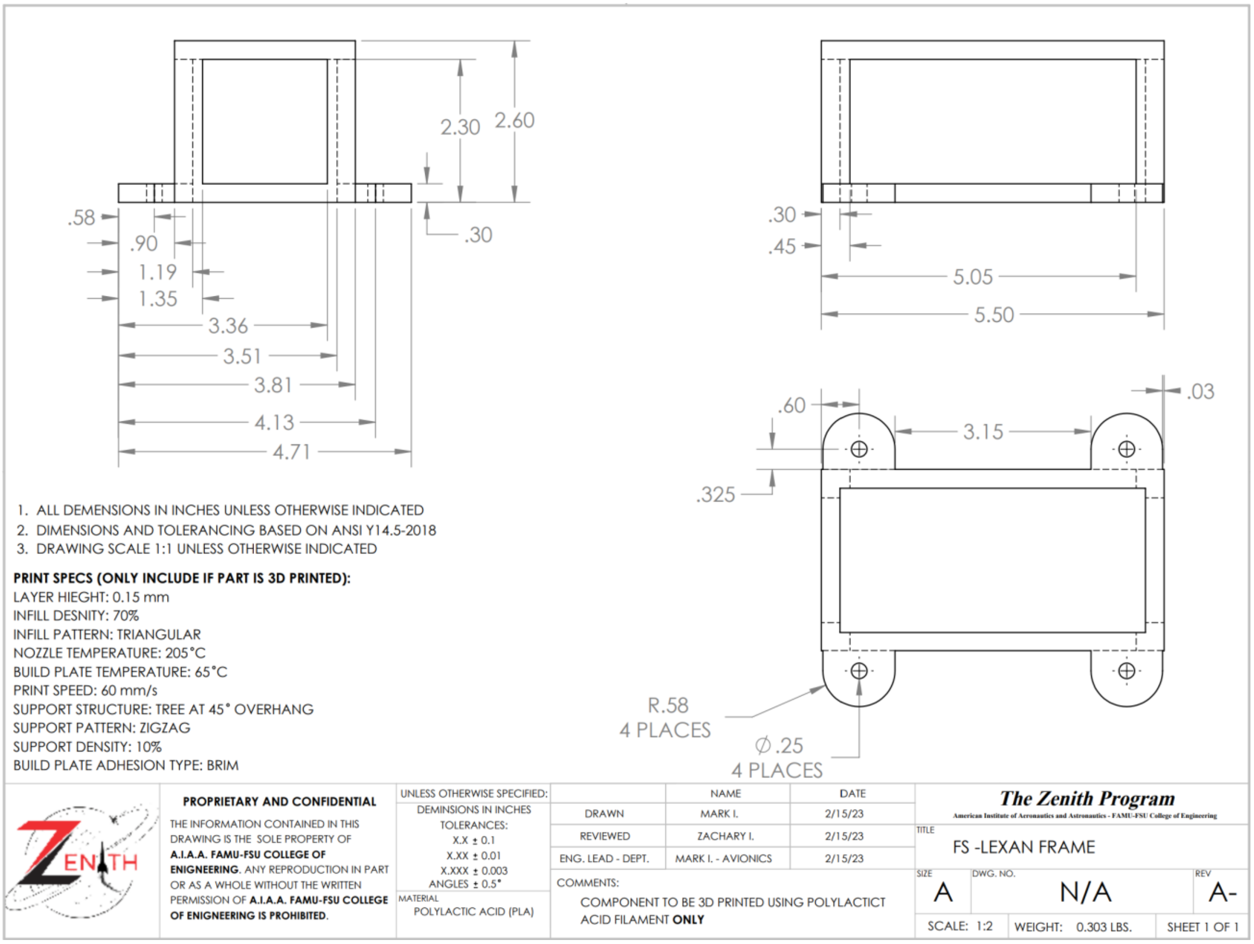


Figure 4-5. Lexan Cover Engineering Drawing

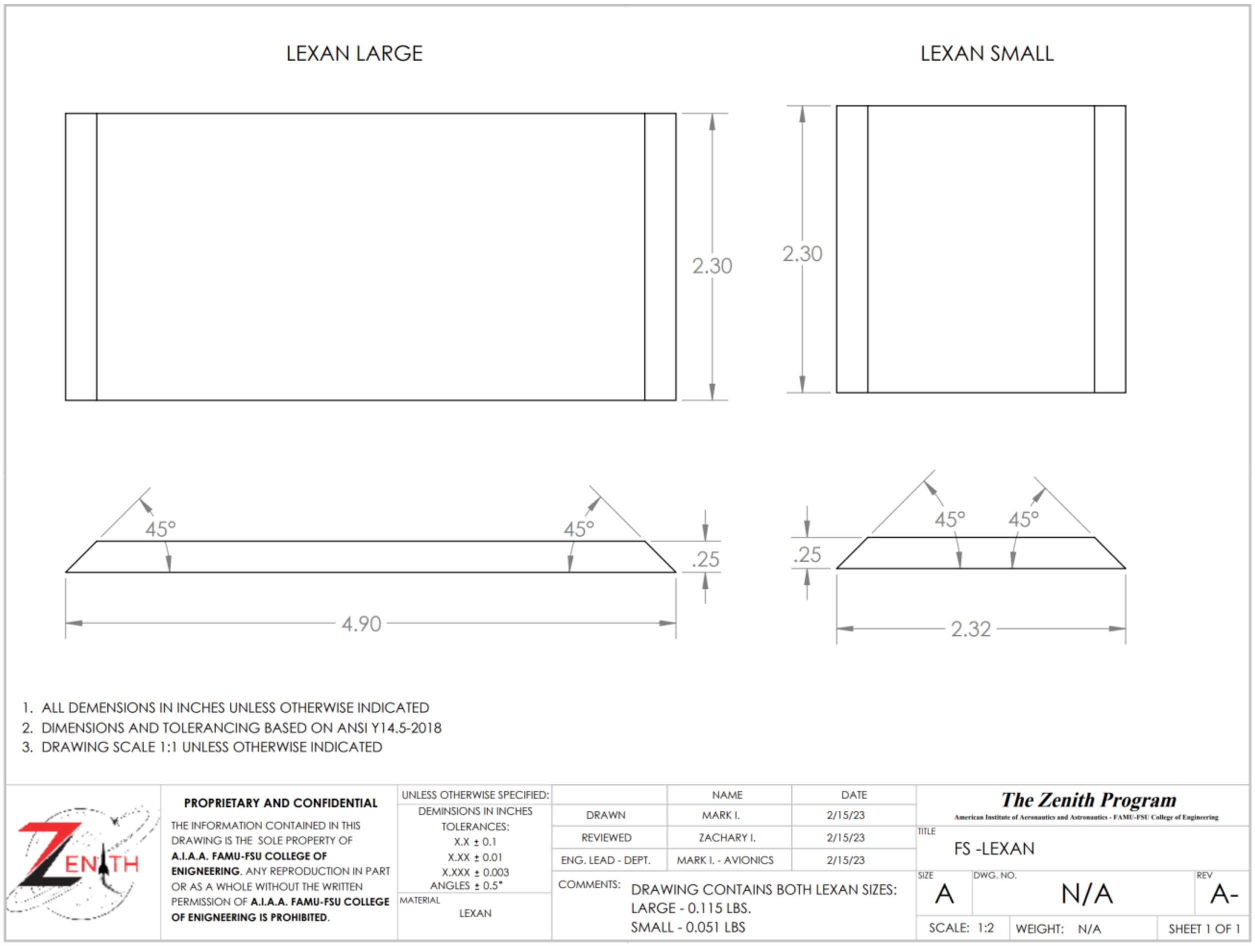


Figure 4-6. Lexan Engineering Drawing

4.2 Design

4.2.1 Structure

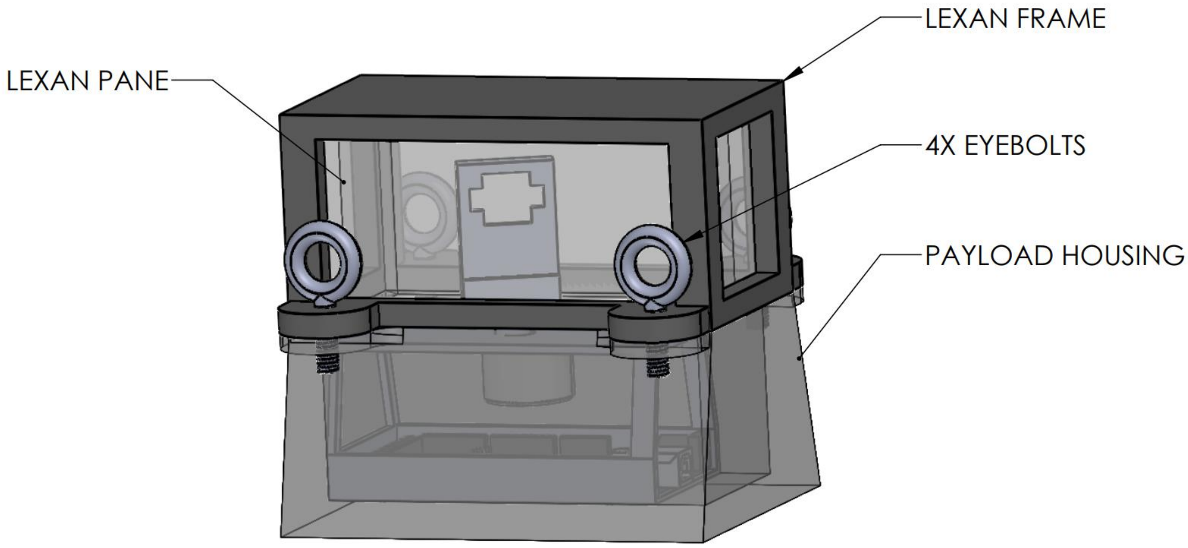


Figure 4-7. Assembled Payload

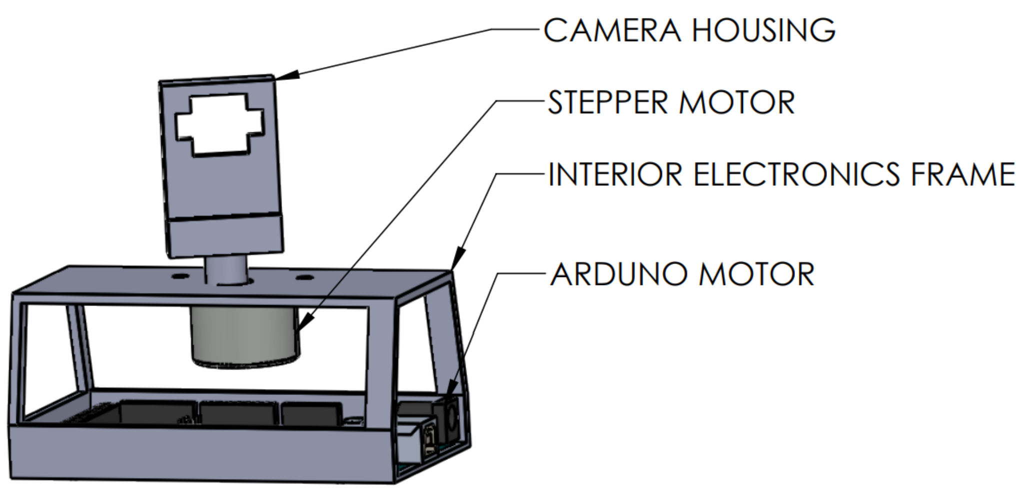


Figure 4-8. Assembled Interior Electronics Frame.

4.2.2 Electrical Diagram

4.2.2.1 Arducam Connection

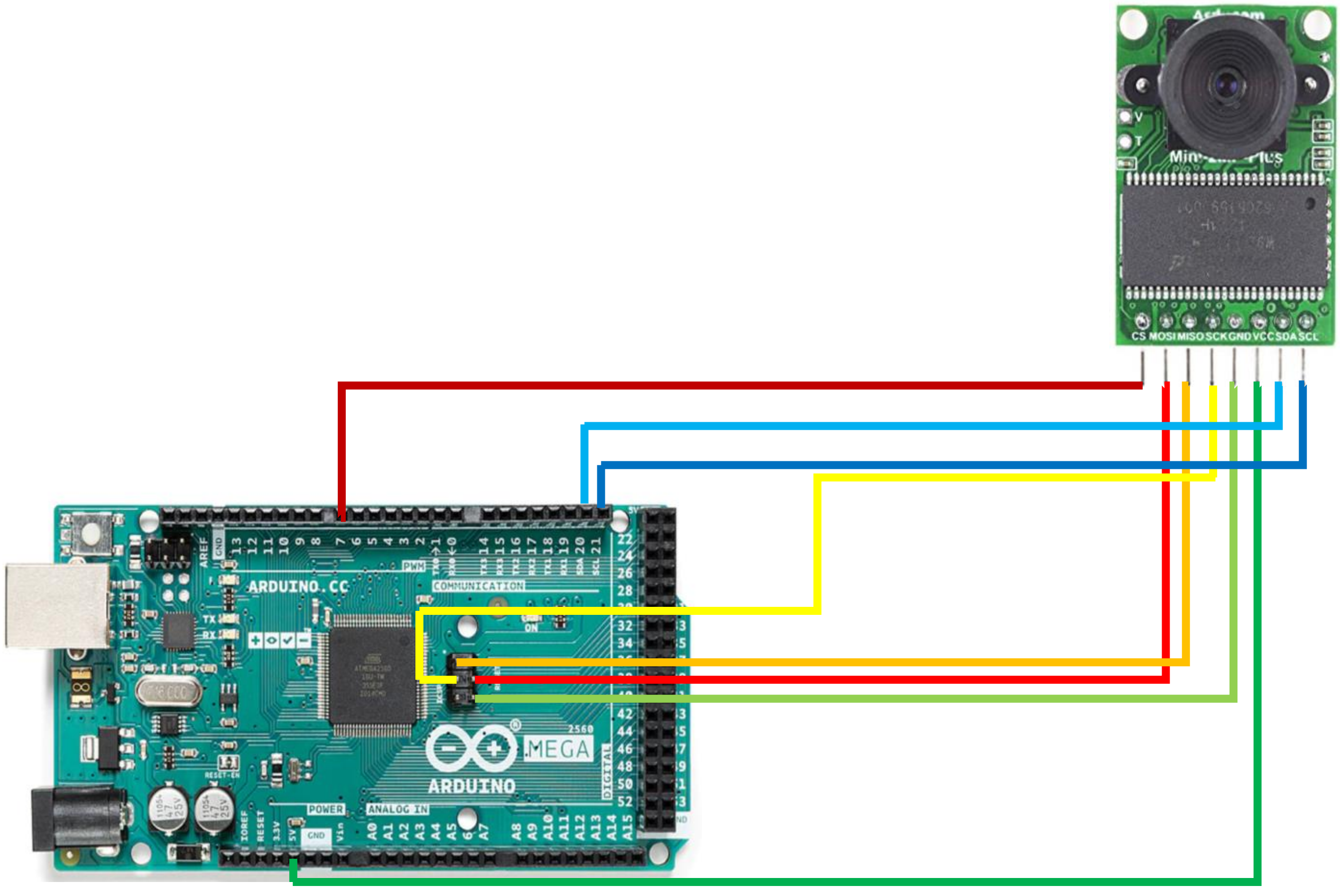


Figure 4-9. Electrical Diagram Part 1

Table 4-1: Arduino mega to Arducam Electrical Connections

Primary Lead Connection	➔ Connected To
GND	GND pin
ISP	CS
ISP	MOSI
ISP	MISO
ISP	SCK
5V	VCC
SDA (pin 20)	SDA pin
SCL (pin 21)	SCL pin

4.2.2.2 Stepper Motor Connection

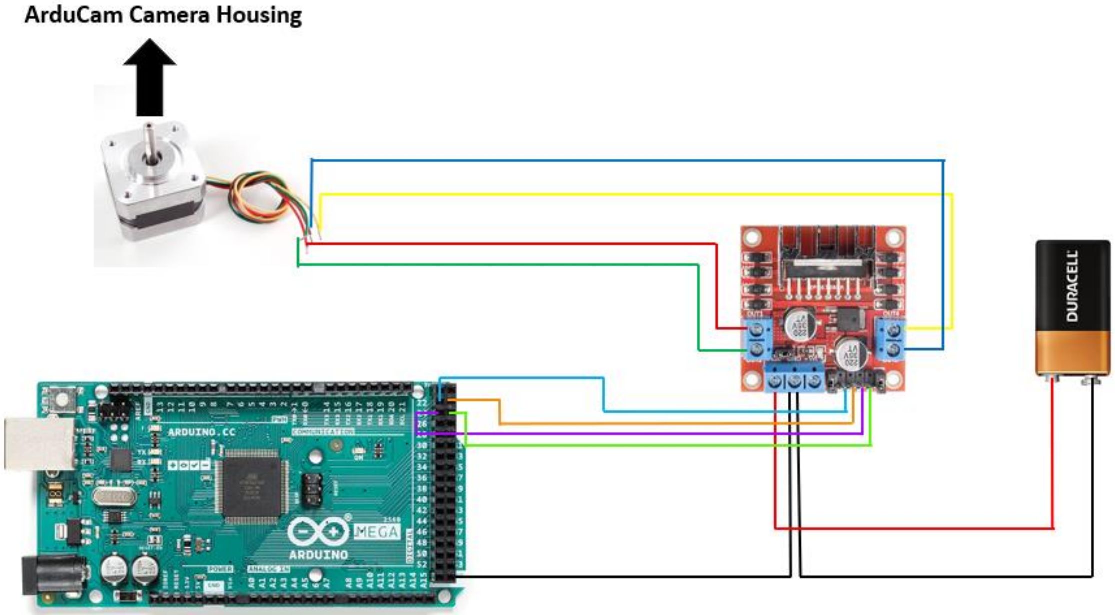


Figure 4-10. Electrical Diagram Part 2

Table 4-2. Payload to Stepper Motor Wiring

Primary Lead Connection	➔ Connected To	➔ Successive Connection
Battery (+)	Motor Driver Power (+) (Bottom Left)	
Battery **(-)**	Motor Driver Power (-) (Bottom Center)	Arduino GND Pin
Arduino Pin 22	Motor Driver Pin 1 **(Leftmost)**	
Arduino Pin 23	Motor Driver Pin 1 **(Left)**	
Arduino Pin 24	Motor Driver Pin 1 **(Right)**	
Arduino Pin 25	Motor Driver Pin 1 **(Rightmost)**	
Motor Pole 1 (+)	Driver Pole 1 (+) Output (Top left)	
Motor Pole 1 (-)	Driver Pole 1 (-) Output (Bottom left)	
Motor Pole 2 (+)	Driver Pole 2 (+) Output (Top Right)	
Motor Pole 2 (-)	Driver Pole 2 (-) Output (Bottom Right)	

4.3 Payload Manufacturing

4.3.1 3D Printing Print Orientation

4.3.1.1 Payload Print Specifications

Table 4-3. Payload 3D Printing Specifications

Ultimaker Cura Printing Specifications	
Material	PLA
Layer Height	0.15 mm
Infill Density	80%
Infill Pattern	Triangular
Nozzle Temperature	205°C
Build Plate Temperature	65°C
Print Speed	60 mm/s
Support Structure	Tree Structure at 45° Overhang
Support Pattern	Zigzag
Support Density	10%
Build Plate Adhesion Type	10 mm thick Brim

Copy what Peyton writes about printing specifications

4.3.1.2 Payload Housing

The payload housing protects the electronics frame inside from impact during landing. The housing is printed of PLA vertically up the z-axis from the wide base, as shown below.

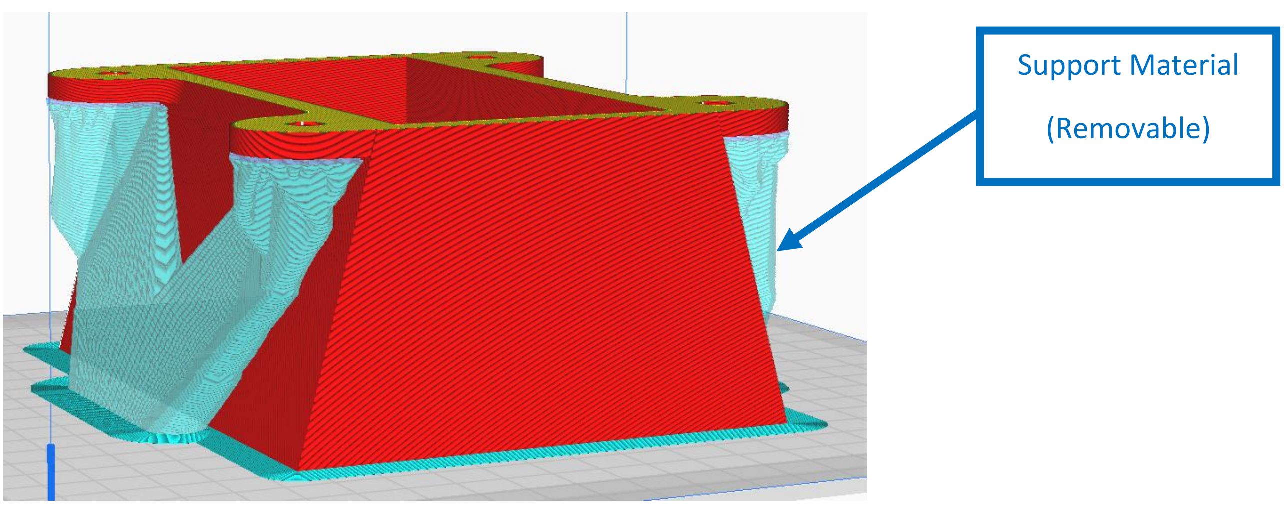


Figure 4-11. Payload Housing Print Orientation

4.3.1.3 Interior Electronics Frame

The electronics frame houses the microcontroller and acts as the attachment point for all payload hardware. The frame is printed of PLA along the longitudinal (x) axis. Print orientation was automatically determined by Cura to limit support material required.

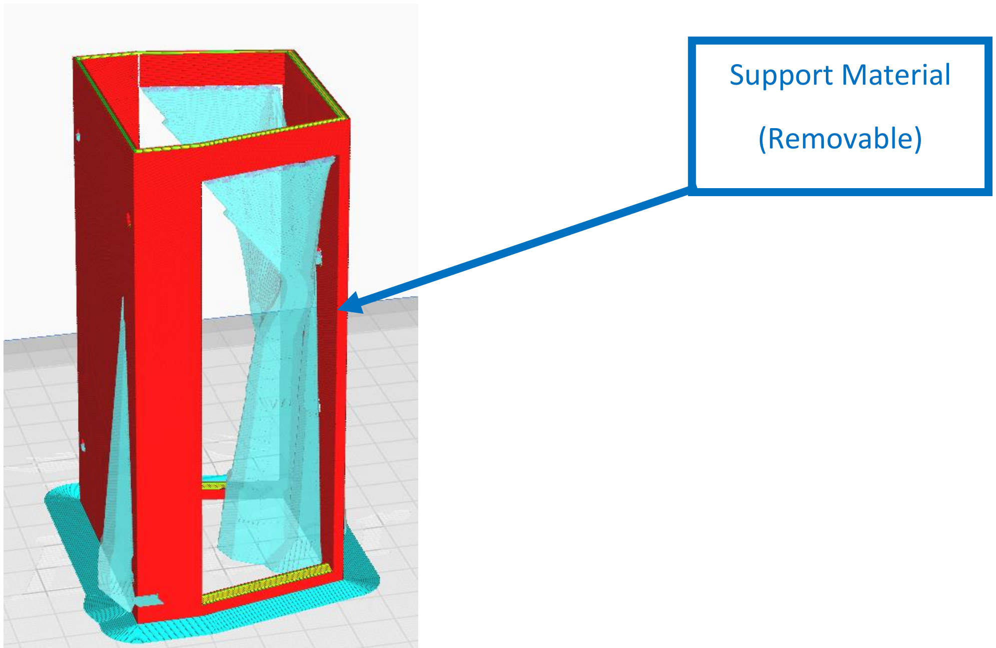


Figure 4-12. Interior Electronics Frame Print Orientation

4.3.1.4 Camera Housing

The camera housing both protects the board of the ArduCam as it retains and acts as the interface joining the camera to the stepper motor to allow rotation. Print orientation was set manually along the depth (y-axis) of the part based on the internal geometry to minimize support material.

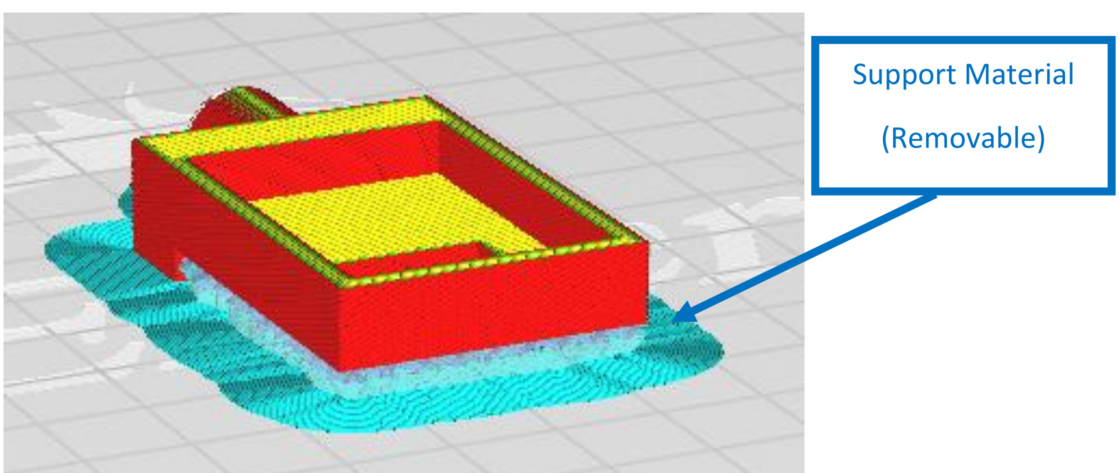


Figure 4-13. Camera Housing Print Orientation

4.3.1.5 Lexan Frame

The Lexan frame holds together the four panes. The frame is printed in this orientation to use less material as support material.

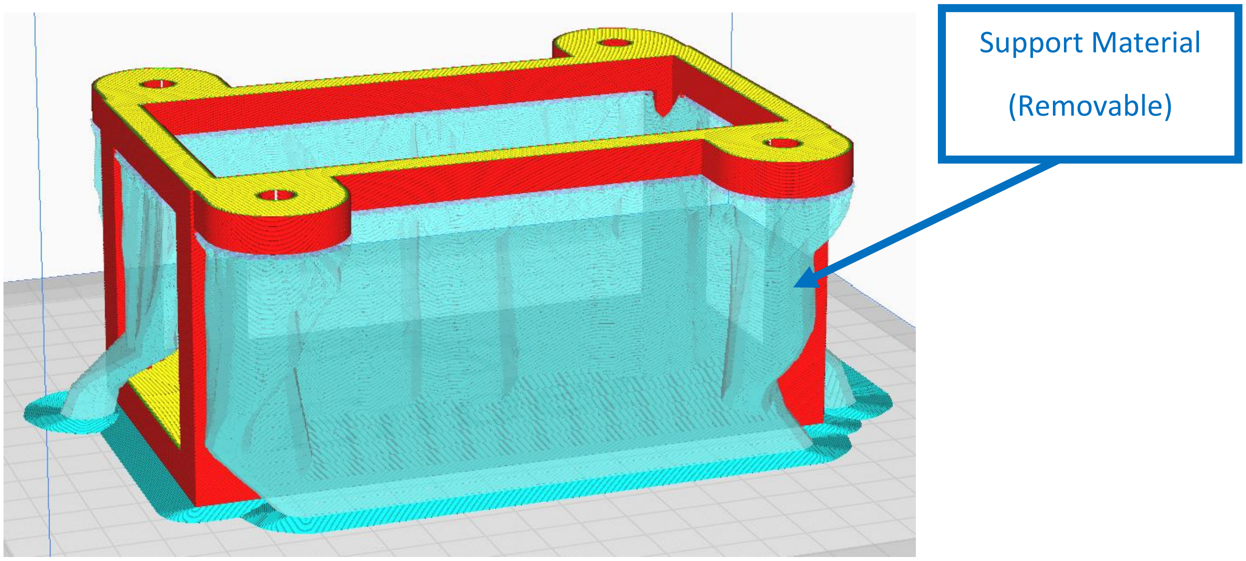


Figure 4-14. Lexan Frame Print Orientation

4.3.2 Payload Cover

Pictured above in the section "4.4.1.5 Lexan Frame" the design has a 3D printed cover and 4 panes of Lexan that have mitered edges. The Lexan panes get epoxied to the 3D printed Frame.

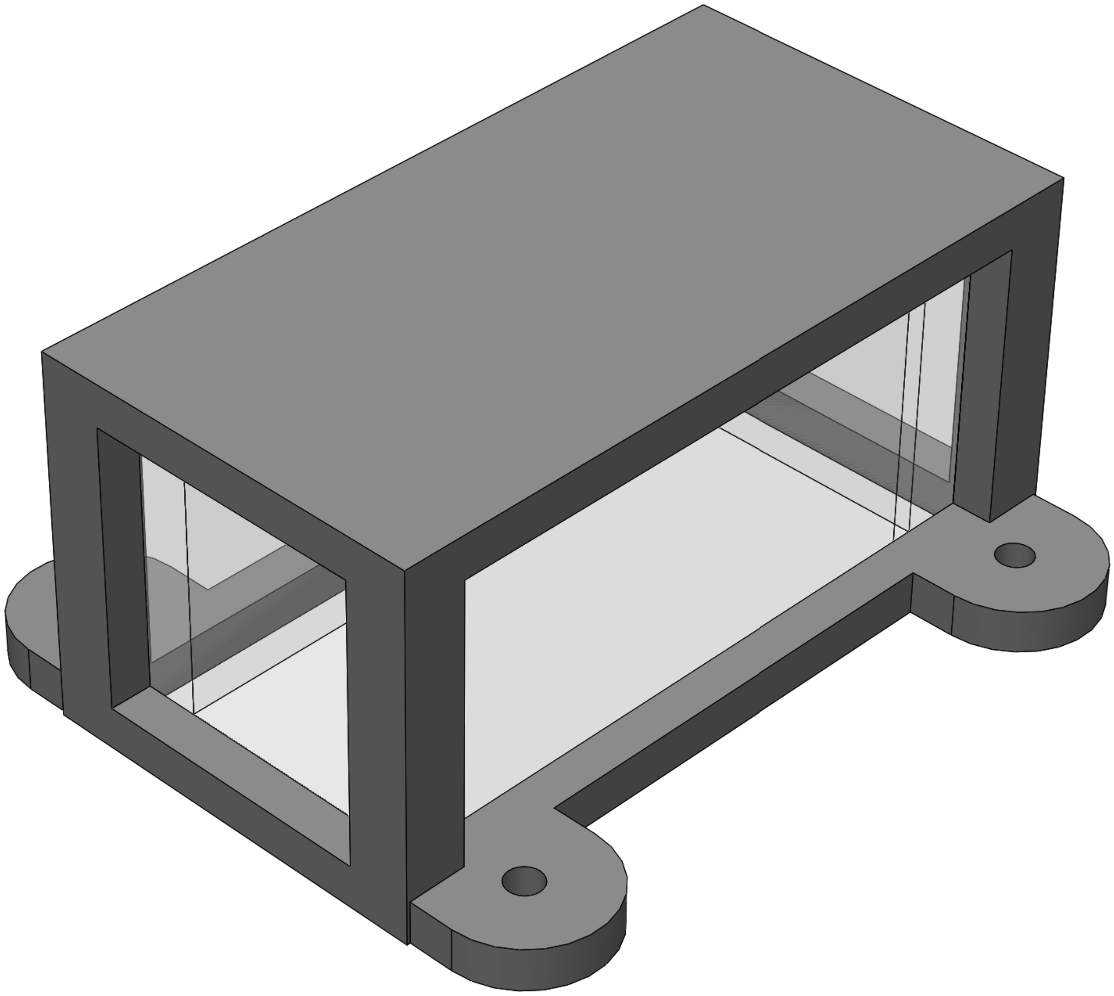


Figure 4-15. Assembled Payload Cover

4.3.3 Interior Electronics Frame

This holds all the electrical components of the design. Those being the Arduino Mega, Arducam, stepper motor, and motor driver. Before assembling the frame, the Arduino Mega was wired and tested. All the individual components slide into place.

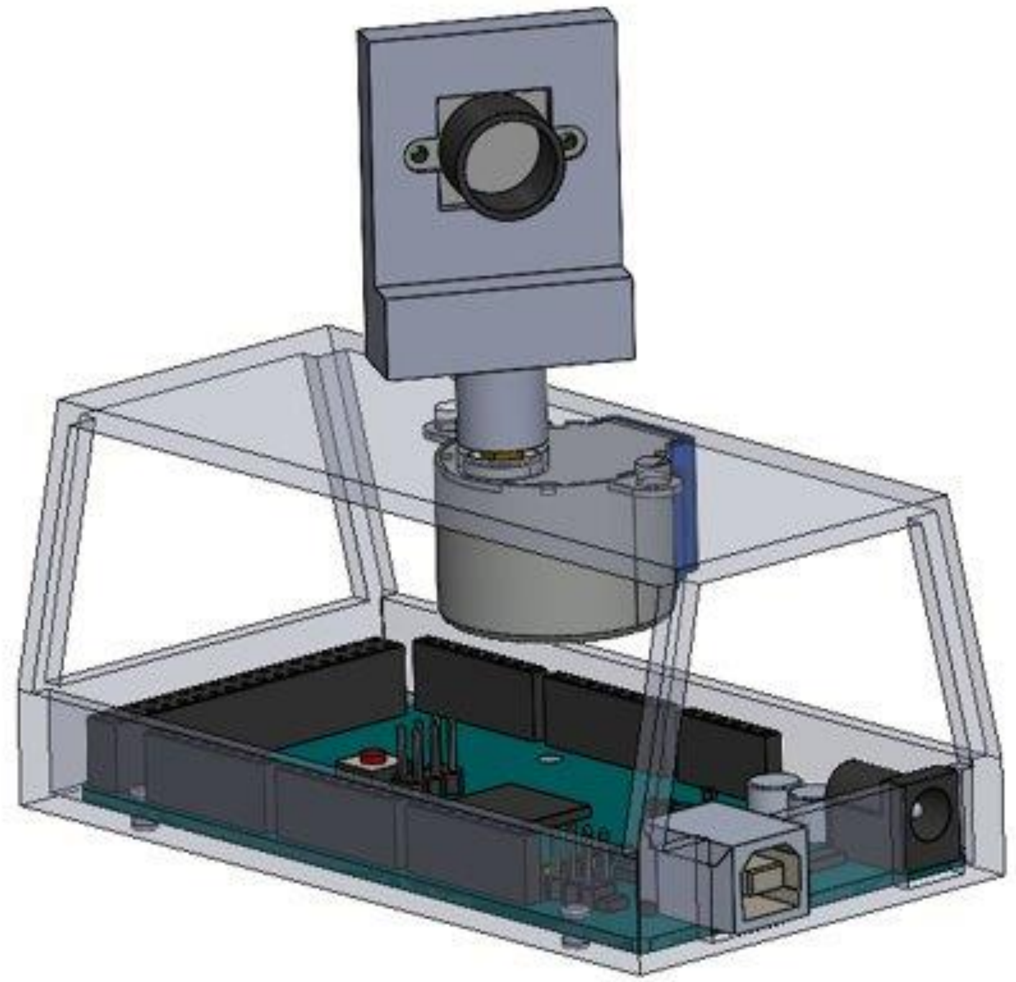


Figure 4-16. Assembled Interior Electronics Frame

4.3.4 Final Assembled Payload

After the Interior Electronics Frame and the Payload cover have been assembled. The electronics frame slides into the payload housing and the payload cover is attached with four 1/4x20 eyebolts.

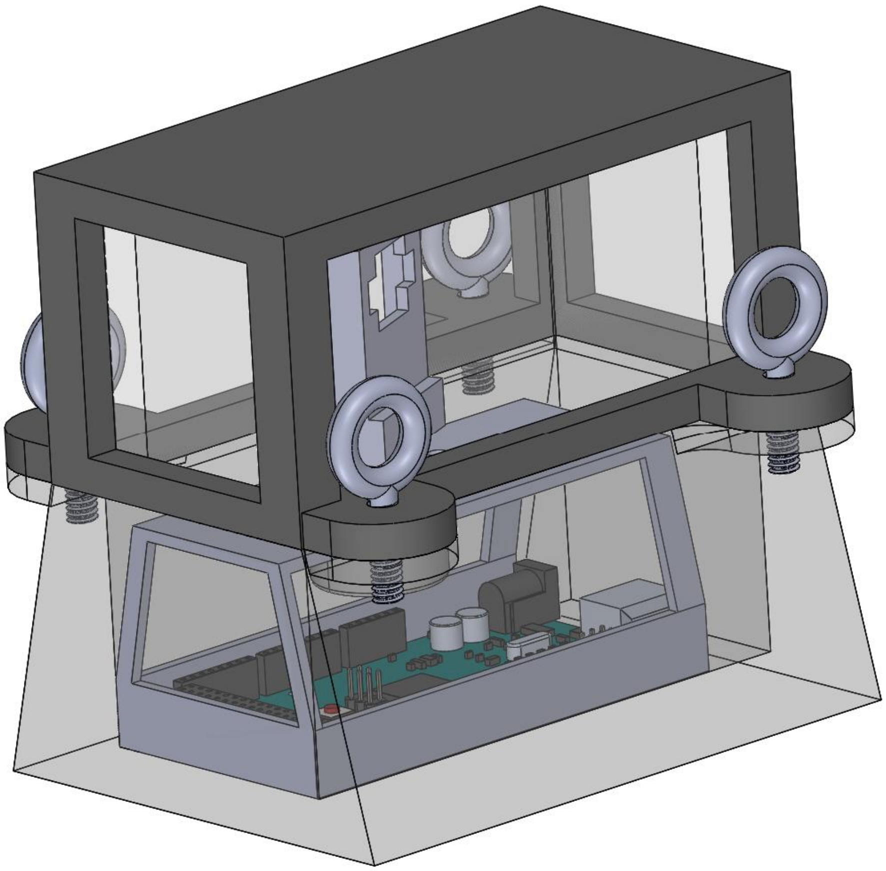


Figure 4-17. Fully Assembled Payload

4.3.5 Payload Vehicle Integration

The payload is connected to the launch vehicle by the 4 eyebolts. Attached to each of the eyebolts are four equal lengths of shock cord. Each of the cords are attached to a quick link on the main section of upper payload bay shock cord. During recovery when the main parachute is deployed it will pull the payload out of the Upper payload bay. By having the four connection points on the payload, it is more likely to land right side up.

Figure 4-18. Payload to Vehicle Integration

4.4 Payload Demonstration

4.4.1 Flight Date

The payload demonstration flight will occur on March 25th, 2023 at the Spaceport Rocketry Association launch site in Palm Bay, Florida. April 1st, 2023 will serve as the backup launch date if necessary.

4.4.2 Flight Success Criteria

1) The payload Deploys with the main parachute.
2) The payload body section lands intact.
3) The payload electrical connections stay connected during flight and landing.
4) The payload receives the entire RAFCO sequence.
5) The payload executes the entire RAFCO sequence.
6) The payload stores all the images successfully with timestamps and proper filters.

5 Full-Scale Demonstration Flight

5.1 Flight Type

The flight discussed in the following section was executed to fulfill the vehicle demonstration flight requirement and utilized a dummy payload to approximate the future payload demonstration flight and competition flight configuration.

5.2 General Flight Information

5.2.1 Date

The flight was conducted on March 4th, 2023, during a 12pm-2pm EST launch window.

5.2.2 Location

The range is in Palm Bay, Florida at the Spaceport Rocketry Association launch area.

Shensi Court Southwest
Palm Bay, FL 32908

5.3 Vehicle Specifications

5.3.1 Motor

Aerotech L850W

5.3.2 Ballast

5lb dummy payload

5.1.1 Payload

The vehicle did not fly the final competition payload in the event of an in-flight failure. The dummy payload was comprised of a 3lb lead shot scuba weight wrapped tightly in two queen size bedsheets to pad the weight and create a firm fit inside the payload bay. The sheets were wrapped tightly in electrical tape to produce a solid, cushioned mass. A cinch knot was used to secure a #1500 shock cord to the mass, which was wrapped several times around and firmly secured. A quick link was used to connect the shock cord to the recovery harness under the main parachute.

The dummy payload is photographed with the upper vehicle section in the post-flight component analysis to follow.

5.3.3 Range Conditions

5.3.3.1 Weather Monitoring Services

The following data is collected for the launch window from the National Weather Service and National Oceanographic and Atmospheric Administrations, taken from Melbourne Orlando International Airport (KLMB), which is 12 linear miles from the range. Windspeeds are measured at ground level. NWS and NOAA do not record or report historical data for high altitude winds.

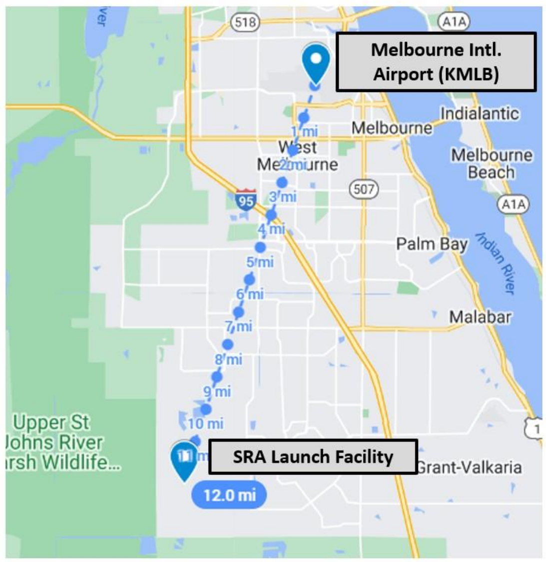


Figure 5-1. Launch Field Distance from Airport

Table 5-1. Wind Speeds at Different Times on Launch Day

Time (EST)	Winds (mph)	Weather Condition
13:53	SW 16	Partly Cloudy
12:53	W 13	Mostly Cloudy
11:53	SW 10	Mostly Cloudy

The NWS/NOAA format for wind reporting is:

(DIR) (##) (*G) (*##)

- DIR = direction of wind by prime compass divisions
- (##) = sustained windspeed
- (*G) = optionally included to indicate gusting winds
- (*##) = maximum instantaneous gusting windspeed

5.1.1.1 SRA Range Safety Officer Monitoring

The RSO the SRA range monitors live ground and high-altitude wind data at the range to clear vehicles for flight. The Zenith I was cleared for flight at approximately 12:30pm EST with liftoff shortly after. The RSO presented the team with the following data:

Table 5-2. Wind Speeds at Different Altitudes on Launch Day

Level	Winds (mph)	Weather Condition
Ground	NW 14	Scattered low level clouds
3,000 ft AGL	NW 18	Clear
6,000 ft AGL	NW 16	Clear

5.4 Flight Overview

5.4.1 Flight Event Outcomes

Descriptions of each event result category include several outcomes. Meeting a single of the criteria in a description box warrants the accompanying failure code. Categorization for each flight event is as follows:

Table 5-3. Event Outcomes

Result	Description
Nominal	As expected
Failure (Class 1)	• Minor vehicle damages. • Deviation from expected performance. • Insignificant impact on flight profile and vehicle performance.
Failure (Class 2)	• Significant vehicle damages • Major deviation from expected performance • Significant impact on flight profile and vehicle performance
Failure (Class 3)	• Catastrophic failure. Total loss of vehicle • Loss of a majority of mission critical components. • Failure resulting in injury

The full-scale demonstration flight saw a successful launch and recovery of the vehicle within 550 feet of the launch pad, although the vehicle underperformed for reasons discussed in the following sections. The order of events for the flight are as follows:

Table 5-4. Full-scale Flight Timeline

Event	Time (s)	Result	Notes
Ignition/liftoff	0	Failure (Class 2)	Performance deviation. Vehicle damage. Loss of tail cone with failure mode resulting in loss of motor efficiency
Burnout	4	Nominal	-
FC read apogee	15	Nominal	-
Fire ejection charge 1	16	Nominal	-
Separation 1	16	Failure (Class 1)	Vehicle damage. Fin lost during collision of airframe segments
Drogue deployment	16	Nominal	-
FC read 550ft AGL	150	Nominal	-
Fire ejection charge 2	57.3	Nominal	-
Separation 2	17	Nominal	-
Ground Impact	78.9	Failure (Class 1)	Vehicle damage. Nosecone split at upper joint.

5.5 Landed Configuration

5.5.1 Recovery Zone Accuracy

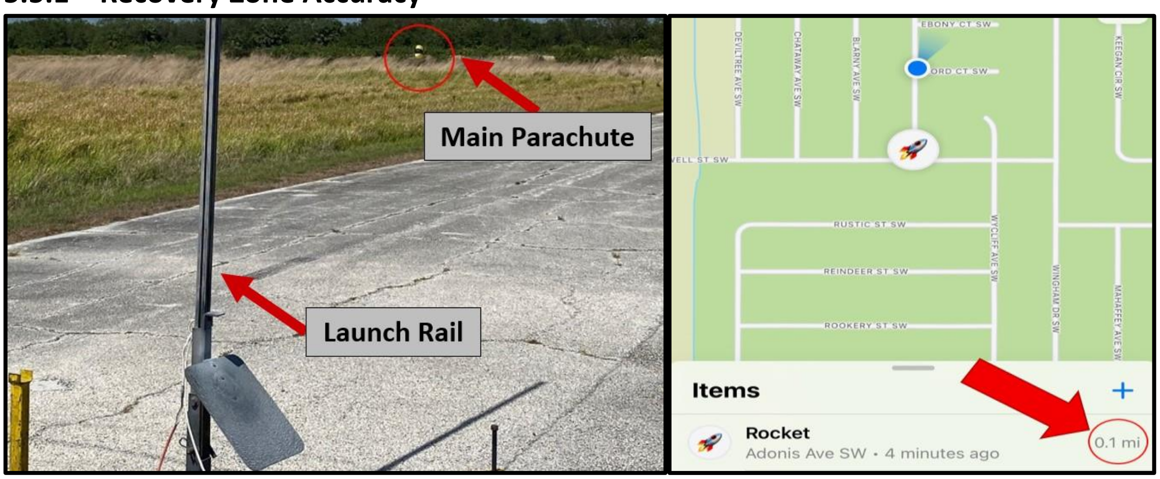
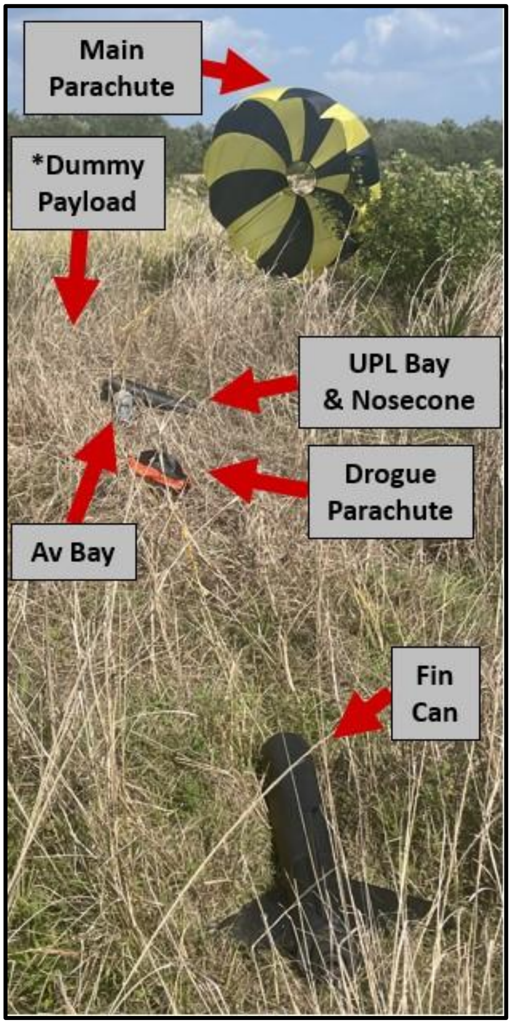


Figure 5-2. Recovery Distance from Launch Rail

The left photo shows the launch rail and fully recovered vehicle in the same frame to visually demonstrate the drift separation. At the time the left photo was taken, the distance to the onboard Apple AirTag was measured from the launch rail, yielding a rough distance of 0.1 miles or 528 feet, well within the required 2500 ft recovery zone.

5.5.2 Vehicle In-Situ



*Dummy payload concealed behind layer of foliage

Figure 5-3. Fully Recovered Vehicle Landing Configuration

The vehicle was recovered in the tall grass fields to the east of the launch rail. At the recovery zone some damage to the vehicle were discovered. Details of the damages, explanation of possible or confirmed failure mechanisms, and discussion of corrective actions will be covered in the flight analysis section.

The vehicle's nosecone lay in two pieces, with the top half shearing at the epoxied shoulder joint and coming to rest under a foot away from the rest of the vehicle section. The fin-can came to rest on its side with one fin facing vertical, two horizontal, and the fourth nowhere to be found. The integrated thrust plate and tail cone component was found to be missing the tail cone, although the thrust plate remained intact. The parachutes, rigging, avionics bay, and any airframe components not discussed were all found to be intact with no failures.

5.6 Flight Analysis

5.6.1 Post-Flight Component Analysis

5.6.1.1 Nosecone and Upper Payload Bay

5.6.1.1.1 Section Damage

Figure 5-4. Upper Payload Bay Sustained Damage

Damage to the upper payload section occurred with a delamination of the nosecone. Also note the dummy payload attached to the recovery harness.

5.6.1.1.2 Damage Mechanism

The nosecone was 3D printed in three separate sections using ABS plastic. The nose failed at the joint between the middle and upper print sections.

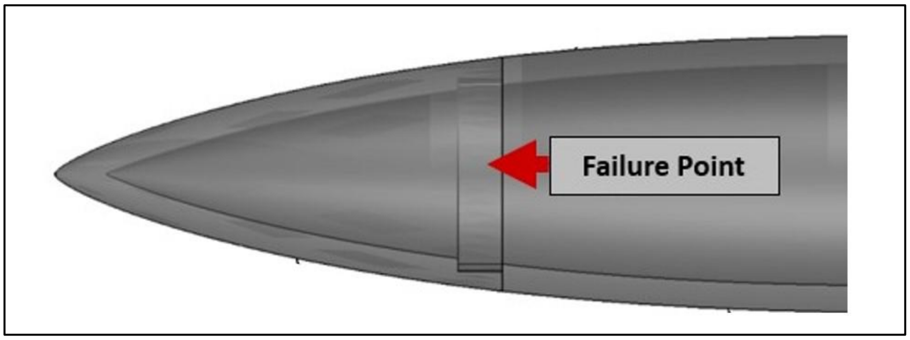


Figure 5-5. Nosecone Shoulder Joint Failure Point 1

The middle nose section attaches to the upper with a printed lip or shoulder which extends into the interior of the upper section. Due to the vertical print orientation, when excessive side load was applied as the nosecone tip hit the ground first, the shoulder, being thinner than the rest of the nosecone, delaminated.

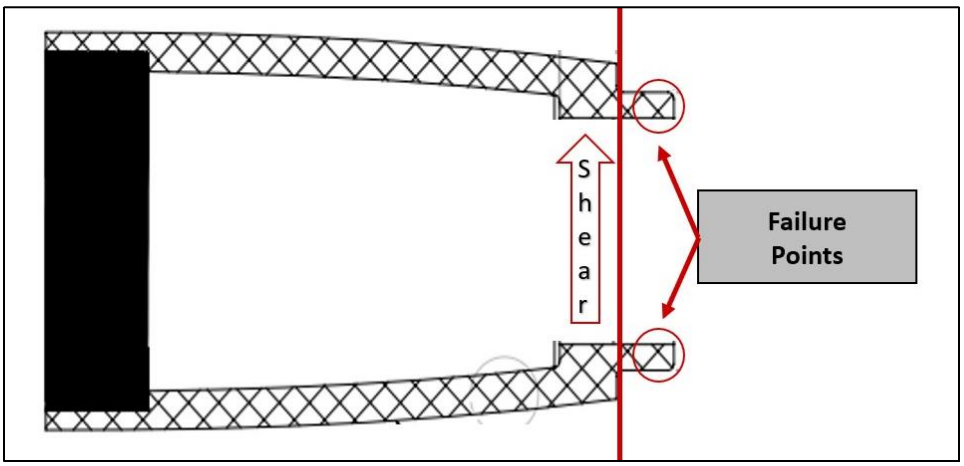


Figure 5-6. Nosecone Shoulder Joint Failure Point 2

5.6.1.1.3 Corrective Action

The simplest corrective action that can be taken to make the vehicle flight ready again is to sand the delaminated surfaces to prepare for reattachment. The team will print a coupler tube component which extends deeper into the inner profiles of the upper and middle nose sections, ensuring adequate thickness to resist shearing under larger loads. Print orientation may also be turned 90 degrees such that the print layers are not parallel to the shear forces exerted at ground impact.

5.6.1.2 Avionics Bay

5.6.1.2.1 Post-Flight Images

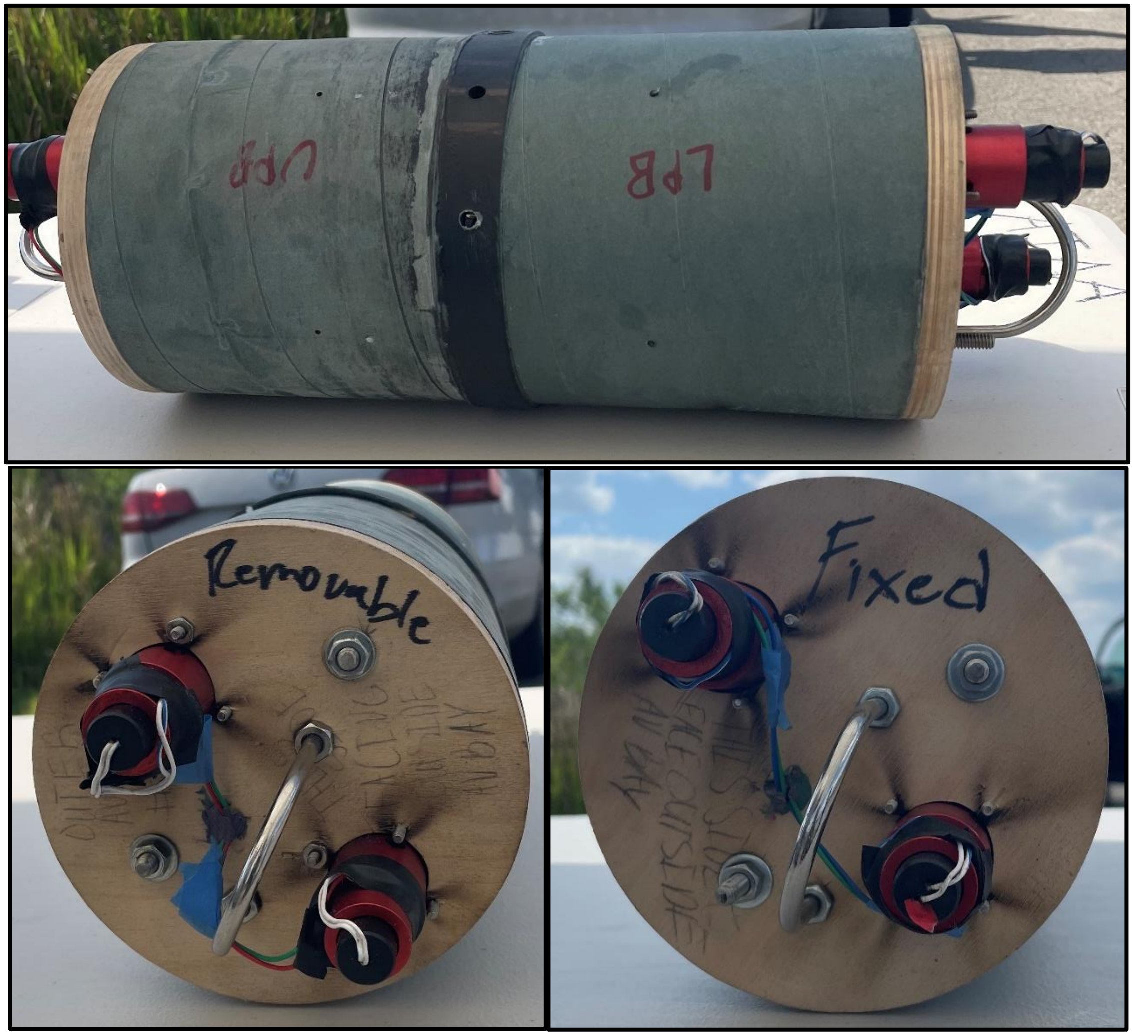


Figure 5-7. Fully Recovered Avionics Bay

5.6.1.2.2 System Success

The avionics bay suffered no structural damage and the internal electronics and wiring were unaffected by flight loads and vibrations or by ground impact. Inspection of the bulkheads showed clear evidence of proper firing of both primary and backup charges at each separation event.

No corrective actions required.

5.6.1.3 Fin Can

5.6.1.3.1 Section Damages

Figure 5-8. Fully Recovered Lower Payload Bay

The fin can was damaged with the shearing of one of the 3D printed fins, and the loss of the cone section of the 3D printed integrate thrust plate / tail cone assembly.

5.6.1.3.2 Fin Damage

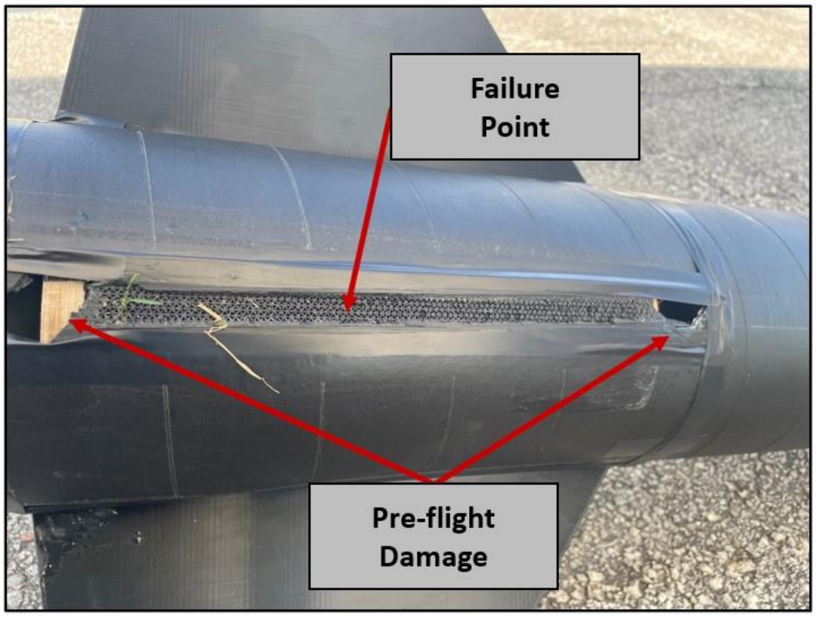


Figure 5-9. Fin Damage

The fin which failed on this flight had pre-existing damage to the rounded plate section on which the aerodynamic surface is printed. The damage which occurred on launch day was a shearing of the fin from the fin plate.

5.6.1.3.3 Failure Mechanisms

For clarity we must state at the outset of this analysis that the fin was not lost to fin flutter or aerodynamic loading during the ascent. Frame by frame video review until separation 1 confirms the vehicle ascended in one piece, and flight data shows no instabilities consistent with what one would expect from such an asymmetry.

During final assembly the fin was dropped, and the plate cracked at either end, but this did not damage the aerodynamic surface. With a 2+ day print time, this fin had to be used for the demonstration flight, as time did now allow for correction of this error. While the extensions of the plate which were damaged are not structural or aerodynamic, the damage sustained may have left the print layers near the top and bottom of the fin compromised and increased the likelihood of failure. Leaving the fin even more prone to failure was the fact this fin set was printed at 70% infill in order to shrink the print time for the same time constraint.

This fin design was also flown with a known design flaw as a result of the same time constraint. The fins were in the process of redesign during final vehicle assembly but material shipment and print time would not allow for implementation for this flight. A detailed explanation of the new fin design can be referred to in Section 3.1.1.

While not a 100% certainty, as the video loses track of the vehicle just before separation 1 and regains track several seconds after drogue deployment, the most reasonable failure mechanism for the fin shear is a collision between airframe segments after drogue deployment. No fin was found within 100 feet of the recovery area, therefore we can infer this was not a result of ground collision. Frame by frame video analysis of the ascent and descent do not show any material separate from the fin can.

The shock cord lengths on either side of the drogue parachute are equal, such that when the parachute catches wind and the airframe sections begin to hang below, a collision is nearly guaranteed. It is therefore most probable that the fin was lost as the camera shot lost vehicle track and the vehicle separated, colliding the upper payload bay with the already weakened fin whose design was prone to shear under large load.

5.6.1.3.4 Corrective Actions

Two corrective actions are being implemented to mitigate this failure mode. First, the fins have been entirely redesigned to remove the design flaw leaving them prone to shearing. The plate has been removed entirely and the fin made to run through the airframe and directly into the clamping centering rings, translating shear loads into the airframe and thrust structure.

Shock cord lengths will be modified such that vehicle sections hang at disparate lengths on either side of the parachute. Cord extensions or parachute relocation will be applied such that when fully hanging beneath the parachutes, vehicle sections will hang with at least 2 lengths of the longest vehicle section in height difference between them. This will prevent future section collisions and has the added benefit of jostling the avionics bay less during parachute deployments.

While these are simple corrections to implement, they are of the utmost priority going into the Payload Demonstration flight, as the risk to personnel on the ground from vehicle components falling unarrested from apogee cannot be understated.

5.6.1.4 Thrust Plate and Tail Cone

5.6.1.4.1 Section Damage

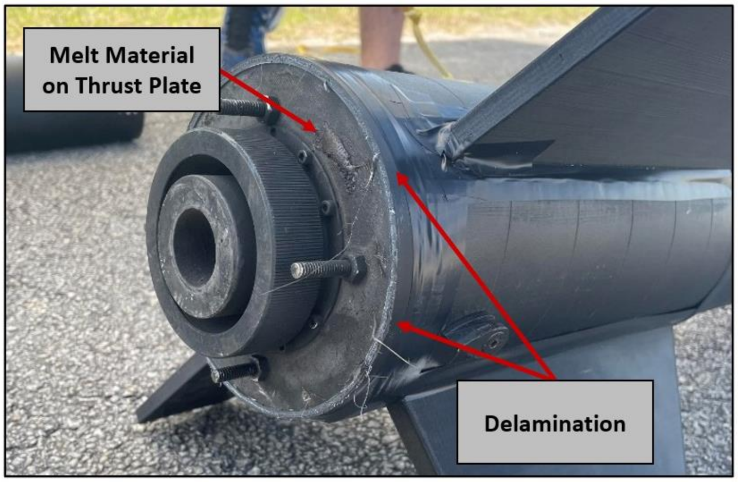


Figure 5-10. Tail Cone Damage

The thrust tail cone damage was by far the most detrimental to the flight, not in that the vehicle performance is much changed by the removal of the tail cone, but because of secondary effects downstream of the failure mode the cone exhibited. The cone entirely delaminated from the thrust plate and was ejected from the vehicle by the motor plume roughly 3 seconds into flight. The failure mechanism and downstream effects are complex.

5.6.1.4.2 Failure Mechanism

Initially the vehicle lifted off the pad and the motor ramped to full thrust without issue. In image 1 the tail cone is present at the base of the vehicle with a well expanded, stable plume from the motor.

Note the sharp 45-degree angles visible as the nozzle expands the flow as designed. At this point the motor has just begun to output max thrust and the vehicle has cleared the rail.

Several frames later the tail cone has been exposed to massive heat load and is beginning to deform. The exit area of the cone has shrunk slightly due to longitudinal stretching. Note the base of the cone is now less than an airframe diameter, its original dimension. The elongation of the cone is likely due to the material becoming pliable as it melts and the vehicle rapidly accelerating upwards, causing a pseudo-force downwards.

Note the degradation of the sharp 45-degree plume expansion because of this constriction. The tail cone is constricting the plume which is attempting to expand as seen in image 1.

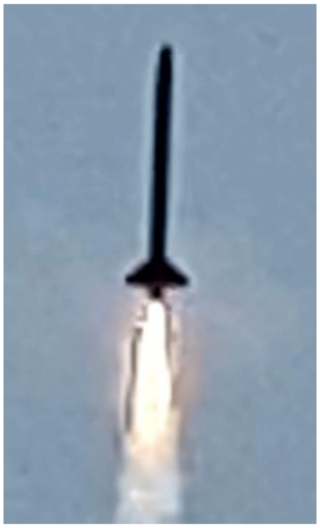


Image 3 shows frames later as the cone continues to melt, deform, and elongate. The exhaust plume is even further constricted and is now an entirely uneven ‘blob’ shape.

This likely results from the bottom edge of the tail cone melting away unevenly and allowing certain parts of the plume to escape earlier and expand.

The flame is now actively burning away the plastic it is in contact with, likely recirculating within the remaining cone, and exerting huge force on this obstruction.

Image 4 shows the moment at which the tail cone delaminates from the thrust plate and is pushed into the exhaust plume. The delamination occurred flush with the thrust plate around the circumference of the plate. The interface print layer was gradually loaded with bending forces as the tail cone stretched and bowed inwards towards the plume. In the previous image, immense downward force was exerted on the cone with this interface layer still under bending stress, causing the separation shown here.

This event occurred less than 30 feet over the pad, and no debris or shattered ABS was found around the area, leading to the inference that the cone was ejected at an angle into the taller grasses nearby, which were not searched as they are nearly impassable, or that the cone was shot down into the pavement and bounced into the same grasses rather than shattering. The former is the more likely scenario of the two.

Immediately following separation of the cone, the vehicle rapidly changed trajectory from near vertical to a fairly horizontal ballistic trajectory. This may be the result of the vehicle being hit by a wind gust or naturally tuning into the oncoming sustained winds, but there is the potential that the interaction of the plume with the cone had a momentary effect on stability.

This final image shows the vehicle several frames later in a stabilized, albeit highly ballistic ascent with the plume also beginning to stabilize once again.

Note the very minor protrusion at the base of the vehicle above the exhaust flame. This is the flanged motor retainer that was shrouded by the tail cone initially.

The vehicle made a stable ascent and successful recovery following this failure mode, but the effects on motor performance were drastic.

5.6.2 Altimeter Data Analysis

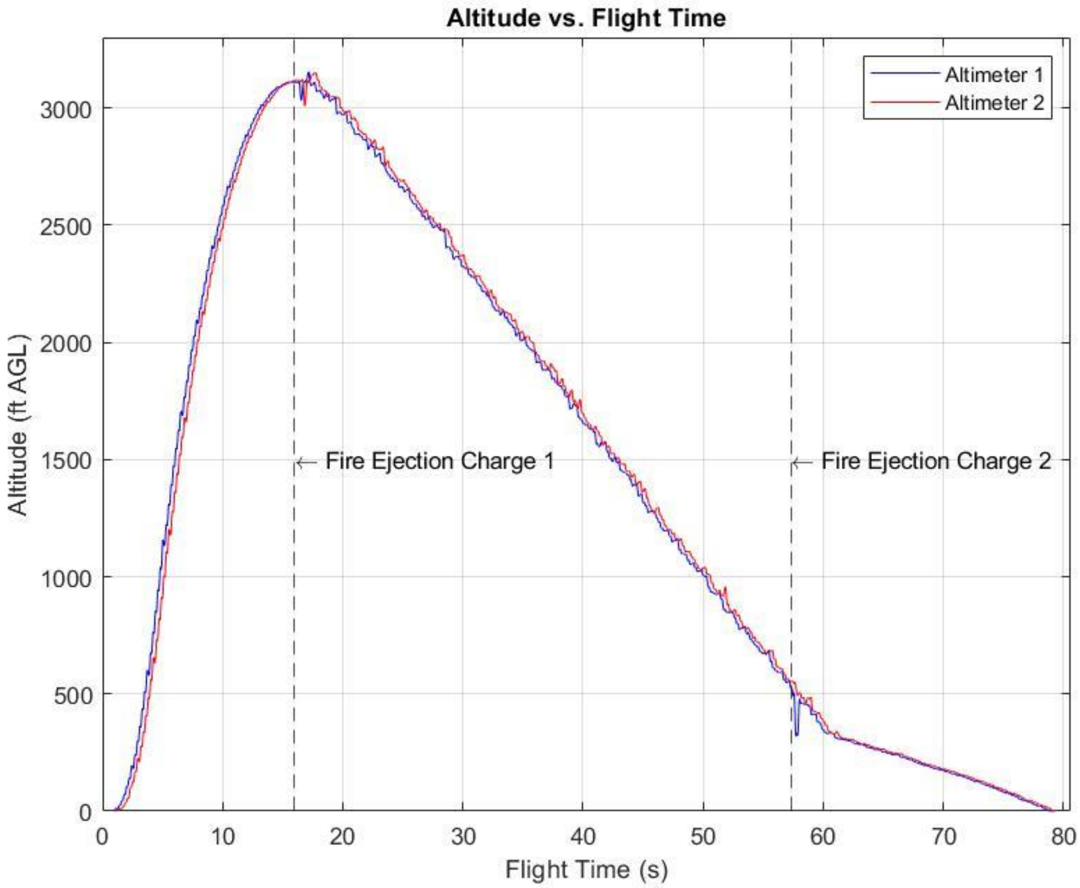


Figure 5-11. Demonstration Flight Altitude vs Time

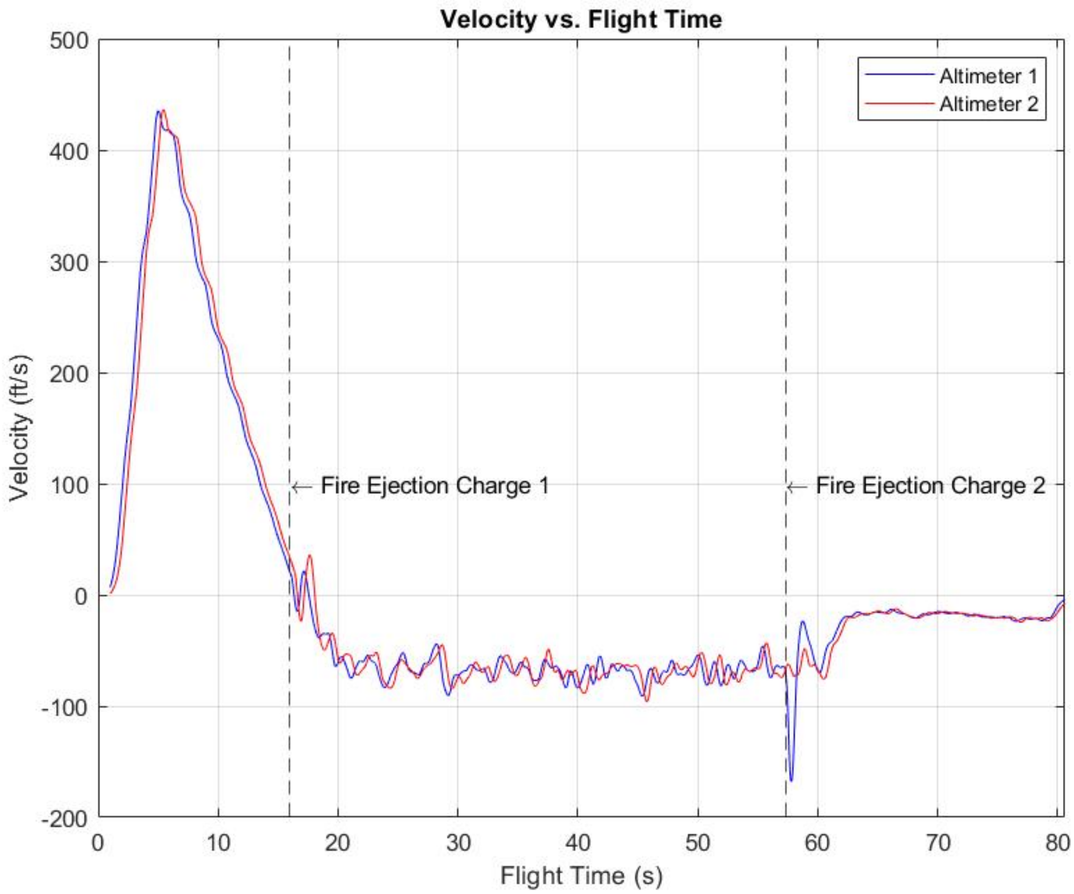


Figure 5-12. Demonstration Flight Velocity vs Time

A MATLAB program was used to process the flight data recovered from both altimeters. Plots were produced using a spline interpolation scheme with a resolution 100x that of the raw data to convert the initial step-plot output to a smooth curve. The program also sought maximums in the data, providing the following outputs for metrics of interest:

Table 5-5. Sub-scale Flight Metrics

	Simulation	Altimeter 1	Altimeter 2
Altitude (ft)	4301	3146	3147
Max Velocity (ft/s)	556	435	436
Max Mach Number	.494	.390	.390

With the data above, a table of deviation and percent error could be generated to evaluate the accuracy of the simulation.

Table 5-6. Sub-scale Flight Error Analysis

	Altimeter 1		Altimeter 2	
	Delta	% Error	Delta	% Error
Altitude (ft)	1,155	26.85	1,154	26.83
Max Velocity (ft/s)	121	21.76	120	21.58
Max Mach Number	0.104	21.05	0.104	21.05

With an error across both altimeters of 26.8% and 21% in the altitude and velocity/Mach, respectively, we can conclude the real vehicle deviated significantly from the model. Given the accuracy of the model in the subscale flight, and the lengthy discussion above of the deviation of the motor exhaust plume from optimal expansion due to the tail cone structural failure, this data serves as concrete proof that the motor efficiency was greatly reduced by the tail cone failure event. Bearing this in mind, the corrective action for the tail cone failure mode must eliminate completely the potential for failure.

5.6.3 Corrective Action Regarding Tail Cone Failure

With the potential for tail cone failures to produce such high errors in our achieved altitude, the decided-on course of action is to drop the component entirely rather than attempt to engineer a component less likely to fail.

<u>The best part is no part.</u>

The risk to vehicle performance cannot be justified for the marginal reduction in wake drag on the vehicle, and the risk to personnel on the ground from a part being violently jettisoned by the full force of an L-class motor is far too great for this program to tolerate.

Bearing in mind the lessons learned from this event, future teams may implement a more robust design with much more time to dedicate towards testing and validation pre-flight.

%% FIXME

- Calculate the kinetic energy at landing for each independent and tethered section of the launch vehicle.
- Estimate the drag coefficient of the full-scale rocket utilizing launch data. Use this value to run a post-flight simulation.
- Discuss the similarities and differences between the full-scale and subscale flight results.

Provide a list of any planned future demonstration flights. Include a summary of the team's objectives for each launch.

6 Safety and Procedures

6.1 Hazard Analysis

6.1.1 Risk Matrix and Definitions

To conduct a Failure Mode and Effects Analysis for each vehicle system, environmental risk assessment, and personnel risk assessment, the risk classification matrix in Table 5-1 was used. Tables 5-1 and 5-2 on the following page define each severity and likelihood class.

Table 6-1. Risk Classification Matrix

Risk Classification Matrix			Event Likelihood			
			Possible	Plausible	Probable	Highly Probably
			A	**B**	**C**	**D**
Event Severity	Marginal	**1**	1A	1B	1C	1D
	Significant	**2**	2A	2B	2C	2D
	Major	**3**	3A	3B	3C	**3D**
	Catastrophic	**4**	4A	4B	4C	**4D**

Table 6-2. Severity Classification Definitions

Severity	Vehicle Outcomes	Personnel Outcomes
Marginal	Little to no impact to vehicle integrity. Flight profile consistent with expectation. Safe recovery. Payload intact and deployed. Vehicle can be reused.	No potential for injury created.
Significant	Vehicle integrity compromised. Minor repair required. Deviation from expected flight profile. Safe recovery. Payload intact and deployable. Vehicle can be reused.	Minor risk of injury created. No injuries.
Major	Vehicle integrity compromised. Substantial repair required. Large deviation from expected flight profile. Recovery may endanger personnel. Payload and deployment mechanism damaged. Vehicle can be reused.	Great risk of injury created. Injuries reported. Injuries are manageable with basic first-aid.
Catastrophic	Vehicle breakup in flight. Irreparable damage. Unarrested descent. Recovery not possible. Payload destroyed. Complete loss of vehicle and payload.	Great risk of injury created. Injuries reported. Injuries require professional medical attention.

Table 6-3. Likelihood Classification Definitions

Likelihood	Definition
Possible	Within the set of all conceivable outcomes. Not likely to occur.
Plausible	Reasonable chance of occurrence due to uncertainty bounds.
Probable	Likley to occur. Uncertainty is now in whether the event will **not** occur.
Highly Probable	Near certainty. Statistical chance of occurrence far outweighs the chance of no occurrence.

6.1.2 Vehicle Systems Failure Mode and Effects Analysis

Table 6-4. Avionics and Power Systems FMEA

Failure Mode	Cause(s)	Hazard Category
(PS.1) Power loss on pad	• Dead battery • Disconnection of leads	1A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of power to flight computer	• Vehicle launch cannot be commanded • Battery replacement required • Personnel must approach cold vehicle – minimal risk	• Ensure battery is charged pre-flight • Have flight computer transmit battery condition • Firm lead attachment • Redundant power/avionics
Failure Mode	**Cause(s)**	**Hazard Category**
(PS.2) Power loss in flight	• Dead battery • Disconnection of leads	4A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of power to flight computer	• Loss of vehicle control • No control authority over recovery system • Unable to measure altitude • Unable to command deployment events • Unarrested descent • Risk to personnel	• Ensure battery is charged pre-flight • Have flight computer transmit battery condition • Firm lead attachment • Redundant power/avionics
Failure Mode	**Cause(s)**	**Hazard Category**
(PS.3) Power loss after recovery	• Dead battery • Disconnection of leads	1A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of power to flight computer	• Loss of control authority over payload deployment mechanism • Unable to deploy payload	• Ensure battery is charged pre-flight • Have flight computer transmit battery condition • Firm lead attachment • Redundant power/avionics

Failure Mode	Cause(s)	Hazard Category
(AV.1) In-flight barometer failure	• Bad component • Poor component calibration • Power loss	2A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Altitude cannot be determined from atmospheric pressure	• Vehicle relies on double integration of accelerometer data for altitude • Large compounding errors in integration may cause off-nominal main deployment • Nominal drogue deployment using accelerometer	• Purchase components from reputable dealer • Test components extensively before flight • Firm electrical lead attachments • Redundant power/avionics
Failure Mode	**Cause(s)**	**Hazard Category**
(AV.2) In-flight accelerometer failure	• Bad component • Poor component calibration • Power loss	2A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Altitude and velocity cannot be determined by integration of acceleration data	• Vehicle relies on inflection of barometric data to determine apogee (pressure begins increasing) • Potential off-nominal drogue deploy • Nominal main chute deployment using barometer	• Purchase components from reputable dealer • Test components extensively before flight • Firm electrical lead attachments • Redundant power/avionics
Failure Mode	**Cause(s)**	**Hazard Category**
(AV.3) Simultaneous in-flight accelerometer/barometer failure	• Power loss	2A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Altitude and velocity cannot be determined	• Recovery events reliant on time-commanded backup charges • Off-nominal drogue deploy • Off-nominal main deploy	• Purchase components from reputable dealer • Test components extensively before flight • Firm electrical lead attachments • Redundant power/avionics

Failure Mode	**Cause(s)**	**Hazard Category**
(AV.4) In-flight/post-flight GPS unit failure	• Bad component • Poor component calibration • Power loss	2A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Vehicle landing site cannot be precisely determined	• Sonic beacon becomes primary locator • Visual tracking to ground aids in recovery	• Purchase components from reputable dealer • Test components extensively before flight • Firm electrical lead attachments • Redundant power/avionics
Failure Mode	**Cause(s)**	**Hazard Category**
(AV.5) Flight computer failure (pre-flight)	• Bad component • Power loss	2A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of control authority over vehicle	• Vehicle launch cannot be commanded • Personnel must approach cold vehicle – minimal risk	• Same as previous
Failure Mode	**Cause(s)**	**Hazard Category**
(AV.6) Flight computer failure (in-flight)	• Bad component • Power loss	4A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of control authority over vehicle	• No control authority over recovery system • Unable to measure altitude • Unable to command deployment events • Unarrested descent • Risk to personnel	• Same as previous
Failure Mode	**Cause(s)**	**Hazard Category**
(AV.7) Flight computer failure (post-flight)	• Bad component • Power loss	1A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of control authority over vehicle	• Loss of control authority over payload deployment mechanism • Unable to deploy payload	• Same as previous

Failure Mode	Cause(s)	Hazard Category
(AV.8) Wire leads disconnect	• Excessive vehicle vibration • Poor terminal connections	4D
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Any combination of AV.1 – AV.4, AV.6, and AV.7 failure modes	• Loss of control authority over vehicle • No control authority over recovery system • Unable to measure altitude • Unable to command deployment events • Unarrested descent • Risk to personnel • Loss of control authority over payload deployment mechanism • Unable to deploy payload	• Ensure proper soldering of terminal leads • Extensively test robustness of connections to tension and vibration • Implement vibration damping measures for electrical components • Redundant power/avionics

Table 6-5. Avionics and Power Systems Risk Matrix

Risk Classification Matrix			Event Likelihood			
			Possible	Plausible	Probable	Highly Probably
			A	B	C	D
Event Severity	Marginal	1	PS.1 1A AV.7 PS.3	1B	1C	1D
	Significant	2	AV.1 AV.4 AV.2 2A AV.3 AV.5	2B	2C	2D
	Major	3	3A	3B	3C	3D
	Catastrophic	4	PS.2 4A AV.6	4B	4C	AV.8 4D

Table 6-6. Energetics and Pyrotechnics FMEA

Failure Mode	Cause(s)	Hazard Category
(PRO.1) Failed motor igniter	• E-match fails to ignite • Black powder pellet fails to ignite after E-match	3B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Vehicle remains on launchpad in unknown state	• E-match/igniter replacement required • Personnel must approach warm vehicle – significant risk • Dud ignition converts vehicle cold • Random ignition in time following dud – significant risk to personnel approaching	• Redundant e-matches • E-match close proximity to black powder pellet
Failure Mode	**Cause(s)**	**Hazard Category**
(PRO.2) Ejection charge initiation failure	• E-match fails to ignite	2B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Body sections do not separate	• Separation dependent on backup charge (time initiated) • Off-nominal parachute deployment	• Redundant e-matches
Failure Mode	**Cause(s)**	**Hazard Category**
(PRO.3) Ejection charge fails to separate sections	• Insufficient black powder load • Excessive friction in coupler • Shock cord entanglement	2B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Body sections do not fully separate	• Structural damage between colliding body sections • Separation dependent on backup charge (time initiated) • Off-nominal parachute deployment	• Redundant ejection charges: • Time-commanded backup charge

Failure Mode	Cause(s)	Hazard Category
(EN.1) Unintentional motor ignition	• Static Discharge • Human Error	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Launch vehicle departs launch rails unexpectedly	• Flight computer not prepared to execute profile • Unable to command recovery sequence • Burns and hearing damage to personnel in immediate vicinity of vehicle	• Ensure vehicle is grounded in prep area and on pad • Ensure proper communication during count sequence
Failure Mode	**Cause(s)**	**Hazard Category**
(EN.2) Unintentional ejection charge initiation (pre-flight)	• Static Discharge • Human Error	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Unexpected black powder detonation	• Creation of large audible signature and expulsion of hot exhaust gasses • Great injury to personnel standing in line with and near charge. Medical emergency • Burns and hearing damage to personnel in immediate vicinity of vehicle • Body section(s) are ejected • Body sections impact nearby personnel. Minor to significant injuries	• Ensure vehicle is grounded in prep area and on pad • Ensure proper communication during count sequence • Implement CO2 ejection system

Failure Mode	**Cause(s)**	**Hazard Category**
(EN.3) Uneven combustion in solid fuel	• Poor mixing of fuel and oxidizer • Poor distribution of propellant in case	4C
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Asymmetric thrust about vehicle z-axis	• Deviation from expected flight path • Loss of vehicle stability • In-flight break up of vehicle. Loss of vehicle • Unarrested descent. Risk to personnel	• Purchase motor from reputable dealer (Cesaroni is the current selection)
Failure Mode	**Cause(s)**	**Hazard Category**
(EN.4) Motor exhaust in body tube	• Motor case rupture • Nozzle foreword of thrust plate	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Damage to body tube • Loss of vehicle integrity	• Mid-flight fin detachment • Catastrophic body rupture • Vehicle in-flight breakup • Loss of vehicle	• Aluminum motor case, thrust plate, and motor retainer • Extensive sealing in motor compartment
Failure Mode	**Cause(s)**	**Hazard Category**
(EN.5) Motor jettison	• Thrust plate or motor retainer failure	3A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Motor and casing separate from launch vehicle after burnout	• Changes to stability margin as Cg shifts towards nose • Deviation from projected flight profile • Risk to personnel from uncontrolled, unarrested descent of metal motor casing	• Aluminum thrust plate and motor retainer to ensure dynamic loading margins are not exceeded

Failure Mode	**Cause(s)**	**Hazard Category**
(EN.6) Avionics damage	• Hot/corrosive ejection charge exhaust gasses	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Development of any AV.1 – AV.4 and AV.6 Failure Modes	• No control authority over recovery system • Unable to measure altitude • Unable to command deployment events • Unarrested descent • Risk to personnel	• Insulate void space in body • Implement CO2 ejection system
Failure Mode	**Cause(s)**	**Hazard Category**
(EN.7) Burned parachute(s)	• Hot/corrosive ejection charge exhaust gasses	4D
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Drogue and/or main parachute unable to provide sufficient drag to slow descent	• Partially or fully unarrested descent • Fire inside body tube • Fire in canopy on descent	• Kevlar blankets to retain chutes • Insulate void space • Implement CO2 ejection system
Failure Mode	**Cause(s)**	**Hazard Category**
(EN.8) Chain detonation of ejection charges	• Hot/corrosive ejection charge exhaust gasses	3B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Multiple separation event at apogee • Simultaneous deployment of drogue and main chute	• Deviation from intended flight profile • Risk to personnel from (4) and (5) • structural damage to colliding body sections • Parachute entanglement. Increased descent rate Uncontrolled descent. • Decreased descent rate. Increased wind drift. Vehicle exits recovery zone	• Insulate void space in body • Implement CO2 cooling system to black powder ejection charges

Table 6-7. Energetics and Pyrotechnics Risk Matrix

Risk Classification Matrix			Event Likelihood			
			Possible	Plausible	Probable	Highly Probably
			A	B	C	D
Event Severity	Marginal	1	1A	1B	1C	1D
	Significant	2	2A	PRO.2 2B PRO.3	2C	2D
	Major	3	EN.5 3A	PRO.1 3B EN.8	3C	3D
	Catastrophic	4	4A	EN.1 EN.4 4B EN.2 EN.6	EN.3 4C	EN.7 4D

Table 6-8. Recovery System FMEA

Failure Mode	Cause(s)	Hazard Category
(RS.1) Drogue parachute entanglement	• Poor shock cord stowage in body • Snag hazards in deployment path	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• High descent rate after apogee • Main parachute deployment at higher speed	• Main parachute canopy damaged in high-speed deployment • Main parachute cords tear or rupture • Partially or fully unarrested vehicle descent • Over tensioning of vehicle shock cord. Cord tearing or rupture • Unarrested descent of body sections • Risk to personnel • Major repair needed	• Design for no snag hazards in deployment path of parachute • Reeve loose shock cord • Implement cord routing solutions
Failure Mode	**Cause(s)**	**Hazard Category**
(RS.2) Main parachute entanglement	• Poor shock cord stowage in body • Snag hazards in deployment path	3B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• High descent rate after main deployment • High ground impact velocity	• Partially arrested descent • Damage to vehicle structures • Damage to internal components • Major repair required	• Design for no snag hazards in deployment path of parachute • Reeve loose shock cord • Implement cord routing solutions
Failure Mode	**Cause(s)**	**Hazard Category**
(RS.3) Shock cord rupture	• Excessive tension on cord	3A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Tether between body sections compromised	• Unarrested descent of body section(s)	• Extensive simulation pre-flight • Select shock cord with large factor of safety

Failure Mode	Cause(s)	Hazard Category
(RS.4) Shock cord entanglement	• Poor shock cord stowage in body • Snag hazards in deployment path	1B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Shock cord unable to extend to full length	• Collision of body sections on descent • Very minor damage to structure	• Reeve loose shock cord • Implement cord routing solutions

Table 6-9. Recovery System Risk Matrix

Risk Classification Matrix			Event Likelihood			
			Possible	Plausible	Probable	Highly Probably
			A	B	C	D
Event Severity	Marginal	1	1A	RS.4 1B	1C	1D
	Significant	2	2A	2B	2C	2D
	Major	3	RS.3 3A	RS.2 3B	3C	3D
	Catastrophic	4	4A	RS.1 4B	4C	4D

Table 6-10. Vehicle Structures FMEA

Failure Mode	Cause(s)	Hazard Category
(STR.1) Melting of fin assembly during motor burn	• Heat transfer from motor case • Lack of heat resistance in fin material	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of flight stability	• Vehicle breakup in-flight • Loss of vehicle • Unarrested descent of body sections • Risk to personnel	• Use heat resistant print material • Treat for heat resistance • Minimize heat transfer
Failure Mode	**Cause(s)**	**Hazard Category**
(STR.2) Fins shear off	• Fin flutter • Aerodynamic loading	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of flight stability	• Vehicle breakup in-flight • Loss of vehicle • Unarrested descent of body sections • Risk to personnel	• Extensive simulation pre-flight • Ensure flutter speed >> max vehicle velocity
Failure Mode	**Cause(s)**	**Hazard Category**
(STR.3) Body tube zippering	• Shock cord contact with body on deployment	3B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of vehicle integrity	• Vehicle damage on descent • Major repair needed	• Implement “bumpers” to avoid cord contact • Implement cord routing
Failure Mode	**Cause(s)**	**Hazard Category**
(STR.4) Damaged motor retainer	• Defect in part • Excessive dynamic loading	3A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Potential motor jettison after burnout	• Unarrested descent of motor casing • Risk to personnel • Minor repair required	• Aluminum motor retainer to absorb far larger loads than necessary

Failure Mode	Cause(s)	Hazard Category
(STR.5) Bulkhead or U-bolt torn loose	• Excessive loading during chute deployment • Late chute deployment	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Body section(s) disconnected from parachute	• Unarrested descent of body section(s) • Risk to personnel • Major repairs required	• Extensive pre-flight simulation • Extra thick bolts and wide bracing on bulkheads
Failure Mode	**Cause(s)**	**Hazard Category**
(STR.6) Dislodged centering ring(s)	• Defect in part(s) • Excessive dynamic loading • Poor connection to threaded rods	3A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Motor long axis no longer colinear with vehicle z-axis	• Deviation from flight profile • Minor loss of stability • Risk to personnel	• Fix centering rings to threaded rods with hex nuts • Use thread lock to fix nuts
Failure Mode	**Cause(s)**	**Hazard Category**
(STR.7) Damaged payload retainer	• Defect in part(s) • Poor 3D print • Excessive dynamic loading • Excessive ground impact velocity	1B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Payload sits loose in bay	• Minor decrease in vehicle stability • Minor camera housing damage • Improper or impossible rover deployment	• Extensive pre-flight testing • Minimize ground impact velocity • Cushion landing
Failure Mode	**Cause(s)**	**Hazard Category**
(STR.8) Damaged avionics sled retainer(s)	• Defect in part(s) • Poor 3D print • Excessive dynamic loading • Excessive ground impact velocity	3B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Avionics sleds sit loose in av bay	• Potential for AV.8 failure mode • Loss of control authority over vehicle	• Extensive pre-flight testing • Minimize ground impact velocity • Cushion landing

Table 6-11. Vehicle Structures Risk Matrix

Risk Classification Matrix			Event Likelihood			
			Possible	Plausible	Probable	Highly Probably
			A	B	C	D
Event Severity	Marginal	1	STR.7 1A	1B	1C	1D
	Significant	2	2A	STR.4 2B	2C	2D
	Major	3	STR.6 3A	STR.3 3B STR.8	3C	3D
	Catastrophic	4	4A	STR.1 4B STR.5 STR.2	4C	4D

Table 6-12. Payload FMEA

Failure Mode	Cause(s)	Hazard Category
(PLD.1) 3-D printed housing damaged	• High ground impact velocity • Defects in 3D print	1C
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Structure of camera housing compromised	• Internal wiring shifted. Leads torn from Arduino	• Extensive pre-flight testing • Minimize ground impact velocity • Cushion landing
Failure Mode	**Cause(s)**	**Hazard Category**
(PLD.2) Payload lands in a manner where camera is not right side up	• Improper connection to shock cord • High lateral landing velocity	1B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Payload may be in a configuration which occludes camera view	• Images gathered during RAFCO sequence are poor	• Design connection points to maximize possibility of landing z-axis up • Implement anti-roll measures in the design
Failure Mode	**Cause(s)**	**Hazard Category**
(PLD.3) Power loss	• Dead battery • Electrical lead disconnection	1B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of control authority over camera	• RAFCO Mission failure	• Charge battery pre-flight • Firm electrical connections
Failure Mode	**Cause(s)**	**Hazard Category**
(PLD.4) Antenna disconnection from microcontroller	• Excessive vibration in flight • Excessive ground impact velocity	1D
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of control authority over camera and antenna	• RAFCO Mission failure	• Firm electrical connections • Pad landing, reduce velocity

Failure Mode	Cause(s)	Hazard Category
(PLD.5) Microcontroller unit failure	• Bad component • Power loss	1A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of control authority over camera and antenna	• RAFCO Mission failure	• Firm electrical connections • Reputable supplier
Failure Mode	**Cause(s)**	**Hazard Category**
(PLD.6) Camera failure	• Broken lens during ground impact • Power loss	1A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of image capturing ability	• RAFCO Mission failure	• Padding or protection around camera assembly • Firm electrical connections
Failure Mode	**Cause(s)**	**Hazard Category**
(PLD.7) Camera actuation system failure	• Motor failure • Obstructed gears • Power loss	1B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Camera cannot swivel camera	• RAFCO Mission failure	• Firm electrical connections • Clean gear mechanism • Use self-contained stepper motor

Table 6-13. Payload Risk Matrix

Risk Classification Matrix			**Event Likelihood**			
			Possible	Plausible	Probable	Highly Probably
			A	**B**	**C**	**D**
Event Severity	Marginal	**1**	PLD.5 1A PLD.6	PLD.2 1B PLD.3 PLD.7	PLD.1 1C	PLD.4 1D
	Significant	**2**	2A	2B	2C	2D
	Major	**3**	3A	3B	3C	**3D**
	Catastrophic	**4**	4A	4B	4C	**4D**

Table 6-14. Environment FMEA

Vehicle Risks to Environment		
Failure Mode	**Cause(s)**	**Hazard Category**
(ENV.1.1) Launch pad/recovery area fire (energetic initiated)	• Dry vegetation in vicinity of motor ignition	3B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Danger to wildlife • Danger to habitat • Danger to personnel	• Potential for fire growth if left unmitigated	• Clear launch area of vegetation
Failure Mode	**Cause(s)**	**Hazard Category**
(ENV.1.2) Launch pad/recovery area fire (LiPo battery initiated)	• Battery overcharge, over discharge, overtemp	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Danger to wildlife • Danger to habitat • Danger to personnel • HazMat release	• Pollution of crops with HazMat • Pollution of groundwater with HazMat	• Clear launch area of vegetation • Do not use battery improperly
Failure Mode	**Cause(s)**	**Hazard Category**
(ENV.1.3) Interstage insulation littered in launch/ recovery area	• Insulation used in body tube to minimize void space and insulate parachutes from ejection charge gasses	1C
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Ingestion of insulation by wildlife	• Disrespectful to property owners to eject litter on their land	• Biodegradable insulation (popcorn)
Failure Mode	**Cause(s)**	**Hazard Category**
(ENV.1.4) Litter spread over launch site by personnel	• Lack of trashcans • Poor team leadership	1D
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Ingestion of litter by wildlife	• Disrespectful to property owners to litter on their land	• Bring trash bags • Firm leadership. Zero tolerance for littering

Environmental Risks to Vehicle		
Failure Mode	**Cause(s)**	**Hazard Category**
(ENV.2.1) Vehicle touches down in nearby trees	• Excessive wind drift	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Difficulty in or inability to recover launch vehicle • Minor damage to vehicle components	• Loss of vehicle • Repairs required	• Extra-long shock cord to bring components closer to ground
Failure Mode	**Cause(s)**	**Hazard Category**
(ENV.2.2) Vehicle touches down in nearby body of water	• Excessive wind drift	3B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Damage to body tube structure • Damage to avionics or payload electronics	• Major repairs required	• Extensive sealing of avionics bay and rover GNC unit
Failure Mode	**Cause(s)**	**Hazard Category**
(ENV.2.3) In-flight Collision	• Tall infrastructure (power lines) • Bird strike	4A
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Loss of stability • Damage to animal or object impacted	• Loss of vehicle • Repair to damaged infrastructure required	• Ensure vehicle is launched away from all infrastructure • Await clear skies
Failure Mode	**Cause(s)**	**Hazard Category**
(ENV.2.4) Vehicle or components dropped	• Uneven launch site terrain causes personnel tripping	3B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Damage to vehicle structures • Damage to payload structures • Damage to avionics • Damage to payload electronics	• Inability to launch • Repairs required	• Practice extreme caution while handling vehicle components

Table 6-15. Environmental Risk Matrix

Risk Classification Matrix			Event Likelihood			
			Possible	Plausible	Probable	Highly Probably
			A	**B**	**C**	**D**
Event Severity	Marginal	**1**	1A	1B	ENV.1.3 1C	ENV.1.4 1D
	Significant	**2**	2A	2B	2C	2D
	Major	**3**	3A	ENV.1.1 3B ENV.2.2 ENV.2.4	3C	3D
	Catastrophic	**4**	ENV.2.3 4A	ENV.1.2 4B ENV.2.1	4C	4D

6.1.3 Personnel Risk Assessment

Personnel risk assessment was conducted using the same FMEA format as was used for vehicle systems and environmental risk assessment.

Table 6-16. Personnel FMEA

Failure Mode	Cause(s)	Hazard Category
(PPL.1) Skin contact with APCP solid propellant	• Improper material handling • Lack of PPE	3D
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Chemical burns • Eye irritation	• None	• Provide safety training • Provide PPE
Failure Mode	**Cause(s)**	**Hazard Category**
(PPL.2) Electrocution	• Improper safety procedures followed • Live electrical while wiring	2D
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Discomfort/pain • Burns	• Greater or grave injury with prolonged exposure	• Provide safety training
Failure Mode	**Cause(s)**	**Hazard Category**
(PPL.3) Proximity to high-pressure burst event (CO_2 charge)	• Overpressure in pressure vessel • Pressure vessel tipping • Human error	3B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Hearing damage • Struck/Impaled by flying object(s)	• None	• Provide safety training • Do not overfill pressure vessels • Pressure vessels chained to walls • Declare all testing and clear area prior to initiation

Failure Mode	Cause(s)	Hazard Category
(PPL.4) Proximity to explosive event (Black powder charge)	• Accidental initiation (human error, static discharge)	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Hearing damage • Burns from expanding hot gasses	• Severity increased with proximity • Severity increased with decreased angle-off-bore of charge	• Ground vehicle components • Minimize personnel handling charges • Isolate firing mechanism until range clear
Failure Mode	**Cause(s)**	**Hazard Category**
(PPL.5) Proximity to combustion event	• Motor ignition (intentional) • Motor ignition (unintentional) • Loose black powder burn	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Hearing damage • Burns from expanding hot gasses	• Severity increased with proximity • Severity increased with decreased angle-off-bore of charge	• Ground vehicle components • Minimize personnel handling motor • Isolate ignition mechanism until range clear
Failure Mode	**Cause(s)**	**Hazard Category**
(PPL.6) Injury: slip and fall, minor cuts, accidental collisions	• Uneven terrain • Tripping hazards on flat ground • Improperly stored sharp objects	3B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Pain/discomfort • Bruises • Small lacerations	• Infection of lacerations not immediately treated	• Situational awareness • Clean lab spaces • Proper safety procedures
Failure Mode	**Cause(s)**	**Hazard Category**
(PPL.7) Dehydration, heat exhaustion, heat stroke	• Lack of water • Lack of adequate sun protection or shade	4B
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Thirst • Disorientation • Loss of consciousness	• None	• Provide ample water • Bring portable awning/tent • Bring sunscreen, hats, etc.

Failure Mode	Cause(s)	Hazard Category
(PPL.8) Soldering iron burns	• Improper use or stowage of soldering iron	3D
Primary Effect(s)	**Secondary Effect(s)**	**Mitigations**
• Minor burns	• Increased severity with prolonged contact	• Proper training in use of soldering iron • Minimize personnel involved

Table 6-17. Personnel Risk Matrix

Risk Classification Matrix			Event Likelihood			
			Possible	Plausible	Probable	Highly Probably
			A	B	C	D
Event Severity	Marginal	1	1A	1B	1C	1D
	Significant	2	2A	2B	2C	PPL.2 2D
	Major	3	3A	PPL.3 3B PPL.6	3C	PPL.1 3D PPL.8
	Catastrophic	4	4A	PPL.4 4B PPL.7 PPL.5	4C	4D

6.2 Verification Strategies for proposed mitigations

Table 6-18. Risks, Mitigations, and Verifications

Avionics and Power Systems			
Failure Mode	**Cause(s)**	**Mitigations**	**Verification Strategy**
(PS.1) Power loss on pad	• Dead battery • Disconnection of leads	• Ensure battery is charged pre-flight • Have flight computer transmit battery condition • Firm lead attachment • Redundant power/avionics	• Assembly/Operations Checklist • Pre-flight hardware testing (vibration test)
(PS.2) Power loss in flight	• Dead battery • Disconnection of leads	• Ensure battery is charged pre-flight • Have flight computer transmit battery condition • Firm lead attachment • Redundant power/avionics	• Assembly/Operations Checklist • Pre-flight hardware testing (vibration test)
(PS.3) Power loss after recovery	• Dead battery • Disconnection of leads	• Ensure battery is charged pre-flight • Have flight computer transmit battery condition • Firm lead attachment • Redundant power/avionics	• Assembly/Operations Checklist • Pre-flight hardware testing (vibration test)
(AV.1) In-flight barometer failure	• Bad component • Poor component calibration • Power loss	• Purchase components from reputable dealer • Test components extensively before flight • Firm electrical lead attachments Redundant power	• Mentor input • Pre-flight hardware testing (vibration test)
(AV.2) In-flight accelerometer failure	• Bad component • Poor component calibration • Power loss	• Purchase components from reputable dealer • Test components extensively before flight	• Mentor input • Pre-flight hardware testing (vibration test)

		• Firm electrical lead attachments • Redundant power/avionics	
(AV.3) Simultaneous in-flight accelerometer/ barometer failure	• Power loss	• Purchase components from reputable dealer • Test components extensively before flight • Firm electrical lead attachments • Redundant power/avionics	• Mentor input • Pre-flight hardware testing (vibration test)
(AV.4) In-flight/post-flight GPS unit failure	• Bad component • Poor component calibration • Power loss	• Purchase components from reputable dealer • Test components extensively before flight • Firm electrical lead attachments Redundant power/avionics	• Mentor input • Pre-flight hardware testing (vibration test)
(AV.5) Flight computer failure (pre-flight)	• Bad component • Power loss	• Same as previous	• Mentor input • Pre-flight hardware testing (vibration test)
(AV.6) Flight computer failure (in-flight)	• Bad component • Power loss	• Same as previous	• Mentor input • Pre-flight hardware testing (vibration test)
(AV.7) Flight computer failure (post-flight)	• Bad component • Power loss	• Same as previous	• Mentor input • Pre-flight hardware testing (vibration test)
(AV.8) Wire leads disconnect	• Excessive vehicle vibration • Poor terminal connections	• Ensure proper soldering of terminal leads • Extensively test robustness of connections to tension and vibration • Implement vibration damping measures for electrical components Redundant power	• Pre-flight hardware testing (vibration test)

Energetics and Pyrotechnics			
Failure Mode	**Cause(s)**	**Mitigations**	**Verification Strategy**
(PRO.1) Failed motor igniter	• E-match fails to ignite • Black powder pellet fails to ignite after E-match	• Redundant e-matches • E-match close proximity to black powder pellet	• Assembly/Operations Checklist
(PRO.2) Ejection charge initiation failure	• E-match fails to ignite	• Redundant e-matches	• Assembly/Operations Checklist
(PRO.3) Ejection charge fails to separate sections	• Insufficient black powder load • Excessive friction in coupler Shock cord entanglement	• Redundant ejection charges • Time-commanded backup charge	• Assembly/Operations Checklist • Pre-flight hardware testing (separation test) • Pre-flight simulations (Shear pin FEA)
(EN.1) Unintentional motor ignition	• Static Discharge • Human Error	• Ensure vehicle is grounded in prep area and on pad • Ensure proper communication during count sequence	• Assembly/Operations Checklist
(EN.2) Unintentional ejection charge initiation (pre-flight)	• Static Discharge • Human Error	• Ensure vehicle is grounded in prep area and on pad • Ensure proper communication during count sequence • CO2 ejection system	• Assembly/Operations Checklist
(EN.3) Uneven combustion in solid fuel	• Poor mixing of fuel and oxidizer • Poor distribution of propellant in case	• Purchase motor from reputable dealer	• Mentor input
(EN.4) Motor exhaust in body tube	• Motor case rupture Nozzle foreword of thrust plate	• Aluminum motor case, thrust plate, and motor retainer • Extensive sealing in motor compartment	• Assembly/Operations Checklist • Mentor input

(EN.5) Motor jettison	• Thrust plate or motor retainer failure	• Aluminum thrust plate and motor retainer to ensure dynamic loading margins are not exceeded	• Pre-flight hardware testing (load test) • Pre-flight simulations (FEA)
(EN.6) Avionics damage	• Hot/corrosive ejection charge exhaust gasses	• Insulate void space in body • Implement CO2 ejection system	• Pre-flight hardware testing (separation test) • Pre-flight simulations (CFD heat transfer)
(EN.7) Burned parachute(s)	• Hot/corrosive ejection charge exhaust gasses	• Kevlar blankets to retain chutes • Insulate void space • Implement CO2 ejection system	• Pre-flight hardware testing (separation test) • Pre-flight simulations (CFD heat transfer)
(EN.8) Chain detonation of ejection charges	• Hot/corrosive ejection charge exhaust gasses	• Insulate void space in body • Implement CO2 cooling system to black powder ejection charges	• Pre-flight hardware testing (separation test) • Pre-flight simulations (CFD heat transfer)
Recovery System			
Failure Mode	**Cause(s)**	**Mitigations**	**Verification Strategy**
(RS.1) Drogue parachute entanglement	• Poor shock cord stowage in body • Snag hazards in deployment path	• Design for no snag hazards in deployment path of parachute • Reeve loose shock cord • Implement cord routing solutions	• Assembly/Operations Checklist • Mentor input
(RS.2) Main parachute entanglement	• Poor shock cord stowage in body • Snag hazards in deployment path	• Design for no snag hazards in deployment path of parachute • Reeve loose shock cord • Implement cord routing solutions	• Assembly/Operations Checklist • Mentor input
(RS.3) Shock cord rupture	• Excessive tension on cord	• Extensive simulation pre-flight • Select shock cord with large factor of safety	• Pre-flight hardware testing (weighted drop test) • Pre-flight simulations
(RS.4) Shock cord entanglement	• Poor shock cord stowage in body • Snag hazards in deployment path	• Reeve loose shock cord • Implement cord routing solutions	Assembly/Operations Checklist • Mentor input

Vehicle Structures			
Failure Mode	**Cause(s)**	**Mitigations**	**Verification Strategy**
(STR.1) Melting of fin assembly during motor burn	• Heat transfer from motor case • Lack of heat resistance in fin material	• Use heat resistant print material • Treat for heat resistance • Minimize heat transfer	• Material testing with High Performance Materials Institute
(STR.2) Fins shear off	• Fin flutter • Aerodynamic loading	• Extensive simulation pre-flight • Ensure flutter speed >> max vehicle velocity	• Aerodynamic testing with Florida Center for Advanced Aero Propulsion
(STR.3) Body tube zippering	• Shock cord contact with body on deployment	• Implement “bumpers” to avoid cord contact • Implement cord routing	• Assembly/Operations Checklist • Mentor input
(STR.4) Damaged motor retainer	• Defect in part • Excessive dynamic loading	• Aluminum motor retainer to absorb far larger loads than necessary	• Pre-flight simulations (FEA)
(STR.5) Bulkhead or U-bolt torn loose	• Excessive loading during chute deployment • Late chute deployment	• Extensive pre-flight simulation • Extra thick bolts and wide bracing on bulkheads	• Pre-flight hardware testing (jerk) • Pre-flight simulations (FEA)
(STR.6) Dislodged centering ring(s)	• Defect in part(s) • Excessive dynamic loading • Poor connection to threaded rods	• Fix centering rings to threaded rods with hex nuts • Use thread lock to fix nuts	• Pre-flight hardware testing (jerk) • Pre-flight simulations (FEA)
(STR.7) Damaged payload retainer	• Defect in part(s) • Poor 3D print • Excessive dynamic loading • Excessive ground impact velocity	• Extensive pre-flight testing • Minimize ground impact velocity • Cushion landing	• Material testing with High Performance Materials Institute • Pre-flight hardware testing (vibration and drop test)
(STR.8) Damaged avionics sled retainer(s)	• Defect in part(s) • Poor 3D print • Excessive dynamic loading • Excessive ground impact velocity	• Extensive pre-flight testing • Minimize ground impact velocity • Cushion landing	• Material testing with High Performance Materials Institute • Pre-flight hardware testing (vibration and drop test)

Payload			
Failure Mode	**Cause(s)**	**Mitigations**	**Verification Strategy**
(PLD.1) 3-D printed housing damaged	• High ground impact velocity • Defects in 3D print	• Extensive pre-flight testing • Minimize ground impact velocity • Cushion landing	• Pre-flight hardware testing (drop test)
(PLD.2) Payload lands in a manner where camera is not right side up	• Improper connection to shock cord • High lateral landing velocity	• Design connection points to maximize possibility of landing z-axis up • Implement anti-roll measures in the design	• Pre-flight hardware testing (drop test)
(PLD.3) Power loss	• Dead battery • Electrical lead disconnection	• Charge battery pre-flight • Firm electrical connections	• Assembly/Operations Checklist • Pre-flight hardware testing (vibration test)
(PLD.4) Antenna disconnection from microcontroller	• Excessive vibration in flight • Excessive ground impact velocity	• Firm electrical connections • Pad landing, reduce velocity	• Pre-flight hardware testing (vibration test)
(PLD.5) Microcontroller failure	• Bad component • Power loss	• Firm electrical connections • Reputable supplier	• Pre-flight hardware testing (vibration test)
(PLD.6) Camera failure	• Broken lens during ground impact • Power loss	• Padding or protection around camera assembly • Firm electrical connections	• Pre-flight hardware testing (drop test) • Pre-flight hardware testing (vibration test)
(PLD.7) Camera actuation system failure	• Motor failure • Obstructed gears • Power loss	• Firm electrical connections • Clean gear mechanism • Use self-contained stepper motor	• Pre-flight hardware testing (drop test) • Pre-flight hardware testing (vibration test)
Vehicle Risks to Environment			
Failure Mode	**Cause(s)**	**Hazard Category**	**Verification Strategy**
(ENV.1.1) Launch pad/recovery area fire (energetic initiated)	• Dry vegetation in vicinity of motor ignition	• Clear launch area of vegetation	• Team has no control over this. NASA has stated pad will be in plowed field (only topsoil)

(ENV.1.2) Launch pad/recovery area fire (LiPo battery initiated)	• Battery overcharge, over discharge, overtemp	• Clear launch area of vegetation • Do not use battery improperly	• Team has no control over this. NASA has stated pad will be in plowed field (only topsoil) • Assembly/Operations Checklist • Battery storage/charging rules implemented in fabrication shop
(ENV.1.3) Interstage insulation littered in launch/recovery area	• Insulation used in body tube to minimize void space and insulate parachutes from ejection charge gasses	• Biodegradable insulation (popcorn)	• Assembly/Operations Checklist • May not need to insulate interstage…subscale did fine without any
(ENV.1.4) Litter spread over launch site by personnel	• Lack of trashcans • Poor team leadership	• Bring trash bags • Firm leadership. Zero tolerance for littering	• Pre-launch week departure purchase and packing list
Environmental Risks to Vehicle			
Failure Mode	**Cause(s)**	**Mitigations**	**Verification Strategy**
(ENV.2.1) Vehicle touches down in nearby trees	• Excessive wind drift	• Extra-long shock cord to bring components closer to ground	• Assembly/Operations Checklist
(ENV.2.2) Vehicle touches down in nearby body of water	• Excessive wind drift	• Extensive sealing of avionics bay and camera system	• Assembly/Operations Checklist
(ENV.2.3) In-flight Collision	• Tall infrastructure (power lines) • Bird strike	• Ensure vehicle is launched away from all infrastructure • Await clear skies	• Assembly/Operations Checklist
(ENV.2.4) Vehicle or components dropped	• Uneven launch site terrain causes personnel tripping	• Practice extreme caution while handling vehicle components	• Assembly/Operations Checklist

Personnel			
Failure Mode	**Cause(s)**	**Mitigations**	**Verification Strategy**
(PPL.1) Skin contact with APCP solid propellant	• Improper material handling • Lack of PPE	• Provide safety training • Provide PPE	• Posted laboratory safety rules • Safety Officer supervision • PPE freely available in fabrication shop • PPE included in launch week packing list
(PPL.2) Electrocution	• Improper safety procedures followed • Live electrical while wiring	• Provide safety training	• Posted laboratory safety rules • Safety Officer supervision • PPE freely available in fabrication shop
(PPL.3) Proximity to high-pressure burst event (CO2 charge)	• Overpressure in pressure vessel • Pressure vessel tipping • Human error	• Provide safety training • Do not overfill pressure vessels • Pressure vessels chained to walls • Declare all testing and clear area prior to initiation	• Posted laboratory safety rules • Safety Officer supervision
(PPL.4) Proximity to explosive event (Black powder charge)	• Accidental initiation (human error, static discharge)	• Ground vehicle components • Minimize personnel handling charges • Isolate firing mechanism until range clear	• Posted laboratory safety rules • Safety Officer supervision
(PPL.5) Proximity to combustion event	• Motor ignition (intentional) • Motor ignition (unintentional) • Loose black powder burn	• Ground vehicle components • Minimize personnel handling motor • Isolate ignition mechanism until range clear	• Posted laboratory safety rules • Safety Officer supervision
(PPL.6) Injury: slip and fall, minor cuts, accidental collisions	• Uneven terrain • Tripping hazards on flat ground • Improperly stored sharp objects	• Situational awareness • Clean lab spaces • Proper safety procedures	• Posted laboratory safety rules • Safety Officer supervision • Fabrication shop manager appointed to

			ensure clean/safe environment
(PPL.7) Dehydration, heat exhaustion, heat stroke	• Lack of water • Lack of adequate sun protection or shade	• Provide ample water • Bring portable awning/tent • Bring sunscreen, hats, etc.	• Safety Officer supervision • Water, sunscreen, portable tents included in launch week packing list
(PPL.8) Soldering iron burns	• Improper use or stowage of soldering iron	• Proper training in use of soldering iron • Minimize personnel involved	• Safety Officer supervision • PPE (thick gloves) provided freely in fab shop • Training offered before use

6.1 Launch Operations Procedures

6.1.1 Operations Manual

The Zenith 1 Vehicle Operations Manual is a one of the core components of program documentation, serving as a full instruction manual written such that any person provided all the components listed in the program budget could manufacture and assemble all vehicle components into the major vehicle sections, systems, and payload. The manual includes troubleshooting and improvement steps for manufacture and assembly of the current vehicle and/or future iterations. Below is the document table of contents to illustrate items covered:

Contents

The manufacturing section of this report was developed using excepts from the operations manual. The following are relevant sections of the operations manual including procedures and methods not described in the manufacturing section of this report or assembly and launch operations checklists.

6.1.1.1 CO2 Ejection System Preparation

- **Step 1:** Run e-match through charge cup until the wire is e-match sits against the bottom of the cup with the wire exposed at the other end.
- **Step 2:** Apply a small amount of epoxy to the bottom of the charge cup so that the e-match does not move around.
- **Step 3:** Pour pre-measured amount of black powder into the charge cup and seal with cover sticker.
- **Step 4:** Use a cotton swab to apply silicon lube on the inside of the pyro housing.
- **Step 5:** Run the charge cup and e-match through the pyro housing up until the charge cup just barely passes the lip of the threaded side of the pyro housing.
- **Step 6:** Place the piston spring on top of the puncture piston.
- **Step 7:** Place the puncture piston on top of the charge cup sitting in the pyro housing (spring should still be on the piston)
- **Step 8:** Turn upside down and push down (spring first) onto a tabletop (or any other flat surface) until the charge cup and e-wire slides through the pyro housing and is slightly exposed on the unthreaded side *This step is crucial as it allows for there to be no air gap between the puncture piston and charge cup. If an air gap were to be present, the piston might not achieve enough force to penetrate the CO2 cartridge*
- **Step 9:** Screw on pyro housing to the bulkhead mount
- **Step 10:** Screw on CO2 cartridge to bulkhead mount
- **Step 11:** Place entire system through both AV Bay bulkheads (where the mount plate sits against the outer bulkhead) and use screws, bolts, and washers to fasten.

Video demonstration of the assembly process:

https://www.youtube.com/watch?v=ibEwm-nd0UU&t=81s&ab_channel=TinderRocketry

6.1.1.2 Parachute Packing Method

The drogue parachute being used for this flight vehicle is the Fruity Chutes 24” Classic Elliptical, and it will be stored in the lower payload bay. The main parachute being used for this flight vehicle is the Fruity Chutes 84” Iris Ultra Standard, and it will be stored in the upper payload bay along with the payload.

To properly pack the drogue parachute into the lower payload bay, first acquire 23 ft of recovery harness. Looking from left to right of the shock cord, at about 11.5 ft tie a loop-knot to allow for a quick link to attach to the drogue parachute. The far end of the harness that is closest to the payload should have a quick link that connects to the eye-bolt of the motor tube. The other end of the harness should have another quick link that attaches to the U-bolt on the outer bulkhead of the avionics bay. Fanfold any “loose” shock cord between each component for more space in the lower bay. Once this is done, the drogue parachute can slide right into the lower bay.

To properly pack the main parachute and payload into the upper payload bay, first acquire 23 ft of recovery harness. Looking from left to right of the shock cord, at about 3 ft tie a loop-knot to allow for a quick link to attach to four 1 ft long shock cords that are tethered to the four eyebolts on the payload. At another 10 ft down the harness tie another loop-knot for a quick link that will attach to the main parachute. The far end of the harness that is closest to the payload should have a quick link that connects to the U-bolt on the bulkhead that is underneath the nosecone. The other end of the harness should have another quick link that attaches to the U-bolt on the outer bulkhead of the avionics bay. Fanfold any “loose” shock cord between each component for more space in the upper bay. Once this is done, the payload and parachute can slide right into the upper bay. The following is steps on how to properly fold the drogue and main parachute (the images provided uses a 48” parachute, but the same method works on any size chute):

- **Step 1:** Detangle all shroud lines (as best as you can)
- **Step 2:** Lay out parachute on flat surface (the ground is fine)

- **Step 3:** Fold the parachute panels over each other (might have to repeat 2-3x depending on how big the parachute is)

- **Step 4:** Take the shroud lines and place them in the center of the folded chute (but leave some of the line extended to be able to connect with recovery harness quick link)

- **Step 5:** Fold/Roll up the parachute (lengthwise) into a burrito-like shape encasing all of the shroud lines

6.1.1.3 Launch Day Assembly Instructions

Avionics Bay Preparation

Charge Loading

Step 1: Execute the ejection charge loading steps from Section 4.3.3 for all charges.
Step 2: Verify that the two charges on each side are wired for the same ejection event. One side should include only the apogee event, the other the main parachute deployment. **COLOR CODING WIRES IS CRITICAL FOR THIS VERIFICATION**
Step 3: Turn on altimeter 1 and listen for the four status tones. Ensure no failure modes.
Step 4: Repeat for altimeter 2.

Fin Can Preparation

Motor Installation:

Propellant Grain Installation to Motor Tube

Defer to TRA/NAR certified mentor for handling/installation of the motor.

Motor Tube Installation

Step 1: Ensure propellant grains are packed in motor and the case's rear retaining ring (gold) is fixed in place.
Step 2: Remove thrust plate retaining ring (black) from retainer in tail cone.
Step 3: Orient the motor casing such that the eye bolt is at the top and nozzle is at the bottom. Slide the case, eye bolt first, into the motor tube.
Step 4: Replace the retaining ring on the thrust plate.

Lower Recovery Harness Installation

Step 1: Lay both recovery harnesses outstretched on the floor.
Step 2: Identify the harness with only three quick links, two at the ends and one in the center. This is the LOWER recovery harness.
Step 3: Connect the quick link on one end of the harness to the motor eye bolt, and the other to the avionics bay U-bolt.
[CRITICAL] ENSURE the lower harness is connected to the APOGEE charge side of the avionics bay!
Step 4: Connect the central quick link to the DROGUE parachute.
Step 5: Ladder fold loose recovery harness beginning at the motor eye bolt and ending at the drogue parachute connection. Use one loop of masking tape to connect.
Step 6: Pack the drogue parachute as outlined in Section 3.3.6. Stow folded recovery harness and packed chute in lower payload bay.
Step 7: Repeat the ladder fold and tape wrap on the remaining loose harness length. Stow in lower payload bay
Step 8: Slide the APOGEE side of the avionics bay into the lower payload bay

Upper Payload Bay Preparation

Upper Recovery Harness Installation

Step 1: Verify the remaining harness with has four quick links, two at the ends, one in the center, and the last 1/4 of the way from the end of the harness.
Step 2: The side with the quick link 1/4 of the way from the harness end is the TOP of the harness.
Step 3: Connect the quick link at the TOP end of the harness to the forward bulkhead U-bolt.
Step 4: Connect the central quick link to the MAIN parachute. The remaining quick link at 1/4 length from the end is the payload attachment point.
Step 5: Connect the bottom quick link to the MAIN CHUTE side of the avionics bay
[CRITICAL] ENSURE the upper harness is connected to the MAIN charge side of the avionics bay!

Payload to Vehicle Integration

Step 1: Take four pre-cut, 1’ lengths of #1500 shock cord.
Step 2: Tie a loop into both ends of each shock cord. Pass each chock cord through an eye bolt such that the midpoint of the cord sits at the eye. Pass one end of the cord through the loop on the other end and allow the tie off to pull tight on the eye.
Step 3: Connect the loop at the other end of the cord to the payload quick link on the recovery harness.
Step 4: Repeat for all eye bolts.
Step 5: Insert the payload and the shock cord into the upper payload bay.

Upper Payload Bay Completion:

Main Parachute Packing

Step 1: Execute steps outlined in Section 3.3.6 to pack the main parachute.
Step 2: Press packed parachute and shock cord into upper payload bay, pushing firmly against the installed payload.
Step 3: Ladder fold remaining loose harness between main chute and avionics bay. Wrap with single wrap of masking tape. Insert into upper payload bay.
Step 4: Slide the MAIN charge side of the avionics bay into the upper payload bay.

Shear Pins

The vehicle is now fully assembled, but the sections must be joined together before flight. This is achieved with nylon shear pins. Align the pre-drilled holes in the upper and lower payload bays with the pre-drilled holes in the avionics bay. Insert a shear pin into each and tighten.

Rail Installation

The vehicle must now be taken to the pad, where the launch rail and igniter system are waiting. Using the pre-installed rail buttons, slide the buttons onto the launch rail. The vehicle may sit atop the rail on the buttons or hang beneath the rail by the buttons.

**If at a test launch:* Adjust the launch rail angle to suit wind and design conditions

**If at the NASA competition range:* DO NOT modify the launch rail. If necessary, request a range safety officer

Once the vehicle is installed on the rail, defer to the TRA/NAR mentor for installation of the igniter system to the rocket motor. **Perform a continuity check after installation.**

6.2 Operations Checklists

The assembly and operation checklists are a chronological order of operations to be executed by the team during launch day assembly, pre-flight, and post-flight. For steps not including the VERIFY notice, any member of the team may participate and/or execute the action under the supervision of the associated technical lead (ex: recovery lead supervises connection of wire leads to ejection charge e-matches).

Items that include the VERIFY label constitute an all-stop until the Program Director and Safety Officer have both conferred with the associated technical lead, inspected the action described in the VERIFY notice, as well as all unverified work up to that point. Both the PD and SO must sign the checklist at each VERIFY notice, with approval of the relevant lead, before the team can resume working down the checklist. The purpose of the VERIFY notice is to halt progress at mission-critical actions that, if executed incorrectly, could or would result in failure to launch or failures during flight.

A table of justifications for each VERIFY notice will follow all checklists.

NOTE:

Checklists announce **Recommended PPE** and **REQUIRED PPE** before an action where PPE would enhance safety or is otherwise required.

While only several checklist steps recommend or require PPE, there are many steps which involve work with low voltage electrical. This is to serve as a blanket notice for all such steps in the interest of brevity on the checklist.

For ANY electrical work, the recommendation of the safety officer is the following:

When making electrical connections – **Recommended PPE: insulating nitrile gloves**

6.2.1 Vehicle Assembly Checklist

Vehicle Assembly Checklist					
Step	**Verification**	**Action**	**Complete**	**Program Director**	**Safety Officer**
		Avionics Bay			
1	**VERIFY**	Batteries are charged prior to assembly. Altimeters turn on when switches pressed			
2		Fasten avionics sled inside bay			
3		Route A channel wires and corresponding grounds to one side			
4		Route B channel wires and corresponding grounds to opposite side			
5		Screw CO2 cartridges into all charge bases on inside of bulkheads			
6		Replace avionics bulkhead and fasten			
7	**VERIFY**	Altimeters turn on when switches are pressed			
	For Today's Launch				
	Channel	**Color**			
	A				
	B				
	G				
		Ejection Charges			
8		Place O-ring around base of e-match			
9		**PPE Recommended: respirator** Place small amount of 5-minute epoxy on O-ring. Pass through charge cup until O-ring contacts bottom			
10		Wait 10 minutes for e-matches to become secured in place			
11		**While waiting:** prepare 4 pre-measured black powder loads			
12		**Once cured:** load black powder into charge cups and cover with red sticker			
13	**VERIFY**	E-matches secure. Charge cups loaded. Stickers firmly attached.			
14		Unscrew red housings from external AV bay. Place charge cups in red housing			

15		Place spring vertical on table. Place steel pointed plug **point side down** on top of screw			
16		Invert red housing w/ charge cup. Push down against plug until cup is seated at bottom of red housing and plug is directly above			
17	**VERIFY**	Charge cups properly seated and plugs flush with cup tops			
18		Keeping spring pressed against plug, screw red housing into base on outside of AV bay			
19		Connect A-channel and ground leads to e-match 1 & 2 leads			
20		Connect B-channel and ground leads to e-match 3 & 4 leads			
21	**VERIFY**	Each e-match is connected to an A/B channel AND a ground wire. **Reference wire color table above**			
22		Tape external wires to red charge housings, clear of recovery harness attachment bolts			
Upper PL Bay, Payload, and Main Chute					
23		Connect recovery harness end to foreword nosecone bulkhead			
24		Connect main parachute to midpoint quick link of recovery harness			
25	**VERIFY**	Payload battery is charged. Payload activates when switch is in "on" position			
26		Switch payload electronics to "on"			
27	**VERIFY**	Payload is on before installation			
28		Place payload in housing. Connect to PL quick link between foreword bulkhead and main chute			
29		Connect recovery harness end to avionics **B side** bulkhead			
30	**VERIFY**	Main parachute harness is connected to avionics **B side**			

31		Lay out horizontal: Nose section, payload, main chute, av bay			
32	**VERIFY**	All quick links are attached to the correct location and fully closed			
33		Insert payload into housing. Insert payload housing into upper payload bay.			
34		Pack main parachute			
35	**VERIFY**	Main parachute is correctly packed			
36		Insert packed main chute into upper payload bay. Reeve remaining shock cord			
37		Join upper payload bay and **B-side** of avionics bay			
38	**VERIFY**	B-channel ejection charges are inserted into upper payload bay			
39		Screw shear pins into pre-drilled holes to join UPLB and AV bay			
Motor Installation					
52		Unscrew motor retainer ring. Remove motor casing and motor tube from vehicle			
53		**PPE Recommended: nitrile gloves** Lightly grease aft, forward, and foreword seal disk O-rings			
54		Install forward seal disk O-ring onto foreword seal disk. Install foreword seal disk in motor case.			
55		Install foreword closure with threaded adapter to receive eye bolt			
56	**VERIFY**	Eye bolt adapter installed to motor case. Install eye bolt			
57		**PPE REQUIRED: nitrile gloves, safety goggles, long sleeves, pants, closed-toed shoes** Insert liner containing propellant grains into motor tube. Insert tube into motor casing			
58	**VERIFY**	No motor delay or ejection charge installed.			

59		Screw on aft closure. Replace motor retainer			
60	**VERIFY**	Motor retainer reinstalled. Nozzle cap fixed over nozzle.			
Lower PL Bay, Fin Can, and Drogue Chute					
40		Connect recovery harness end to eye bolt at top of motor case			
41		Connect drogue parachute to midpoint quick link of recovery harness			
42		Connect recovery harness end to avionics **A side** bulkhead			
43	**VERIFY**	Drogue parachute harness is connected to avionics **A side**			
44		Lay out horizontal: av bay, drogue chute, LPLB/fin can			
45	**VERIFY**	All quick links are attached to the correct location and fully closed			
46		Pack drogue parachute			
47	**VERIFY**	Drogue parachute is correctly packed			
48		Insert packed drogue chute into lower payload bay. Reeve remaining shock cord			
49		Join lower payload bay and **A-side** of avionics bay			
50	**VERIFY**	A-channel ejection charges are inserted into upper payload bay			
51		Screw shear pins into pre-drilled holes to join LPLB and AV bay			
Final Sign-Off					
61	**VERIFY**	All checklist steps completed. Vehicle prepared for pre-flight.			

6.2.2 Pre-flight Checklist

Pre-Flight Checklist					
Step	**Verification**	**Action**	**Complete**	**Program Director**	**Safety Officer**
1		Confirm launch group/time with NASA RSO			
2		Confirm launch pad with NASA RSO			
3	**VERIFY**	Cleared by RSO to approach pad			
4		Inspect launch rail cant. Note and refer to simulations			
5		Install vehicle on 1515 launch rail			
6		Switch on flight computers			
7	**VERIFY**	Flight computers both active			
8		Connect 12V launch leads to igniter leads on vehicle			
9		Continuity check			
10	**VERIFY**	Good continuity			
Final Sign-Off					
11	**VERIFY**	All checklist steps completed. Vehicle prepared for flight.			

6.2.3 Terminal Count and In-Flight Checklist

Terminal Count and In-Flight Checklist					
Step	**Verification**	**Action**	**Complete**	**Program Director**	**Safety Officer**
1		Ensure active communication with TeleMega flight computer			
2		Ensure avionics batteries sufficiently charged (live telemetry from TeleMega)			
3	**VERIFY**	Cleared for launch by RSO			
4		Begin terminal count			
Launch					
5		Avionics lead, using live telemetry, confirms apogee charges fire. **Callout: "Sep 1"**			
6		Team visually confirms drogue deployment. **Callout: "Good drogue"**			
7		Avionics lead confirms reduction in descent velocity from telemetry. **Callout: "Av Concurs"**			
8		Team maintains visual on vehicle during descent			
9		Avionics lead, using live telemetry, confirms 550ft charges fire. **Callout: "Sep 2"**			
10		Team visually confirms main deployment. **Callout: "Good main"**			
11		Avionics lead confirms reduction in descent velocity from telemetry. **Callout: "Av Concurs"**			
12		Team maintains visual on vehicle during descent			

13		Team visually confirms landing. **Callout: "Impact"**			
14		Avionics lead confirms zero descent velocity from telemetry. **Callout: "Av Concurs"**			
Final Sign-Off					
15	**VERIFY**	All checklist steps completed. Vehicle successfully recovered.			

6.2.4 Post-Flight Checklist

Post-Flight Checklist					
Step	**Verification**	**Action**	**Complete**	**Program Director**	**Safety Officer**
1	**VERIFY**	Cleared by RSO to approach pad			
2		Measure distance from pad to point of vehicle impact			
3		Disconnect recovery harnesses from both sides of avionics bay			
4	**VERIFY**	Avionics bay turned over to avionics lead			
5		Vehicle components returned to staging area. Lay upper and lower sections on table and inspect for damage			
6	**VERIFY**	Visual confirmation that all ejection charges fired before work on av bay begins			
7		Detach removable av bay bulkhead and remove sled			
8		Avionics team connects to flight computers and downloads data			
9	**VERIFY**	Data has been downloaded and saved before computer shutdown			
10		Flight computers shutdown			
Final Sign-Off					
11	**VERIFY**	All checklist steps completed			

6.2.5 Verify Notice Justifications

Table 6-19. Safety Verification(s) Rationale

Checklist	Step #	Step to Verify	Reason for Verification
Assembly	1	Batteries are charged prior to assembly. Altimeters turn on when switches pressed	Uncharged batteries could cause failure to launch or loss of power in flight
Assembly	7	Altimeters turn on when switches are pressed	Ensures connections were not disturbed after av bay is sealed
Assembly	13	E-matches secure. Charge cups loaded. Stickers firmly attached.	Ensures e-matches will not slip from cups and cups cannot spill out black powder. Either case can cause failed separation.
Assembly	17	Charge cups properly seated and plugs flush with cup tops	Improper cup and plug seating can fail to puncture CO2 cartridge, causing failed separation
Assembly	21	Each e-match is connected to an A/B channel AND a ground wire. **Reference wire color table above**	Ensures a complete circuit is connected to each e-match. Failure to connect ground would mean a dud charge and no separation
Assembly	25	Payload battery is charged. Payload activates when switch is in "on" position	Ensures payload will not experience power loss during mission and switch is functional
Assembly	27	Payload is on before installation	Ensures payload is powered up to execute RAFCO mission
Assembly	30	Main parachute harness is connected to avionics **B side**	B side is 550ft deployment. If connected to A side, main would deploy at apogee
Assembly	32	All quick links are attached to the correct location and fully closed	Ensures chute is in proper position and connected to sections to avoid entanglement
Assembly	35	Main parachute is correctly packed	Poor packing can cause entanglement
Assembly	38	B-channel ejection charges are inserted into upper payload bay	B side is 550ft deployment. If A side was inserted, upper payload bay would separate at apogee and deploy main chute
Assembly	44	Eye bolt adapter installed to motor case. Install eye bolt	Ensures non-stock foreword closure is used and connection point to fin can is present

Checklist	Step #	Step to Verify	Reason for Verification
Assembly	46	No motor delay or ejection charge installed.	Installation of ejection charge could cause motor jettison, or explosion in airframe if motor was retained after blast
Assembly	48	Motor retainer reinstalled. Nozzle cap fixed over nozzle.	Ensures motor cannot be jettisoned and nozzle will remain debris free
Assembly	52	Drogue parachute harness is connected to avionics **A side**	A side is apogee deployment. If connected to B, drogue would deploy at 550ft
Assembly	54	All quick links are attached to the correct location and fully closed	Ensures chute is in proper position and connected to sections to avoid entanglement
Assembly	56	Drogue parachute is correctly packed	Poor packing can cause entanglement
Assembly	59	A-channel ejection charges are inserted into upper payload bay	A side is apogee deployment. If B side was inserted, lower payload would separate at 550ft and deploy drogue chute
Pre-Flight	3	Cleared by RSO to approach pad	Ensures range is safe for team members to exit the viewing and staging areas
Pre-Flight	7	Flight computers both active	Failure to activate computers would result in failure to launch
Pre-Flight	10	Good continuity	Bad continuity in 12V igniter system would result in failure to launch
Terminal Count	3	Cleared for launch by RSO	Ensures pad and range are clear for flight
Post-Flight	1	Cleared by RSO to approach pad	Ensures pad and range are safe to approach
Post-Flight	4	Avionics bay turned over to avionics lead	Ensures hand-off of flight data to the team responsible for download and storage
Post-Flight	6	Visual confirmation that all ejection charges fired before work on av bay begins	Prevents possibility of team accidentally triggering charges that failed to fire
Post-Flight	9	Data has been downloaded and saved before computer shutdown	Data may be deleted if power to computer is cut

7 Project Plan

7.1 Testing

7.1.1 Testing Regime

Table 7-1. Vehicle Component and Assembly Tests

Test	Objective	Items of Interest	Success Criteria	Methodology
Drop	Determine impact resistance of vehicle sections	Breaks or cracks in vehicle or payload structures Electrical leads disconnecting	Fins do not crack Airframe does not crack or rupture Payload housing and clear camera shroud do not crack or break	Simple Newtonian physics calculation using known mass and desired impact velocity to determine drop height. Drop vehicle sections and/or payload. Inspect
Jerk	Verify connection points do not fail under large instantaneous load	Bending, warping, or cracking of bulkheads, hardware, or epoxied connections	U-bolts and eyebolts remain intact No deformation of bulkheads No dislodging of epoxied bulkhead to airframe connections No unintended breakage of shear pins	Connect recovery harness to U-bolt or eyebolt. Affix end of harness to test rig on second story of college. Drop individual vehicle sections, multiple tethered sections, or avionics pinned to upper or lower payload bay. Inspect airframe structures. Inspect shear pins.

Vibration	Ensure electrical and hardware connections are robust enough to withstand powered ascent phase	Severity of vibration or RPM of spin before connections begin to loosen	No electrical leads disconnect No nuts, bolts, etc begin to come undone	Affix vehicle section to LabView controlled electric motor. Spin section about z-axis to max RPM from simulations. Inspect hardware connections and av bay. Physically shake vehicle section (by hand) progressively harder and faster so as to well exceed the minor vibrations experienced on launch. Inspect hardware connections and av bay.
Separation	Validate that shear pins will break and sections will forcefully separate when ejection charges are fired	Grams in CO2 cartridge Shear pin size Distance vehicle section is ejected from avionics bay	Shear pins break Vehicle section is ejected from test rig Vehicle section travels far enough to deploy parachute from payload bay	Load ejection charges in av bay with long alligator leads to be manually attached to a battery. Join av bay to vehicle section with shear pins. Place joined sections onto “test sled” rig, ensuring the rear of the av bay is firm against the rear brace. Connect alligator clips to battery terminal to light charge. Inspect result.

7.1.2 Detailed Testing Methodologies and Results

7.1.2.1 Fin Drop Test

Method

1. Newtonian equations of motion solved for drop height to produce final velocity of 19 ft/s at impact.
2. Calculated value of 5.6 ft
3. Use shop ladder and grass patch outside shop for testing area.
4. Using 3 team members, have one ascend the ladder, another act as a spotter for the person on ladder, and the third to pass the fin can to the member on the ladder.
5. Member 3 collects a tape measure and extends to calculated drop height. Ensure end of tape is in contact with ground.
6. Align bottom of vehicle section with top of tape measure.
7. Orient vehicle section vertical or angled off vertical to produce desired ground strike orientation. For fin testing, desired impact is fin tip striking ground at 45 degree angle to test fin resistance to shear.
8. Drop vehicle section. Inspect for damage.
9. Repeat in varying configurations if desired or necessary.

Result

No fin breakage observed after impact with grass. Single run executed.

Interpretation

Fins will survive ground impact during flight test.

7.1.2.2 Tail Cone Drop Test

Method

1. Repeat steps of above test with exception of step seven.
2. Desired impact orientations for tail cone testing are vertical and 45 degrees.
3. Execute steps 8 and 9 of drop test described above

Result

Tail cone survived ground impact intact. 2 runs executed at vertical and 45 deg impact.

Interpretation

Tail cone will survive ground impact during flight test.

7.1.2.3 Shear Pin Number and Sizing – UPLB Jerk Test

Method

1. Load dummy payload and main parachute into upper payload bay, connect shock cord to avionics bay, insert avionics bay into UPLB. Install shear pins in flight configuration (7 pins)
2. Proceed to third floor of College of Engineering in A Building exterior stairwells.
3. Loop second shock cord around upper railing and pass through end loop of shock cord, creating an attachment to the railing.
4. Connect loose end to exposed end of avionics bay.
5. Use 4 team members to secure area. Post one at stairwell door of each level and one in the parking lot well clear of the drop zone. Ensure area is free of pedestrian and vehicle traffic before preparing and executing following steps.
6. Request "CLEAR BELOW?" and await response "CLEAR" from all four safety spotters.
7. Two team members carefully lift pinned vehicle sections over railing, ensuring all shock cord hangs to the outside of the railing and is not entangled with any personnel.
8. Load is transferred to single team member, who holds vehicle by the shock cord just above the avionics bay U-bolt. Vehicle sections are held with outstretched arms clear of railing.
9. Lower the load gently until the shock cord is fully extended. Ensure vehicle body is not against railings or concrete landings of lower levels. The vehicle should hand such that the section will swing into a lower landing when the shock cord is fully extended post-drop.
10. Resize initial attachment loop if necessary. Repeat step 8-10 if load is brought back over railing.
11. Call "LANDING CLEAR?" and await "LANDING [Floor #] CLEAR" response from spotter on landing where the vehicle will swing in.
12. Drop the pinned vehicle sections and assess if shear pins withstand jerk.
13. Hoist sections back up, over railing, and onto landing. Call "CLEAR ABOVE" and await "SAFE" response from spotters. Spotters discontinue blocking doors and drop area.

Result

Shear pins broke during jerk. Vehicle bypassed second floor landing and upper recovery harness let out main chute and dummy payload. Bottom edge of upper payload bay contacted cement floor of second floor landing as vehicle section swung into first floor landing. No damage caused to vehicle.

Interpretation

Load calculations suggest the pins would not break under force of drogue parachute deployment. The testing method over exaggerates the simulated deployment force as the shock cord is connected to a fixed object, rather than a parachute dragging through air. Test was called a success in that all components survived exaggerated loads without failure, and the decision to trust that the load calculations were accurate and that pins would not shear at deployment was made.

7.1.2.4 Separation Test

Method

1. Prime single ejection charge and route leads through bulkhead into the bay and out the opposite bulkhead.
2. Load dummy payload and main parachute into upper payload bay, connect shock cord to avionics bay, insert avionics bay into UPLB. Install shear pins in flight configuration (7 pins).
3. Ensure primed charge is facing into attached UPLB and initiator wires are accessible through exposed bulkhead.
4. Assemble testing rig: collect the two scrap 6” diameter blue tube airframe sections and use masking tape to connect side by side at one end. Lay assembly horizontally in grass outside of shop. Using 4 cinder blocks and the plywood sheet in the shop, create a blast shield by propping the sheet up with the blocks. Place the tube assembly up against one side of the barrier.
5. Place the joined UPLB and avionics bay atop the tube assembly in the groove created by the union of the two tubes. Attach long wires to the e-charge leads with alligators clips from the avionics dept. Slide the exposed bulkhead of the avionics bay back so that it contacts the plywood barrier. Run alligator leads over or around barrier.
6. Clear area for 100 feet in front of the direction of fire. Use 2-4 spotters at sufficient distance and displacement off line of fire to maintain control of the safe area. Remove all non-essential personnel from cordon area and relocate to 25 feet to the rear of the plywood blast shield.
7. Allow a single team member to approach the rear of the blast shield. Be seated with back into the barrier and head down. Ensure no body parts extend past the sides or over the top of the shield. Using a 9V battery, attach one alligator clip to the battery. Call “CORDON CLEAR?” and await “CLEAR” response from spotters.
8. Deliver an audible 3-count and attach the second lead. Brace.
9. Inspect for good separation and full deployment of at least the main parachute from the UPLB. Preferable result is full shock cord extension and deployment of parachute and dummy payload.

Result

Sections separated with sufficient velocity to fully extend shock cord, deploying main parachute and dummy payload from the UPLB.

Interpretation

Parachute will easily clear payload bay at charge firing. Force of ejection alone is enough to extend shock cord and remove payload from bay, making the parachute pulling the shock cord a redundant removal mechanism.

7.2 Requirements Compliance

7.2.1 Team Derived Requirements

Table 7-2. Team Derived Requirements

Requirement	Verification	Implementation
Safety and Operations		
All team members complete safety training by attending or reviewing safety officer's presentation	Review completion records	Require completion prior to work
Wear safety glasses at all times in workshop	Visual check by safety officer	Require glasses and provide them
Wear closed-toe shoes in workshop	Visual check by safety officer	Remind team members to wear closed-toe shoes
Wear hearing protection when using power tools	Visual check by safety officer	Provide hearing protection and remind team members to use it
Regularly inspect and maintain power tools	Visual inspection and maintenance log	Assign team member for inspections and maintenance
Wear appropriate PPE when working with tools, pyrotechnics, or rocket motors	Visual check by safety officer	Provide appropriate PPE and remind team members to use it
All personnel must conduct themselves in accordance with best safety practices while handling solid motors	Safety training and compliance inspection	Ensure that all team members receive training on the proper handling, storage, transportation, and disposal of rocket motors and propellants, and that the team complies with all relevant safety regulations and guidelines for high power solid rocketry. Team members should always wear personal protective equipment (PPE), including eye protection and ear protection, when handling rocket motors and propellants.

All personnel must conduct themselves in accordance with best safety practices while using power tools in the fabrication shop	Safety training and compliance inspection	Ensure that all team members receive training on the proper use of power tools, including saws, drills, and sanders, and that the team complies with all relevant safety regulations and guidelines for the use of power tools in a workshop setting. Team members should always wear PPE, including eye protection, ear protection, and gloves, when using power tools.
Any team member who is unsure of how to operate a power tool or machinery in the rocket fabrication workshop must receive proper training from a qualified team member or supervisor before use	Verification by fabrication shop manager or safety officer	Establish a clear protocol for training and verify that it is followed for all team members
Any team members working directly with high-power solid rocket motors must complete Level 2 High-Power Rocket Certification through the National Association of Rocketry (NAR) or equivalent certification program	Verification of completed certification through documentation	Ensure all team members who will work with high-power solid rocket motors complete the certification requirement prior to beginning work on the rocket
Launch site operations must be FAA compliant and abide by NAR HPR Safety Code	Compliance Inspection	Ensure that the launch site complies with all relevant safety regulations and guidelines, including those related to launch site distance and safety clearances. A safety officer should be designated and present on-site during the entire launch operation.

Launch Vehicle		
The launch vehicle shall not exceed 16Gs of acceleration during ascent	Pre-flight Simulation	Simulations are done in OpenRocket to verify acceleration
The launch vehicle shall have symmetrical fins	Visual Inspection by structures team lead	The launch vehicle has four fins equally spaced from each other
The lower payload bay shall have at least 6 inches of interior length	Physical Measurement Assembly checklist	The lower payload bay is designed to have 6 inches of interior length
The airframe shall be capable of launching in temperatures between 20- and 100-degrees Fahrenheit	Pre-flight Testing	The airframe material selected must be rated for high operating temperatures
The launch vehicle shall not go above Mach 0.7	Pre-flight Simulation	Simulations are done in OpenRocket to confirm the launch vehicles maximum velocity
The launch vehicle shall use at least 2 centering rings to support the motor tube	Visual inspection by structures team lead Assembly checklist	Vehicle design incorporates two centering rings. Fabrication shop manager and structures lead ensure proper installation
The launch vehicle shall have a stability margin between 2.5 and 3.5 calibers	Pre-flight Simulation	Simulations are done in OpenRocket to confirm the launch vehicle’s stability across the flight profile

Avionics		
Fully charged batteries shall be used for the altimeters before every flight	Operations checklist	New batteries will be chosen and verified to be full before being placed on the AV sled
U-bolts shall be used for all shock cord connections	Assembly Checklist	U-bolts are installed on the bulkheads as anchor points for the recovery harness
All electronic components in the launch vehicle shall be removable.	Pre-flight testing Assembly checklist	None of the electronic components in the launch vehicle are permanently fixed in place
There shall be no more than 4 sections of the vehicle recovered	Pre-flight simulation Pre-flight testing Assembly checklist	The vehicle will be designed to have only 4 sections
The secondary ejection charges shall be based off a configured time set on the redundant altimeter	Pre-flight testing Sub-scale flight result verification	Both altimeters are completely independent of each other
Payload		
The payload vehicle shall have a diameter of less than 4.5 inches.	Physical measurement Assembly checklist	The Payload fits inside of its housing.
The payload shall be supported within the launch vehicle	Pre-flight testing	The payload housing supports the payload in the airframe so that it is not dislodged before deployment
The payload integration system shall be a maximum of 7 inches long	Visual inspection Physical measurement	Payload lead ensures design meets dimension requirement. Fabrication shop physically measures final component.

7.2.2 NASA Derived Requirements

Table 7-3. NASA Derived Requirements

Requirement ID	Requirement	Verification	Implementation
2.1.	Deliver payload to apogee altitude of 4,000-6,000 ft AGL	Pre-flight simulation and inspection of flight data	Design rocket to reach target altitude range
2.2.	Teams declare target altitude goal at PDR milestone	Review PDR documentation	Communicate requirement to team members and set clear deadline for declaration
2.3.	Rocket designed to be recoverable and reusable	Conduct test launches to verify recoverability and reusability	Incorporate design features that promote recoverability and reusability
2.4.	Rocket has a maximum of 4 independent sections	Design stage verification	Conduct regular checks during design and build process to ensure compliance
2.4.1.	Coupler/airframe shoulders at in-flight separation points ≥ 2 airframe diameters	Measure shoulder length during design and build process	Incorporate design features that ensure shoulders meet length requirement
2.4.2.	Nosecone shoulders at in-flight separation points ≥ ½ body diameter	Design stage verification Measure shoulder length during design and build process	Incorporate design features that ensure shoulders meet length requirement
2.5	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.	Time the assembly procedure to prepare for flight during test launches	Develop a launch preparation plan that includes all necessary tasks and timelines to prepare the launch vehicle for flight within 2 hours of the FAA flight waiver opening.
2.6	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	Functionality test after 2 hours on the pad	Conduct a thorough analysis of all critical on-board components to identify potential failure points and develop mitigation strategies to prevent failures

	on-board components, although the capability to withstand longer delays is highly encouraged.		during extended periods of readiness.
2.7	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.	Compatibility test with the firing system	Execute test launches on a 12V DC igniter system
2.8	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).	Design stage verification	Initiate test launch motor ignitions solely with a 12V igniter system
2.9	Each team shall use commercially available ematches or igniters. Hand-dipped igniters shall not be permitted.	Verification of ematch/igniter source.	Order from commercial material provider
2.10	The launch vehicle will use a commercially available solid motor propulsion system using ammonium perchlorate composite propellant (APCP) which is approved and certified by the National Association of Rocketry (NAR), Tripoli Rocketry Association (TRA), and/or the Canadian Association of Rocketry (CAR).	Review documentation of the selected propulsion system	Review documentation of the selected propulsion system from the team and verify that it meets the requirements and that it is certified by NAR, TRA, and/or CAR for use with APCP.
2.10.1	Final motor choices will be declared by the Critical Design Review (CDR) milestone.	Review team documentation	Review documentation from the team indicating the date of their CDR milestone and the final motor choices declared at that time.
2.10.2	Any motor change after CDR shall be approved by	Review team documentation	Review documentation from the team indicating that any

	the NASA Range Safety Officer (RSO). Changes for the sole purpose of altitude adjustment will not be approved. A penalty against the team's overall score will be incurred when a motor change is made after the CDR milestone, regardless of the reason.		motor changes made after the CDR milestone have been approved by the NASA RSO. Ensure written approval for clarity.
2.11	The launch vehicle will be limited to a single motor propulsion system.	Design stage verification	Design the launch vehicle to ensure that only one motor propulsion system is used.
2.12	The total impulse provided by a College or University launch vehicle will not exceed 5,120 Newton seconds (L-class).	Verify motor documentation	Review documentation from the manufacturer regarding the total impulse of the launch vehicle and verify that it does not exceed 5,120 Newton-seconds.
2.14	The launch vehicle will have a minimum static stability margin of 2.0 at the point of rail exit. Rail exit is defined at the point where the forward rail button loses contact with the rail.	Pre-flight simulation	Analyze the rocket's design using an OpenRocket model to determine its static stability margin and ensure that it meets the requirement.
2.15	The launch vehicle will have a minimum thrust to weight ratio of 5.0 : 1.0.	Physical measurement of weight. Review motor documentation Pre-flight simulation	Determine the thrust of the rocket's motor from manufacturer and weigh the rocket to determine its weight. Then, divide the thrust by the weight to obtain the thrust to weight ratio and ensure that it meets the requirement.
2.16	Any structural protuberance on the rocket will be located aft of the burnout center of gravity. Camera housings will be exempted, provided the team can show that the housing(s) causes minimal	Design stage verification	Review the rocket's design and inspect it to ensure that any protruding structures are located aft of the burnout center of gravity. Analyze the effect of camera housings on the rocket's stability an OpenRocket model and ensure that they

	aerodynamic effect on the rocket's stability.		do not cause significant aerodynamic effect.
2.17	The launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit.	Pre-flight simulation Post-flight data analysis	Analyze the rocket's design using an OpenRocket model to determine rail exit velocity ensure that it meets the requirement. Cross check with flight data
2.18	All teams will successfully launch and recover a subscale model of their rocket prior to CDR.	NASA review panel approval of the subscale flight results	Review and approval of subscale flight data by NASA review panel.
2.18.1	The subscale model should resemble and perform as similarly as possible to the full-scale model; however, the full-scale will not be used as the subscale model.	Design stage verification	Compare designs and performances of subscale and full-scale models.
2.18.2	The subscale model will carry an altimeter capable of recording the model's apogee altitude.	Visual inspection Post-flight data review	Check the presence and functionality of the altimeter in the subscale model. Check the recorded apogee altitude of the subscale model during flight.
2.18.3.	The subscale rocket shall be a newly constructed rocket, designed and built specifically for this year's project.	Review team's construction documentation	Verify construction date and previous use
2.18.4.	Proof of a successful flight shall be supplied in the CDR report.	Review CDR report	Check for evidence of successful flight
2.18.4.1.	Altimeter flight profile graph(s) OR a quality video showing successful launch, recovery events, and landing as deemed by the NASA management panel are acceptable methods of proof. Altimeter flight profile graph(s) that are not complete (liftoff through	Review team's proof of flight	Verify complete profile or video of successful launch, recovery, and landing

	landing) shall not be accepted.		
2.18.4.2.	Quality pictures of the as landed configuration of all sections of the launch vehicle shall be included in the CDR report. This includes but not limited to nosecone, recovery system, airframe, and booster.	Review team's pictures in the CDR report	Verify pictures of all sections of the landed vehicle
2.18.5.	The subscale rocket shall not exceed 75% of the dimensions (length and diameter) of your designed full-scale rocket	Review team's documentation and measurements	Verify that the subscale rocket does not exceed 75% of the dimensions of the full-scale rocket
2.21.	The team's name and Launch Day contact information shall be in or on the rocket airframe.	Visual inspection	Check if the team's name and Launch Day contact information are clearly visible on the rocket airframe.
2.22.	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.	Visual inspection	Check if all Lithium Polymer batteries are adequately protected, brightly colored, clearly marked as a fire hazard, and easily distinguishable from other payload hardware.
2.23.1	The launch vehicle will not utilize forward firing motors.	Inspection of motor orientation in the launch vehicle.	Check the orientation of all motors in the launch vehicle to ensure none of them are forward firing.
2.23.2	The launch vehicle will not utilize motors that expel titanium sponges.	Inspection of the motor specifications and design.	Verify that the motor specifications and design do not include the use of titanium sponges.
2.23.3	The launch vehicle will not utilize hybrid motors.	Inspection of the motor specifications and design.	Verify that the motor specifications and design do not include the use of hybrid motors.
2.23.4	The launch vehicle will not utilize a cluster of motors.	Inspection of the motor configuration in the launch vehicle.	Design a vehicle using a single motor and motor case

2.23.5	The launch vehicle will not utilize friction fitting for motors.	Inspection of the motor attachment method in the launch vehicle.	Implement a motor retaining ring at the base of the vehicle to positively retain motor case
2.23.6	The launch vehicle will not exceed Mach 1 at any point during flight.	OpenRocket simulation of the flight Verification by altimeter data	Analyze flight simulation data to ensure that the launch vehicle will not exceed Mach 1 at any point during flight. Cross check with actual flight results
2.23.7.	Vehicle ballast will not exceed 10% of the total unballasted weight of the rocket as it would sit on the pad.	Measure the rocket and ballast weight	Use a scale to weigh the rocket and ballast and verify that the total weight of ballast does not exceed 10% of the total unballasted weight.
2.23.8.	Transmissions from onboard transmitters, which are active at any point prior to landing, will not exceed 250 mW of power (per transmitter).	Measure power output of transmitters	Use a power meter to measure the power output of each transmitter.
2.23.9.	Transmitters will not create excessive interference. Teams will utilize unique frequencies, handshake/passcode systems, or other means to mitigate interference caused to or received from other teams.	Inspection of transmission bands of selected antennas	Use a spectrum analyzer to monitor the frequency spectrum during the flight and ensure there is no excessive interference.
2.23.10	Excessive and/or dense metal will not be utilized in the construction of the vehicle	Design stage verification	Review the design and construction materials of the rocket to ensure they do not contain excessive or dense metal.
3.1	The full-scale launch vehicle will stage the deployment of its recovery devices	Visual inspection	Observe the rocket during launch to ensure the recovery devices are deployed as specified.
3.1 (cont.)	A drogue parachute is deployed at apogee, and a main parachute is deployed at a lower altitude.	Visual inspection Post-flight data analysis	Observe the rocket during launch and ensure the drogue and main parachutes are deployed at

			the specified altitudes with flight data
3.1.1	The main parachute shall be deployed no lower than 500 feet.	Post-flight data analysis	Review flight data of subscale and full scale demonstration flights to ensure proper parachute deployment heights
3.1.2	The apogee event may contain a delay of no more than 2 seconds.	Design stage verification Physical measurement Post-flight data review	Design for >2s apogee event delay. Measure the delay time between apogee and drogue parachute deployment and ensure it is no more than 2 seconds. Cross check with flight data.
3.1.3	Motor ejection is not a permissible form of primary or secondary deployment.	Design stage verification	Review the rocket design and ensure that motor ejection is not used as a primary or secondary form of deployment.
3.2	Each team will perform a successful ground ejection test for all electronically initiated recovery events prior to the initial flights of the subscale and full scale vehicles.	Visual inspection Pre-flight testing	Observe the ground ejection test and ensure that all electronically initiated recovery events are successful.
3.3	Each independent section of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf at landing.	Pre-flight simulation	Calculate the kinetic energy for each independent section of the rocket and ensure that it does not exceed 75 ft-lbf at landing.
3.4	The recovery system will contain redundant, commercially available barometric altimeters that are specifically designed for initiation of rocketry recovery events. The term "altimeters" includes both simple altimeters and more sophisticated flight computers.	Design stage verification Visual inspection	Design for redundant avionics system. Inspect the recovery system to confirm the presence of redundant barometric altimeters that are specifically designed for rocketry recovery events. Verify the commercial origin of the altimeters
3.5	Each altimeter will have a dedicated power supply,	Design stage verification Visual inspection	Inspect the recovery system to confirm that each

	and all recovery electronics will be powered by commercially available batteries.		altimeter has a dedicated power supply and that all recovery electronics are powered by commercially available batteries.
3.6	Each altimeter will be armed by a dedicated mechanical arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the pad.	Function test	Test the accessibility and operation of the dedicated mechanical arming switch for each altimeter when the rocket is in the launch configuration on the launch pad.
3.7	Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	Visual inspection and functional testing.	Check that each arming switch can be locked in the ON position and perform a functional test to ensure that the switch remains in the ON position during simulated flight forces.
3.8	The recovery system, GPS and altimeters, electrical circuits will be completely independent of any payload electrical circuits.	Electrical continuity test	Use a multimeter or continuity tester to check for continuity between the recovery system, GPS and altimeters, and the payload electrical circuits. Ensure that there is no electrical connection between these circuits.
3.9	Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.	Visual inspection and functional testing.	Check that each compartment uses removable shear pins, and perform a functional test to ensure that the pins can be removed without damage.
3.10	The recovery area will be limited to a 2,500 ft. radius from the launch pads.	Physical measurement	Use a GPS device to measure the distance from the launch pad to the farthest point within the designated recovery area. Ensure that the distance does not exceed 2,500 ft.

3.11	Descent time of the launch vehicle will be limited to 90 seconds (apogee to touch down).	Physical measurement	Measure descent time using a clock or timer during the demonstration flight.
3.12	An electronic GPS 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.	Visual inspection	Check that GPS tracking device is installed in launch vehicle and verify that it is transmitting position data to a ground receiver.
3.12.1	Any rocket section or payload component, which lands untethered to the launch vehicle, will contain an active electronic GPS tracking device.	Visual inspection	Inspect each rocket section and payload component to confirm the presence of an electronic GPS tracking device.
3.12.2	The electronic GPS tracking device(s) will be fully functional during the official competition launch.	Pre-flight testing	Test the electronic GPS tracking device(s) prior to launch to ensure that they are fully functional.
3.13	The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight.	Pre-flight testing	Test the recovery system electronics with other on-board electronic devices to confirm that they are not adversely affected during flight.
3.13.1	The recovery system altimeters will be physically located in a separate compartment within the vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.	Pre-flight inspection	Inspect the rocket airframe to confirm that the recovery system altimeters are located in a separate compartment from other electronic devices that produce radio frequency and/or magnetic waves.
3.13.2	The recovery system electronics will be shielded from all onboard transmitting devices to avoid inadvertent excitation of the recovery system electronics.	Measurement and testing	Use an electromagnetic interference meter to measure the electromagnetic emissions of all transmitting devices onboard and ensure that they are below a certain threshold.

3.13.3	The recovery system electronics will be shielded from all onboard devices which may generate magnetic waves (such as generators, solenoid valves, and Tesla coils) to avoid inadvertent excitation of the recovery system.	Measurement and testing	Use a magnetic field meter to measure the magnetic fields generated by all devices that may generate magnetic waves onboard and ensure that they are below a certain threshold.
3.13.4	The recovery system electronics will be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	Pre-flight testing and review of flight data	Review the specifications and manuals of all onboard devices to identify any that may adversely affect the recovery system electronics and ensure they are physically located in a separate compartment from the recovery system electronics or develop proper shielding between components in same compartment. Review flight data for abnormalities indicating interference.
4.1	Teams shall design a payload capable upon landing of autonomously receiving RF commands and performing a series of tasks with an on-board camera system.	Functionality test	The team shall demonstrate the ability of the payload to receive RF commands and perform the required tasks.
4.2.1	Launch Vehicle shall contain an automated camera system capable of swiveling 360° to take images of the entire surrounding area of the launch vehicle.	Functionality test	The team shall demonstrate that the camera system is capable of swiveling 360° and taking images of the surrounding area.
4.2.1.1	The camera shall have the capability of rotating about the z axis. The z axis is perpendicular to the ground plane with the sky oriented up and the	Functionality test	The team shall demonstrate that the camera can rotate about the z-axis as specified.

	planetary surface oriented down.		
4.2.1.2	The camera shall have a FOV of at least 100° and a maximum FOV of 180°.	Measurement test	The team shall measure the FOV of the camera and verify that it meets the specified requirements.
4.2.1.3	The camera shall time stamp each photo taken. The time stamp shall be visible on all photos submitted to NASA in the PLAR.	Visual inspection	Review the submitted photos and check if the time stamp is visible on each photo.
4.2.1.4	The camera system shall execute the string of transmitted commands quickly, with a maximum of 30 seconds between photos taken.	Testing using a timer	Observe the payload demo launch and use a timer to measure the time between each photo taken by the camera system.
4.2.3.1	NASA will operate on the 2-Meter amateur radio band between the frequencies of 144.90 MHz and 145.10 MHz.	Monitor outgoing transmissions and frequencies	Verify that the team is not transmitting on any frequency in the range of 144.90 MHz to 145.10 MHz during the competition launch by monitoring radio frequency usage.
4.2.3.3	The payload system shall not initiate and begin accepting RAFCO until AFTER the launch vehicle has landed on the planetary surface	Review of flight data.	Verify that the payload system is not initiating and accepting RAFCO until the launch vehicle has landed on the planetary surface by reviewing flight data.
4.2.4	The payload shall not be jettisoned.	Review video of footage and flight data.	Verify that the payload is not jettisoned during the launch and landing by reviewing video footage and flight data.
4.2.5	The sequence of time-stamped photos taken need not be transmitted back to ground station and shall be presented in the correct order in your PLAR.	Review of PLAR submission	NASA personnel will review the PLAR submission to ensure that the time-stamped photos are presented in the correct order.
4.3.1	Black Powder and/or similar energetics are only	Inspection of payload	NASA personnel will inspect the payload to ensure that

	permitted for deployment of in-flight recovery systems. Energetics shall not be permitted for any surface operations.		no black powder or similar energetics are present and will monitor the payload during surface operations to ensure compliance.
4.3.2	Teams shall abide by all FAA and NAR rules and regulations.	NASA Review of team documentation	NASA personnel will review team documentation to ensure compliance with FAA and NAR rules and regulations. Any questions or concerns will be addressed with the team prior to the competition.

7.3 Budgeting and Funding Summary

7.3.1 Budget

The following table summarizes the remaining project expenses for the Payload Demonstration Flight and travel to Huntsville for launch week.

Table 7-4. Updated Project Budget Summary (Rev 3/2023)

Project Component	Expected Cost
Transportation (SRA Payload Demo)	$100.00
Transportation and Lodging (Huntsville)	$1,800.00
Remaining Project Cost:	**$1,900.00**

7.3.2 Funding and Material Acquisition

7.3.2.1 Funding Sources

Funding for this year's project team is being graciously provided through two sources: the Aero-Propulsion, Mechatronics, and Energy (AME) center has diverted $2,000 of their NASA MUREP Grant funding to Zenith Program to facilitate Florida A&M student involvement with a complex aerospace project, and to facilitate FAMU student relations with NASA at large, in hopes of creating a firm feeder pipeline of underrepresented minority students to NASA. This MUREP funding has been expended to cover a portion of the subscale vehicle. The FAMU-FSU COE Mechanical Engineering Department has generously agreed to cover the difference in project costs for parts and material, transportation to the test launch facility, and transportation and lodging for Launch Week. At this point in the project the ME Department has taken on material acquisition after the AME funding was expended.

7.3.2.2 Funding Allocation

Using the budget sheet, now excluding all purchased components, a determination of project cost vs. time (or milestones) was made. These expenditures to achieve each milestone will be used to coordinate with the project sponsors for funding disbursement and to monitor the whether the project is on-budget when reaching each milestone.

Table 7-5. Updated Expenditure by Flight Milestone

Expenditure by Milestone		
Sub-scale	$425.47	
Recovery	$1,487.99	
Avionics	$362.25	
General Mats.	$323.52	
Logistics/Transport	$293.70	
Propellant	$205.98	**Subscale Test Flight (PDR)**
Initial:	**$3,098.91**	**$3,098.91**
Full-scale	$1,264.75	
Propellant	$289.99	
Logistics/Transport	$293.70	**Fullscale Test Flight (FRR)**
Add:	**$1,848.44**	**$4,947.35**
Payload	$158.56	
Propellant	$289.99	
Logistics/Transport	$293.70	**Payload Demo Flight**
Add:	**$742.25**	**$5,689.60**
Propellant	$289.99	
Transport/Lodging	$1,911.55	**Competition Flight**
Add:	**$2,201.54**	**$7,891.14**

7.3.2.3 Full Budget Spending Distribution by System

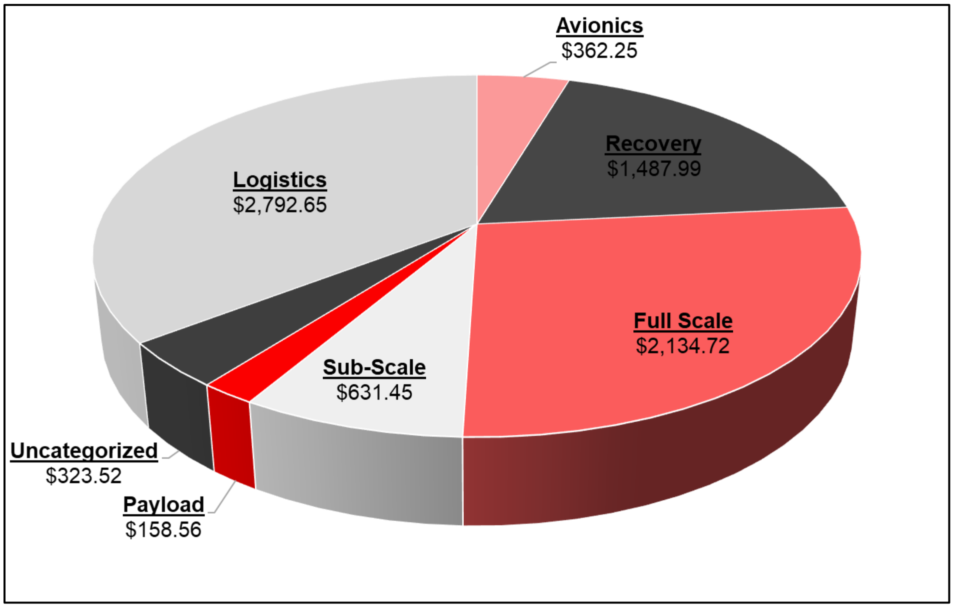


Figure 7-1. Expense Distribution

7.3.2.4 Material Acquisition Plan

Material acquisition has been streamlined with the initial investment by the AME Center expended and the full and sole support of the Mechanical Engineering department. The Program Director was given a point of contact in the office of the ME Department Chair to relay all material requests. The department then either places online orders with expedited shipping using Florida State P-Cards (university issued credit cards) or coordinates with local hardware vendors who have existing accounts with the University.

Appendix A. Program Budget

2022/23 Zenith Program Budget			
Part Name	**Quantity**	**Cost**	**Quantity Cost**
Avionics System			
Aim-4 Altimeter	1	$121.15	$121.15
900 mAh LiPo Battery	4	$12.12	$48.48
SMA-BNC Connector	2	$9.99	$19.98
Push button kill switches	2	$9.53	$19.06
Eggfinder TX Transmitter (GPS)	1	$75.00	$75.00
Eggfinder RX Reicever (GPS)	1	$25.00	$25.00
12V DC brushed motor	3	$17.86	$53.58
		Subtotal:	**$362.25**
Recovery System			
Tinder Rocketry Raptor CO2 Ejection System	4	$145.00	$580.00
1500lb rated Kevlar Shock Cord ($/ft)	30	$1.30	$39.00
9/16 in. 3000lb rated Nylon webbed Shock Cord (16 ft)	2	$14.95	$29.90
Cartridge Adapters	4	$7.99	$31.96
Firewire Electric Match	40	$0.75	$30.00
AeroTech Mini Initiators	2	$15.49	$30.98
CO2 Canister - 25g - 50pk	1	$39.99	$39.99
24" Fruity Chutes: Drogue Chute	1	$75.33	$75.33
84" Fruity Chutes: Iris Ultra Main Chute	1	$367.01	$367.01
4-40 Nylon Shear Pins - 20 ct.	1	$6.37	$6.37
2-56 Nylon Shear Pins - 20 ct.	1	$3.00	$3.00
15" Fruity Chutes: Drogue Chute	1	$63.41	$63.41
48" Fruity Chutes: Class Eliptical Main Chute	1	$149.99	$149.99
Fire Retardant Blanket 3"x3" for 3" AF	2	$4.95	$9.90
Fire Retardant Blanket 18"x18" for 6" AF	2	$11.60	$23.20
2-56 Drill and Tap Set (for shear pins)	1	$7.95	$7.95
		Subtotal:	**$1,487.99**
Full-Scale Vehicle			
ID = 6.0", L = 48" Airframe - Blue Tube 2.0	2	$77.42	$154.84
6" to 75mm Centering Ring - Baltic Birch	3	$9.50	$28.50
6" Airframe Bulkhead	6	$8.95	$53.70
OD = 6", L = 12" Avionics Bay w/ Hardware	1	$72.00	$72.00
1515 Rail Button (large)	2	$4.00	$8.00
AeroPack 75mm retainer (flanged)	1	$75.83	$75.83
AeroTech 75mm Hardware Kit	1	$629.99	$629.99

Aerotech L850-W	3	$289.99	$869.97
3D Print Section	**1kg spool - ABS**	--	--
Fins (PETG)	6	--	--
Tail cone (ABS)	1		
Nosecone (ABS)	4	--	--
Sum of Spools:	11	$21.99	$241.89
		Subtotal:	**$2,134.72**
Sub-Scale Vehicle			
ID = 3.0", L = 48" Airframe - Blue Tube 2.0	1	$31.00	$31.00
3" to 38mm Centering Ring - Baltic Birch	3	$3.49	$10.47
3" Airframe Bulkhead w/ eyebolt	4	$3.69	$14.76
OD = 3", L = 8" Avionics Bay w/ Hardware	1	$40.00	$40.00
Aerotech 38mm Hardware Kit	1	$179.99	$179.99
Aerotech J575FJ-14	2	$102.99	$205.98
AeroTech 38mm Drill Kit	1	$25.29	$25.29
AeroPack 38mm retainer (Blue Tube)	1	$25.00	$25.00
3D Print Section	**1kg spool - ABS**	--	--
Fins (ABS)	3	--	--
Tail cone (ABS)	1		
Nosecone (ABS)	1	--	--
Sum of Spools:	5	$21.99	$98.96
		Subtotal:	**$631.45**
Payload			
12V DC brushed motor	3	$17.86	$53.58
Arduino Mega	1	$48.40	$48.40
900 mAh LiPo Battery	1	$12.12	$12.12
Jumper Wire (F2M)(5.9in)	1	$7.49	$7.49
Jumper Wire (M2M)(5.9in)	1	$7.49	$7.49
Jumper Wire (F2F)(5.9in)	1	$7.49	$7.49
3D Print Section	**1kg spool - ABS**	--	--
Payload (PLA)	1	--	--
Sum of Spools:	1	$21.99	$21.99
		Subtotal:	**$158.56**
General/Uncategorized Components			
1/4" Stainless Steel Quick Link	8	$4.47	$35.76
1/4in x 1in Stainless Steel Eye Bolt w/Hardware (4pk)	2	$5.38	$10.76
LipoCharger V2	1	$27.00	$27.00

Shipping and handling - all budget bulk estimate	1	$250.00	$250.00
		Subtotal:	**$323.52**
Transportation and Logistics			
Gas reimbursement - SRA test launches (Tal --> PB) (44.5¢/mil)	3	$293.70	$881.10
Gas reimbursement - NASA MSFC (Tal --> Hunts) (44.5¢/mil)	1	$351.55	$351.55
Student IHG Hotel Rooms (4 days, $90/day)	2	$360.00	$720.00
Food Stipend (4 comp. days, $35/day/student)	6	$140.00	$840.00
		Subtotal:	**$2,792.65**
Redundant or Erroneous Purchases			
24" Fruity Chutes: Drogue Chute [Redundant]	1	$75.33	$75.33
Arrow Antenna 3E Yagi [Redundant]	1	$74.99	$74.99
Jolly Logic Chute Release 5X-series [Redundant]	4	$141.52	$566.08
Aerotech L1150-R [Redundant]	3	$289.99	$869.97
		Subtotal:	**$1,586.37**

Appendix B. Flutter Speed and Stability Calculation MATLAB Code

Flutter Speed (Material --> PETG Filament)

```
% Defining Initial parameters

E = 416258.3; % Young's Modulus  (psi) v = 0.37; %
Poisson's Ratio (unitless) Cr = 11; % Root Chord in
inches Ct = 3; % Tip Chord in inches t = 0.47; %
Thickness in inches b = 9;  % Fin Height relative to the
root chord in inches


% The following equations will calculate the fin flutter speed for the
% leading design's fin configuration using ABS filament as the
material of % choice

G = E/(2*(1+v));
h = 4600; % Max height the rocket will reach in feet S = (1/2)*(Cr +
Ct)*b; % Wing Area (inches squared) AR = (b.^2)/S; % Aspect Ratio
(unitless) lambda = Ct/Cr; % Taper Ratio (unitless) T = 59 -
0.00356*h; % Temp (Fahrenheit) P = (2116/144)*((T +
459.7)/518.6).^5.256; % Pressure (also converts  to lb/in^2)
a = sqrt(1.4*1716.59*(T + 459.7)); %Speed of sound (ft/s)
Vf = a*sqrt((G/(1.337*(AR.^3)*P*(lambda+1)))*(2*(AR+2)*((t/
Cr).^3))); % Fin Flutter Speed


% The follwing code is written simply to output the answers on the
script % publication

fprintf('The flutter speed of the fin is %.4f ft/s. \n\n', Vf)
```

The flutter speed of the fin is 1446.2738 ft/s

```
L_N = 20;
S = 9;
X_R = 6.773;
C_T = 3;
C_R = 11;
R = 3.077;
N = 4;
X_B = 83.5;
X_CG = 58.575;
C_N = 2; d =
6.154;

X_N = 0.466*L_N;

theta = 37; L_F = sqrt(S^2 + ((1/2)*C_T - (1/2)*C_R +

(S/tan(theta))));

% C_F = (1 + (R/(S+R)))*((4*N*(2/(2*R))^2)/ (1 + sqrt(1+((2*L_F)/C_R
+C_T)^2)));

a = 1 + ((R)/(S+R)); b =
4*N*((S/d)^2); c =
((2*L_F)/(C_R + C_T))^2; g =
1+sqrt(1+c); C_F = a*(b/g);




X_F = X_B + (X_R/3)*((C_R + 2*C_T)/(C_R + C_T)) + (1/6)*(C_R + C_T -
 (C_R*C_T/(C_R + C_T))); X_CP =

(C_N*X_N + C_F*X_F)/(C_N + C_F);

S_M = (X_CP - X_CG)/d;
```

Published with MATLAB® R2021a

Appendix C. Apogee Calculation MATLAB Code

```
clc
clear all
```

Finding Apogee of Zenith Rocket with air resistance and changing mass

```
burntime = 4.4; %burntime in seconds
totimp = 3646.2; %Total impulse in Newton seconds
m_takeoff = 17.368; %Mass of rocket at takeoff in kg
m_prop = 2.095; %Mass of propellant in kg


M = m_takeoff - 0.5*m_prop; %average weight of the rocket in kg
M2 = m_takeoff - m_prop; %weight of the vehicle after powered ascentg in kg

g = 9.81; %acceleration of gravity m/s^2
rho = 1.2; %density of the atmosphere kg/m^3
Cd = 0.5; %coefficient of drag
A = 0.0182; %cross sectional area of the rocket m^2

k = 0.5*rho*Cd*A;
T = totimp/burntime; %thrust forc

v_burnout = (sqrt((T - M*g) / k))*(1-exp(-(2*k*(sqrt((T - M*g) / k)) / M)*burntime)) / (1+exp(-(2*k*(sqrt((T - M*g) / k)) / M)*burntime));
alt_burnout = (-M / (2*k))*log((T - M*g - k*v_burnout^2) / (T - M*g));


alt_coast = (+M2 / (2*k))*log((M2*g + k*v_burnout^2) / (M2*g));

totalalt = alt_burnout + alt_coast;

fprintf('The apogee of the rocket is %.2f ft  \n', 3.28*totalalt)
```

```
The apogee of the rocket is 4431.75 ft
```

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Appendix D. Flight Data Analysis MATLAB Code

```
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%                                                           %
%                  Test Flight Processing                   %
%                                                           %
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% Helps for when I run this a million times to debug
clc;
clear all;
format short;
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%


% Ejection Charge Firing Times
% Input time of first of the two to shoot @ event

c1 = 16;   % Apogee
c2 = 57.3; % Main chute

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

            %%%%% Import Data %%%%%
 alt1 = readmatrix('FRR_1.csv');
 alt2 = readmatrix('FRR_2.csv');

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
           %%%%%%% Define Data Struct %%%%%%%%%

 % Altimeter 1
 data.time1 = alt1(:,1); % Time in second
 data.AGL1 = alt1(:,5);  % Altitude AGL in meter
 data.vel1 = alt1(:,6);  % Velocity in km/h

 % Altimeter 2
 data.time2 = alt2(:,1); % Time in second
 data.AGL2 = alt2(:,5);  % Altitude AGL in meter
 data.vel2 = alt2(:,6);  % Velocity in km/h


%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
```

```
 % Convert units (altitude to ft) (velocity to ft/s)
 for i = 1:length(data.vel1)
     data.vel1(i) = data.vel1(i)*0.911344;
     data.AGL1(i) = data.AGL1(i)*3.28;

     data.vel2(i) = data.vel2(i)*0.911344;
     data.AGL2(i) = data.AGL2(i)*3.28;
 end

%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
          %%%%% Interpolate to Smooth Data %%%%%%%

% Define end of time axis for auto plot
 t.end = max([max(data.time1) max(data.time2)]);

% Interpolation points to end of time axis
 int = [1:.001:t.end];
% Interpolate altitude and velocity
 data.AGL1int = interp1(data.time1, data.AGL1, int,
'spline');
 data.AGL2int = interp1(data.time2, data.AGL2, int,
'spline');
 data.vel1int = interp1(data.time1, data.vel1, int,
'spline');
 data.vel2int = interp1(data.time2, data.vel2, int,
'spline');


%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%

 % Print Result Summary
 fprintf('Vehicle apogee // Alt1: %4.0f ft // Alt2: %4.0f
ft // \n', ...
     max(data.AGL1), max(data.AGL2))
 disp(' ')
 fprintf('Max Velocity // Alt1: %4.0f ft/s // Alt2: %4.0f
ft/s // \n',...
     max(data.vel1), max(data.vel2))
 disp(' ')
 fprintf('Mach Number // Alt1: %1.3f // Alt2: %1.3f // \n',
...
     (max(data.vel1)/1116.4), (max(data.vel2)/1116.4))
```

```
  disp(' ')
 fprintf('Charge Firings // Drogue: %1.1fs // Main Chute: 
%1.1fs // \n', ...
     c1, c2)


%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
           %%%%%%% Auto Plot Params %%%%%%%%%

 % altitude y axis max
 a.max = max([max(data.AGL1int) max(data.AGL2int)]);
 a.max = a.max + 100;
 a.max = round(a.max, 2, 'significant');

 % velocity y axis min
 v.min = min([min(data.vel1int) min(data.vel2int)]);
 v.min = v.min -15;
 v.min = round(v.min, 1, 'significant');

 % velocity y axis max
 v.max = max([max(data.vel1int) max(data.vel2int)]);
 v.max = v.max + 15;
 v.max = round(v.max, 1, 'significant');

 % end of time axis
 t.end = t.end + 5;
 t.end = round(t.end, 1, 'significant');


%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
                      %%%%% Plots %%%%%

 % Labels for e-charge lines
 e1 = '\leftarrow Fire Ejection Charge 1';
 e2 = '\leftarrow Fire Ejection Charge 2';

 % Altitude plot
 figure(1)
 plot(int, data.AGL1int, 'b-')
 hold on
 plot(int, data.AGL2int, 'r-')
 xline(c1, 'k--')
 xline(c2, 'k--')
```

```
hold off
ylabel('Altitude (ft AGL)')
xlabel('Flight Time (s)')
legend('Altimeter 1', 'Altimeter 2')
grid on
axis([0 int(end) 0 a.max])
title('Altitude vs. Flight Time')
text(c1,1500,e1)
text(c2,1500,e2)

% Vel plot
figure(2)
plot(int, data.vel1int, 'b-')
hold on
plot(int, data.vel2int, 'r-')
xline(c1, 'k--')
xline(c2, 'k--')
hold off
ylabel('Velocity (ft/s)')
xlabel('Flight Time (s)')
legend('Altimeter 1', 'Altimeter 2')
grid on
axis([0 int(end) v.min v.max])
title('Velocity vs. Flight Time')
text(c1,100,e1)
text(c2,100,e2)

%
%   % Detailed View of Altitude
%   figure(2)
%   % Apogee
%   subplot(1,3,1)
%   plot(int, data.AGL1int, 'b-')
%   hold on
%   plot(int, data.AGL2int, 'r-')
%   xline(16, 'k--')
%   xline(155.4, 'k--')
%   hold off
%   ylabel('Altitude (ft AGL)')
%   xlabel('Flight Time (s)')
%   legend('Altimeter 1', 'Altimeter 2')
%   grid on
```

```
%   axis([5 30 2800 3300])
%   text(16.5,2900,e1)
%   title('Apogee and Separation 1')
%   % Sep 2
%   subplot(1,3,2)
%   plot(int, data.AGL1int, 'b-')
%   hold on
%   plot(int, data.AGL2int, 'r-')
%   xline(16, 'k--')
%   xline(155.4, 'k--')
%   hold off
%   ylabel('Altitude (ft AGL)')
%   xlabel('Flight Time (s)')
%   legend('Altimeter 1', 'Altimeter 2')
%   grid on
%   axis([140 170 200 800])
%   title('Ejection Charge 2')
%   text(156,650,e2)
%   % Impact
%   subplot(1,3,3)
%   plot(int, data.AGL1int, 'b-')
%   hold on
%   plot(int, data.AGL2int, 'r-')
%   xline(16, 'k--')
%   xline(155.4, 'k--')
%   hold off
%   ylabel('Altitude (ft AGL)')
%   xlabel('Flight Time (s)')
%   legend('Altimeter 1', 'Altimeter 2')
%   grid on
%   axis([170 185 0 150])
%   title('Ground Impact')
%   sgtitle('Altitude vs. Flight Time')
%
%   % Velocity
%   figure(3)
%   plot(int, data.vel1int, 'b-')
%   hold on
%   plot(int, data.vel2int, 'r-')
%   xline(16, 'k--')
%   xline(155.4, 'k--')
%   hold off
```

```
%   xlabel('Flight Time (s)')
%   ylabel('Velocity (ft/s)')
%   title('Velocity vs. Flight Time')
%   legend('Altimeter 1', 'Altimeter 2')
%   axis([0 200 -75 575])
%   grid on
%   text(16.5,350,e1)
%   text(156,350,e2)
%
%    % Detailed view of velocity
%   figure(4)
%   % Liftoff to burnout
%   subplot(1,3,1)
%   plot(int, data.vel1int, 'b-')
%   hold on
%   plot(int, data.vel2int, 'r-')
%   xline(16, 'k--')
%   xline(155.4, 'k--')
%   hold off
%   xlabel('Flight Time (s)')
%   ylabel('Velocity (ft/s)')
%   legend('Altimeter 1', 'Altimeter 2')
%   grid on
%   axis([0 10 0 550])
%   title('Liftoff and Burnout')
%   % Apogee and chutes
%   subplot(1,3,2)
%   plot(int, data.vel1int, 'b-')
%   hold on
%   plot(int, data.vel2int, 'r-')
%   xline(16, 'k--')
%   xline(155.4, 'k--')
%   hold off
%   xlabel('Flight Time (s)')
%   ylabel('Velocity (ft/s)')
%   legend('Altimeter 1', 'Altimeter 2')
%   grid on
%   axis([14 30 -60 30])
%   title('Apogee and Parachute Deployment(s)')
%   text(16.5,-10,e1)
%   % E2 to impact
%   subplot(1,3,3)
```

```
%   plot(int, data.vel1int, 'b-')
%   hold on
%   plot(int, data.vel2int, 'r-')
%   xline(16, 'k--')
%   xline(155.4, 'k--')
%   hold off
%   xlabel('Flight Time (s)')
%   ylabel('Velocity (ft/s)')
%   legend('Altimeter 1', 'Altimeter 2')
%   grid on
%   axis([150 185 -30 5])
%   title('Ejection Charge 2 to Impact')
%   sgtitle('Velocity vs. Flight Time')
%   text(156,-10 ,e2)
```