Project Photogator

NASA Student Launch 2023 Critical Design Review

University of Florida

Swamp Launch Rocket Team

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Gainesville, FL 32611

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Table of Contents

Table of Contents 1

Table of Figures 4

Table of Tables 6

1. Summary of Report 9

1.1 Team Summary 9

1.1.1 Team Mentor 9

1.1.2 Launch Plans 9

1.1.3 Hours 9

1.2 Launch Vehicle Summary 9

1.3 Payload Summary 9

2. Changes Made Since PDR 10

2.1 Changes Made to Vehicle Criteria 10

2.2 Changes Made to Payload Criteria 10

2.3 Changes Made to Project Plan 11

3. Vehicle Criteria 12

3.1 Design and Verification of Launch Vehicle 12

3.1.1 Mission Statement 12

3.1.2 Final Design 12

3.1.3 Separation Points 16

3.1.4 Manufacturing Methods 18

3.1.5 Design Justification 42

3.1.6 Design Integrity 47

3.2 Subscale Flight Results 49

3.2.2 Launch Conditions and Simulation 49

3.2.1 Scaling Factor 51

3.2.3 Subscale Flight Analysis 52

3.2.4 Impact on Full-Scale Design 53

3.2.5 Descent Analysis 53

3.3 Recovery Subsystem 55

3.3.1 Selected Components 55

3.3.2 System Redundancy 59

3.4 Mission Performance Predictions 59

3.4.1 Flight Profile Simulations 59
3.4.2 Motor Selection 61
3.4.3 Stability Margin 62
3.4.4 Simulation Verifications 63
3.4.5 Descent Predictions 64
4. Payload Criteria 69
4.1 Design of Payload Equipment 69
4.2 Selected Design Alternatives 70
4.2.1 Selected Mechanical Components 70
4.2.2 Selected Electronic Components 71
4.2.3 Selected Software Components 75
4.3 System Level Design 76
4.3.1 Payload Mechanical Design 76
4.3.2 Payload Electronic Design 91
4.3.3 Payload Software Design 94
4.3.4 Payload Integration 100
4.3.5 Payload Mass Table 100
5. Safety 101
5.1 Safety Procedures 101
5.1.1 Avionics and Recovery Preparation 101
5.1.2 Camera Preparation 103
5.1.3 Payload Preparation 104
5.1.4 Ejection Charge Preparation 105
5.1.5 Rocket Assembly Preparation 107
5.1.6 Motor Preparation 109
5.1.7 Setup on Launch Pad 110
5.1.8 Igniter Installation 110
5.1.9 Launch Procedure 111
5.1.10 Troubleshooting 112
5.1.11 Post-Flight Inspection 114
5.2 Personnel Hazards Analysis 114
5.2.1 Chemical Hazards 115
5.2.2 Physical Hazards 117

5.2.3 Biological Hazards 120
5.3 Failure Mode and Effects Analysis 121
5.3.1 Structures 122
5.3.2 Payloads 123
5.3.3 Avionics and Recovery 125
5.3.4 Flight Dynamics 127
5.4 Environmental Concerns 128
5.4.1 Effect of Environment on Launch Vehicle 128
5.4.2 Effect of Launch Vehicle on Environment 129
6. Project Plan 130
6.1 Launch Vehicle Testing 130
6.1.1 Testing Rationale and Resultant Effects 130
6.1.2 Testing Plan and Description 136
6.1.3 Completed Launch Vehicle Testing 142
6.2 Payload Testing 161
6.2.1 Testing Rationale and Resultant Effects 161
6.2.2 Testing Plan and Description 165
6.2.3 Completed Payload Testing 169
6.3 Completed Subscale Testing 172
6.3.1 Subscale Center of Gravity Inspection 173
6.3.2 Subscale Parachute Ejection Demonstration 173
6.3.3 Subscale Parachute Unfolding Demonstration 175
6.3.4 Subscale Demonstration Launch 176
6.4 Requirements Compliance 179
6.4.1 Competition Requirements 179
6.4.2 Team Derived Requirements 208
6.5 Budgeting and Timeline 212
6.5.1 Funding 212
6.5.2 Budget 213
6.5.3 Timeline 215
7. Launch Plans 222
7.1 Preliminary Launch Roster 222
8. Conclusion 222

Table of Figures

Figure 1: Launch Vehicle modeled in Fusion360 (above) & OpenRocket (below) 12
Figure 2: Assembly Drawing of Nosecone Section 13
Figure 3: Assembly Drawing of Forward Section 14
Figure 4: Aft Section Assembly Drawing 15
Figure 5: Payload Airframe Assembly Drawing 15
Figure 6: Separation Points 17
Figure 7: First Separation Event 17
Figure 8: Second Separation Event 17
Figure 9: Vehicle Orientation After Parachute Deployment 18
Figure 10: Nosecone Drawing 19
Figure 11: Nosecone Bulkhead Drawing 20
Figure 12: Forward Airframe Drawing 22
Figure 13: Avionics Switchband Drawing 23
Figure 14: Avionics Coupler Drawing 24
Figure 15: Avionics Bulkhead Drawing 25
Figure 16: Central Airframe Drawing 27
Figure 17: Payload Airframe Drawing 28
Figure 18: Payload Airframe Drawing Side View 29
Figure 19: Payload Bay Drawing 30
Figure 20: Payload Bay Forward Bulkhead Drawing 31
Figure 21: Payload Aft Bulkhead 33
Figure 22: Payload Aft Coupler 34
Figure 23: Fin Drawing 35
Figure 24: Aft Airframe Drawing 36
Figure 25: Centering Rings Drawing 38
Figure 26: Motor Tube Drawing 39
Figure 27: Subscale Altitude vs. Time 50
Figure 28: Subscale Total Velocity vs. Time 50
Figure 29: Subscale Acceleration vs. Time 51
Figure 30: Subscale Resulting Velocity and Altitude vs. Time 52
Figure 31: Subscale Flight Data with Programmed (thick line) and Actual (thin line) Main Parachute Deployment Altitudes 54
Figure 32: GPS Receiver 54
Figure 33: GPS Transmitter in Launch Vehicle 54
Figure 34: Avionics Bay with Key Switches 58
Figure 35: Altimeter Wiring Diagram 59
Figure 36: Altitude vs. Time 60
Figure 37: Total Velocity vs. Time 60
Figure 38: Total Acceleration vs. Time 61
Figure 39: Aerotech L1090 Thrust vs. Time 62
Figure 40: Stability vs. Time 63

Figure 42: A Plot of Lateral Distance vs. Time from an OpenRocket Simulation 66
Figure 43: Distance Traveled during Main Parachute Deployment (distance between blue lines) 69
Figure 44: Full-scale Assembly Payload aligned with fins 70
Figure 45: Camera System with Labeled Components 76
Figure 46: Camera Mount Drawing 77
Figure 47: Camera Case 78
Figure 48: Camera Mount CAD 78
Figure 49: Payload Camera System Side View 79
Figure 50: Payload Camera System Isometric View 79
Figure 51: Camera Mount includes Locking Lug 80
Figure 52: Spring-Loaded Hinge Drawing 81
Figure 53: Stepper Motor Hub 81
Figure 54: Payload System Rotated 90° Out of Airframe 82
Figure 55: Payload System Rotated Isometric View 82
Figure 56: Direction of Rotation of Stepper for Camera System 83
Figure 57: Motor Hub and Mount Assembly 83
Figure 58: Payload Housing Drawing 84
Figure 59: Payload Housing Top View 85
Figure 60: Payload Housing Isometric View 85
Figure 61: Radio Housing in Central Payload Housing 86
Figure 62: Radio Housing Cover 86
Figure 63: Payload Electronics Sled Drawing 87
Figure 64: Battery Housing Drawing 88
Figure 65: Battery Housing Drawing Sheet 2 88
Figure 66: Isometric View of Payload Coupler Assembly 89
Figure 67: Side View of Payload Coupler Assembly 89
Figure 68: Payload Electronics Assembly Drawing 90
Figure 69: Payload Electronics Assembly Sheet 2 90
Figure 70: Payload Electronic Components Diagram 92
Figure 71: Schematics of Payload Electronics 93
Figure 72: Payload Electronics PCB Layout 94
Figure 73: Software State Flowchart 94
Figure 74: Sensor System Flowchart 95
Figure 75: Radio System Flowchart 97
Figure 76: Linux Command to Initialize Radio Listening 98
Figure 77: Motor System Flowchart 98
Figure 78: Camera System Flowchart 99
Figure 79: Payload Section in Airframe 100
Figure 80: Knot Strength Test Stand 144
Figure 81: Failure at Knot Location 145
Figure 82: Measurement of Weight at Tensile Failure 145
Figure 83: From Left to Right: The Figure-Eight Knot, the Bowline, and the Slip Knot 146
Figure 84: Tabulated Knot Strength Data 146
Figure 85: Graphed Knot Strength Data 146

Figure 86: Phenolic (top left), Quantum Tube (top right), Blue Tube (bottom left), and G-12 Fiberglass (bottom right) Airframes After Drop Test 148
Figure 87: Airframe Material Zippering Resistance Test 149
Figure 88: Epoxy Fillet 150
Figure 89: Battery Voltage 153
Figure 90: Drogue Parachute Drag Testing 154
Figure 91: Main Parachute Drag Testing 156
Figure 92: Method 1 Folding Steps 158
Figure 93: Method 2 Folding Steps 159
Figure 94: Method 3 Folding Steps 159
Figure 95: Control Circuit Installed on Breadboard 171
Figure 96: Circuit Diagrams with and without Control Circuit 171
Figure 97: Camera Housing Failure 172
Figure 98: Failed Drogue Ejection 174
Figure 99: Successful Main Ejection 175
Figure 100: Successful Drogue Ejection 175
Figure 101: Parachute Unfolding Demonstration Set-Up 176
Figure 102: Central Airframe Immediately Following Landing 177
Figure 103: Main Parachute Immediately Following Landing 177
Figure 104: Aft Section (Containing Payload) Immediately Following Landing 178
Figure 105: Drogue Parachute Immediately Following Landing 178
Figure 106: 2022-2023 Budget Breakdown 213
Figure 107: Full Project Schedule 216
Figure 108: Structures Schedule 218
Figure 109: Full-Scale Manufacturing Schedule 219
Figure 110: Payload Schedule 220
Figure 111: Payload Manufacturing Schedule 221

Table of Tables

Table 1: Vehicle Dimensions 9
Table 2: Ejection Charge Masses 16
Table 3: Qualitative Scores 42
Table 4: Nosecone Decision Matrix 43
Table 5: Airframe and Couplers Decision Matrix 43
Table 6: Motor Tube Decision Matrix 44
Table 7: Fins Decision Matrix 44
Table 8: Centering Rings Decision Matrix 45
Table 9: Bulkheads Decision Matrix 46
Table 10: Full-scale Final Masses 49
Table 11: Subscale Launch Conditions 49
Table 12: Subscale Flight Data 52
Table 13: Main Parachute Decision Matrix 55
Table 14: Drogue Parachute Decision Matrix 56

Table 15: Altimeter Decision Matrix 57
Table 16:Launch Conditions used in Simulations 60
Table 17: Tabulated Monte Carlo Results 63
Table 18: Parachute Descent Rates 64
Table 19: Descent Time Calculation Results 65
Table 20: Drift Results 67
Table 21: Kinetic Energy Results 67
Table 22: Predicted Deployment Load 68
Table 23: 3D Print Material Decision Matrix 71
Table 24: Fastener Decision Matrix 71
Table 25: Payload Microprocessor Decision Matrix 72
Table 26: Payload Camera Decision Matrix 73
Table 27: Payload IMU Decision Matrix 73
Table 28: Payload Barometer Decision Matrix 74
Table 29: Payload Solenoid Decision Matrix 75
Table 30: Payload Stepper Motor Decision Matrix 75
Table 31: Primary and Secondary I2C Device Addresses 95
Table 32: IMU Acceleration Data Registers 95
Table 33: Barometer Pressure Registers 96
Table 34: rtl_fm Configuration Parameters and Values 97
Table 35: Dire Wolf Configuration Parameters and Values 97
Table 36: Payload Component Masses 101
Table 37: Risk Assessment Chart (RAC) 115
Table 38: Risk Assessment Score Chart 115
Table 39: Chemical Hazard Identification 117
Table 40: Manufacturing Hazard Identification 119
Table 41: Launch Preparation Hazard Identification 119
Table 42: Launch Operations Hazard Identification 120
Table 43: Biological Hazard Identification 121
Table 44: Risk Priority Number Chart 121
Table 45: RPN Score Chart 122
Table 46: Structures FMEA 123
Table 47: Payload Mechanics FMEA 123
Table 48: Payload Electronics FMEA 124
Table 49: Payload software FMEA 125
Table 50: Avionics and recovery FMEA 127
Table 51: Flight dynamics FMEA 128
Table 52: Effects of Environment on Launch Vehicle 129
Table 53: Effects of Launch Vehicle on Environment 129
Table 54: Test ID Abbreviations 130
Table 55: Launch Vehicle Testing Plan: Objectives, Rationales, and Resultant Effects 136
Table 56: Launch Vehicle Testing Plan: Variables, Success Criteria, Methods, and Materials 142
Table 57: Completed Launch Vehicle Tests 143
Table 58: Epoxy Density Inspection Results 150

Table 59: Barometric Formula Variables and Constants 151
Table 60: Altimeter Accuracy Inspection Results 152
Table 61: Subscale Drogue Coefficient of Drag 155
Table 62: Full-Scale Drogue Coefficient of Drag 155
Table 63: Subscale Main Coefficient of Drag 157
Table 64: Full-Scale Main Coefficient of Drag 157
Table 65: Lengths of Packed Parachutes 160
Table 66: Deployment Times for Folding Methods 161
Table 67: Payload Testing Plan: Objectives, Rationales, and Resultant Effects 165
Table 68: Payload Testing Plan: Variables, Success Criteria, Methods, and Materials 169
Table 69: Completed Payload Testing 170
Table 70: Control Circuit Resistance Inspection Results 171
Table 71: Ejection Charge Masses 175
Table 72: General Requirements 184
Table 73: NASA Vehicle Requirements 198
Table 74: NASA Recovery Requirements 202
Table 75: NASA Payload Requirements 206
Table 76: NASA Safety Requirements 208
Table 77: Team Vehicle Requirements 208
Table 78: Team Recovery Requirements 210
Table 79: Team Payload Requirements 212
Table 80: Budget Breakdown by Category 213
Table 81: Subscale Itemized Budget 214
Table 82: Full-Scale Itemized Budget 215
Table 83: General Itemized Budget 215
Table 84: Testing Itemized Budget 215
Table 85: Project Milestone Status 217
Table 86: Project Activity Status 217
Table 87: Preliminary Launch Roster 222

1. Summary of Report

1.1 Team Summary

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1.1.2 Launch Plans

The team's final launch of the competition is anticipated to take place on 4/15/2023 at the primary competition launch field in Huntsville, AL. The team's local launch site will be the Tripoli Tampa Rocketry Association (Prefecture #17) site in Plant City, FL.

1.1.3 Hours

The total number of hours spent developing the CDR milestone was 515 hours.

1.2 Launch Vehicle Summary

Section	Length (in)	Mass (oz)
Nosecone	18.0	26.9
Forward	44.0	126.8
Aft	53.0	266.4
Total	**115.0**	**420.0**

Table 1: Vehicle Dimensions

The targe altitude of the launch vehicle is 4600 ft. The motor selected is the Aerotech L1090W. The vehicle does not have any ballast. The dry mass of the launch vehicle is 334.0 oz. The wet mass of the launch vehicle is 420.0 oz. The burnout mass is 371 oz. A 12 ft 1515 launch rail will be used. Two rail buttons will be used, one located at the center of gravity and another 4 in from the aft.

1.3 Payload Summary

Payload Title: InvestiGator

The payload contains three camera systems aligned with each of the three fins, spaced 120° apart, so that one will always be normal to the ground upon landing. This is achieved through the natural orientation of the launch vehicle upon landing, as the vehicle will come to rest on two fins, leaving one oriented normal to the ground. Each camera system will have a camera housing that holds a camera, camera mount, and motors. Additionally, the payload includes an electronics sled that retains electronics for controlling the payload. An Inertial Measurement Unit (IMU) will detect the orientation of the launch vehicle and determine which camera to activate so that only one camera will be in use when taking photos of the surroundings. A Software Defined Radio (SDR) dongle will be incorporated in the payload to receive Automatic Package Reporting System (APRS) commands from NASA. The Raspberry Pi processes the digitalized APRS command and then sends the commands to the upright camera system.

2. Changes Made Since PDR

2.1 Changes Made to Vehicle Criteria

The length of the forward payload coupler was increased to 9.0 in. Previously, the forward payload coupler was 8.0 in long. This change was made to provide more space for the payload electronics and its retention system in the payload forward coupler.

The payload aft coupler's length was increased from 7.0 in to 8.0 in. The length of the aft payload coupler was increased to improve the structural integrity of the connection point between the payload and aft airframes.

The length of the payload airframe increased from 25.0 in to 29.0 in to provide enough space for the payload housing since changing the couplers resulted in them occupying more space in the airframe. The length of the forward airframe was decreased from 27.0 in to 23.0 in to maintain the original length of launch vehicle. The nosecone coupler length was decreased from 6 in to 4 in. in order to maintain the same amount of space in the forward section for the parachute and recovery harness.

The dimensions of the slots for the payload housings in the payload airframe were changed from 4.52 in by 1.73 in to 4.84 in by 1.65 in. This was altered so the camera system movement does not interfere with the airframe when it rotates out of the housing. The distances between each of the payload housing slots were decreased from 1.73 in to 1.16 in to fit all components in the payload airframe without interfering with the couplers. In addition, a 0.5 in thick bulkhead was added to the aft end of the forward payload coupler as part of the payload electronics retention system.

The motor was changed from an Aerotech K1000 to an Aerotech L1090 to achieve the designated target apogee altitude of 4600 ft. The Aerotech K1000 did not supply the impulse needed to achieve this target apogee altitude, so it was determined that a higher impulse motor, the L1090, was necessary. A 2.242 in diameter motor tube was selected to contain the motor in the aft airframe. Since the motor tube's length was increased from 18 in to 26 in, a fourth centering was added to the forward end. This centering ring is in the aft payload coupler, so it has an inner diameter of 2.2 in and outer diameter of 3.8 in. The inner diameter of all centering rings was changed to 2.2 in.

The location of the rail buttons changed as well. The forward rail button was moved to be located at the center of gravity, which is 8.80 in from the forward end of the payload airframe. The aft rail button was placed 4 in away from the aft end of the aft airframe instead of 12 in.

The location of the main parachute separation point also changed from between the avionics bay and forward airframe to between the forward airframe and nosecone. This change was made so that the ejection charges could be positioned directly next to the avionics bay, minimizing the risk of them moving unexpectedly during flight or disconnecting the backup charges.

2.2 Changes Made to Payload Criteria

The payload housing design has changed to fit the true dimensions of the motors. The accommodated change in length includes a 0.62 in tall lip, extruded from the base of the housing. This change caused an addition to the payload housing to maximize available space. The drawing schematics provided by Uxcell did not include wiring dimensions, and so the design of the housing did not account for the protruding wires at the side of the motor. The new design accounts for those wires.

A push-button was added to each payload assembly to detect when the camera system rotates out of the airframe. If the solenoid failed and the stepper motor attempted to rotate the system about the z-axis, the stepper could break the camera system if attempting this. Upon releasing the solenoid, the button will be unpressed as the camera system rotates outward. Using this mechanism, the software will be able to include a safety feature that ensures the stepper will not attempt to rotate the camera system until it has rotated out of the airframe. This change is reflected in the Payload Housing drawing (Figure 19).

The electrical schematic of the payload was changed due to the addition of 3 push buttons and a piezo. Three push buttons are used as feedback to make sure stepper motors do not turn the camera when they are locked inside the payload coupler. The piezo is used as output device to announce the state of payload. Three push buttons are simply connected to Raspberry Pi through GPIO (General Purpose Input Output) pins and a 10K pull-down resistor. The piezo is controlled through an N-Channel MOSFET gate that is connected to Raspberry Pi through a GPIO pin.

The camera housing now features two rectangular holes for the electronics wires to fit through instead of one, since the motor drawings did not include schematics for the wiring. The dimensions for the second rectangular hole are 0.59 x 0.55 in. through the wall in the Payload Housing drawing (Figure 19).

The spring-loaded hinge previously selected from McMaster-Carr did not include details of load capacity or spring constant. It was decided to manufacture a spring-hinge by modifying hinge pieces and winding a coil. After testing, it was determined that the solenoid has a safe maximum load capacity of 7.94 oz. The minimum torque required for the spring was calculated to be 0.375 lb-in. This was calculated after finding a safe distance of 3.94 in and load mass of 1.5238e-3 oz. The maximum torque allowed for the given solenoid was found using the maximum load capacity of 7.94 oz. From this range of tolerances, it was decided to make a spring-hinge using spring wire for the coil and steel plates for the hinge to weld together.

The footprints of electrical parts are also changed due to the inaccurate dimensions provided by the manufacturers. Each electrical part is measured using a caliper and a custom footprint is created in Altium to make sure the PCB (Printed Circuit Board) will fit all the components.

The shape and layout of the PCB are also changed due to the height constraints of payload coupler. The XL6009 Boost Power Converter and 3 ULN2003 Stepper Motor Controllers are placed in the middle of the PCB. The hole location of the bulkhead is also moved down by 0.2 inch to give more space for these electrical parts.

2.3 Changes Made to Project Plan

Various changes to the project plan have taken place since the PDR milestone regarding team structure and funding.

With respect to the team structure, the team's advisor is now solely Dr. Sean Niemi, instead of both Dr. Niemi and Dr. Richard Lind. Also, the team is welcoming a new subteam lead to the flight dynamics team: Mary Rowe.

Additionally, one of the team's sponsors, Blue Origin, has increased their initial sponsorship by $1000, which allocates $500 to the USLI project budget. The team's budget also decreased to $6800.00 on account of eliminating the necessity of new motor hardware.

3. Vehicle Criteria

3.1 Design and Verification of Launch Vehicle

3.1.1 Mission Statement

The mission of the launch vehicle is to perform a safe and successful flight and carry the payload and electronics without any damage. The mission success criteria includes that the launch vehicle will reach an altitude of 4600 ft., deploy recovery systems upon descent, and complete the payload mission once the vehicle has safely landed. For the flight to be a success, it must be recoverable, reusable, and not sustain irreparable damage.

3.1.2 Final Design

The final design of the launch vehicle is composed of three sections: a nosecone section, a forward section, and an aft section (Figure 1). An avionics bay, which contains components relating to the recovery system, is located in the forward section. There is also a payload airframe, which contains the payload housings, payload electronics, and payload retention system, in the aft section.

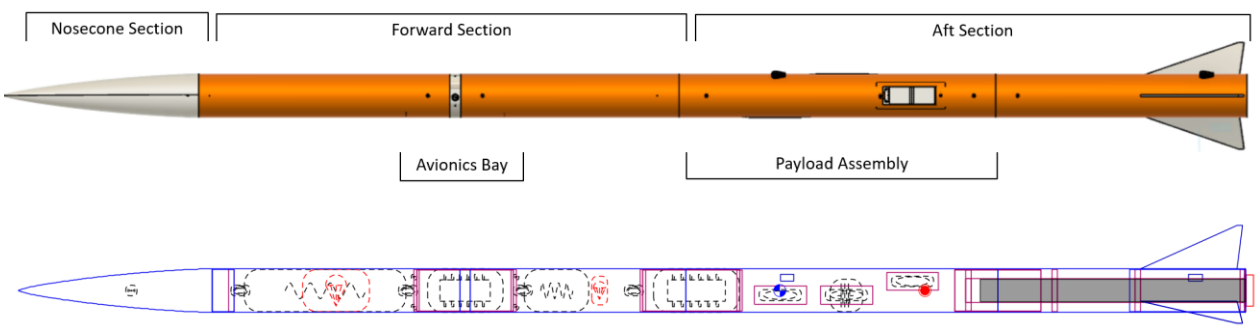

Figure 1: Launch Vehicle modeled in Fusion360 (above) & OpenRocket (below)

One alternative design that was considered included externally mounted camera systems located in the aft section. While the payload design would have been mechanically simpler compared to the final design, it would have caused issues with aerodynamics and become over stable. Another design that was considered included three cameras in the aft section aligned with the fins, similar to the final design. Upon landing, the cameras would extend radially outward using linear actuators. This design was not selected due to the extra weight from the linear actuators and insufficient space to store the camera system and motors.

3.1.2.1 Nosecone Section

The nosecone section consists of a 4 in diameter 4.5:1 Von Karman nosecone, a nosecone coupler, a bulkhead, and a Big Red Bee 900 GPS (Figure 2). The nosecone is made of G12 fiberglass and has a metal tip. The G12 fiberglass nosecone coupler is 4 in long and connects to the nosecone with three plastic rivets. A type II PVC bulkhead is epoxied in place inside of the nosecone coupler. A ¼-20 steel eyebolt is secured to the bulkhead using a hex nut and is epoxied in place. The eyebolt attaches to the recovery harness to keep the nosecone section connected to the forward section after main parachute deployment. A Big Red Bee 900 GPS is located in the nosecone and secured using Velcro tape.

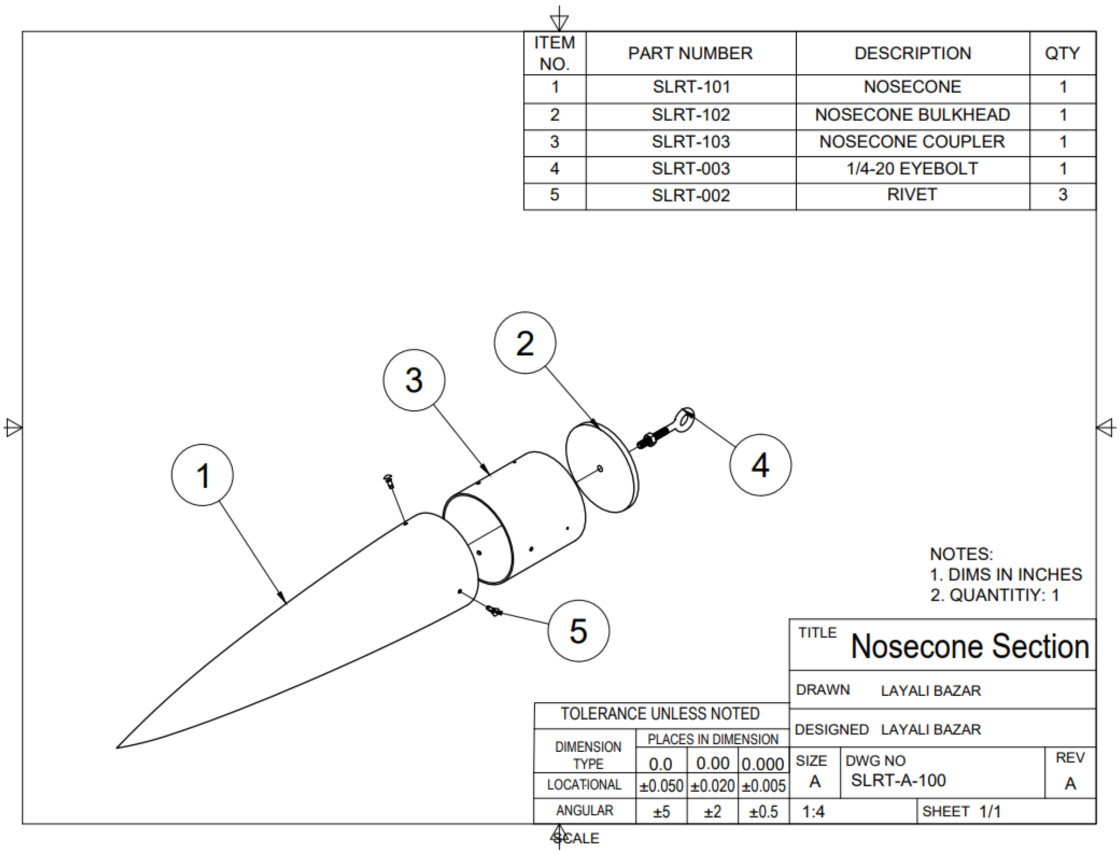

Figure 2: Assembly Drawing of Nosecone Section

3.1.2.2 Forward Section

The forward section consists of a forward airframe, avionics bay, and central airframe (Figure 3). The airframe and couplers are made of G12 fiberglass. The two avionics bay bulkheads are made of type II PVC. The forward section will connect to the nosecone section using three 2-56 nylon shear pins. The avionics bay consists of a coupler, a switchband, two type II PVC bulkheads, two ¼-20 eyebolts, and avionics hardware. The switchband is epoxied onto the coupler with RocketPoxy. Each end of the coupler is sealed with a bulkhead. The ¼-20 eyebolts are attached to the bulkheads and secured in place with a ¼-20 hex nut and RocketPoxy. The eyebolts connect to the recovery harnesses to keep the forward section connected to the nosecone section and aft section upon separation. Inside of the avionics bay there is a 3D printed PETG sled that secures the avionics components in place. The sled is connected to the two avionics bay bulkheads using two ¼-20 threaded rods. The threaded rods also keep the avionics bay bulkheads secure. The avionics bay is connected to the forward and central airframe with three equally spaced rivets on each side.

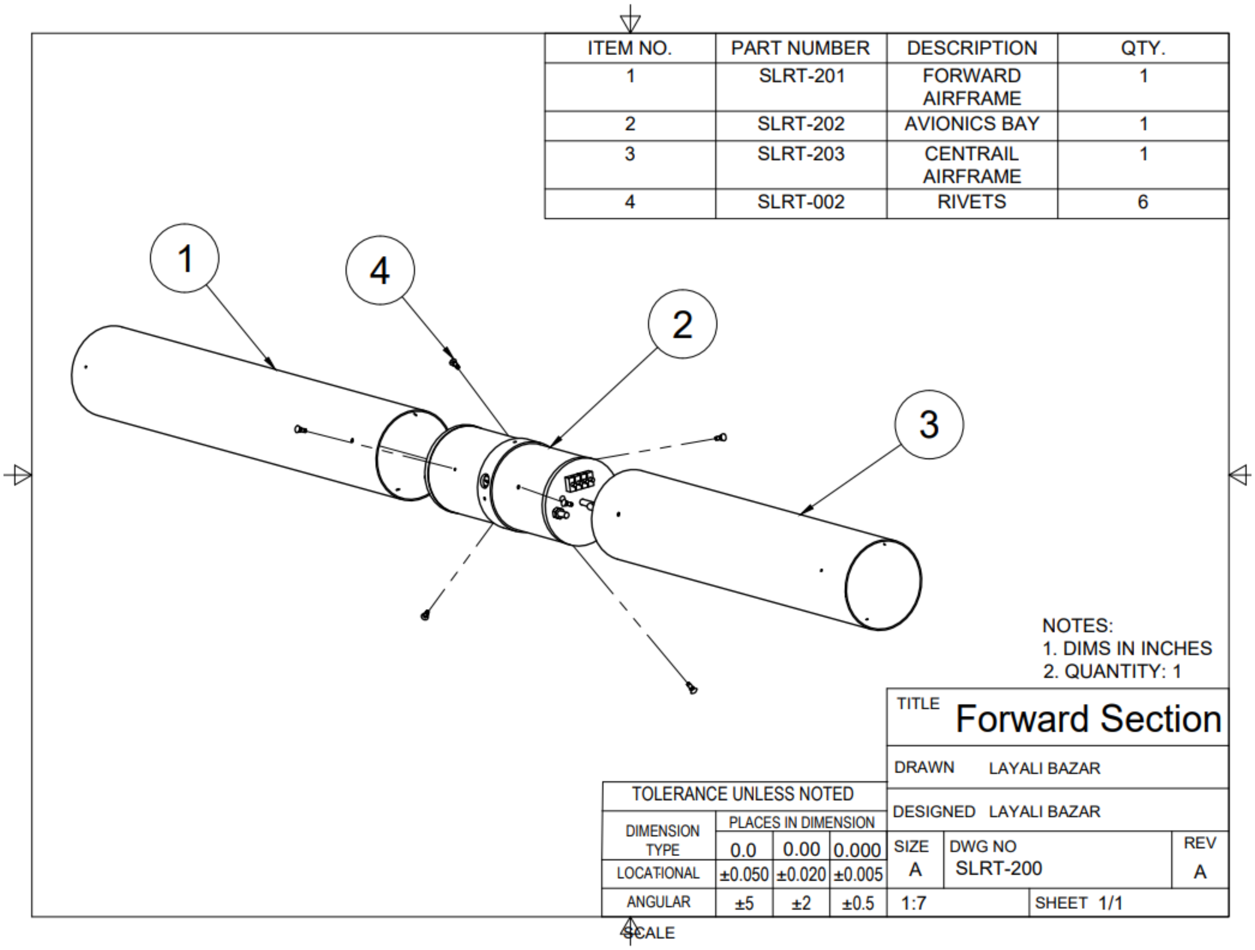

Figure 3: Assembly Drawing of Forward Section

3.1.2.3 Aft Section

The aft section consists of a payload bay, a payload airframe, three payload housings, a payload aft coupler, an aft airframe, a motor assembly, three fins, and two rail buttons (Figure 4). The aft section is connected to the forward section using three 2-56 nylon shear pins. The payload bay is secured to the payload airframe using three plastic rivets. The payload bay contains a sled for the electronics retention system. This sled is retained in the coupler using two ¼-20 threaded rods that are secured to the bulkheads. The forwardmost payload bay bulkhead has an eyebolt that is epoxied in place. This eyebolt will be used to attach the recovery harness. The payload airframe features three rectangular slots that are equally spaced radially and are linearly offset from each other (Figure 5). The airframe contains three payload housings, that will be attached to the airframe using 8-32 screws. The payload aft coupler connects the payload airframe to the aft airframe with three rivets. The coupler and airframe are attached using three rivets. The aft end of the aft payload coupler is epoxied into the aft airframe. The aft airframe contains the motor assembly and fins. The motor assembly consists of a G12 fiberglass motor tube and four plywood centering rings. The three G10 fiberglass fins are attached through the wall of the airframe and are secured in place both externally and internally using RocketPoxy and JBWeld, respectively. The forward and aft rail buttons are attached to the payload airframe and the aft airframe, respectively.

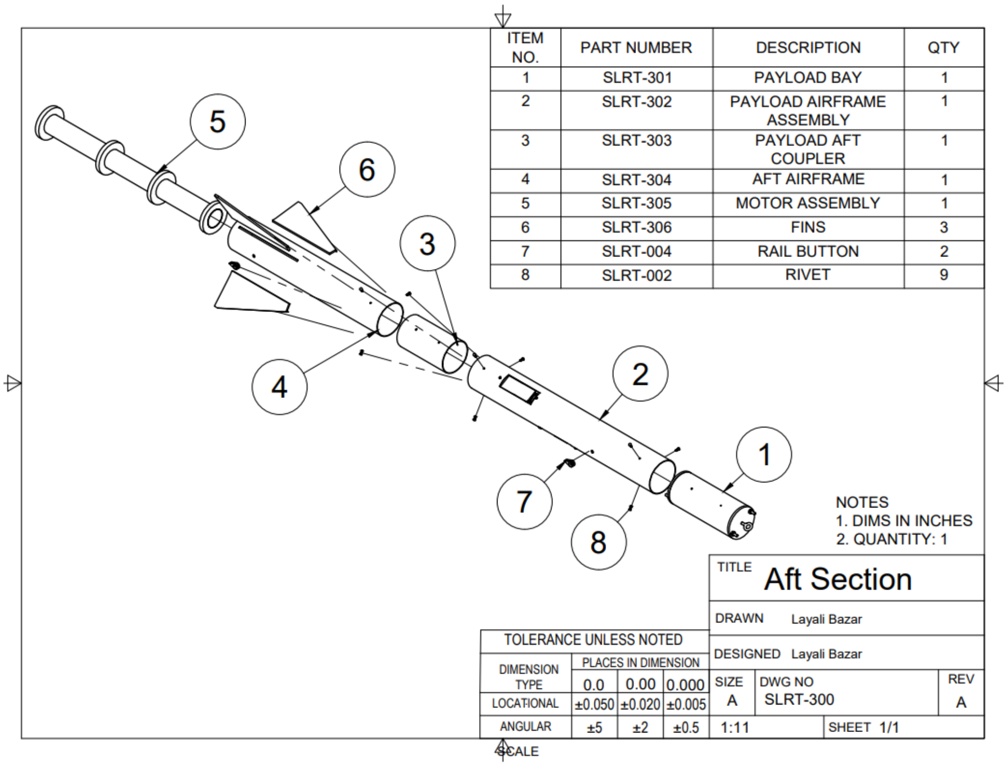

Figure 4: Aft Section Assembly Drawing

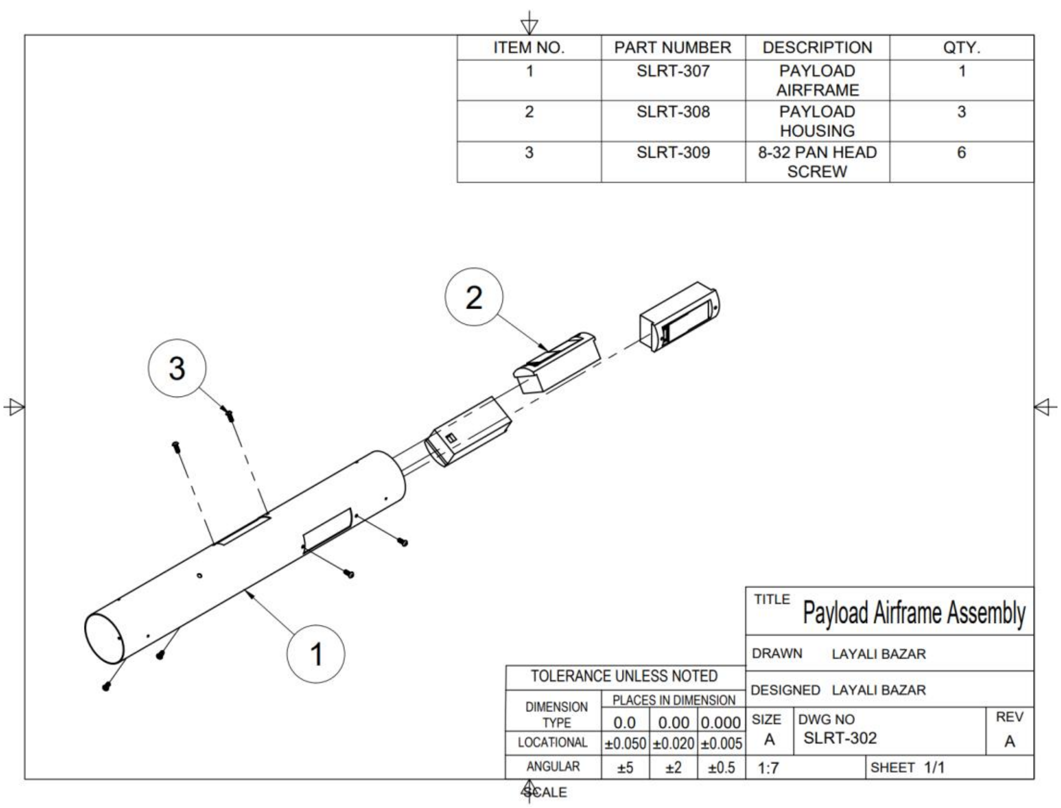

Figure 5: Payload Airframe Assembly Drawing

3.1.3 Separation Points

The launch vehicle will have two separation points. Black powder ejection charges will be used to cause separation. Each separation point will use two ejection charges as a redundancy, and each secondary ejection charge will be 25% larger than its respective primary charge to ensure that separation occurs. Expected masses for all ejection charges are presented (Table 2).

Ejection Charge Masses		
	Primary (g)	Secondary (g)
1st separation point	1.7	2.1
2nd separation point	2.8	3.5

Table 2: Ejection Charge Masses

The ejection charge masses were calculated using the ideal gas law, which relates pressure in the parachute compartment (*P*) or the force required to eject the parachute per unit of cross-sectional bulkhead area (*F*/*A*), volume of the parachute compartment (*V*), ejection charge mass (*m*), the specific gas constant for black powder (R), and the burning temperature (*T*) for black powder (Equation 1). The specific gas constant is 5.979 $\frac{ft \cdot lb}{slug \cdot {}^{\circ}R}$ and the burning temperature for black powder is 3307 °R. The force required to separate was set to 96.5 lb, which is enough force to shear three shear pins.

$$PV = \frac{F}{A}V = mRT$$

Equation 1: Ideal Gas Law

Parachute compartment volume was calculated using the formula for a cylinder, and Equation 1 was rearranged to solve for *m*.

The force required to shear a single shear pin was calculated with Equation 2, where σ_s is the tensile strength of the material and A_s is the minimum cross-sectional area. The shear pins that will be used are 2-56 nylon fasteners. They have a minor diameter of 0.064 in and a shear strength of 10,000 psi, which yields a required force of 32.17 lbs for a single shear pin.

$$F = \sigma_s A_s$$

Equation 2: Shear Force

The first separation point is between the central airframe and aft section. This separation will deploy the drogue parachute. The second separation point is between the nosecone and forward section (Figure 6). This separation will deploy the main parachute. The second separation point was changed from between the avionics bay and forward airframe to between the forward airframe and nosecone so that the ejection charges could be positioned directly next to the avionics bay, minimizing the risk of them moving unexpectedly during flight or separation events.

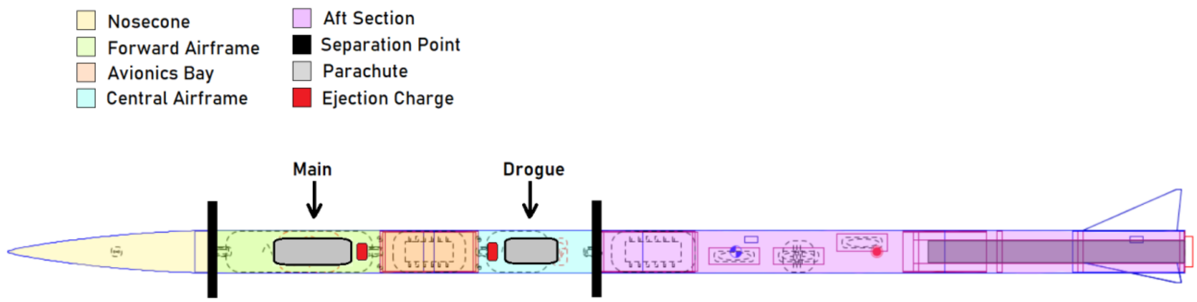

Figure 6: Separation Points

The first separation event will separate the central airframe and aft section, deploying the drogue parachute (Figure 7). This event will occur at apogee. The ejection charges for this event will be directly aft of the avionics bay and forward of the drogue parachute. The secondary charge will go off 1 second after apogee. A recovery harness will be used to connect the separated sections of the launch vehicle to each other and to the drogue parachute.

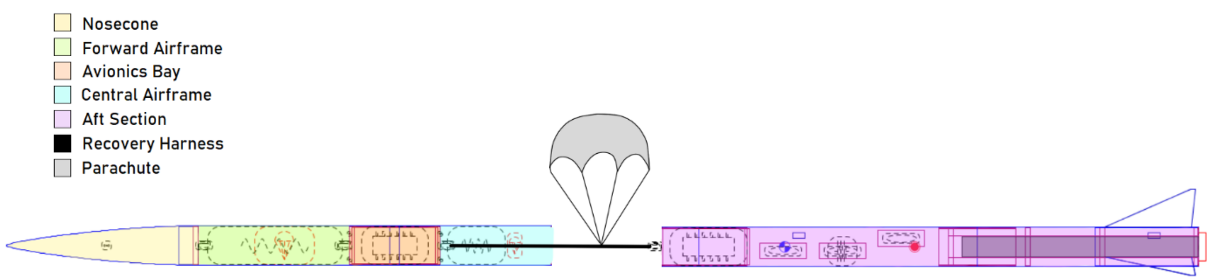

Figure 7: First Separation Event

The second separation event will separate the forward airframe and nosecone, deploying the main parachute (Figure 8). This event will occur at an altitude of 600 ft AGL, and the secondary ejection charge will go off at a delay of 50 ft, or an altitude of 550 ft. Both the primary and secondary ejection charges for this event will be directly forward of the avionics bay.

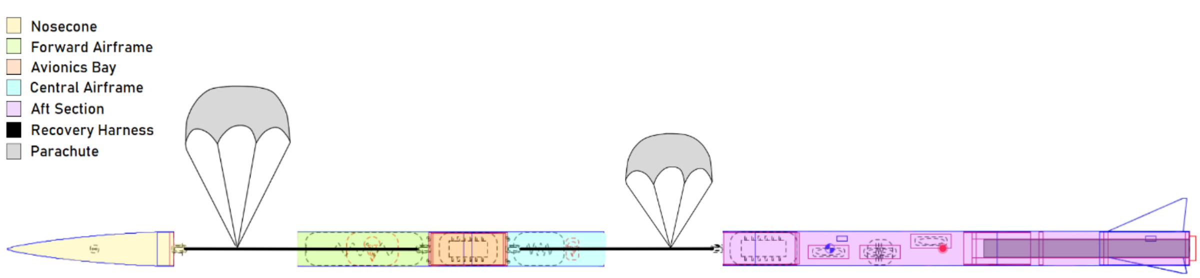

Figure 8: Second Separation Event

Quick links will be used to connect the recovery harness to eyebolts on the avionics bulkheads, payload bulkhead, nosecone bulkhead, and to the parachutes. The quick links will be oval shaped with a length of 2 in, a thickness of 9/32 in, and a carrying capacity of 1000 lbs. The quick link for the parachutes will be tied to the recovery harness 1/3 of the way from forward section to prevent the separated sections from colliding with each other during descent (Figure 9).

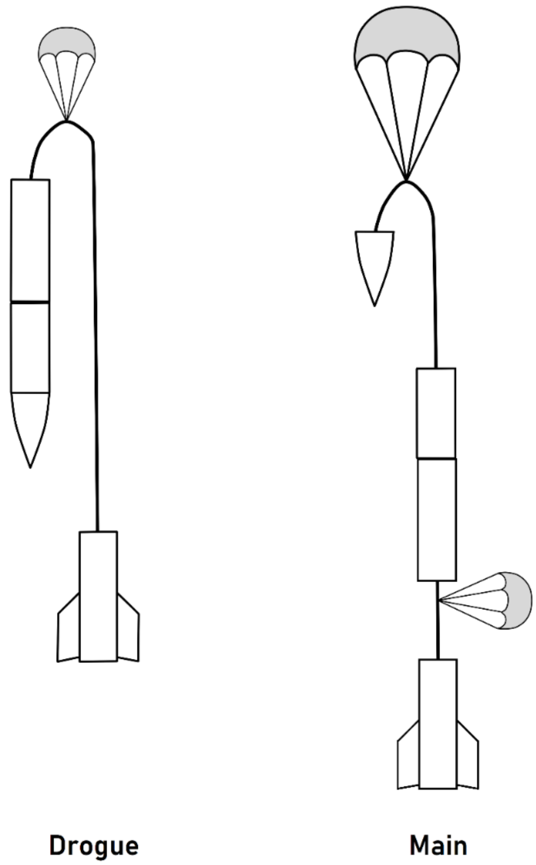

Figure 9: Vehicle Orientation After Parachute Deployment

3.1.4 Manufacturing Methods

The launch vehicle was modeled using Fusion360 and detailed drawings were created to aid with manufacturing. Assembly drawings were created to aid in building the launch vehicle once components are manufactured. All members who participated in manufacturing must have worn proper PPE, which included safety glasses, long pants, and closed-toe shoes. To use the machinery in the workspace, participants completed the required training which explains the safety hazards, proper use of the machinery, and emergency procedures. Depending on the material being used, other additional safety measures were taken.

3.1.4.1 Nosecone Section

To construct the nosecone section, an off-the-shelf nosecone needs to be purchased and the nosecone bulkhead and shoulder need to be manufactured. Rivet holes are drilled into both the nosecone and nosecone shoulder, so rivets can be installed to attach them together. Shear pin holes need to be drilled into the nosecone coupler so it can be attached to the forward airframe. To manufacture the nosecone section, a lathe, Roll-In bandsaw, and power tools are used.

3.1.4.1.1 Nosecone

The nosecone shoulder is manufactured using the Roll-In bandsaw and power tools. (Figure 10).

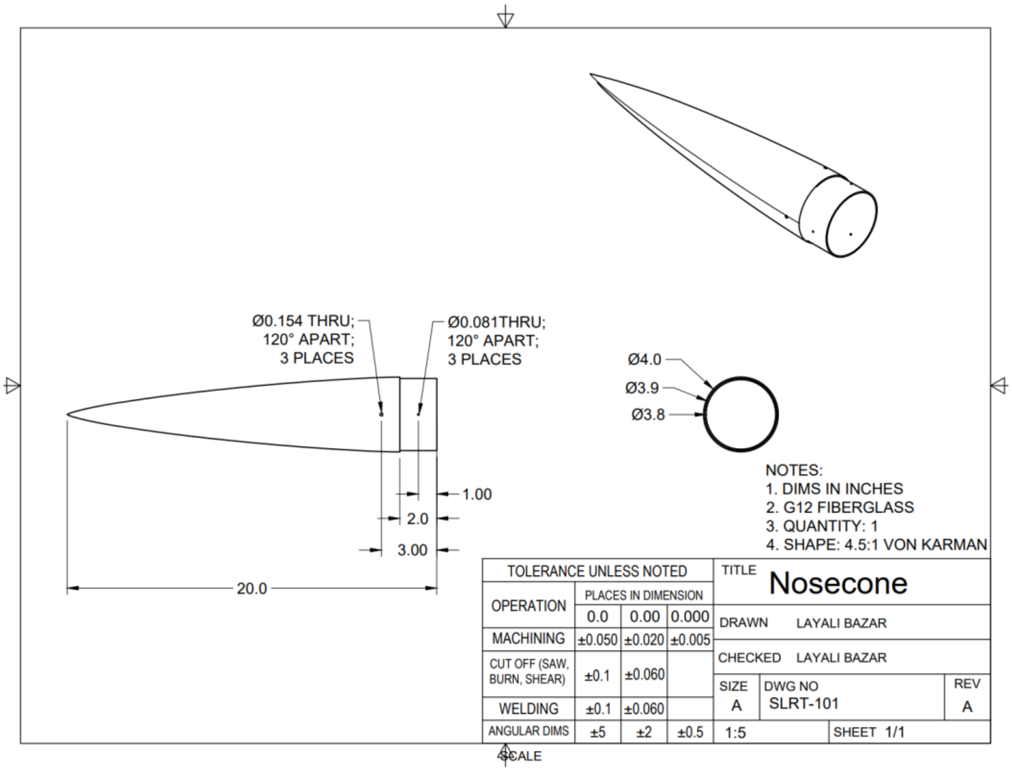

Figure 10: Nosecone Drawing

1. Put on safety glasses and respirator.
2. Mark three rivet holes 120.0° apart and 17.00 in from the tip of the nosecone.
3. Measure and mark 4.0 in of the nosecone coupler.
4. Clamp nosecone coupler in the Roll-In bandsaw vise.
5. Turn on vacuum and place near cutting area for fiber evacuation.
6. Cut nosecone coupler to size using the Roll-In bandsaw.
7. Debur edges of the coupler using 80-100 grit sandpaper.
8. Insert nosecone shoulder into the nosecone.
9. Clamp nosecone in a vise.
10. Drill one 0.154 in hole at a marked rivet hole.
11. Sand holes until smooth using sandpaper.
12. Insert a rivet into the drilled hole.
13. Repeat steps 12-14 until all three holes are drilled.
14. Clean machinery, work area, and workpiece.

3.1.4.1.2 Nosecone Bulkhead

A 0.25 in thick nosecone bulkhead was manufactured by using a lathe, Roll-In bandsaw, and power tools (Figure 11).

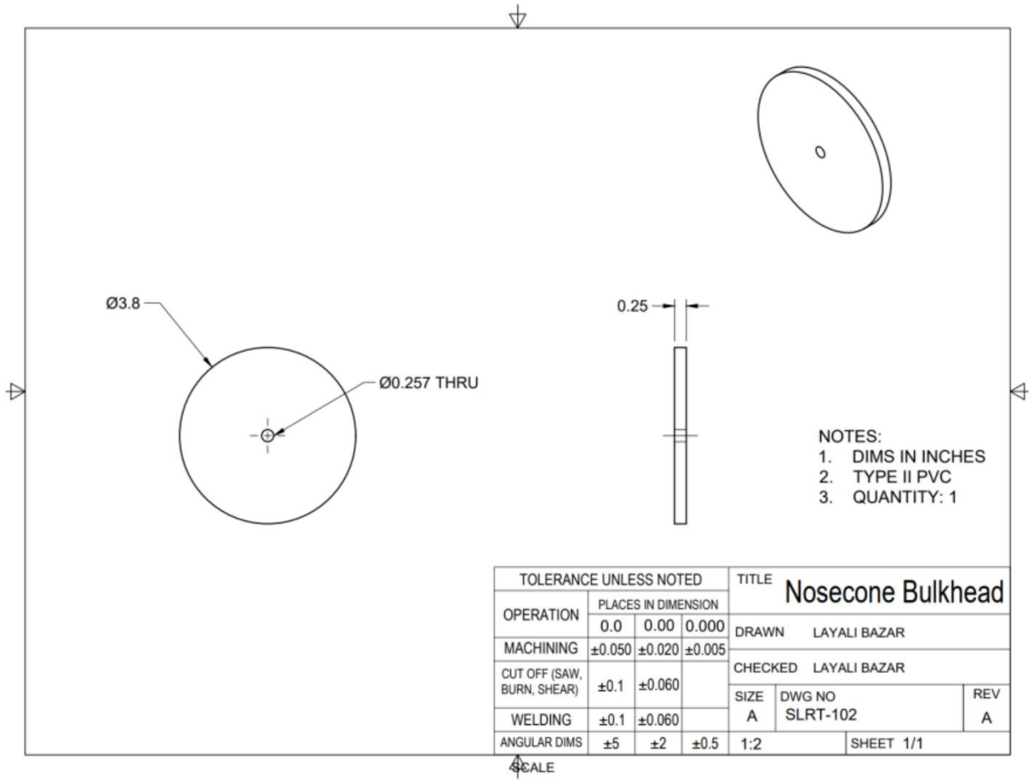

Figure 11: Nosecone Bulkhead Drawing

1. Put on safety glasses.
2. Measure raw type II PVC stock. Ensure there is at least 2.0 in of material to machine and at least 1 in to clamp into the chuck.
3. Measure nosecone shoulder inner diameter and PVC stock to determine the amount of material to remove.
4. Load stock into chuck and clamp material securely.
5. Load turning/facing tool into lathe tool post and ensure that it is aligned with spindle axis.
6. Lower safety guard to cover stock and chuck.
7. Turn lathe into high range and turn on machine.
8. Set speed to 800 RPM.
9. Turn the stock 0.25 in deep using 0.050 in roughing passes with auto feed. Use oil for all roughing passes.
10. Measure diameter of stock with calipers and test the fit of the stock with the nosecone shoulder.
11. Repeat steps 8-10 until stock's diameter is about 0.1 in larger than inner diameter of the shoulder.
12. Turn the stock using 0.020 in or less finishing passes with auto feed.
13. Measure the diameter of the stock with calipers and test the fit of the stock with nosecone shoulder.
14. Repeat steps 12-13 until stock fits in nosecone shoulder.
15. Load the center drill into the tailstock and center drill the part. Oil must be used during all drilling operations.
16. Remove the center drill and load the 0.25 in drill bit into tailstock. Drill 0.25 in deep through the center.

17. Remove the 0.25 in drill bit and load the 0.257 in drill bit into tailstock. Drill 0.257 in deep through the center.
18. Remove the workpiece and the tools.
19. Measure and mark 0.25 in. in length from the end of the material.
20. Clamp material into bandsaw vise.
21. Place oil onto material where the blade will cut and cut to size using the Roll-In bandsaw.
22. Clean machines and work area.
23. Clean workpiece and sand edges.

3.1.4.1.3 Nosecone Section Assembly

The steps necessary to assembly the nosecone section is listed (Figure 2).

1. Put on gloves in preparation for epoxying.
2. Attach an eyebolt to the bulkhead and secure with a nut.
3. Apply RocketPoxy around the eyebolt and nut.
4. Allow epoxy to cure for at least 6 hours.
5. Sand interior nosecone shoulder and nosecone bulkhead. Clean sanded area with paper towel to remove extra dust.
6. Apply RocketPoxy to the sanded area in the nosecone.
7. Insert bulkhead and twist into place.
8. Apply RocketPoxy between coupler and bulkhead to create a fillet.
9. Allow epoxy to cure for at least 6 hours.

3.1.4.2 Forward Section

The forward section consists of the forward airframe, the avionics bay coupler, avionics bay switchband, and two avionic bulkheads (Figure 3). The components are manufactured using a milling machine, lathe, Roll-In bandsaw, and power tools.

3.1.4.2.1 Forward Airframe

The forward airframe is manufactured using a bandsaw and power tools (Figure 12).

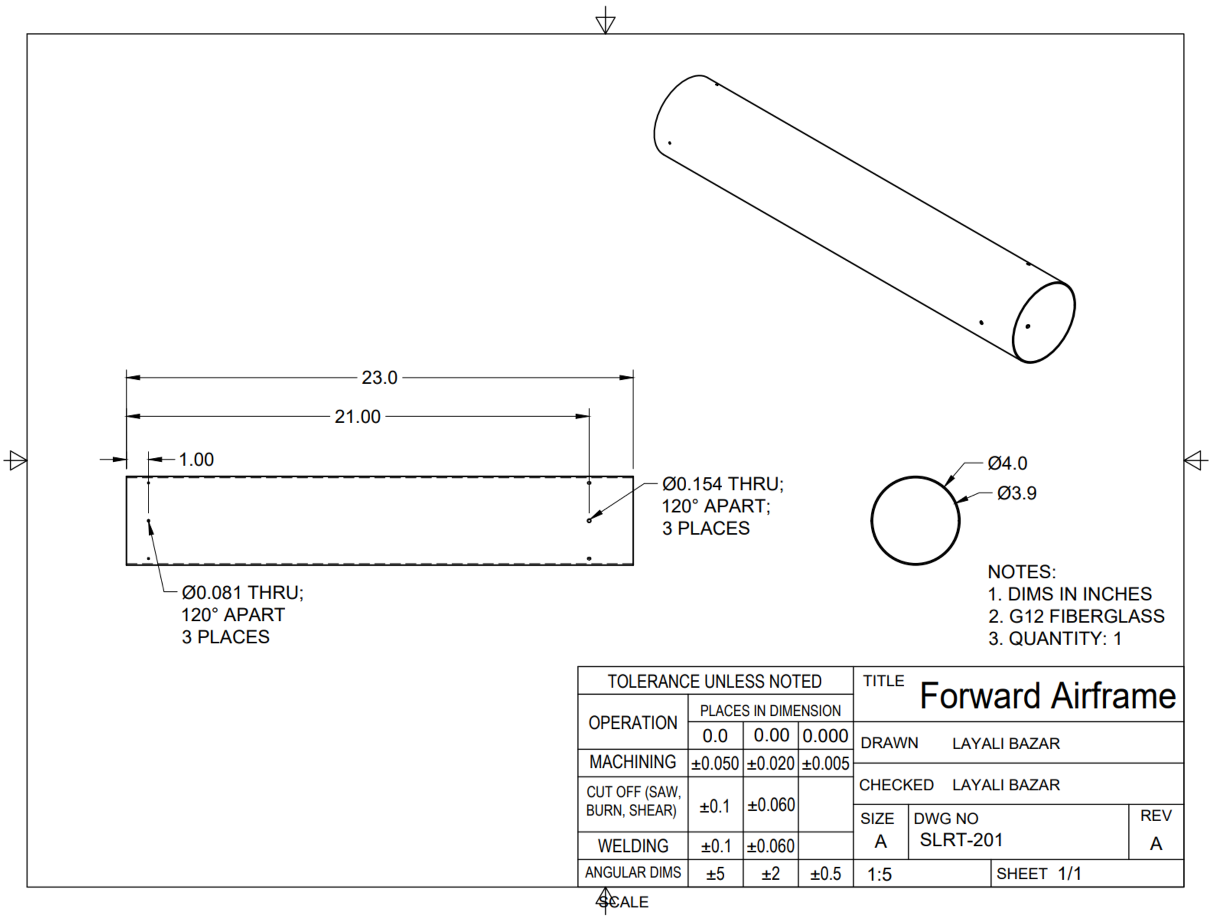

Figure 12: Forward Airframe Drawing

1. Put on safety glasses.
2. Measure and mark a length of 23.0 in on a G12 fiberglass airframe.
3. Put on respirator. Turn on vacuum and place near Roll-In bandsaw blade.
4. Cut airframe to size using bandsaw.
5. Debur edges of the airframe using 80-100 grit sandpaper.
6. Mark three shear pin holes that are 120.0° apart and 1.00 in from the forward end of the airframe.
7. Mark three rivet holes that are 120.0° apart and 21.00 in from the forward end of the airframe.
8. Clean machinery, work area, and workpiece.

3.1.4.2.2 Avionics Switchband

The avionics switchband is manufactured using the Roll-In bandsaw and power tools (Figure 13).

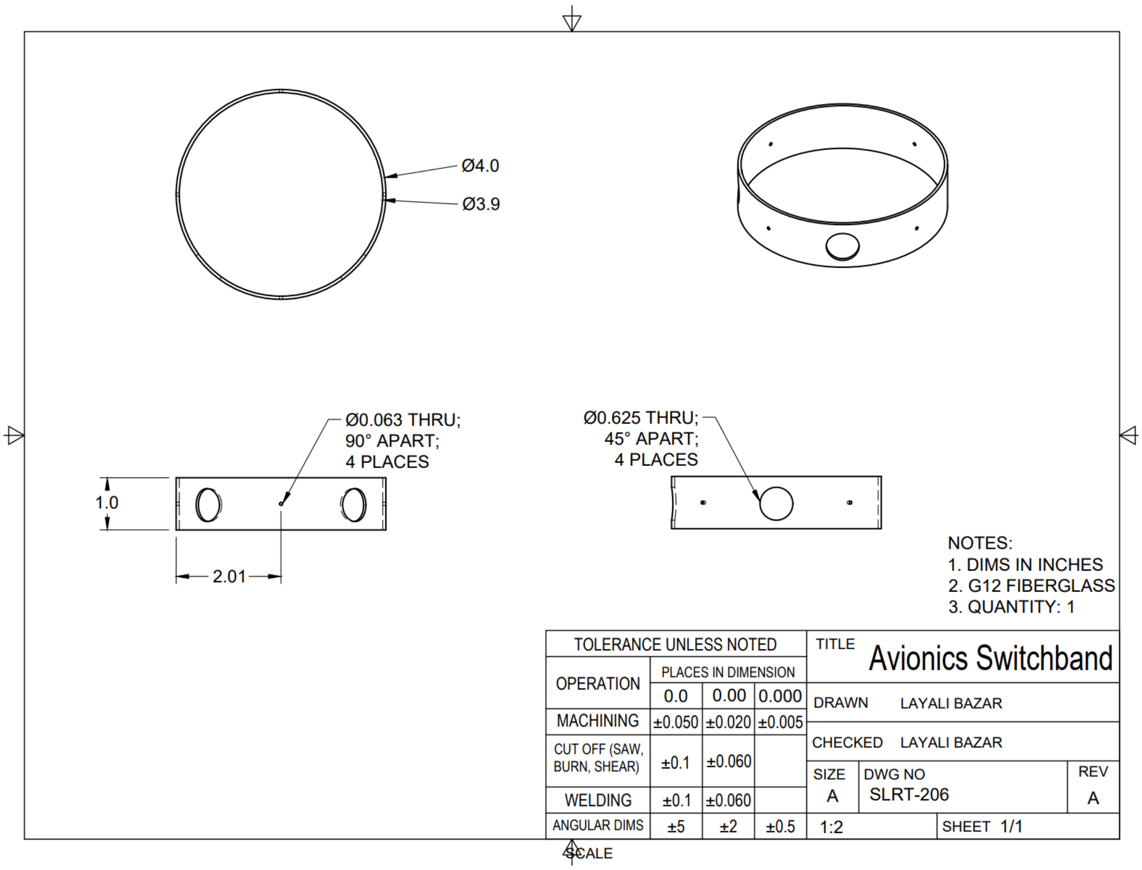

Figure 13: Avionics Switchband Drawing

1. Put on safety glasses.
2. Measure and mark 1.0 in of airframe.
3. Put on respirator. Turn on vacuum and place near cutting area.
4. Cut airframe to size using bandsaw.
5. Clean machinery and workpiece.
6. Debur edges using 80-100 grit sandpaper.
7. Mark two 0.625 in diameter holes, 90.0° apart, and 0.5 in from bottom of airframe.
8. Mark four 0.063 in diameter holes, 90.0° apart, and 0.5 in from bottom of part.
9. Clean work area.

3.1.4.2.3 Avionics Coupler

The avionics coupler is made by using the Roll-In bandsaw to cut to length. Holes for the keylock switches, rivets, and pressure points are also drilled (Figure 14).

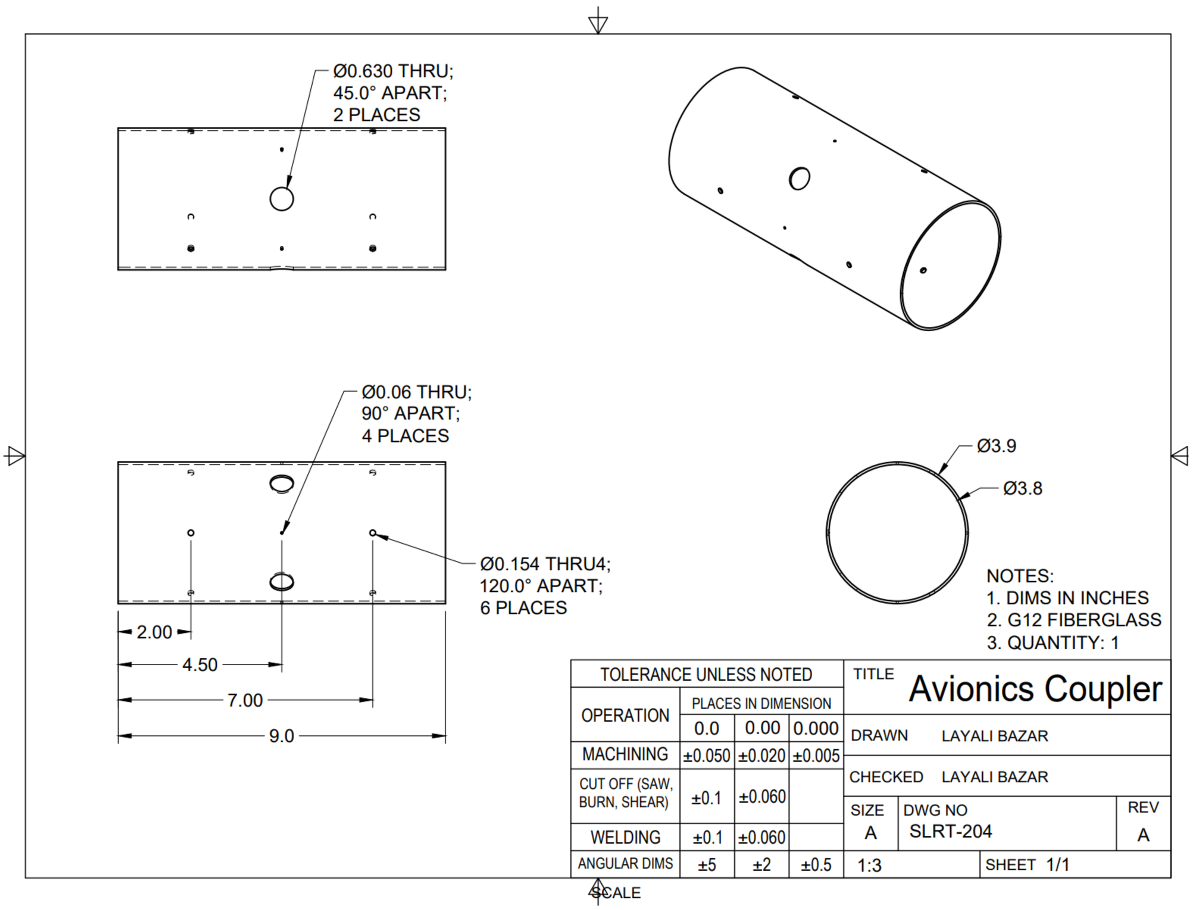

Figure 14: Avionics Coupler Drawing

1. Put on safety glasses.
2. Measure and mark 9.0 in of coupler.
3. Put on respirator. Turn on vacuum and place near cutting area.
4. Cut coupler to size using Roll-In bandsaw.
5. Clean machinery and workpiece.
6. Debur edges of coupler using 80-100 grit sandpaper.
7. Mark two lines onto the coupler: one at 4.0 in and another at 5.0 in from the aft end of the coupler.

3.1.4.2.4 Avionics Bulkheads

The avionics bulkheads are manufactured using the lathe, Roll-In bandsaw, and mill (Figure 15).

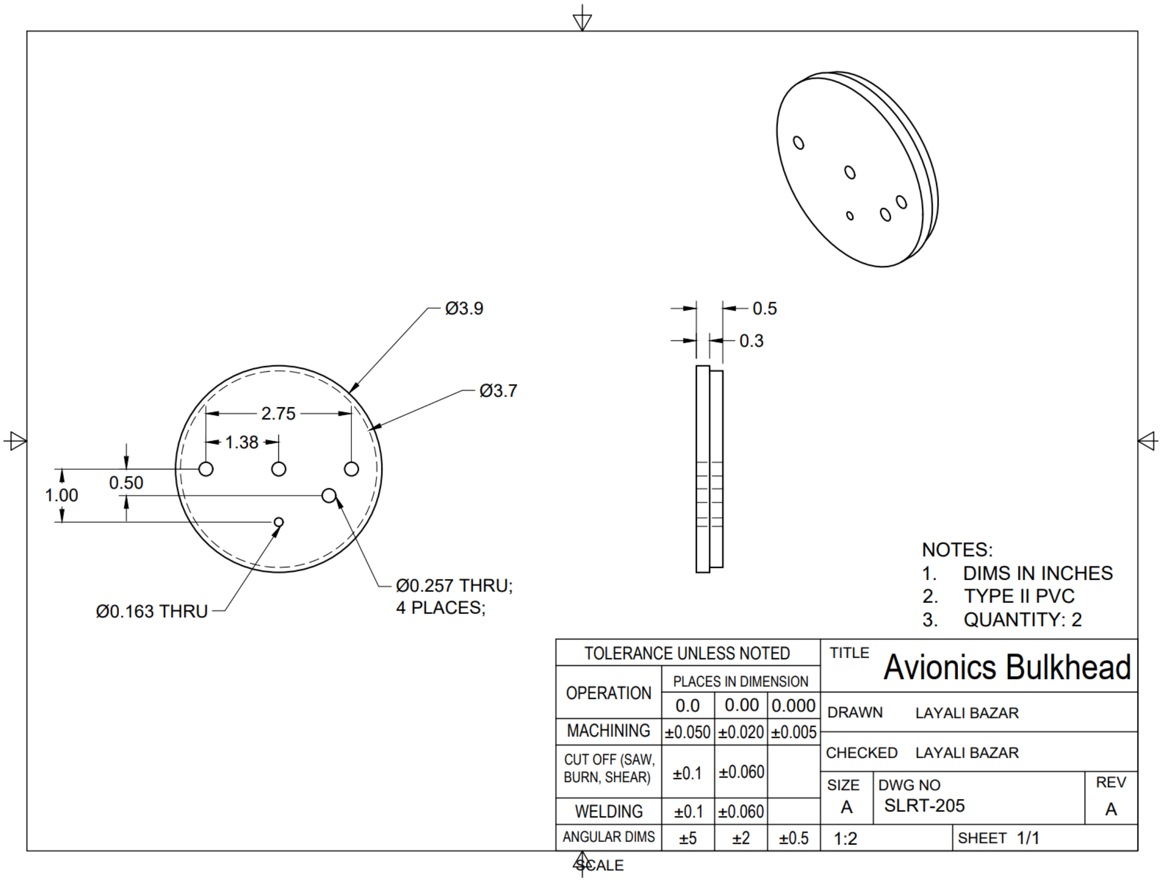

Figure 15: Avionics Bulkhead Drawing

1. Put on safety glasses.
2. Measure raw type II PVC stock. Ensure there is at least 2.0 in of material to machine and at least 1 in to insert into chuck.
3. Measure inner diameter of forward airframe and diameter of PVC to know how much material to remove.
4. Load stock into chuck and clamp material securely.
5. Load turning/facing tool into lathe tool post and ensure that it is aligned with spindle axis.
6. Lower safety guard to cover stock and chuck.
7. Turn lathe into high range and turn on machine.
8. Set speed to 800 RPM.
9. Turn the stock 0.5 in using 0.050 in rough passes with auto feed. Use oil during rough passes.
10. Turn off machine, measure the diameter of the stock using calipers, and test the fit of the stock with the airframe.
11. Repeat steps 8-9 until material is about 0.1 in away from the inner diameter of the airframe.
12. Turn the stock using 0.020 in or less finishing passes with auto feed.
13. Measure stock with calipers and test the fit of the stock with airframe.
14. Repeat steps 11-13 until stock fits in airframe.
15. Turn the stock 0.25 in using 0.05 in rough passes with auto feed. Use oil during rough passes.
16. Turn off machine and test the fit of the stock with the avionics coupler.

17. Repeat steps 15-16 until material is about 0.1 in away from the diameter of the avionics coupler.
18. Turn the stock using 0.020 in or less finishing passes with auto feed.
19. Measure the diameter of the stock using calipers and test the fit of the stock with avionics coupler.
20. Repeat steps 18-19 until stock fits in avionics coupler.
21. Load the center drill into the tailstock and center drill the part. Oil must be on drill bit during all drilling operations.
22. Remove the center drill and load the 0.5 in drill bit into tailstock. Drill 0.5 in deep through the center.
23. Remove the 0.25 in drill bit and place 0.257 in drill bit into tailstock. Drill 0.5 in deep through the center.
24. Remove the workpiece and the tools.
25. Measure and mark 0.5 in on the material.
26. Clamp material into bandsaw vise.
27. Place oil onto material where the blade will cut and cut into size using the Roll-In bandsaw.
28. Remove workpiece from vise.
29. Load the workpiece into milling machine vise using v-blocks.
30. Load a keyless chuck into the spindle with a cylindrical edge finder.
31. Zero the part with cylindrical edge finder using center hole.
32. Remove the edge finder and load a center drill into chuck. Place oil on bulkhead face.
33. Locate and center drill all holes.
34. Remove the center drill and load a 0.25 in drill bit into the chuck. Drill four thru holes.
35. Remove the 0.25 in drill bit and load a 0.257 in drill bit into the chuck. Drill four thru holes.
36. Remove the 0.257 in drill bit and load a 0.136 in drill bit into the chuck. Drill a hole.
37. Remove the part and tools.
38. Clean machines and work area.
39. Clean workpiece and sand edges.
40. Repeat steps 1-39 for second avionics bulkhead.

3.1.4.2.5 Avionics Bay Assembly

The procedure to assembly the avionics bay is listed. To assemble the avionics bay power tools are needed.

1. Wear gloves in preparation for epoxy.
2. Sand outer diameter of avionics coupler and inner diameter of the switchband.
3. Clean sanded area.
4. Place tape on the aftmost marked switchband line.
5. Apply RocketPoxy between the marked switchband lines.
6. Twist the switchband into place.
7. Remove tape and clean any excess epoxy.
8. Allow RocketPoxy to cure for at least six hours.
9. Clamp the component into the vise.
10. Drill a 0.25 in hole in one of the marked keylock switches on the avionics bay switchband.
11. Drill the hole up to 0.368 in.
12. Drill the hole up to 0.625 in.
13. Debur the hole.

14. Repeat steps 10-13 for the second keylock switch hole.
15. Drill a 0.060 in hole into one of the marked pressure holes.
16. Debur the hole.
17. Repeat steps 15-16 until all four pressure holes are drilled.
18. Clean the part and tools.

3.1.4.2.6 Central Airframe

The central airframe is made by using the Roll-In bandsaw and power tools (Figure 16).

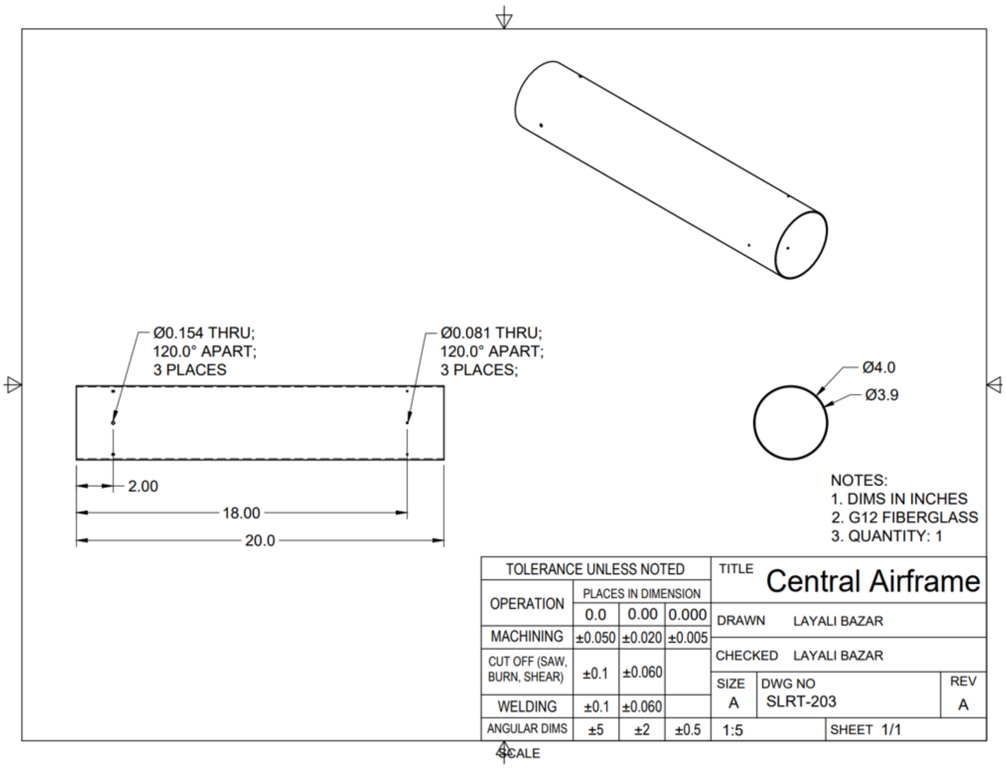

Figure 16: Central Airframe Drawing

1. Put on safety glasses.
2. Measure and mark 20.0 in on a G12 fiberglass airframe.
3. Put on respirator. Turn on vacuum and place near Roll-In bandsaw blade.
4. Cut airframe to size using bandsaw.
5. Sand edges of the airframe until smooth using 80-100 grit sandpaper.
6. Mark three rivet holes that are 120.0° apart and 2.00 in from the forward end of the airframe.
7. Mark three rivet holes that are 120.0° apart and 18.00 in from the forward end of the airframe.
8. Clean machinery, work area, and workpiece.

3.1.4.2.7 Forward Assembly

The procedure to assembly the forward section is provided below (Figure 3).

1. Insert the avionics bay 4.5 in into the forward airframe.
2. Secure component into a vise.
3. Drill a 0.154 in hole into a marked rivet hole.

4. Debur the hole.
5. Insert a rivet into the drilled hole.
6. Repeat steps 3-5 until all rivet holes are drilled.
7. Remove from vise.
8. Insert the aft end of avionics bay coupler 4.5 in into the central airframe.
9. Secure component into a vise.
10. Drill a 0.154 in hole into a marked rivet hole.
11. Debur the hole.
12. Insert a rivet into the drilled hole.
13. Repeat steps 10-12 until all rivet holes are drilled.
14. Remove from vise.

3.1.4.3 Aft Section

The aft section consists of a payload bay, payload airframe, payload aft coupler, aft airframe, three fins, and a motor assembly. The payload bay has two bulkheads and a coupler. The motor assembly has four centering rings and a motor tube. The tools and machines that will be used to manufacture the parts include a milling machine, lathe, bandsaw, abrasive waterjet, and power tools.

3.1.4.3.1 Payload Airframe

The payload airframe is manufactured using a Roll-In bandsaw, Dremel, and a power drill (Figure 17, Figure 18).

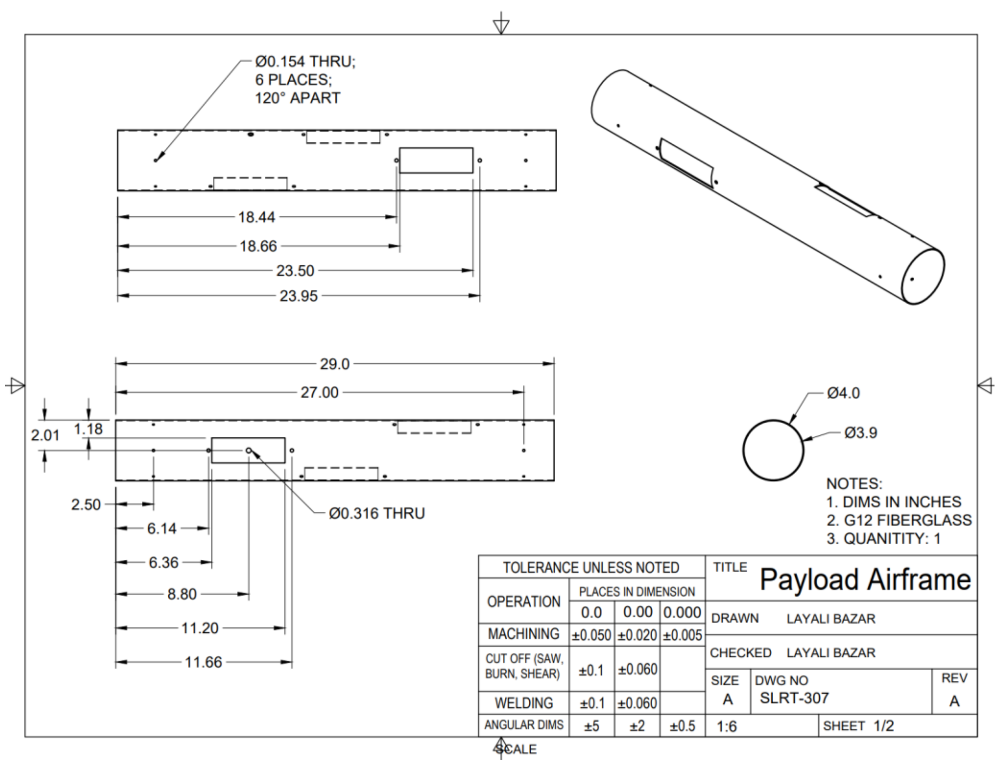

Figure 17: Payload Airframe Drawing

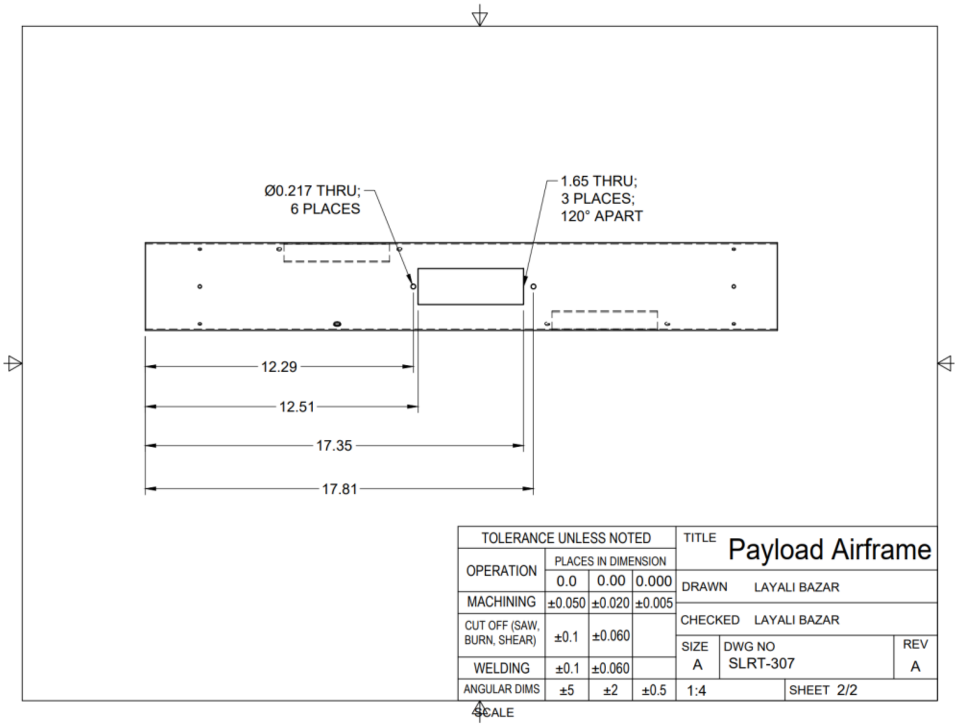

Figure 18: Payload Airframe Drawing Side View

1. Put on safety glasses.
2. Measure and mark 29.0 in on a G12 fiberglass airframe.
3. Put on respirator. Turn on vacuum and place near Roll-In bandsaw blade.
4. Cut airframe to size using bandsaw.
5. Sand edges of the airframe until smooth using 80-100 grit sandpaper.
6. Mark three rivet holes that are 120.0° apart and 2.50 in from the forward end of the airframe.
7. Mark three rivet holes that are 120.0° apart and 27.00 in from the forward end of the airframe.
8. Mark a rail button hole 8.80 in away from the forward end of the payload airframe.
9. Drill a 0.250 in diameter hole through the rail button hole.
10. Drill a 0.316 in through the same hole.
11. Debur the hole.
12. Mark the three slots 120.0° apart.
13. Use the Dremel to cut the marked slots.
14. Mark the six screw holes for the payload housings.
15. Drill a 0.217 in diameter hole through the screw hole mark using a power drill.
16. Repeat step 14 until all drill holes are made.
17. Debur all holes.
18. Sand inner diameter of airframe near the rail button hole using 80-100 grit sandpaper.
19. Clean the airframe.
20. Put on gloves in preparation for epoxy.
21. Apply RocketPoxy on a 1/4-20 t-nut.
22. Insert t-nut into drilled hole.
23. Clean machinery, work area, and workpiece.

3.1.4.3.2 Payload Bay Coupler

The payload coupler is manufactured using a Roll-In bandsaw and power tools (Figure 19).

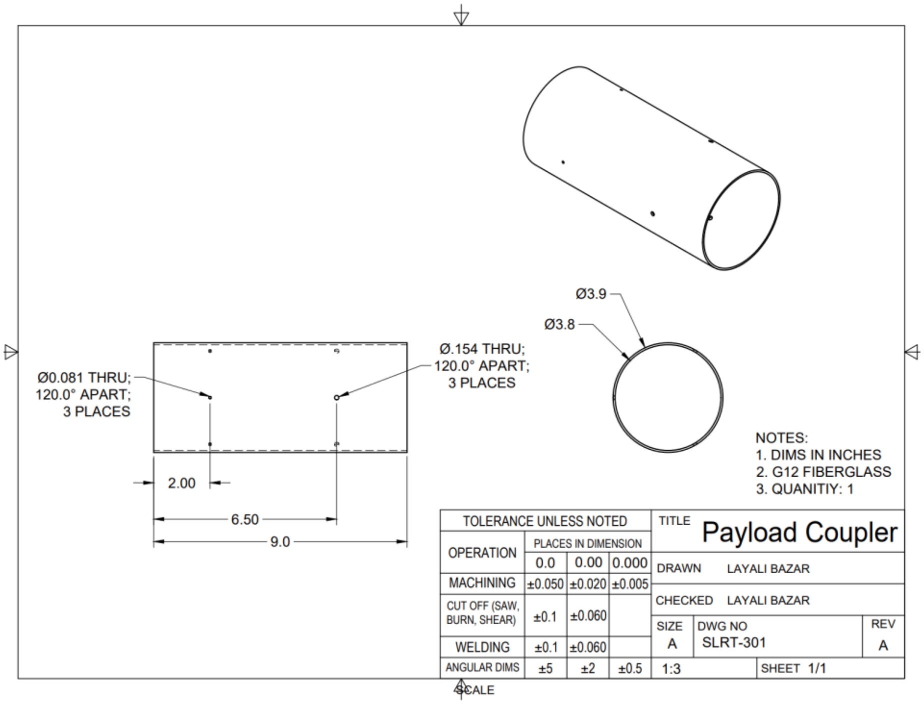

Figure 19: Payload Bay Drawing

1. Put on safety glasses.
2. Measure and mark 9.0 in on 3.898 in diameter coupler.
3. Put on respirator. Turn on vacuum and place near cutting area.
4. Cut coupler to size using Roll-In bandsaw.
5. Clean machinery and workpiece.
6. Sand edges of coupler using 80-100 grit sandpaper until smooth.

3.1.4.3.3 Payload Bay Forward Bulkhead

The payload bay forward bulkhead is manufactured using a lathe, Roll-In bandsaw, and milling machine (Figure 20).

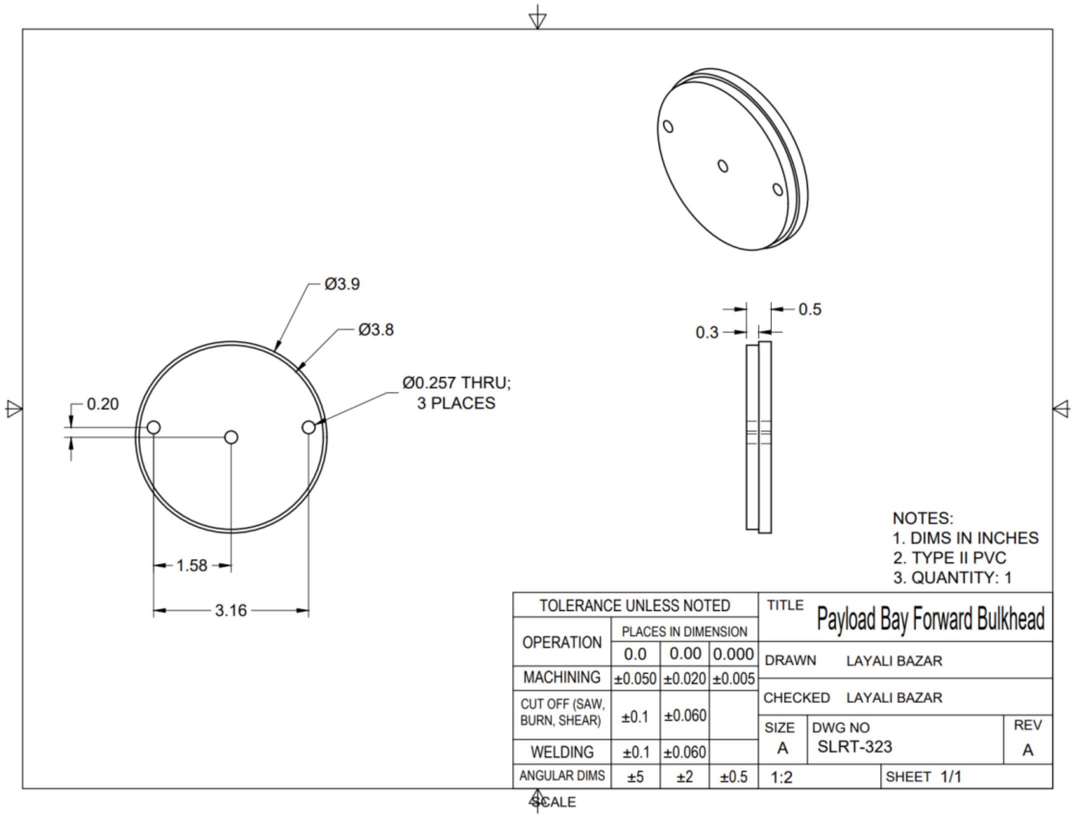

Figure 20: Payload Bay Forward Bulkhead Drawing

1. Put on safety glasses.
2. Measure raw type II PVC stock. Ensure there is at least 2.0 in of material to machine and at least 1 in to insert into chuck.
3. Measure inner diameter of central airframe and diameter of PVC to know how much material to remove.
4. Load stock into chuck and clamp material securely.
5. Load turning/facing tool into lathe tool post and ensure that it is aligned with spindle axis.
6. Lower safety guard to cover stock and chuck.
7. Turn lathe into high range and turn on machine.
8. Set speed to 800 RPM.
9. Turn the stock 0.5 in deep using 0.05 in rough passes with auto feed. Use oil during rough passes.
10. Turn off machine, measure the stock using calipers, and test the fit of the stock with the airframe.
11. Repeat steps 8-9 until material is about 0.1 in larger than the inner diameter of the airframe.
12. Turn the stock using 0.020 in or less finishing passes with auto feed.
13. Test the fit of the stock with airframe.
14. Repeat steps 12-13 until stock fits in airframe.
15. Turn the stock 0.3 in deep using 0.05 in rough passes with auto feed. Use oil during rough passes.
16. Turn off machine and test the fit of the stock with the payload bay coupler.
17. Repeat steps 15-16 until material is close to size of the inner diameter of the payload bay coupler.
18. Turn the stock using 0.020 in or less finishing passes with auto feed.
19. Measure the stock using calipers and test the fit of the stock with payload bay coupler.

20. Repeat steps 18-19 until stock fits in payload coupler.
21. Load the center drill into the tailstock and center drill the part. Oil must be used during all drilling operations.
22. Remove the center drill and load the 0.5 in drill bit into tailstock. Drill 0.5 in deep through the center.
23. Remove the 0.25 in drill bit and place 0.257 in drill bit into tailstock. Drill 0.5 in deep through the center.
24. Remove the workpiece and the tools.
25. Measure and mark 0.5 in on the material.
26. Clamp material into bandsaw vise.
27. Place oil onto material where the blade will cut and cut into size using the Roll-In bandsaw.
28. Remove workpiece from vise.
29. Load the workpiece into milling machine vise using v-blocks.
30. Load a keyless chuck into the spindle with a cylindrical edge finder.
31. Zero the part with cylindrical edge finder using center hole.
32. Remove the edge finder and load a center drill into chuck. Place oil on bulkhead face.
33. Locate and center drill all holes.
34. Remove the center drill and load a 0.25 in drill bit into the chuck. Drill two thru holes.
35. Remove the 0.25 in drill bit and load a 0.257 in drill bit into the chuck. Drill two thru holes.
36. Remove the part and tools.
37. Clean machines and work area.
38. Clean workpiece and sand edges.

3.1.4.3.4 Payload Bay Aft Bulkhead

The payload aft bulkhead was manufactured using the lathe, Roll-In bandsaw, and a milling machine (Figure 21).

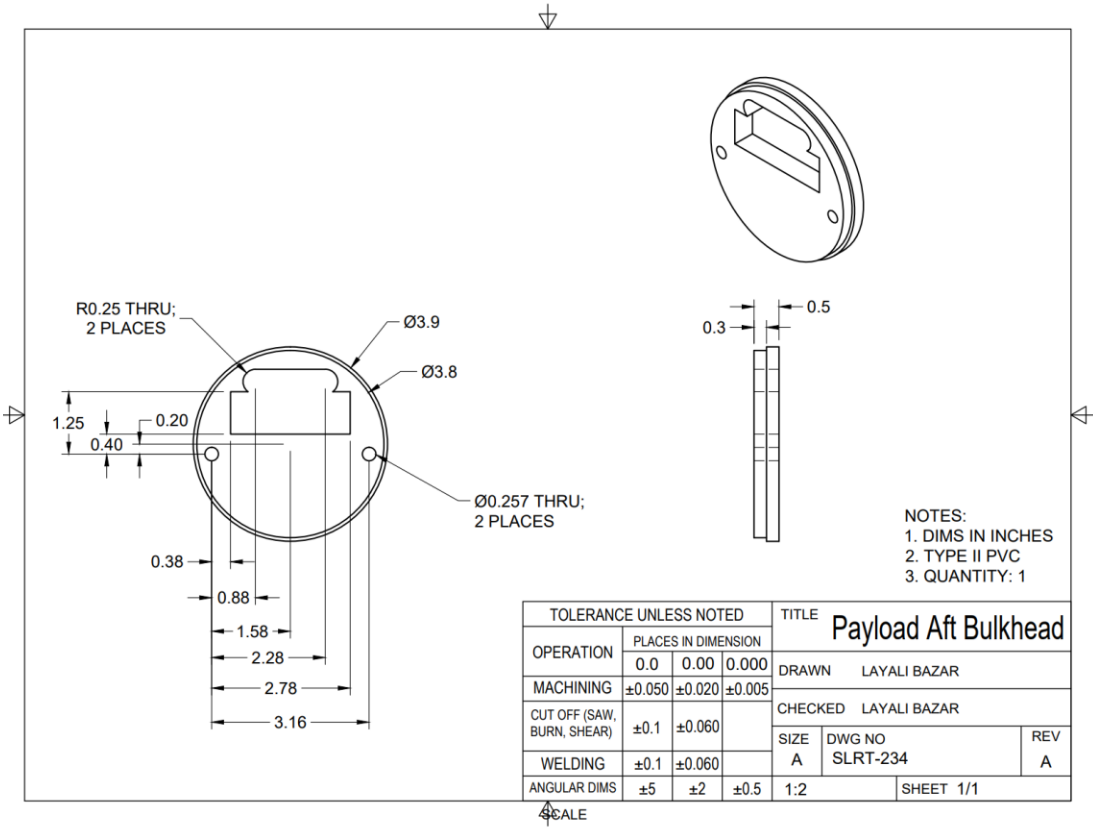

Figure 21: Payload Aft Bulkhead

1. Put on safety glasses.
2. Measure raw type II PVC stock. Ensure there is at least 2 in of material to machine.
3. Measure inner diameter of payload airframe and diameter of PVC to know how much material to remove.
4. Load stock into chuck and clamp material securely.
5. Load turning/facing tool into lathe tool post and ensure that it is aligned with spindle axis.
6. Lower safety guard to cover stock and chuck.
7. Turn lathe into high range and turn on machine.
8. Set speed to 800 RPM.
9. Turn the stock 0.5 in deep using 0.05 in rough passes with auto feed. Use oil during rough passes.
10. Turn off machine and test the fit of the stock with the airframe.
11. Repeat steps 9-10 until material is close to size of the inner diameter of the airframe.
12. Turn the stock using 0.020 in or less finishing passes with auto feed.
13. Test the fit of the stock with airframe.
14. Repeat steps 12-13 until stock fits in airframe.
15. Turn the stock 0.3 in deep using 0.05 in rough passes with auto feed. Use oil during rough passes.
16. Turn off machine, measure the stock using calipers, and test the fit of the stock with the payload bay coupler.
17. Repeat steps 15-16 until material is 0.1 in larger than the inner diameter of the payload bay coupler.
18. Turn the stock using 0.020 in or less finishing passes with auto feed.

19. Measure the stock using calipers and test the fit of the stock with payload bay coupler.
20. Repeat steps 18-19 until stock fits in payload coupler.
21. Load the workpiece into milling machine vise using v-blocks.
22. Load a keyless chuck into the spindle with a cylindrical edge finder.
23. Zero the part with cylindrical edge finder.
24. Remove the edge finder and load a center drill into chuck. Place oil on bulkhead face.
25. Locate and center drill all holes.
26. Remove the center drill and load a 0.25 in drill bit into the chuck. Drill two thru holes.
27. Remove the 0.25 in drill bit and load a 0.257 in drill bit into the chuck. Drill two thru holes.
28. Zero the part using the center of the left 0.257 in hole.
29. Remove 0.257 in drill bit and load an endmill into the chuck.
30. Cut slots on part using endmill.
31. Remove the part and tools.
32. Clamp material into bandsaw vise.
33. Place oil onto material where the blade will cut and cut to size using the bandsaw.
34. Remove the part from the vise.
35. Clean machines and work area.
36. Clean workpiece and sand edges with 80-100 grit sandpaper.

3.1.4.3.5 Payload Aft Coupler

The payload aft coupler is manufactured using a Roll-In bandsaw and power tools (Figure 22).

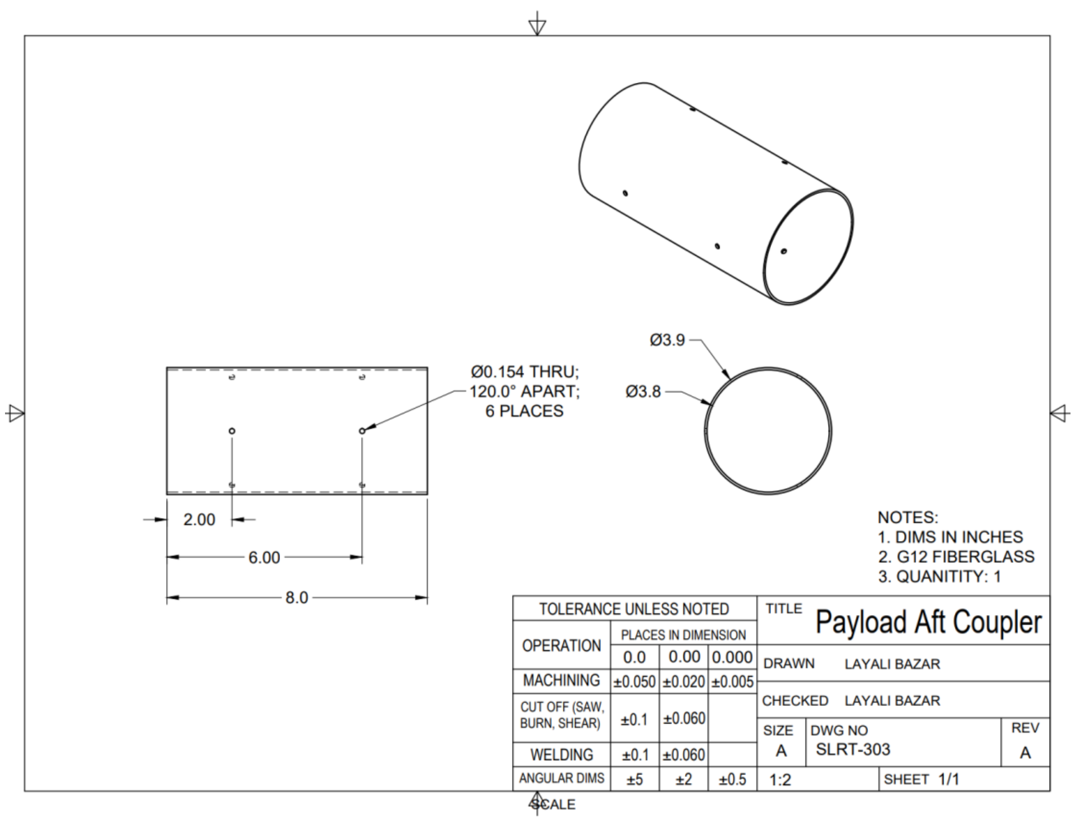

Figure 22: Payload Aft Coupler

1. Put on safety glasses.
2. Measure and mark 8.0 in on 3.898 in diameter coupler.
3. Put on respirator.
4. Turn on vacuum and place near cutting area.
5. Cut coupler to size using Roll-In bandsaw.
6. Clean machinery and workpiece.
7. Sand edges of coupler using 80-100 grit sandpaper until smooth.

3.1.4.3.6 Fins

The three fins are manufactured using an abrasive waterjet (Figure 23).

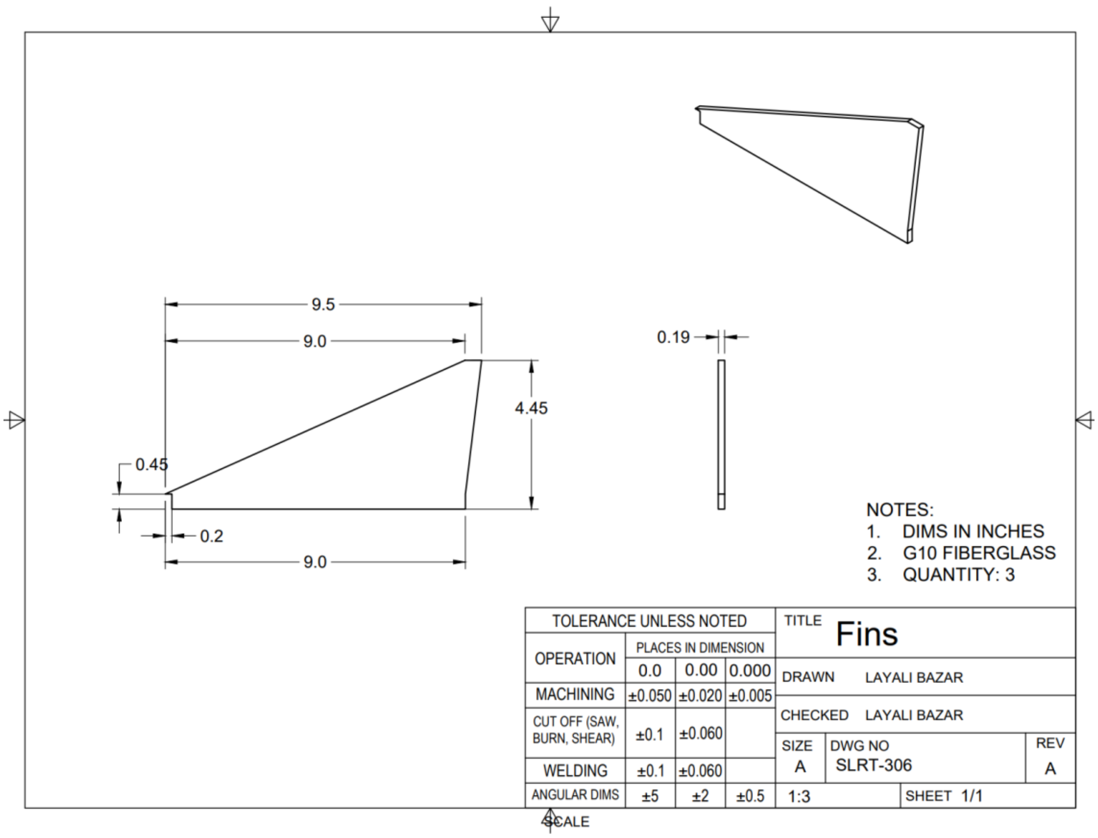

Figure 23: Fin Drawing

1. Put on safety glasses.
2. Model a fin in Solidworks and export file to a 1:1 .dxf file.
3. Put on ear protection.
4. Create cutting path for three fins and save as .ord file.
5. Open .ord file and select material type and thickness.
6. Place and clamp material in cutting area.
7. Verify material is aligned with x and y axes of work area.
8. Set z-axis of water jet nozzle.
9. Run the waterjet and cut the path.
10. Remove material.

11. Clean machinery, work area, and workpiece.
12. Sand fins using 80-100 grit sandpaper.

3.1.4.3.7 Aft Airframe

The aft airframe was manufactured using a Roll-In bandsaw, milling machine, and power tools (Figure 24).

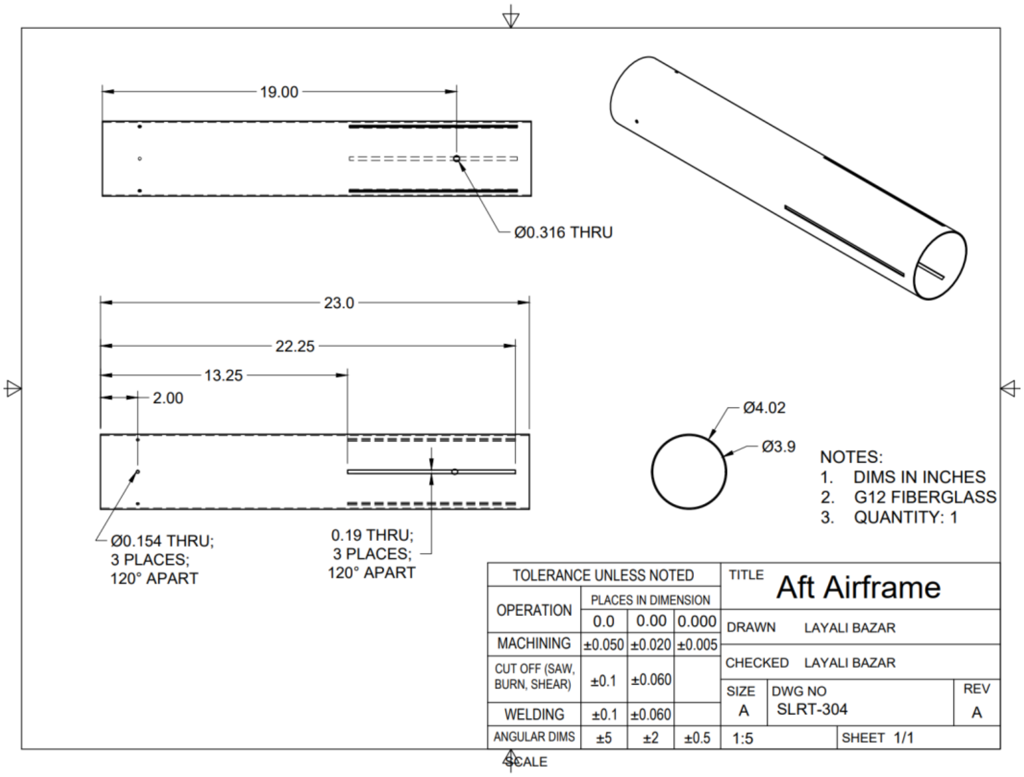

Figure 24: Aft Airframe Drawing

1. Put on safety glasses.
2. Measure and mark 23.0 in on a G12 fiberglass airframe.
3. Put on respirator. Turn on vacuum and place near Roll-In bandsaw blade.
4. Cut airframe to size using bandsaw.
5. Sand edges of the airframe until smooth using 80-100 grit sandpaper.
6. Mark three rivet holes that are 120.0° apart and 2.00 in from the forward end of the airframe.
7. Mark a rail button hole 19.00 in from the forward end of the airframe between two fins.
8. Ensure both rail button holes align.
9. Clamp component into vise.
10. Drill a 0.25 in diameter hole through the rail button hole.
11. Drill the hole up to 0.316 in.
12. Debur the hole.
13. Remove airframe from the vise.
14. Mark three fin slots, 120.0° apart and 22.25 in from the forward end of the airframe.
15. Load workpiece into milling machine custom vise.
16. Load a keyless chuck into the spindle with a cylindrical edge finder.
17. Zero the part using the v-blocks and aft end of airframe.

18. Remove the cylindrical edge finder and load a 1/8 in endmill into the chuck.
19. Cut the fin slot with the endmill along the markings.
20. Debur the slot.
21. Measure thickness of fins and slot using calipers.
22. Test the fit of the fins in slot.
23. Remove endmill from spindle.
24. Repeat steps 11-18 until all three fin slots are completed.
25. Remove part and tools from milling machine.
26. Sand inner diameter of airframe near the rail button hole using 80-100 grit sandpaper.
27. Clean the airframe.
28. Put on gloves in preparation for epoxying.
29. Apply RocketPoxy on a 1/4-20 t-nut.
30. Insert t-nut into drilled hole and allow RocketPoxy to cure for 6 hours.
31. Clean machinery, work area, and workpiece.

3.1.4.3.8 Centering Rings

The four centering rings are manufactured using an abrasive waterjet (Figure 25).

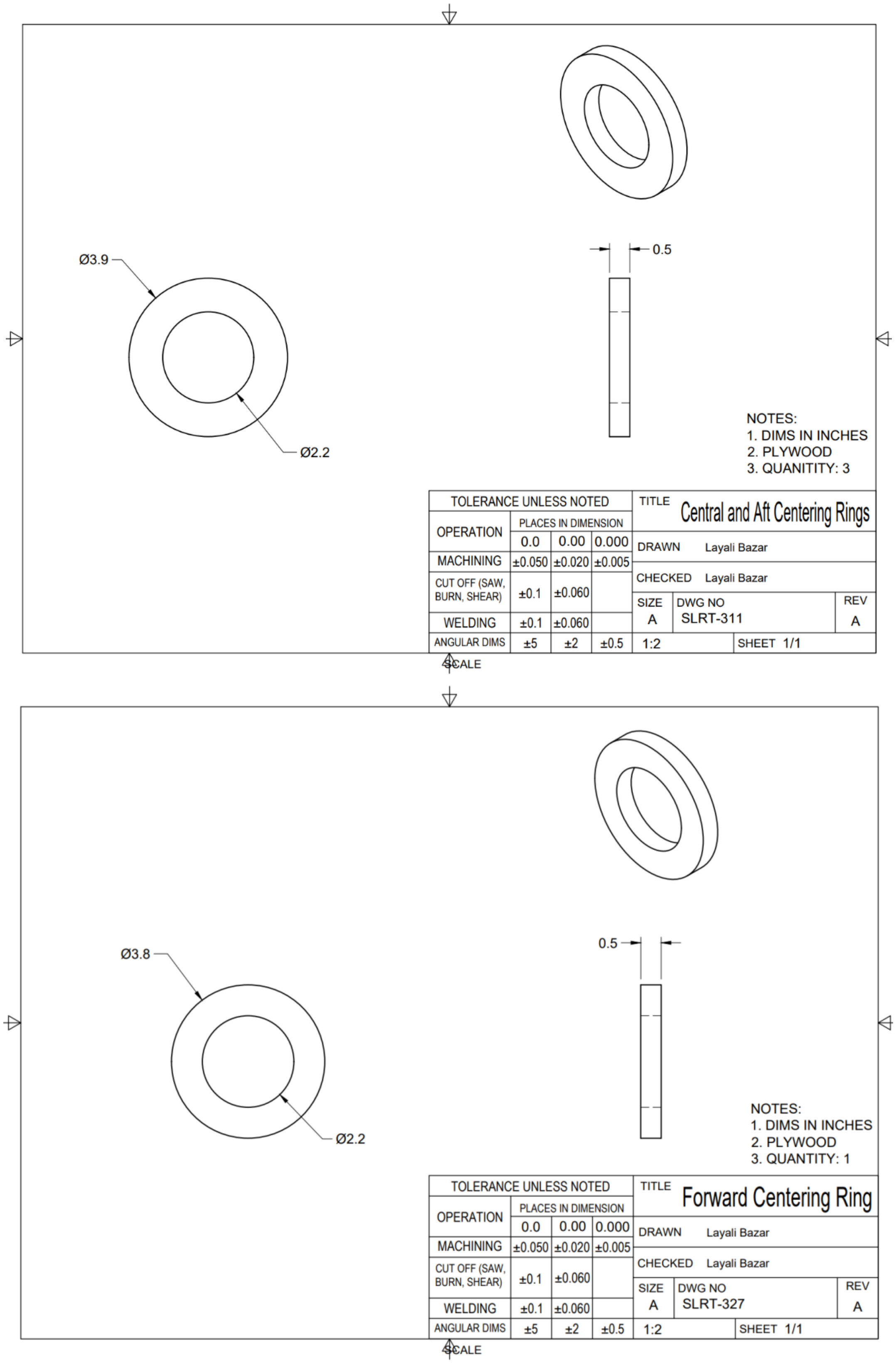

Figure 25: Centering Rings Drawing

1. Put on safety glasses.
2. Model a centering ring in Solidworks and export file to a 1:1 .dxf file.
3. Put on ear protection.
4. Create cutting path for three centering rings and save as .ord file.
5. Open .ord file and select material type and thickness.
6. Place and clamp material in waterjet area.
7. Verify material is aligned with x and y axes of work area.
8. Set z-axis of water jet nozzle.
9. Run the waterjet and cut the path.
10. Repeat steps 4-9 for the forward centering ring.
11. Remove material.
12. Clean machinery, work area, and workpiece.
13. Sand centering rings using 80-100 grit sandpaper.

3.1.4.3.9 Motor Tube

The motor tube was manufactured using a bandsaw (Figure 26).

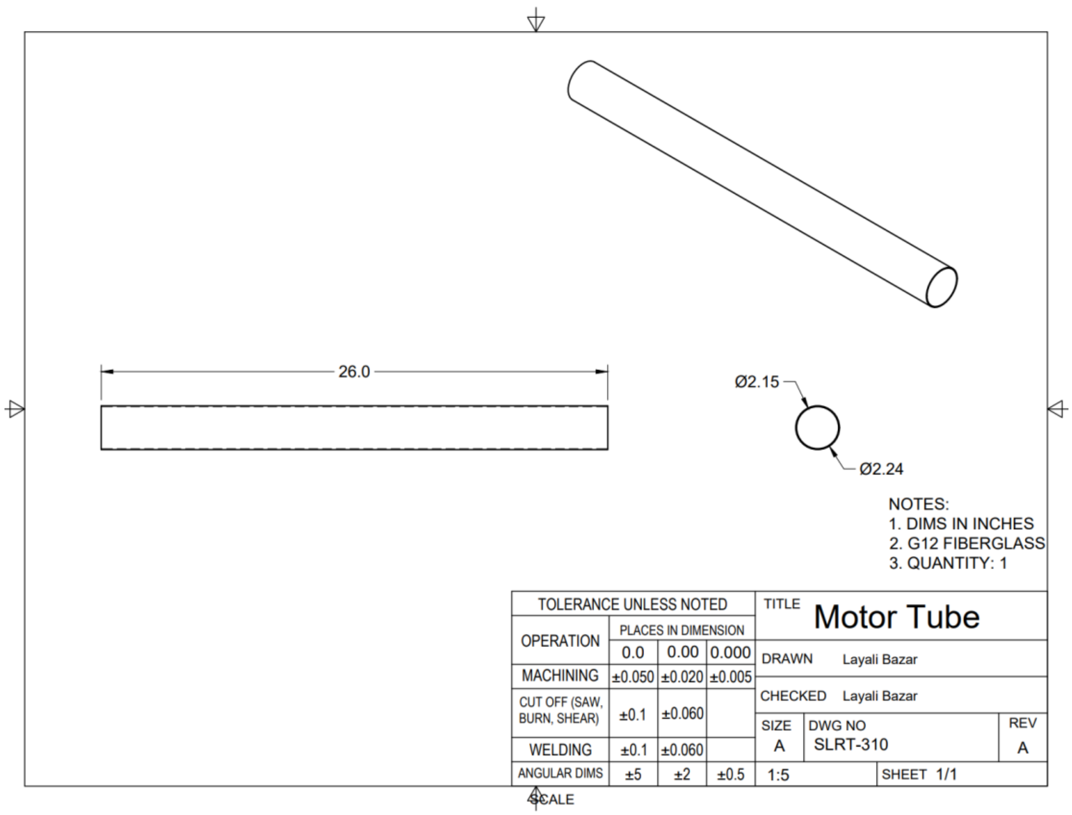

Figure 26: Motor Tube Drawing

1. Put on safety glasses.
2. Measure and mark 26.0 in on 2.24 in diameter fiberglass motor tube.
3. Put on respirator. Turn on vacuum and place near Roll-In bandsaw blade.
4. Cut motor tube to size using bandsaw.
5. Sand edges of the motor tube until smooth using 80-100 grit sandpaper.

3.1.4.3.10 Motor Tube Assembly

1. Put on gloves in preparation for epoxying.
2. Mark a line 9.75 in away from the aft end of the motor tube for the central centering rings.
3. Apply a layer of epoxy.
4. Twist the central centering ring in place 10.25 in away from the aft end of the motor tube.
5. Mark a line 17.50 in away from the aft end of the motor tube for the second central centering rings.
6. Apply a layer of epoxy.
7. Twist the second central centering ring in place 17.50 in away from the aft end of the motor tube for the second central centering rings.
8. Apply epoxy at the forward end of the motor tube.
9. Twist the forward centering in place flush with the end of the motor tube.
10. Allow epoxy to cure for at least 6 hours.

3.1.4.3.11 Aft Assembly

1. Secure three payload housings in payload airframe with 8-32 screws.
2. Insert the payload bay coupler 5.0 in into the payload airframe.
3. Secure component into a vise.
4. Drill a 0.154 in hole into a marked rivet hole.
5. Debur the hole.
6. Insert a rivet into the drilled hole.
7. Repeat steps 4-6 until all rivet holes are drilled.
8. Remove from vise.
9. Insert the payload aft coupler 4.0 in into the aft end of the payload airframe.
10. Secure component into vise.
11. Drill a 0.154 in hole into a marked rivet hole.
12. Debur the hole.
13. Insert a rivet into the drilled hole.
14. Repeat steps 11-13 until all rivet holes are drilled.
15. Remove from vise.
16. Put on gloves in preparation for epoxying.
17. Sand outer diameter of aft payload coupler and inner diameter of aft airframe with 80-100 grit sandpaper.
18. Clean all sanded surfaces.
19. Apply layer of JBWeld epoxy 4.0 in from aft end of aft payload coupler.
20. Insert the payload aft coupler 4.0 in into the aft airframe.
21. Allow epoxy to cure for at least 6 hours.
22. Secure component into vise.
23. Drill a 0.154 in hole into a marked rivet hole.
24. Debur the hole.
25. Insert a rivet into the drilled hole.
26. Repeat steps 23-25 until all rivet holes are drilled.
27. Remove from vise.
28. Sand forward end of the aft airframe with 80-100 grit sandpaper.
29. Clean all sanded surfaces.

30. Put on gloves in preparation for epoxying.
31. Remove rivet from aft airframe.
32. Apply JBWeld on the rivet and aft airframe.
33. Allow epoxy to cure for 6 hours.
34. Repeat steps 31-33 until the three aft airframe rivets are epoxied in place.
35. Put on gloves in preparation for epoxying.
36. Sand all fins, motor tube, and inner and outer diameter of aft airframe using 80-100 grit sandpaper.
37. Clean all sanded surfaces.
38. Apply a layer of JB Weld to the inside of the aft payload coupler 1.0 in from the forward end of coupler.
39. Apply a layer of JB Weld to the inside of the aft airframe 5.0 in from the forward end of airframe.
40. Apply a layer of JB Weld to the inside of the aft airframe 12.25 in from the forward end of airframe.
41. Insert motor assembly, so motor tube is flush with aft end of the airframe.
42. Allow epoxy to cure for at least 6 hours.
43. Apply cyanoacrylate adhesive to root of a fin and attach to the motor tube through the fin slots.
44. Allow to dry for 15 minutes.
45. Repeat steps 43-44 until all three fins are set.
46. Apply painter's tape 0.25 in above and below the fin on the aft airframe.
47. Apply painter's tape on both sides of the fins, 0.25 in above the fin slot.
48. Apply painter's tape 0.25 in from the fin slot to the airframe.
49. Repeat steps 46-48 for the three fins.
50. Apply JB Weld to create two fillets where the two upper fins contact the motor tube.
51. Apply JB weld to create two fillets where the lower fin contacts the aft airframe.
52. Apply RocketPoxy to the exterior of the aft airframe where the upper fins touch the airframe.
53. Smooth out exterior fillets with popsicle stick.
54. Allow epoxy to dry for at least 6 hours.
55. Repeat steps 50-54 two more times while rotating the airframe to ensure all fillets are made at each connection point.
56. Apply JB Weld onto end of motor tube and the inner aft airframe.
57. Insert aft centering ring until flush with fins.
58. Allow epoxy to cure for 6 hours.
59. Install three #10 threaded inserts into aft centering ring, spaced 120 degrees apart.
60. Insert thrust plate into aft airframe and screw in three #10 flathead fasteners.

3.1.4.4 Overall Vehicle Assembly

1. Insert nosecone shoulder 2.0 in into the forward airframe.
2. Secure components into a vise.
3. Drill a 0.081 in hole at the marked shear pin location.
4. Debur the hole.
5. Insert a shear pin into the drilled hole.
6. Repeat steps 3-5 until all three marked shear pin holes are drilled.
7. Remove from vise.
8. Insert payload bay 4.0 in into the central airframe.
9. Secure components into a vise.

10. Drill a 0.081 in hole into a marked shear pin hole.
11. Debur the hole.
12. Insert a shear pin into the drilled hole.
13. Repeat steps 10-12 until all three marked shear pin holes are drilled.
14. Remove from vise.
15. Clamp forward rail button into the vise.
16. Drill 0.316 in diameter hole into the rail button.
17. Repeat step 23 for the aft rail button.
18. Secure both rail buttons into the t-nut using a ¼-20 flat head screw.
19. Wet sand the exterior of the launch vehicle.
20. Prime and paint the launch vehicle.

3.1.5 Design Justification

3.1.5.1 Material Justification

Alternative materials were evaluated using decision matrices and a set of objectives for each component of the launch vehicle. Qualitative score assessments were derived from a predetermined qualitative score assignment (Table 3).

Qualitative Score	
Great	10
Good	8
Okay	6
Fair	4
Poor	2

Table 3: Qualitative Scores

3.1.5.1.1 Nosecone

The two materials considered for the nosecone were polypropylene and G12 fiberglass (Table 4). Each material was evaluated based on its cost, density, and tensile strength. The selected material was G12 fiberglass since the tensile strength and compressive strength is much higher than polypropylene. The nosecone must be able to withstand tensile and compressive forces from the ejection events and landing. G12 fiberglass scored well, in terms of density, because it caused the center of gravity to move forward making the launch vehicle more stable. Even though the cost of the G12 fiberglass nosecone is greater, the vehicle performance relies on the density, tensile strength, and compressive strength. Therefore, the tensile strength, compressive strength, and density of G12 fiberglass were the reasons it was the selected material for the nosecone.

Nosecone			Polypropylene			G12 Fiberglass		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.1	USD	24.75	10.0	1.00	75.9	3.3	0.33
Density	0.30	lb/in^3	0.03	4.5	1.34	0.067	10.0	3.00
Tensile Strength	0.30	ksi	6.50	0.6	0.18	115	10.0	3.00
Compressive Strength	0.30	ksi	5.80	1.9	0.58	30.00	10.0	3.00

Overall value			3.10			9.33

Table 4: Nosecone Decision Matrix

3.1.5.1.2 Airframe and Couplers

For the airframe and coupler, four different materials were evaluated based on their cost, density, compressive strength, and machinability (Table 5). These include blue tube, G12 fiberglass, phenolic, and quantum tube. The airframe and couplers need to be able to withstand compressive forces caused by launch and landing. The main components of the launch vehicle are its airframe and couplers, so it is important for machining to not be time consuming and for the overall mass to minimized. Cost was also evaluated to stay within the team's budget. The cost, density, and machinability of G12 fiberglass do not impact the performance of the launch vehicle so the material was selected based on its compressive strength. Since G12 fiberglass has a high compressive strength and scored the highest compared to the other materials, it was selected as the material for airframe and couplers.

Airframe			**Blue Tube**			**G12 Fiberglass**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.17	USD/in	0.92	8.04	1.34	2.12	3.49	0.58
Density	0.17	lb/in^3	0.05	10.00	1.67	0.07	6.19	1.03
Compressive Strength	0.50	ksi	4.28	1.43	0.71	30.00	10.00	5.00
Machinability	0.17	experience	good	8.00	1.34	okay	6.00	1.00
Overall value					**5.1**			**7.6**
Airframe			**Phenolic**			**Quantum Tube**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.17	USD/in	0.74	10.00	1.67	0.87	8.51	1.42
Density	0.17	lb/in^3	0.05	9.83	1.64	0.05	9.42	1.57
Compressive Strength	0.50	ksi	13.50	4.50	2.25	18.20	6.07	3.03
Machinability	0.17	experience	good	8.00	1.34	good	8.00	1.34
Overall value					**6.9**			**7.4**

Table 5: Airframe and Couplers Decision Matrix

3.1.5.1.3 Motor Tube

Three different materials were considered for the motor tube. These include G12 fiberglass, blue tube, and phenolic (Table 6). The material selected for the motor tube needs to be able to resist compressive forces from the motor upon launch. G12 fiberglass scored the highest overall compared to all the materials, especially in compressive strength. The cost, density, and machinability of the motor tube does not impact the overall performance of the launch vehicle, so they were weighed less compared to compressive strength. G12 fiberglass was therefore selected as the material for the motor tube due to its high compressive strength.

Motor Tube			**G12 Fiberglass**			**Blue Tube**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.17	USD/in	1.20	3.33	0.56	0.40	10.00	1.67

Density	0.17	lb/in^3	0.07	6.87	1	0.06	7.80	1.30
Compressive Strength	0.50	ksi	37.10	10.00	5	5.08	1.37	0.68
Machinability	0.17	experience	fair	4.00	0.67	good	8.00	1.34
Overall value					**7.37**			**4.99**
Motor Tube			**Phenolic**					
Objective	Weighting Factor	Parameter	Mag.	Score	Value			
Cost	0.17	USD/in	0.71	5.63	0.94			
Density	0.17	lb/in^3	0.05	10.00	1.67			
Compressive Strength	0.50	ksi	13.50	3.64	1.82			
Machinability	0.17	experience	fair	4.00	0.67			
Overall value					**5.10**			

Table 6: Motor Tube Decision Matrix

3.1.5.1.4 Fins

Three materials were considered for the fins. These include structural FRP fiberglass, plywood, and G10 fiberglass (Table 7). These materials were evaluated and compared based on their shear strength, cost, density, and impact strength. The cost and density of the fins does not impact the vehicle greatly due to there being only three fins. The fins must be able to withstand the shear forces experienced during landing. Since G10 fiberglass has the highest shear and impact strength and overall score in the decision matrix, it was selected for the fin material.

Fins			**Structural FRP**			**Plywood**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Shear Strength	0.33	ksi	21.50	10.0	3.3	2.00	0.9	0.3
Cost	0.17	USD/ft^2	27.89	1.5	0.2	4.10	10.0	1.7
Density	0.17	lb/in^3	0.06	4.0	0.7	0.02	10.0	1.7
Impact Strength	0.33	ft-lb/in	8.00	6.7	2.2	3.70	3.1	1.0
Overall value					6.5			**4.7**
Fins			**G10 Fiberglass**					
Objective	Weighting Factor	Parameter	Mag.	Score	Value			
Shear Strength	0.33	ksi	21.50	10.0	3.3			
Cost	0.17	USD/ ft^2	62.75	0.7	0.1			
Density	0.17	lb/in^3	0.07	3.7	0.6			
Impact Strength	0.33	ft-lb/in	12.00	10.0	3.3			
Overall value					**7.4**			

Table 7: Fins Decision Matrix

3.1.5.1.5 Centering Rings

Three materials that were evaluated for the centering rings (Table 8). These materials include structural FRP fiberglass, plywood, and type II PVC. These materials were evaluated based on their density, cost, shear strength and machining time. Based on the decision matrix, the selected material for the centering rings was plywood. While the shear strength of plywood is not the strongest, it will still be able to withstand the shear forces from launch. Thicker centering rings allow for more contact to be made with the airframe and centering rings when epoxying. Since plywood has the lowest density, the thickness will have minimal impact on the overall mass of the vehicle. Plywood was also the cheapest material compared to the other options and is least time consuming to machine. For these reasons, plywood was the selected material for the centering rings.

Centering rings			**Structural FRP Fiberglass**			**Plywood**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Density	0.30	lb/in^3	0.06	3.3	0.99	0.02	10.0	3.00
Cost	0.10	USD/in^2	4.67	0.7	0.07	0.31	10.0	1.00
Shear Strength	0.30	ksi	21.50	10.0	3.00	2.00	0.9	0.28
Machining Time	0.30	mins	13.00	3.8	1.15	5.00	10.0	3.00
Overall value					**5.2**			**7.3**

Centering rings			**Type II PVC**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value
Density	0.30	lb/in^3	0.05	4.0	1.21
Cost	0.10	USD/in^2	2.18	4.7	0.47
Shear Strength	0.30	ksi	1.50	0.7	0.21
Machining Time	0.30	mins	25.00	2.0	0.60
Overall value					**2.5**

Table 8: Centering Rings Decision Matrix

3.1.5.1.6 Bulkheads

Three materials for the bulkheads were considered. These include type II PVC, plywood, and FRP fiberglass (Table 9). These materials were compared based on their density, tensile strength, assembly, and cost. The selected material for the bulkheads was type II PVC. Even though type II PVC did not score the highest in tensile strength, it is still able to withstand the forces caused during ejection. Another advantage of type II PVC is that the bulkheads can be made of one component, while the other materials require two components that are epoxied together, which could result in misalignment issues. The density and cost of type II PVC do not have a large impact on the launch vehicle's performance; therefore, they were weighted lower than the other objectives. Type II PVC was the selected material for the bulkheads due to its ease of assembly and tensile strength.

Bulkhead			**Type II PVC**			**Plywood**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Density	0.1	lb/in^3	0.05	4.0	0.4	0.02	10.0	1.0
Tensile Strength	0.5	ksi	7.6	5.6	2.8	4	3.0	1.5
Assembly	0.3	experience	Great	10.0	3.0	Fair	4.0	1.2
Cost	0.1	USD/ft^2	8.42	4.9	0.5	4.10	10.0	1.0
Overall value					**6.70**			**4.68**

Bulkhead			FRP Fiberglass		
Objective	Weighting Factor	Parameter	Mag.	Score	Value
Density	0.1	lb/in^3	0.06	3.3	0.3
Tensile Strength	0.5	ksi	13.5	10.0	5.0
Assembly	0.3	experience	Fair	4.0	1.2
Cost	0.1	USD/ft^2	27.89	1.5	0.1
Overall value					**6.68**

Table 9: Bulkheads Decision Matrix

3.1.5.1.7 Epoxy

For applications involving the exterior fillets for fins, epoxying eyebolts, and epoxying bulkheads, the selected epoxy was RocketPoxy. Other epoxies, like JBWeld, harden quickly and are more difficult to smooth out once applied. For exterior fin fillets RocketPoxy was selected since it is strong enough to secure the fins and does not need to withstand a large amount of heat. Due to JBWeld's ability to withstand heat produced by the motor, it was selected for interior fillets and epoxying the centering rings to the motor tube.

3.1.5.2 Dimensions Justification

The launch vehicle has a nominal outer diameter of 4.0 in It was selected as it is a sufficient size to fit all the internal components, such as the parachutes, payload, avionics bay, electronics, and motor system, inside the vehicle. The launch vehicle's airframe has an outer diameter of about 4.02 in and inner diameter of about 3.90 in.

The nosecone shape and length were constrained by available off the shelf parts. As a result, a 4.02 in diameter 4.5:1 Von Karman nosecone was selected. The nosecone coupler was provided with the nosecone, however, was cut to a length of 4.0 in, resulting in a 2.0 in shoulder, which meets the requirements of the separation point being at least ½ of the body's diameter.

The forward airframe length of 23.0 in was selected because it has enough space to house half the nosecone shoulder, the nosecone ejection charges, half of the avionics bay, the main parachute, insulation material, and the recovery harness. The avionics bay has a length of 9.0 in, which provides sufficient space to house the avionics sled, batteries, and wiring. The avionics bay bulkheads are 0.5 in thick which is sufficiently strong to protect the avionics systems from black powder ejection charges. The central airframe has a length of 20.0 in, which provides enough space to hold the drogue parachute, insulation material, the recovery harness, half of the avionics bay, and 4.0 in of the payload bay.

The forward payload bay length of 9.0 in was selected to house the payload sled, batteries, two bulkheads, and electronics. The payload bay length needed to be at least twice the diameter of the launch vehicle, due to being a separation point. The payload airframe length of 29.0 in was selected to contain the three payload housings, 5.0 in of the payload bay, 4.0 in of the payload aft coupler, and wiring. The aft airframe has a length of 27.0 in, which provides enough space to hold 4.0 in of the payload aft coupler, motor, motor tube, and centering rings.

All bulkheads have a thickness of 0.5 in, aside from the 0.25 in thick nosecone bulkhead, which was selected so that they would be sufficiently strong in order to be able to withstand forces from the ejection charges and separation events.

3.1.5.3 Component Placement Justification

Three fins are in the aft end of the aft airframe. The three fins are 120.0˚ apart to ensure that there will be one fin pointed upward upon landing. To keep the fins in place, the fin tabs are flush with two aftmost centering rings. The fins are placed in the aft end of the launch vehicle to have the center of pressure behind the center of gravity.

The payload is in the payload airframe since the payload housings need to be in the same section as the fins after separation and upon landing. Due to the payload housings needing to stay oriented with the fins it must be connected to the airframe that contains the fins.

The drogue parachute is located in the central airframe and the main parachute is in the forward airframe. Due to the main parachute being heavier than the drogue parachute, it allows the center of gravity to be closer to the forward end of the launch vehicle.

3.1.6 Design Integrity

3.1.6.1 Fin Suitability

A trapezoidal fin design was selected to allow stability for the vehicle and balance the payload. There will be three fins on the launch vehicle so upon landing a payload housing with be oriented normal to the ground. To minimize the number of payload housings only three fins were necessary. 3/16 in thick G10 fiberglass was selected because it can withstand the shear forces upon landing. The fins will be attached using a through-the-wall attachment method. This is when the fins are attached to the exterior and interior of the airframe and to the exterior of the motor tube allowing for three points of connection on each side of the fins, a total of six per fin. The fins will be flush with two centering rings to secure the fins.

3.1.6.2 Material Suitability

G12 fiberglass is the material for the nosecone, airframe, couplers, and motor tube. G12 fiberglass has high compressive strength, high tensile strength, and durability, allowing for the launch vehicle to be recoverable with minimal damage. G12 fiberglass is water resistant so if the launch vehicle were to land in a wet area the launch vehicle would not be damaged. G10 fiberglass is the selected material for the fins due to its high shear, flexural, and impact strength that will be able to sustain the forces experienced upon landing.

Type II PVC is the material used for all bulkheads in the launch vehicle due to its ease of machinability and tensile strength large enough to withstand separation. During launch, bulkheads undergo many tensile forces during ejection and from the recovery harness.

The three centering rings located in the aft airframe are made of plywood. Plywood has a shear strength large enough to withstand the shear stresses from the motor’s thrust. Plywood also has a low density which minimally impacts the mass of the launch vehicle and still center the motor within the airframe.

3.1.6.3 Motor Retention

An Aerotech 54/2800 motor casing is used and secured inside the motor tube. The motor tube is centered using four centering rings. The forward centering ring is flush with the motor tube, the forward central centering ring is in the middle of the forward end of the motor tube and fins, the aft central centering ring is flush with the forward end of the fins, and the aftmost centering ring is flush with the aft end of the fins. The four centering rings are used to center the motor tube in the airframe. The forward centering ring is epoxied onto the motor tube and aft payload coupler and the other centering rings are epoxied onto the motor tube and aft airframe to secure the motor tube during launch. An Aerotech 4 in 54 mm thrust plate is screwed into the aft centering using three #10 screws. An Aerotech 54 mm motor retainer was secured to the thrust plate using six 6-32 screws.

3.1.6.4 Final Mass Estimates

A final estimate of the vehicle's component masses was developed (Table 10).

Nosecone Section		
Subteam	**Component**	**Mass (oz)**
Structures	Nosecone	21.1
Structures	Nosecone Bulkhead	4.8
Structures	Eyebolt	1.0
	Total	**26.9**
Forward Section		
Subteam	**Component**	**Mass (oz)**
Structures	Switchband	0.8
Structures	Forward Airframe	18.4
Avionics and Recovery	Main Parachute	13.4
Avionics and Recovery	Recovery Harnesses	32.0
Avionics and Recovery	Eyebolts	2.0
Structures	Bulkheads	9.6
Avionics and Recovery	Sled and Electronics	25.1
Structures	Central Airframe	16.0
Structures	Coupler	8.3
Avionics and Recovery	Drogue Parachute	1.5
	Total	**126.8**
Aft Section		
Subteam	**Component**	**Mass (oz)**
Structures	Payload Airframe	23.1
Structures	Rail Buttons	0.7
Structures	Couplers	15.6
Avionics and Recovery	Eyebolt	1.0
Payload Electronics	Electronics	21.0
Payload Mechanics	Payload Housing	36.1
Structures	Bulkheads	9.6
Structures	Aft Airframe	18.3
Structures	Centering Rings	5.8
Structures	Epoxy	22.2

Structures	Motor Retainer	4.9
Structures	Thrust Plate	2.5
Structures	Fins	11.4
Flight Dynamics	Motor and Motor Casing	86.0
Structures	Motor Tube	8.6
	Total	**266.4**
Overall Total		**420.0**

Table 10: Full-scale Final Masses

3.2 Subscale Flight Results

3.2.2 Launch Conditions and Simulation

The successful subscale launch demonstration occurred on the first attempt on December 3, 2022, under the following launch conditions (Table 11):

Subscale Launch Conditions	
Launch angle	5°
Launch rod length	39 in
Latitude	28.6 °N
Longitude	-80.6 °N
Elevation	85.3 ft
Temperature	75 °F
Pressure	1 atm

Table 11: Subscale Launch Conditions

An OpenRocket simulation was performed under these conditions and yielded an apogee altitude of 3543 ft. The simulated values for altitude over time (Figure 27), total velocity over time (Figure 28), and total acceleration over time (Figure 29) were plotted.

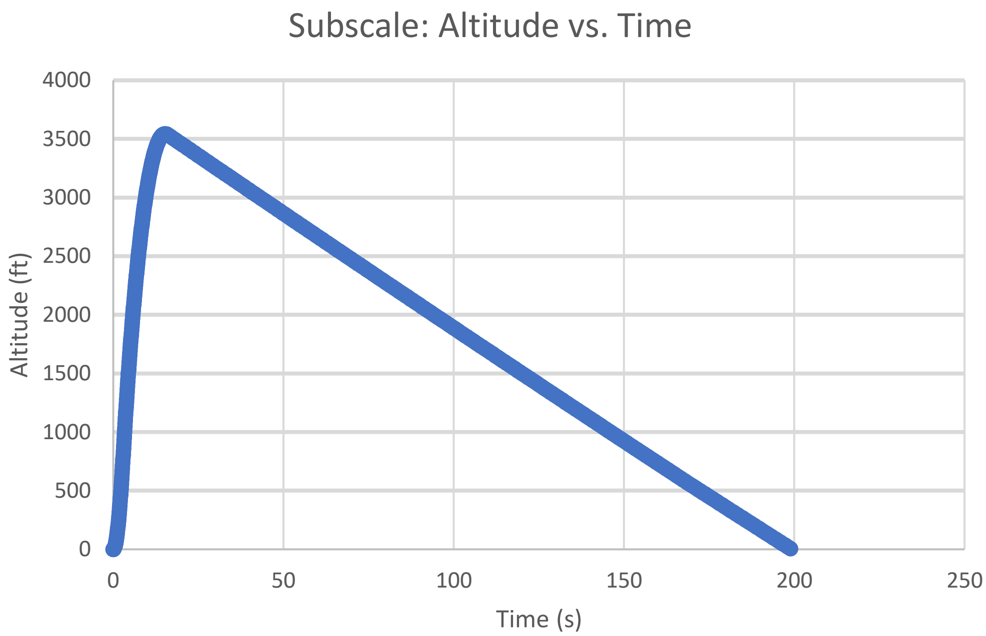

Figure 27: Subscale Altitude vs. Time

In the velocity profile (Figure 28), the maximum thrust from the motor ignition is indicated by the initial spike in the graph. This spike indicates a predicted maximum velocity of 484 ft/s, equivalent to Mach 0.63. The predicted ground hit velocity was 18.5 ft/s.

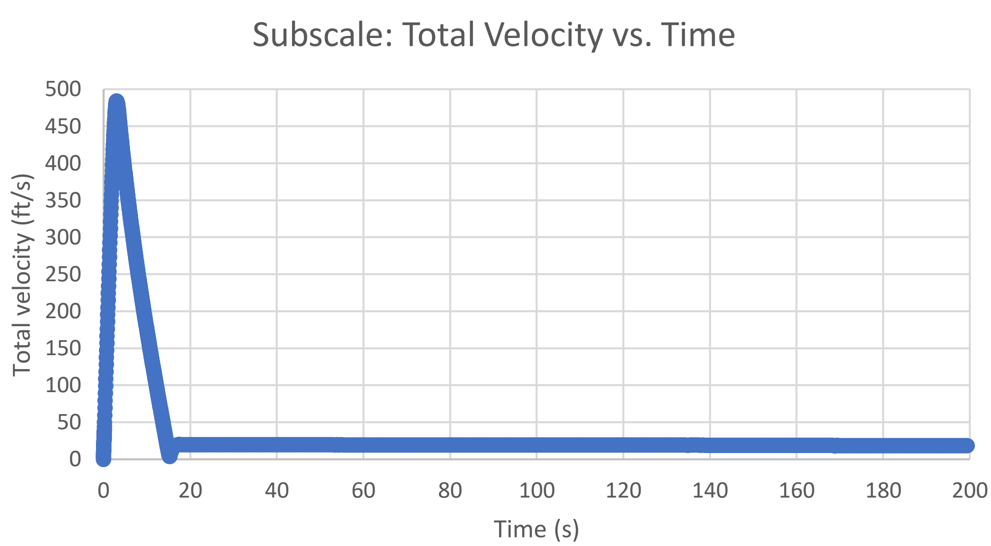

Figure 28: Subscale Total Velocity vs. Time

In the acceleration profile (Figure 29), the maximum thrust from the motor ignition is indicated by the initial spike. The second maximum indicates the deployment of the main parachute. The maximum acceleration was predicted to be 237 ft/s^2.

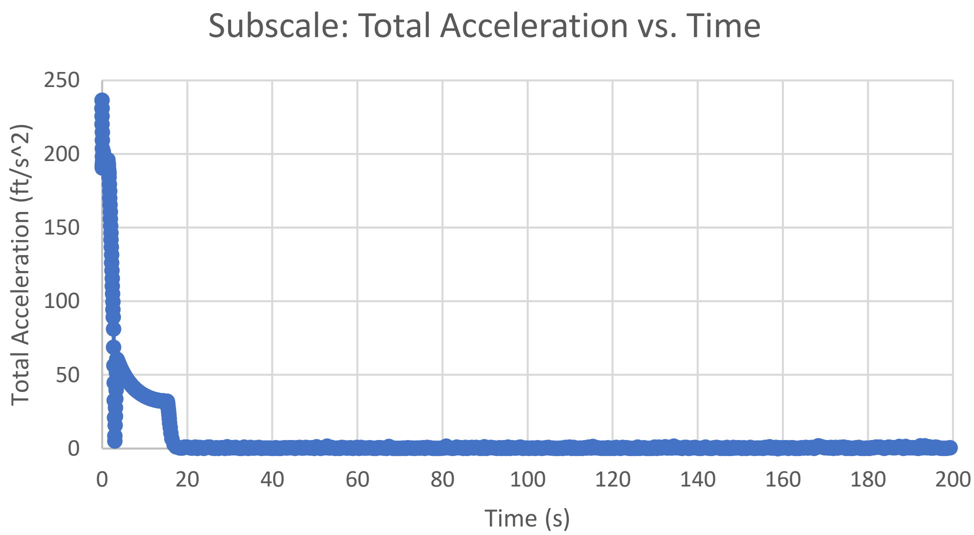

Figure 29: Subscale Acceleration vs. Time

3.2.1 Scaling Factor

To maintain geometric similarity and launch model integrity, the overall size of the subscale launch vehicle was scaled to 75% of its full-scale size to have a nominal diameter of 3 in and an overall length of 86.25 in. To keep the aerodynamic shape of the launch vehicle constant, the dimensional proportions of the fins and overall launch vehicle remained the same as for full-scale. Additionally, the subscale launch vehicle's rivets, shear pins, eyebolts, and rail buttons remained the same as the full-scale vehicle. The nosecone was not scaled to exactly 75% of its full-scale size, because it is an off-the-shelf part, and only a 5:1 nosecone was available for the airframe diameter. To account for this larger nosecone, the overall airframe length was shortened below 75% of its scaling factor to maintain the aerodynamic similarity to the full-scale launch vehicle. The payload components were scaled just below 75% of their overall sizes to ensure that the structural integrity and surface textures of the subscale launch vehicle would be representative of the full-scale launch vehicle. The parachutes that simulated the closest descent rates to the simulated decent rates for the full-scale launch vehicle were selected. These parachutes were a 60 in Rocketman elliptical for the main and an unmarked 24 in parachute for the drogue. The main parachute has a coefficient of drag of 1.5, and OpenRocket simulations of the subscale launch vehicle with this parachute gave a predicted descent rate of 18.5 ft/s, which is about 0.5 ft/s greater than the simulated main descent rate for the full-scale launch vehicle. The coefficient of drag for the drogue parachute was determined by parachute drag testing (LV-R-1) and was 0.75. The simulated drogue descent rate was 70 ft/s, which was about 13 ft/s less than the simulated drogue descent rates for the full-scale vehicle. The Aerotech J415-W motor was selected because during the subscale vehicle's design phase, it produced the same thrust to weight ratio of 8.53:1 as the Aerotech L1090 motor did for the full-scale vehicle. When ballast had to be added to the subscale vehicle after it was manufactured, its thrust to weight ratio decreased to 5.34:1, which still meets the minimum requirement of 5:1. The resulting subscale apogee altitude was scaled 77% as a result of the vehicle's 75% scale factor and the motor selection.

3.2.3 Subscale Flight Analysis

3.2.3.1 Flight Data

The predicted apogee altitude of 3543 ft was 14 ft lower than the experimental apogee altitude of 3557 ft (Table 12). This yielded a negligible percent error of 0.395%. However, there was a higher error yield of 6.86% between the predicted (52.5 in) and experimental (56.1 in) values of the center of gravity. This was likely due to the imperfect calculation of ballast that was added to the payload section to compensate for both the scaled mass difference and the mass difference for replacing the 48 in main parachute with the 60 in main parachute. This calculation error most likely contributed to the 3.10% error between the predicted (484 ft/s) and experimental (469 ft/s) values for maximum velocity as well.

Subscale Flight Data			
Data Type	**Predicted**	**Experimental**	**Percent Error**
Apogee Altitude	3543 ft	3557 ft	0.395%
Center of Gravity	52.5 in	56.1 in	6.86%
Maximum Velocity	484 ft/s	469 ft/s	3.10%

Table 12: Subscale Flight Data

The experimental altitude and velocity profiles recorded by the altimeter are shown in Figure 30. Compared to the simulated altitude profile (Figure 27), the experimental altitude profile agrees with the simulation data. However, the simulation's velocity profile (Figure 28) does not fully align with the experimental velocity profile. The experimental velocity profile shows oscillations between 15 and 75 seconds, but the simulated velocity profile does not show these oscillations. This discrepancy is likely the result of the inability of OpenRocket to accurately simulate the variable acceleration, or jerk, that occurs upon deployment of the drogue parachute. The altimeter can detect these velocity oscillations, however, due to the sensitivity of its sensors.

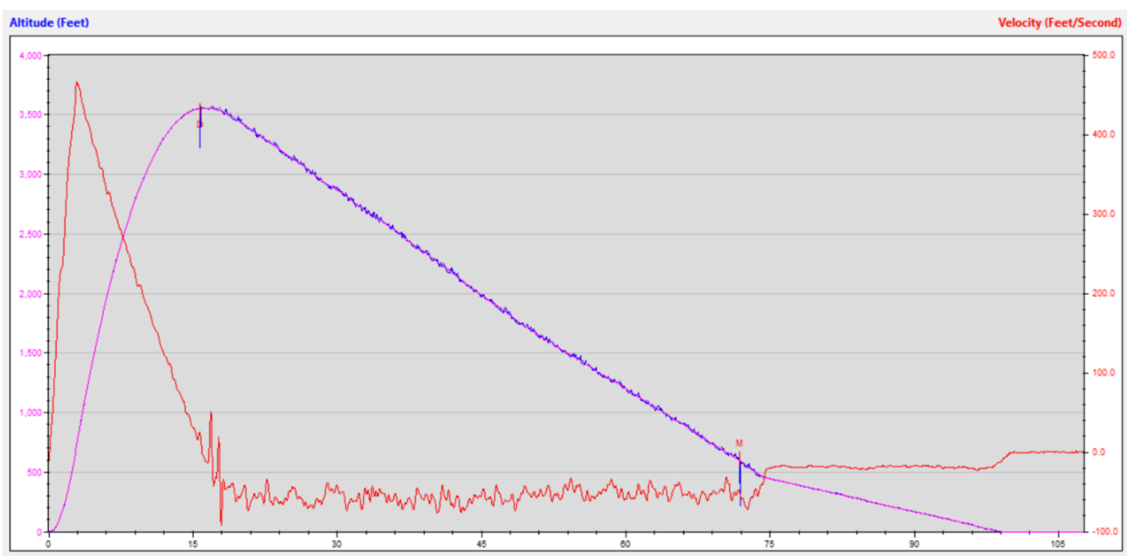

Figure 30: Subscale Resulting Velocity and Altitude vs. Time

3.2.3.2 Drag Coefficient Calculations

To determine the full-scale drag coefficient, the method of altitude backtracking was implemented using OpenRocket. First, the experimental apogee altitude was compared against the predicted apogee altitude. Because the experimental apogee altitude was greater than the predicted apogee altitude, the modeled coefficient of drag was determined to be less than what was predicted in the OpenRocket simulations. The total coefficient of drag was then manually lowered as a fixed constant, and a new simulation was performed using the same subscale launch conditions used in the previous simulations. The resulting predicted apogee altitude was then compared to the experimental apogee altitude given by the altimeter data. The coefficient of drag was then adjusted again according to this comparison. If the predicted apogee altitude was too high, the total coefficient of drag was increased; if the predicted apogee altitude was too low, the total coefficient of drag was decreased. This process was iterated until the predicted apogee altitude matched the experimental apogee altitude. Using this method, the full-scale launch vehicle's coefficient of drag was estimated to be 0.93.

3.2.4 Impact on Full-Scale Design

The success of the subscale flight demonstration confirmed the integrity of the design, modeling, and manufacturing practices the team implemented during the subscale phase. Therefore, most aspects of the team's practices will remain unchanged. The precision with which the team determined necessary ballast mass and placement, considering updates to recovery component masses, will improve to mitigate similar full-scale modeling errors. Since the subscale flight demonstration, improvements to the full-scale vehicle design were made. A bulkhead was added to the forward payload coupler to allow for the implementation of a payload electronics retention system. It was also determined that a longer payload aft coupler would be a necessary addition to improve the structural integrity of the launch vehicle. The aft bulkhead of the aft coupler was removed, because it was determined unnecessary, since the payload housings and electronics were protected without it.

3.2.5 Descent Analysis

Both parachutes were deployed successfully. The altimeter data suggests that the drogue parachute deployed at apogee, the programmed time for the primary altimeter, since the descent rate is mostly constant between apogee and main deployment. However, taking the time derivative of this data suggests an average drogue descent rate of 55 ft/s, which is significantly slower than the predicted drogue descent rate of 70 ft/s. This difference may be due to wind and uncertainty in the coefficient of drag of the drogue parachute, which had to be determined experimentally since it was unmarked.

Additionally, it is likely that the primary ejection charge for the second separation event was not large enough to cause separation. The slope of the altitude vs. time curve does not change until an altitude of roughly 460 ft is reached (Figure 31). The primary altimeter set off the primary ejection charge for this event at the correct altitude of 600 ft and no change in descent rate occurred, so the secondary ejection charge caused separation at the lower altitude. The ejection charge sizes will be increased for full scale to account for this risk. The average main descent rate was 19 ft/s, which is close to the predicted main descent rate of 18.5 ft.

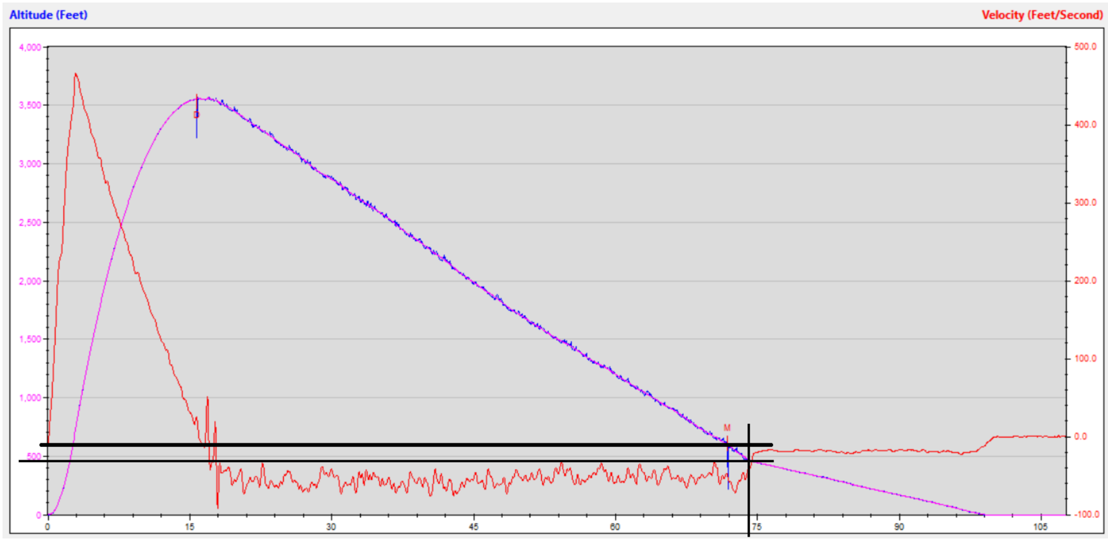

Figure 31: Subscale Flight Data with Programmed (thick line) and Actual (thin line) Main Parachute Deployment Altitudes

The GPS was functional during and after the subscale flight (Figure 32).

Figure 32: GPS Receiver

The GPS was installed in the nosecone shoulder. It was secured with Velcro attached to the inside of the nosecone shoulder and to the GPS battery. The inside of the nosecone bulkhead was covered in aluminum foil to minimize RF interference with the altimeters (Figure 33).

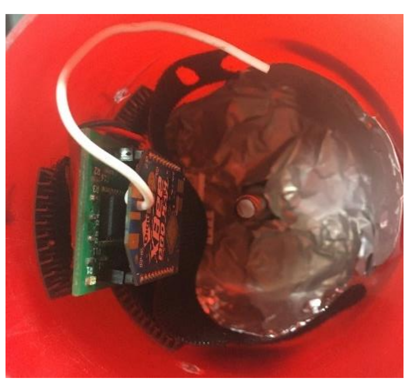

Figure 33: GPS Transmitter in Launch Vehicle

3.3 Recovery Subsystem

The recovery subsystem will use two parachutes to recover the launch vehicle. Appropriately sized parachutes for recovery were selected to allow for a safe recovery and to meet all descent requirements.

3.3.1 Selected Components

Final selections for components in the recovery system are presented in the following sections, and include the main and drogue parachutes, altimeters, altimeter arming switches, the recovery harness, GPS, and recovery hardware. These selections are also compared to alternatives from the Preliminary Design Review.

3.3.1.1 Main Parachute

The main parachute was selected with a decision matrix (Table 13). Updated descent rate predictions from OpenRocket simulations were used to verify this selection.

Main Parachute			**Fruity Chutes 84" Iris Ultra**			**Rocketman 72" Elliptical**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.2	USD	326.66	4.1	0.8	135	10	2
Weight	0.1	oz	19	6.6	0.7	12.6	10	1
Packed Length	0.1	in	8.8	5.7	0.6	5	10	1
Descent Rate	0.35	ft/s	14.7	8.5	3	19.8	8.7	3
Shroud Line Quality	0.25	qualitative	great	10	2.5	okay	6	1.5
Overall value					**7.5**			**8.5**
Main Parachute			**SkyAngle Cert 3 X Large**			**Fruity Chutes 72" Iris Ultra**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.2	USD	189	7.1	1.4	248.33	5.4	1.1
Weight	0.1	oz	34	3.7	0.4	13.4	9.4	0.9
Packed Length	0.1	in	12	4.2	0.4	6.2	8.1	0.8
Descent Rate	0.35	ft/s	16.4	9.5	3.3	17.5	9.9	3.5
Shroud Line Quality	0.25	qualitative	good	8	2	great	10	2.5
Overall value					**7.6**			**8.8**

Table 13: Main Parachute Decision Matrix

The Fruity Chutes 72” Iris Ultra parachute scored the highest overall and will thus be used as the main parachute. It has a coefficient of drag of 2.2 and a weight of 13.4 oz. It has a predicted descent rate of 17.5 ft/s from the OpenRocket simulations, which is closest to the target descent rate of 18 ft/s. Its shroud lines are made of 400 lb flat Nylon, which is strong enough to support the weight of the launch vehicle and withstand forces during deployment.

3.3.1.2 Drogue Parachute

Similar to the main parachute, the drogue parachute selection was verified with an updated decision matrix (Table 14). The parachute from the decision matrix with the highest overall score was the Rocketman 24" Standard parachute, and it will be used as the drogue parachute.

Drogue Parachute			**Rocketman 24" Standard**			**24" Spherachute**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.2	USD	33.5	6.6	1.3	22	10	2
Weight	0.1	oz	1.5	7.3	0.7	1.1	10	1
Packed Length	0.1	in	1.5	6.7	0.7	1.75	5.7	0.6
Descent Rate	0.35	ft/s	81.8	10	3.5	94.5	2	0.7
Shroud Line Quality	0.25	qualitative	great	10	2.5	okay	6	1.5
Overall value					**8.7**			**5.8**
Drogue Parachute			**Rocketman 24" Elliptical**			**Fruity Chutes 18" Elliptical**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.2	USD	50	4.4	0.9	62.89	3.5	0.7
Weight	0.1	oz	2.1	5.2	0.5	1.7	6.5	0.6
Packed Length	0.1	in	1	10	1	3.5	2.9	0.3
Descent Rate	0.35	ft/s	64.7	0.9	0.3	87.9	7	2.5
Shroud Line Quality	0.25	qualitative	okay	6	1.5	good	8	2
Overall value					**4.2**			**6.1**
Drogue Parachute			**Skyangle C3 Drogue**					
Objective	Weighting Factor	Parameter	Mag.	Score	Value			
Cost	0.2	USD	27.5	8	1.6			
Weight	0.1	oz	3.1	3.5	0.4			
Packed Length	0.1	in	6	1.7	0.2			
Descent Rate	0.35	ft/s	98.7	1.4	0.5			
Shroud Line Quality	0.25	qualitative	great	10	2.5			
Overall value					**5.1**			

Table 14: Drogue Parachute Decision Matrix

The predicted descent rate for the drogue parachute is 81.8 ft/s, which is closest to the target descent rate and will meet all descent requirements. The shroud lines are made of 220 lb nylon and are sufficiently strong to support the launch vehicle.

3.3.1.3 Recovery Harness

The recovery harness will be 7/16 in wide, 25 ft long tubular Kevlar. The recovery harness width was decreased from 5/8 in to 7/16 in to ensure that there will be enough space in the parachute compartments to fit the recovery harness, parachute, and recovery wadding. Also, the length of the recovery harness was increased from 24 ft to 25 ft to account for the recovery harness length that will be used to secure to quick links on both ends while staying above the minimum length of 2.5 times the launch vehicle length. The recovery harness material has a strength of 5300 lb, which is sufficiently strong to withstand forces during separation and parachute deployment. Two identical recovery harnesses will be used, with one for each separation event.

3.3.1.4 Altimeters

Two altimeters will be used for redundancy. The primary and secondary altimeters were the two highest-scoring altimeters in the following decision matrix (Table 15).

Altimeter			**Eggtimer Classic**			**Stratologger CF**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.2	USD	35	10	2	54.96	6.4	1.3
Resolution	0.3	ft	1	5	1.5	1	5	1.5
PCB Size	0.3	in^2	3.9	4.3	1.3	1.68	10	3
Weight	0.2	oz	0.705	5.4	1.1	0.38	10	2
Overall value					**5.9**			**7.8**
Altimeter			**Altus Metrum Telemega**			**Entacore AIM**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.2	USD	400	0.9	0.2	115	3	0.6
Resolution	0.3	ft	0.5	10	3	1	5	1.5
PCB Size	0.3	in^2	4.063	4.1	1.2	2.519	6.7	2
Weight	0.2	oz	1	3.8	0.8	0.4	9.5	1.9
Overall value								**6**
Altimeter			**Altus Metrum Easymega**			**RRC3 "Sport"**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.2	USD	341.3	1	0.2	96.5	3.6	0.7
Resolution	0.3	ft	1	5	1.5	unknown	0	0
PCB Size	0.3	in^2	2.9	5.8	1.7	3.626	1	0.3
Weight	0.2	oz	0.5	7.6	1.5	0.6	6.3	1.3
Overall value					**5**			**2.3**

Table 15: Altimeter Decision Matrix

The primary altimeter will be a Perfectflite Stratologger CF and the secondary altimeter will be an Entacore AIM. These altimeters scored the highest overall in the decision matrix, with the Stratologger CF having an overall score of 7.8 and the Entacore AIM having an overall score of 5.9. Both altimeters will record flight data and set off ejection charges. They will be located next to each other in the avionics bay and will be powered by separate 9V batteries.

3.3.1.5 Switches

Keylock switches will be used to arm the altimeters from outside the launch vehicle. Keyed switches were chosen over other alternatives such as push buttons and pull-switches due to their low risk of unintended closing or opening of the switch and ease of access from outside the launch vehicle. A keyed switch can only be closed or opened by inserting and turning a key, minimizing the risk of premature arming or disconnection of avionics circuitry due to flight forces. This makes additional preventative devices such as covers for pushbuttons unnecessary, enabling the switches to be accessed quickly.

The key switches will be secured to the avionics sled so that the keys line up with drilled holes in the avionics coupler, allowing a team member to arm them from outside the launch vehicle (Figure 34).

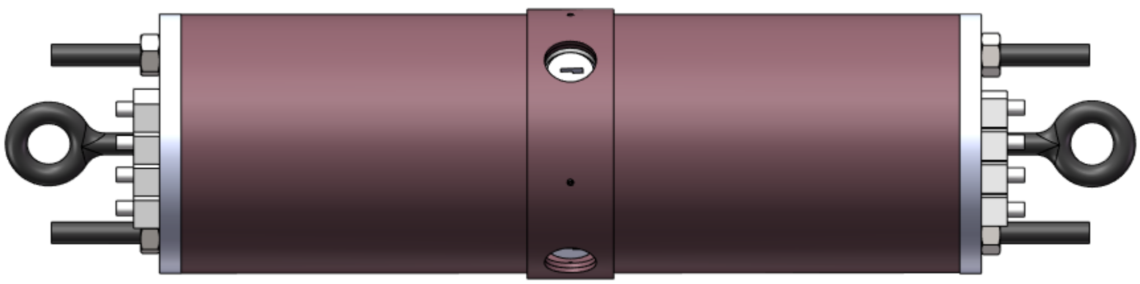

Figure 34: Avionics Bay with Key Switches

3.3.1.6 GPS

The GPS selected for the launch vehicle is a Big Red Bee 900. Location data is sent from the transmitter to the receiver with a 900 MHz spectrum transmitter. This GPS was chosen for its long transmission range of 6 miles, its high resolution of 8.2 ft, and because it does not require any radio licensing to operate, enabling any team member to use it. The GPS transmitter will be powered by a single cell LiPo battery of 3.5-4.2 V and has an expected battery life of 6 hours starting from a full charge. It will not be connected to the avionics or payload electronics in any way and will be inside the nosecone shoulder. Additionally, the nosecone bulkhead will be covered in aluminum foil to minimize interference from the GPS with other electronics in the launch vehicle.

3.3.1.7 Attachment Hardware

The recovery hardware will be a galvanized steel eyebolt with an inner diameter of 0.5 in. The eyebolt has a width of 1 in, a total length of 3 in, a mass of 0.998 oz, and a carrying capacity of 500 lbs. The outer dimensions are smaller among the alternative bolts, which allows for more space on the bulkheads to be used for threaded rods and terminal blocks, and reduces the space taken up by the bolt in the parachute compartment. Additionally, the bolt is sufficiently strong to withstand forces during separation and parachute deployment. Eyebolts will be secured in the center of bulkheads at separation points and recovery harness will be secured to them with quick links. The quick links will be oval shaped with a length of 2 in and a thickness of 9/32 in. The quick links have a carrying capacity of 1000 lbs, which is also sufficient.

3.3.2 System Redundancy

To minimize the impact of an altimeter or ejection charge failure, a redundant system of two altimeters and two sets of ejection charges will be used. Each altimeter will control a main and drogue ejection charge, so a total of four ejection charges will be used for each flight. Delays between the primary and secondary ejection charges will be used to prevent damage to the launch vehicle. A delay of 1s after apogee will be used for the first separation event and a delay of 50 ft will be used for the second separation event. The secondary charges will also be 25% larger than their respective primary charge to ensure that the launch vehicle separates. The altimeters will be wired separately (Figure 35), and each will be powered by a separate 9V battery. This wiring ensures that one of the altimeters can separate the launch vehicle and collect flight data if the other loses power.

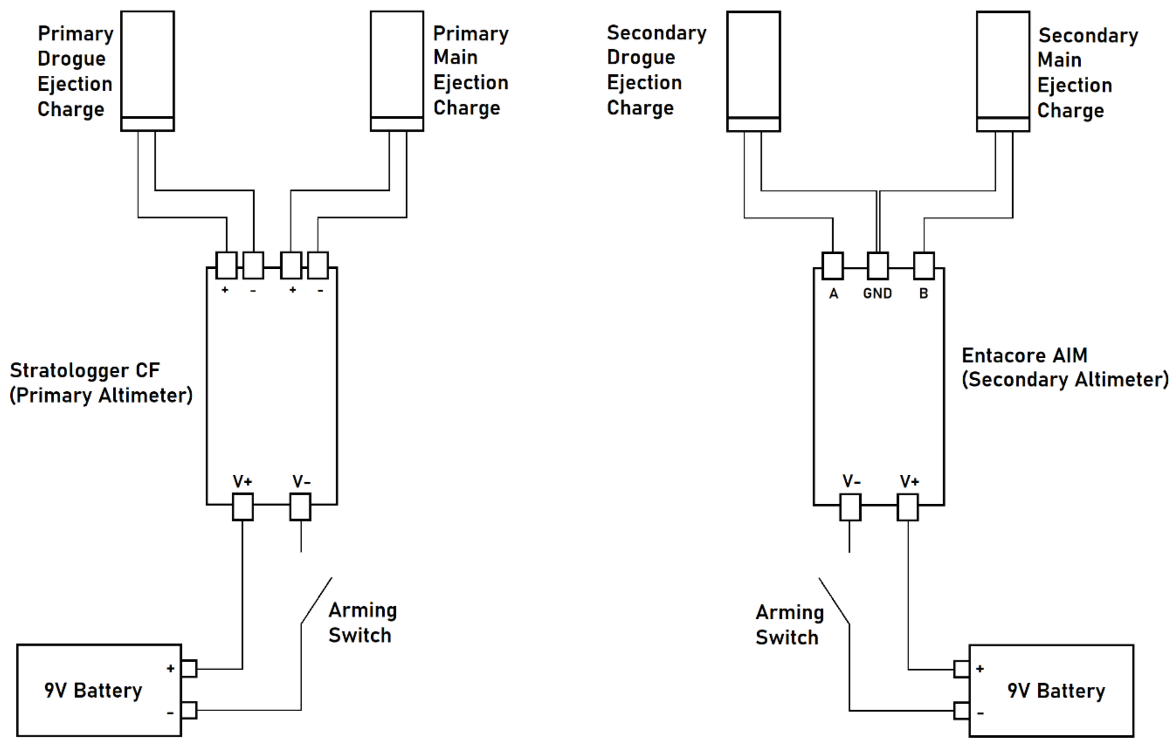

Figure 35: Altimeter Wiring Diagram

3.4 Mission Performance Predictions

3.4.1 Flight Profile Simulations

The launch conditions used in the simulation are shown in Table 16. Based off these conditions, the vehicle's apogee is predicted to be 4780 ft in the simulation. The official target apogee is predicted to be 4600 ft. The drogue is expected to deploy at apogee and the main is expected to deploy at 600 ft. The total flight time is expected to be 99.8 s. The flight profile of the launch is shown in Figure 36.

Launch Conditions in Huntsville, Alabama	
Wind	5 mph
Launch Angle	10°
Launch Rod Length	144 in

Latitude	34.6 °N
Longitude	-86.7 °E
Altitude	800 ft
Temperature	80 °F
Pressure	1 atm

Table 16:Launch Conditions used in Simulations

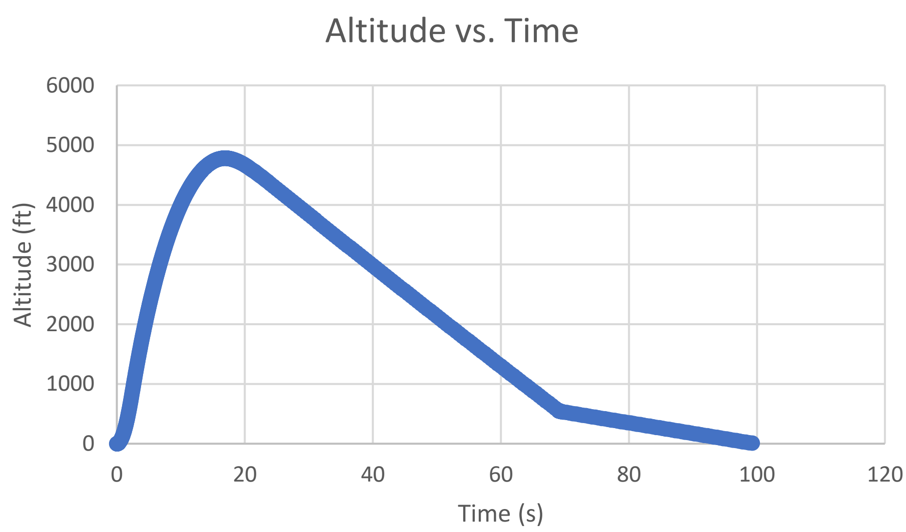

Figure 36: Altitude vs. Time

The velocity profile of the launch vehicle is shown in Figure 37. The maximum thrust from the motor ignition is indicated by the initial spike in the graph. The initial velocity off the rail is expected to be 87 ft/s, meeting the competition requirement of the 52 ft/s minimum. The maximum velocity is expected to be 646 ft/s, equivalent to Mach 0.57. The ground hit velocity is expected to be 18 ft/s.

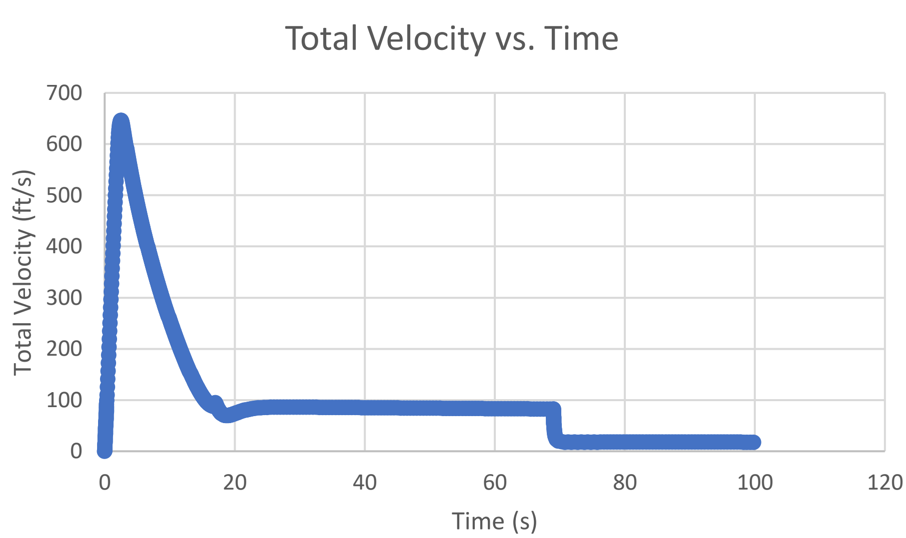

Figure 37: Total Velocity vs. Time

The acceleration profile of the launch vehicle is shown in Figure 38. The maximum thrust from the motor is indicated by the initial spike. The second spike is due to the main parachute deploying. Because it is instantaneous, it can be considered negligible to the overall acceleration of the vehicle. The maximum acceleration is expected to be 332 ft/s^2.

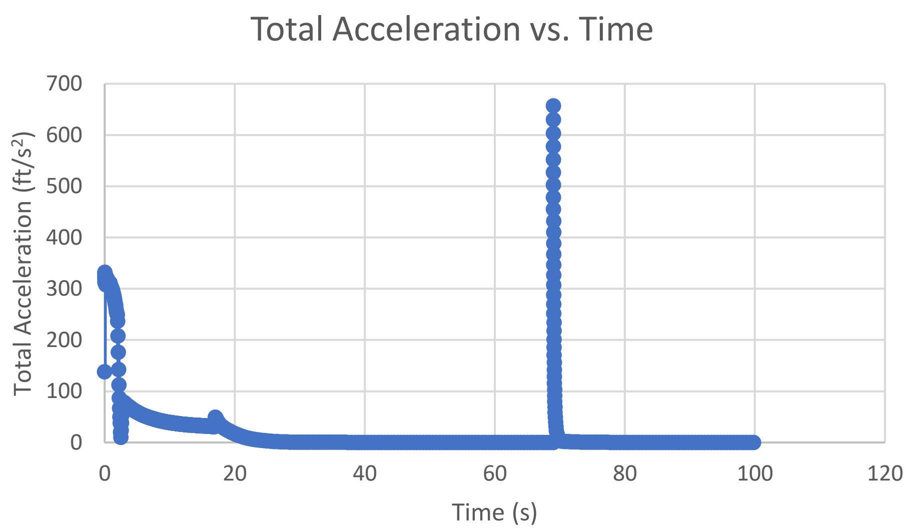

Figure 38: Total Acceleration vs. Time

3.4.2 Motor Selection

The official motor selected for the competition is the Aerotech L1090. It uses 1400 grams of propellant to produce a maximum thrust of 1487 N and a total impulse of 2671 N-s. The maximum thrust occurs at the launch rod and allows the vehicle to propel off the rail at a velocity of 87 ft/s. The Aerotech L1090 has a burn time of 2.5 seconds and provides a thrust to weight ratio of 8.53:1, fulfilling the competition requirement of 5:1. The thrust curve data of the motor is shown in Figure 39.

This motor was selected due to the ability to provide the necessary thrust to weight ratio, as well as the short burn time. It allowed the team to prioritize a motor that can safely launch the vehicle in a consistent manner. The size restraint of the motor tube was an important consideration in choosing the motor, so the other motors that were considered included an Aerotech K780, L850, L1420, K1000, and L1170.

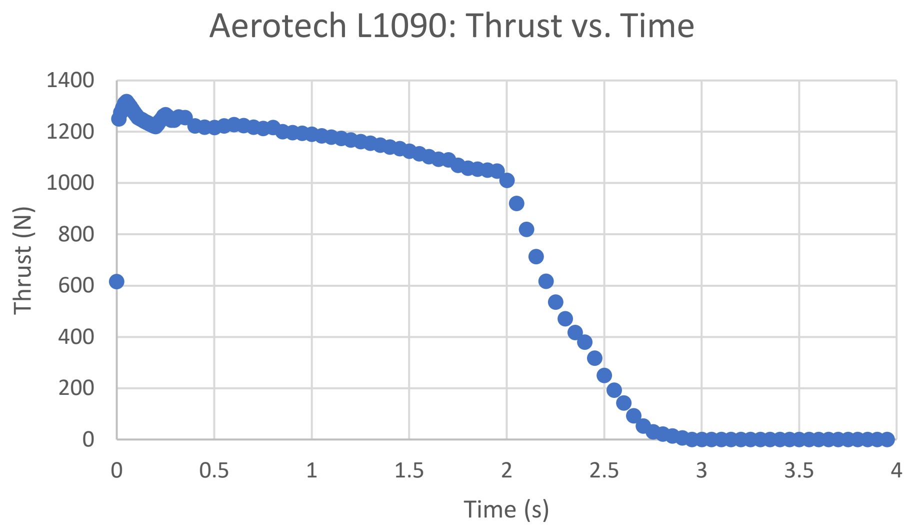

Figure 39: Aerotech L1090 Thrust vs. Time

3.4.3 Stability Margin

The vehicle is statically stable when the center of pressure is located at least 1 body caliber behind the center of gravity. However, it is recommended to have a stability margin within 7-15% of the length of the rocket. In other words, at least a minimum of 2 body calibers as per competition guidelines. The center of gravity of the rocket with the motor is 71 in from the tip of the nosecone. The center of pressure of the rocket with the motor is 84 in from the tip of the nosecone. Therefore, the static stability of the vehicle (loaded) on the launch pad is 3.27 calibers. The static stability at the rail exit is 2.13 calibers. During flight, the stability gradually increases to a maximum of 4.39 calibers, due to the overall vehicle mass decreasing from propellant burnout. After the motor burns out, the stability decreases to a minimum of 2.44 as the velocity decreases as the launch vehicle approaches apogee. Figure 40 displays the change in stability over time where the oscillations are an indication of simulated wind gusts.

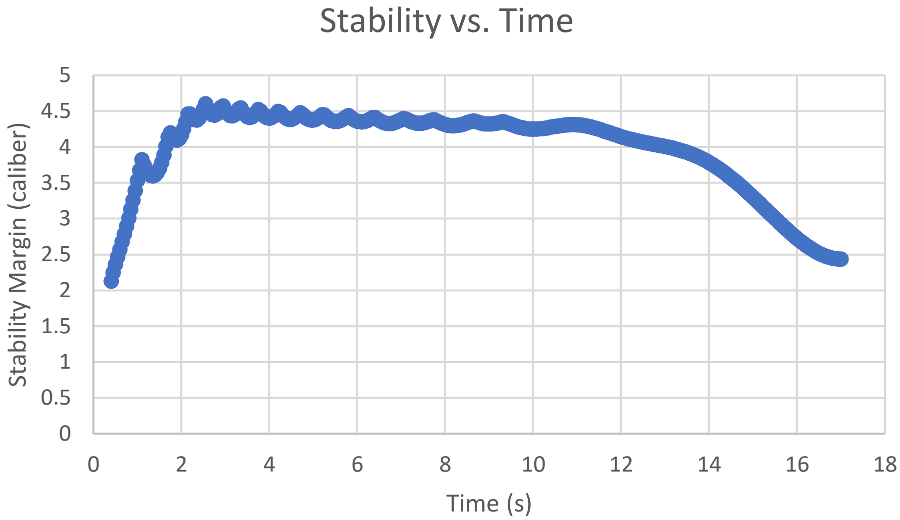

Figure 40: Stability vs. Time

3.4.4 Simulation Verifications

To verify the OpenRocket simulation for accuracy and precision, a Monte Carlo Simulation was conducted in MATLAB to test the sensitivity of the launch vehicle to wind conditions and launch angle and their effects on apogee. A total of 10000 simulations were conducted with wind profiles based on the probability weight as shown in the table. The probability weights were assigned after reviewing the launch site's climate trend around the time of the competition. Table 17 displays the Monte Carlo results to quantify the average altitude for each wind condition and the most probable altitude given the range of launch conditions. The results produce that the most probable altitude is 4718 ft which is close to the predicted apogee from OpenRocket. This indicates ballast is expected to be added for the launch vehicle to reach the previously declared target apogee of 4600 ft.

The differences in the results between OpenRocket and MATLAB are minimal and due to the variations in wind measurement that OpenRocket does not account for. Both simulations are relatively accurate as they use the same foundational mathematics. However, there are differences in the atmospheric model used in the simulations as well as drag coefficient estimates that can only be predicted accurately by experimentation.

Monte Carlo Simulation: Altitude			
Launch Angle	Wind Condition	Probability Weight	Predicted Average Altitude (ft)
0	0 mph	5%	4931
5 deg	5 mph	10%	4780
5 deg	10 mph	70%	4707
10 deg	15 mph	10%	4659
10 deg	20 mph	5%	4656
Most Probable Altitude			**4718 ft**

Table 17: Tabulated Monte Carlo Results

3.4.5 Descent Predictions

Three methods were used to predict descent values for the launch vehicle: simple calculations in Excel, OpenRocket simulations, and MATLAB simulations. Multiple methods were used to verify the accuracy of predicted values.

3.4.5.1 Descent Time Calculations

Descent times in Excel were calculated for each parachute (Equation 3). The times were then summed to find a total descent time from apogee to landing.

$$descent\ time = \frac{apogee - main\ deployment\ height}{drogue\ descent\ rate} + \frac{main\ deployment\ height}{main\ descent\ rate}$$

Equation 3: Total Descent Time (Excel)

Equation 3 depends only on the height traveled under each parachute and the parachutes' descent rates, so time spent accelerating to the final descent rates and delay in the parachutes opening are unaccounted for. As a result, these calculations made with this method are less accurate. Descent rates for the parachutes were obtained from the OpenRocket simulation (Table 18). Apogee for these spreadsheet calculations was set to 4780 ft, the target apogee, and the main deployment height was 600 ft.

Descent Rates from OpenRocket	
Drogue Parachute	81.8 ft/s
Main Parachute	17.5 ft/s

Table 18: Parachute Descent Rates

OpenRocket simulations were also used to estimate descent time. OpenRocket does not give times under each parachute, instead providing the total flight time and time to apogee. The total descent time is the difference between these times (Equation 4).

$$descent\ time\ = flight\ time - time\ to\ apogee$$

Equation 4: Total Descent Time (OpenRocket)

The MATLAB simulation uses an ordinary differential equation solver to solve the differential equations created when balancing the vertical forces on the launch vehicle and parachute, which are weight and drag force (Equation 5).

$$\begin{cases} 0 = \dfrac{1}{2}\rho v^2 S_{drogue} C_{D,drogue} - mg, & h_{main} < h \leq h_{apogee} \\ 0 = \dfrac{1}{2}\rho v^2 S_{main} C_{D,main} - mg, & h_{main} \leq h \leq h_{apogee} \end{cases}$$

Equation 5: Forces on the Parachute During Descent

In Equation 5, ρ is air density, v is velocity, S is the area of the drogue or main parachute, C_D is the coefficient of drag

Air density (ρ) is determined for each step in the simulation calculations with Equation 6, where ρ_1 is density at sea level (1.225 $\frac{kg}{m^3}$), T_1 is the temperature at sea level (288.16 K), *R* is the specific gas constant for air (287 $\frac{J}{kg \cdot K}$), and *a* is the variation of temperature with altitude (-0.0065 $\frac{K}{m}$).

$$\rho = \rho_1 \left(\frac{T}{T_1}\right)^{-\left(\frac{g}{Ra}+1\right)}$$

Equation 6: Air Density at Altitude

T, the temperature for each step in the simulation, was determined with Equation 7.

$$T = 288.16 - 0.0065h$$

Equation 7: Temperature at Altitude

The MATLAB simulation stops calculating values when the altitude *h* is at ground level, and the total descent time is the simulated time for the duration of the simulation (Table 19).

Descent Time Results		
Excel Calculations	Time under drogue	51.1 s
	Time under main	34.3 s
	Descent time	**85.4 s**
OpenRocket Simulation	Flight time	99.8 s
	Time to apogee	16.9 s
	Descent time	**82.9 s**
MATLAB Simulation	**Descent time**	**84.1 s**

Table 19: Descent Time Calculation Results

The Excel calculations yield the longest descent time values of 85.4 s. This difference is most likely because the Excel calculations don't account for acceleration times or variations in air density, and the difference between the OpenRocket and MATLAB simulations may be a result of the different methods used to calculate descent rate. Additionally, the OpenRocket simulations calculate an apogee instead of using the apogee of 4780 ft that was used for the other methods. All of the descent rate predictions are below the 90s maximum.

3.4.5.2 Drift Calculations

Excel spreadsheet, Openrocket, and MATLAB methods were used to calculate drift. For each method, drift was calculated for 0-mph wind, 5-mph wind, 10-mph wind, 15-mph wind, and 20-mph wind. It was assumed that apogee was reached above the launch pad.

Drift was calculated in Excel with Equation 8 for the drogue and main parachutes, where descent time is the descent times determined in Section 3.4.5.1 Descent Time Calculations.

$$drift = descent\ time * \ wind\ speed$$

Equation 8: Drift

Drift in the MATLAB simulation was determined with the same equation, where descent time is the descent time calculated from the simulation. Apogee was assumed to be a constant 4780 ft for all drift values calculated with these methods as a result of setting the apogee in Section 3.4.5.1 Descent Time Calculations.

OpenRocket includes lateral distance from the launch pad in its simulation but does not assume that apogee is reached directly above the launch pad. As a result, the launch vehicle may drift back towards the launch pad for part of the simulation (Figure 41). To determine drift with the assumption that apogee was reached directly above the launch pad, the largest lateral distance near apogee was added to the final lateral distance. The launch rod angle for these simulations was set to 0° to minimize lateral motion before apogee.

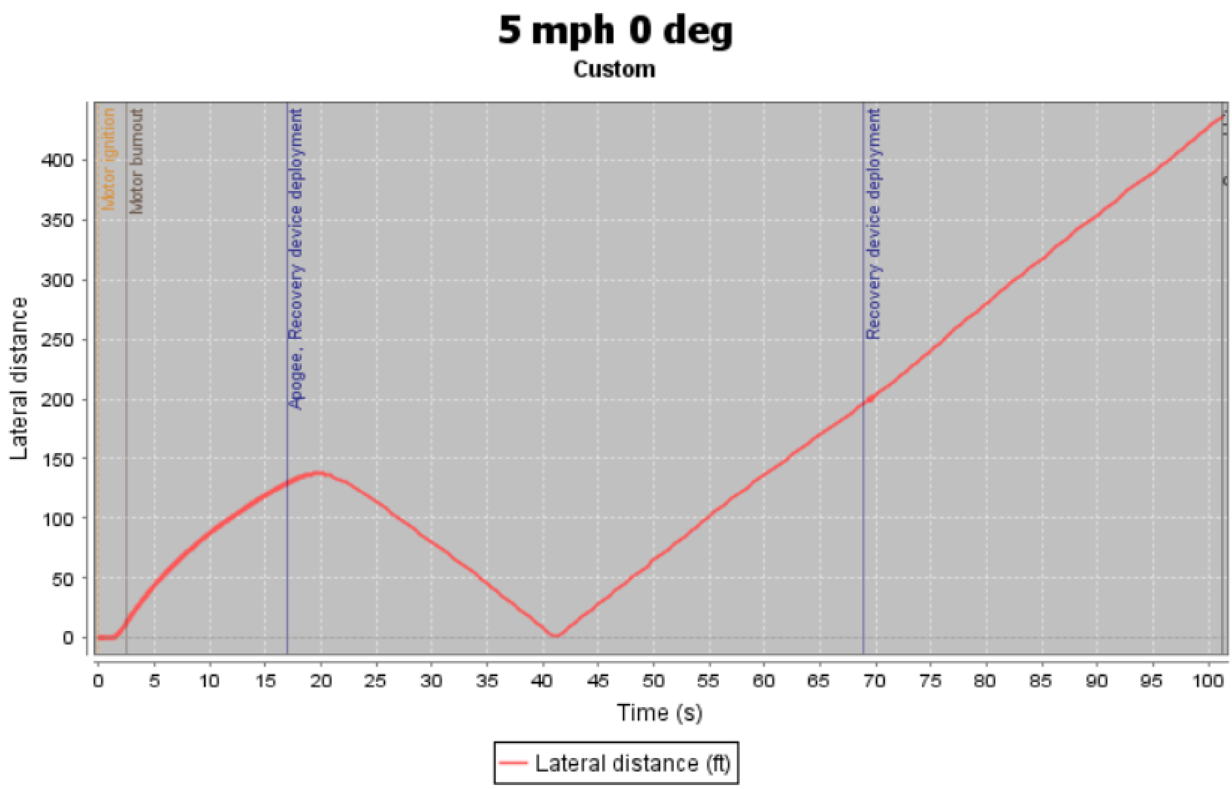

Figure 41: A Plot of Lateral Distance vs. Time from an OpenRocket Simulation

The final drift calculations were tabulated and compared (Table 20).

Drift Calculation Results						
Wind Speed		**0 mph**	**5 mph**	**10 mph**	**15 mph**	**20 mph**
Excel Calculations	Drift under drogue	0 ft	375 ft	749 ft	1124 ft	1499 ft
	Drift under main	0 ft	251 ft	503 ft	754 ft	1006 ft
	Drift	**0 ft**	**626 ft**	**1252 ft**	**1878 ft**	**2505 ft**
OpenRocket Simulation	Peak lateral distance	8 ft	166 ft	294 ft	337 ft	355 ft
	Total lateral distance	7 ft	416 ft	868 ft	1421 ft	2028 ft
	Drift	**15 ft**	**582 ft**	**1162 ft**	**1758 ft**	**2383 ft**

MATLAB Simulation	Drift	0 ft	616 ft	1233 ft	1849 ft	2466 ft

Table 20: Drift Results

The OpenRocket simulations consistently gave lowest drift values of the methods used. This may be due to errors when indirectly determining drift by addition and a result of setting the simulation launch rod angle to 0°, which might lead to less lateral motion than the other methods predict. The Excel calculations and MATLAB simulation results were closer, and the Excel results were larger. The Excel calculations were likely the largest because they assumed that the launch vehicle instantaneously reached its final velocity for each parachute. The largest drift prediction was from the Excel calculations at wind speeds of 20 mph with a drift of 2505 ft, which is 5 ft greater the maximum drift allowed. However, both simulation methods predict drift values less than 2500 ft in 20 mph winds. The OpenRocket simulation accounts for variations in wind speed, air density, and gradual changes in velocity during parachute deployment, and the MATLAB simulation accounts for changes in air density. These simulations are more accurate to real world conditions than the Excel calculations, so the launch vehicle should drift less than the 2500 ft maximum.

3.4.5.3 Kinetic Energy Calculations

Kinetic energy at landing was calculated for each section of the launch vehicle (Equation 9). OpenRocket does not calculate kinetic energy, so the spreadsheet and MATLAB methods were used. The spreadsheet used the predicted main descent rate from Table 18, and the MATLAB simulation used its simulated velocity before ground hit. For both methods, it was assumed that the sections were traveling at the main parachute's descent rate (Table 21).

$$kinetic\ energy = \frac{1}{2} * m_{section} * main\ descent\ rate^2$$

Equation 9: Kinetic Energy at Ground Hit

Kinetic Energy at Ground Hit Calculation Results			
Section	Nosecone	Forward	Aft
Excel Calculations (ft-lbs)	8.0	37.7	64.5
MATLAB (ft-lbs)	8.2	38.6	66.1

Table 21: Kinetic Energy Results

Both calculation methods yielded similar kinetic energy values, and differences between the predictions were larger for more massive sections. This difference may be because the MATLAB simulations use a different calculation method to determine the main descent rate. Both kinetic energy predictions are below the 75 ft-lb maximum.

3.4.5.4 Predicted Deployment Load

The predicted deployment load was determined with Equation 10, where ΔKE is the difference between kinetic energy during drogue and main descent and *d* is the vertical distance traveled during parachute deployment. The kinetic energies used to determine ΔKE were found using descent rates from the OpenRocket and MATLAB simulations (Table 22). It was assumed that all the sections of the launch vehicle traveled at the same speed, so the section with the largest kinetic energy change is the aft due to its larger mass.

$$Load = \frac{\Delta KE}{d}$$

Equation 10: Deployment Load

Predicted Deployment Load	
KE of Aft Section Under Drogue (OpenRocket)	1409.2 ft-lbs
KE of Aft Section Under Main (OpenRocket)	64.5 ft-lbs
ΔKE (OpenRocket)	1344.7 ft-lbs
d	75 ft
Load (OpenRocket)	**287 oz (17.9 lb)**
KE of Aft Section Under Drogue (MATLAB)	1438.1 ft-lbs
KE of Aft Section Under Main (MATLAB)	66.1 ft-lbs
ΔKE (MATLAB)	1372.0 ft-lbs
Load (OpenRocket)	**293 oz (18.3 lb)**

Table 22: Predicted Deployment Load

The distance traveled during parachute deployment was set to 75 ft and was determined from data from OpenRocket simulations (Figure 42), where *d* is the distance between 600 ft, when the main parachute deploys, and the altitude at which the launch vehicle reaches the main parachute descent rate. This distance was determined for OpenRocket simulations at a variety of wind speeds and averaged.

The largest predicted deployment load is 293 oz or 18.3 lb of force on the recovery equipment. All components in the recovery system (hardware, quick links, recovery harness, parachute shroud lines) are rated at significantly higher capacities and will withstand the forces on them during deployment.

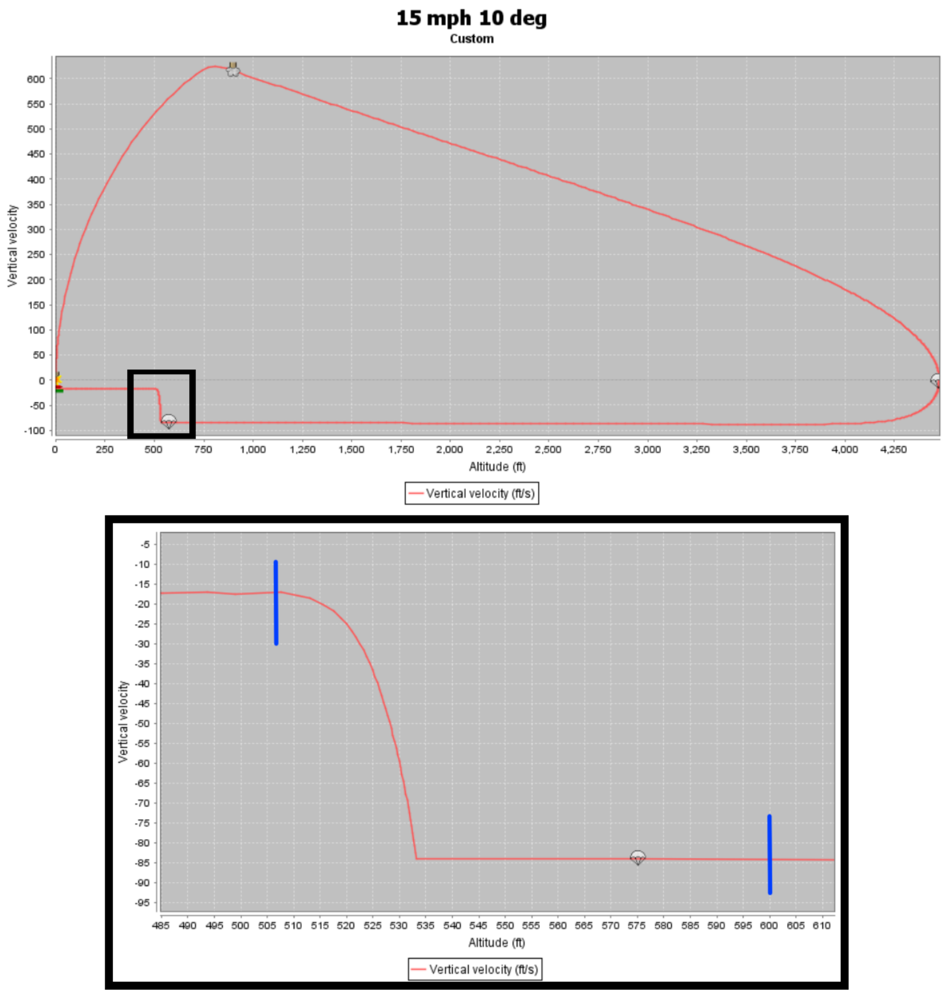

Figure 42: Distance Traveled during Main Parachute Deployment (distance between blue lines)

4. Payload Criteria

4.1 Design of Payload Equipment

The payload design consists of three camera systems and their associated electronics. The camera systems will be positioned 120° apart and aligned with the fins so that one will always be aligned with the z-axis upon landing (Figure 43). This is because the vehicle will have to land on two fins. Therefore, one fin, and the corresponding camera system, will be aligned normal to the ground. The airframe has three rectangular cut-outs that each camera system can deploy out of, should it be the system aligned with the z-axis. The camera mount sits almost flush with the cut-out airframe. They are positioned so that the camera system can deploy out of the airframe upon landing and subsequently rotate about the z-axis to image the surrounding environment. An Inertial Measurement Unit (IMU) and a barometer will be used for launch and landing detection. The IMU will also detect the orientation of the launch vehicle upon landing and determine which camera to activate so that only one camera will be in use when taking photos of the surroundings. Push buttons are used as feedback to make sure the camera will not rotate before the camera is released from camera housing.

A Software Defined Radio (SDR) dongle will be incorporated in the payload to receive Automatic Package Reporting System (APRS) commands from NASA. The payload will use a Raspberry Pi and custom software to manage and control all the motors, cameras, sensors, and radios. The control circuits of all the motor and feedback circuits from the camera housing are all integrated into a custom Printed Circuit Board (PCB) designed by the team. The software will begin running when the Raspberry Pi is powered on and will be a state-based system interacting with specific software subsystems.

Figure 43: Full-scale Assembly Payload aligned with fins

4.2 Selected Design Alternatives

4.2.1 Selected Mechanical Components

4.2.1.1 First Design

The first concept was an externally mounted camera design. It consisted of clear teardrop-shaped housings containing cameras mounted on the outside of the airframe. Each one would have been aligned with a fin so that one of the mounts would be vertically oriented upon landing. This is because the launch vehicle would land with two fins contacting the ground and with the third fin and its corresponding camera oriented normal to the ground. The design advantages were its low motor count, complexity, and cost. It only required three motors which would minimize the cost and number of components required to implement the concept. The most significant disadvantage for the externally mounted camera idea was the effects it had on the aerodynamics of the launch vehicle. Due to the negative aerodynamics effects, this design alternative was not chosen.

4.2.1.2 Second Design

The second concept involved three cameras located within the airframe, each aligned with a fin. The cameras would radially extend out of the airframe using linear motion. It would have used linear actuators to push the cameras out of the airframe through milled cut outs in the airframe using a lead screw. The linear extension design had no aerodynamic risks, which was one of its main advantages. The primary disadvantage for the linear extension concept is that it would require more room for the linear actuator which ultimately would not fit and intrinsically increases complexity and cost; the design introduces too many potential points of failure. Because of the design’s complexity, it was not selected as the final design.

4.2.1.3 Third Design

The third concept required a spring hinge to rotate a camera and a camera housing out of the airframe. It would have cameras mounted inside the airframe in housings that would be flush with milled cut-outs in the airframe. The camera aligned with the vertical fin would spring out of the airframe on a hinge. The spring hinge design's advantages solve the challenges posed by the other ideas. The design protects the cameras during flight, has no external protrusions during flight, and could be designed such that the holes in the airframe are covered by the camera mount flush with the airframe. Additionally, the hinge is oriented in the direction opposite of the flight direction to prevent premature extension of the cameras during flight. The design only requires two motors per subsystem totaling six motors. This was the selected design. All other decisions, including material selection and electronics, were then made based on this design.

4.2.1.4 Material Alternatives Given Leading Orientation Design

Material durability was determined from open-source data regarding resistance to fatigue, heat, and chemicals. According to the Material Selection matrix, the best choice for the material for 3D printed components is PET-G. Its durability was well suited for the design than any other choice (Table 23).

Test	Measurement	PLA	PET-G
Stiffness	Difficult bend /10	7.5	5
Durability	Fatigue, heat, chemical resistance /10	4	8
Max Surface Temperature	Before bending under load in Celsius	52	73

Table 23: 3D Print Material Decision Matrix

4.2.1.5 Fastening Alternatives Given Leading Orientation Design

Threaded inserts were chosen as the best option for fastening the payload systems to the airframe as shown in the matrix (Table 24). The thickness for the airframe was determined to be too small for countersunk fasteners. The threaded inserts also mitigate any concern with external protrusion's effects on stability since the inserts do not require bolts on opposing sides.

Fasteners			Threaded Inserts			Countersunk			Exterior Nut		
Objective	Weight	Parameter	Mag.	Score	Value	Mag.	Score	Value	Mag.	Score	Value
Cost	0.2	Cost	9.00	2	0.4	8.00	6	1.2	5	10	2
Aero Impact	0.5	Area	Great	10	5	Fair	4	2	Poor	2	1
Thickness	0.3	Binary	1	10	3	1	10	3	2	10	3
Overall value					**8.4**			**5.2**			**6.0**

Table 24: Fastener Decision Matrix

4.2.2 Selected Electronic Components

4.2.2.1 Microprocessor Selection

Raspberry Pi 4 Model B, Arduino Uno Rev3, and Libre Computer Board AML-S905X-CC were evaluated by performance, number of IO (Input-Output) pins, ease of use, and cost (Table 25). The Raspberry Pi 4 Model B features the highest performance of 9.45 billion instructions per second as measured by the Dhrystone integer benchmark, and this satisfies the requirement of Software Defined Radio. The Raspberry Pi 4 has 40 IO pins, is considered easy to use due to the amount of documentation available and has a cost of

$163.00. The Raspberry Pi 4 Model B was chosen because it provides the best performance with thorough documentation.

Microprocessor			Raspberry Pi 4 Model B			Arduino Uno Rev3		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Performance	0.5	DIMPs	9450	10	5	8	0.08	0.04
IO	0.2	# of pins	40	10	2	14	3.5	0.7
Ease of use	0.2	Experience	Good	8	1.6	Great	10	2
Cost	0.1	dollars	163.00	1.7	0.2	27.60	10	1
Overall value					**8.8**			**3.7**

Microprocessor			Libre Computer Board AML-S905X-CC		
Objective	Weighting Factor	Parameter	Mag.	Score	Value
Performance	0.5	DMIPs	3450	3.65	1.83
IO	0.2	# of pins	40	10	2
Ease of use	0.2	Experience	Fair	6	1.2
Cost	0.1	dollars	60.00	4.6	0.5
Overall value					**5.5**

Table 25: Payload Microprocessor Decision Matrix

4.2.2.2 Camera Selection

16MP Autofocus Camera for Raspberry Pi, Arducam 8MP IMX179 USB M12 Lens Camera, and Spedal Wide Angle USB Webcam were evaluated based on cost, complexity, area, and resolution (Table 26). The complexity is measured by the number of GPIO (General Purpose Input-Output) pins needed. The Arducam 8MP (MegaPixels) IMX179 USB camera uses 0 GPIO pins, making it the easiest to interface. It also has a cost of $61.00, an area of 0.088 in^2, and a resolution of 8 MP. This camera was chosen because it provides a high resolution while maintaining an easy interface.

Camera			16MP Autofocus Camera for Raspberry Pi			Arducam 8MP IMX179 USB M12 Lens Camera Module		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.3	USD	126.0	3.5	1.0	61.0	7.2	2.2
Complexity	0.3	# of GPIO	6	1.7	0.5	0	10	3
Area	0.2	In^2	0.037	10	2	0.088	4.2	0.8
Resolution	0.2	megapixels	16	10	2	8	5.0	1.0
Overall value					5.5			**7.0**

Camera			Spedal Wide Angle USB Webcam		

Objective	Weighting Factor	Parameter	Mag.	Score	Value
Cost	0.3	USD	44.0	10.0	3
Complexity	0.3	# of GPIO	0	10	3
Area	0.2	In^2	0.797	3.0	0.6
Resolution	0.2	megapixels	2	1.3	0.3
Overall value					**6.9**

Table 26: Payload Camera Decision Matrix

4.2.2.3 IMU Selection

BNO055, LSM9DS1, and LSM6DSOX + LIS3MDL were evaluated by their cost, size, and data variety (Table 27). Data varieties refer to the different types of output data IMU can output. The Adafruit BNO055 IMU features a cost of $33.40, a size of 0.8 in^2. Furthermore, it can also provide 8 different types of data (absolute orientation in both quaternion and Euler vector, angular velocity vector, acceleration vector, magnetic field strength vector, linear acceleration vector, gravity vector, and temperature). BNO055 was chosen because it provides the necessary gravity vector and has a reasonable size and cost.

IMU			**Adafruit BNO055**			**Adafruit LSM9DS1**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.2	USD	33.4	6.0	1.2	40.0	5.0	1.0
Size	0.4	In^2	0.8	9.6	3.9	1.0	7.8	3.1
Data variety	0.4	Number of data types	8	10.0	4.0	3	3.8	1.5
Overall value					**9.1**			**5.6**

IMU			**Adafruit LSM6DSOX + LIS3MDL**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value
Cost	0.2	USD	20.0	10.0	2.0
Size	0.4	In^2	0.8	10.0	4.0
Data variety	0.4	Number of data types	3	3.8	1.5
Overall value					**7.5**

Table 27: Payload IMU Decision Matrix

4.2.2.4 Barometer Selection

Adafruit BMP390, MPL3115A2 and Adafruit BME280 were evaluated by their cost, size, accuracy, and number of extra components needed (Table 28). The Adafruit BMP390 features a cost of $10.95, a size of 468 mm^2, accuracy to 0.25 meters, and no extra components are needed. BMP390 was chosen because it has the best accuracy while requiring no extra components, making it the best component and easiest to interface at the same time.

Barometer	**Adafruit BMP390**	**MPL3115A2 Barometer**

Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Cost	0.2	dollars	10.95	10.00	2.00	15.49	6.00	1.20
Size	0.2	mm^2	468.00	7.31	0.73	342.00	10.00	2.00
Accuracy	0.4	meters	0.25	10.00	4.00	0.30	8.33	3.33
Extra Requirements	0.2	components	0.00	10.00	2.00	3.00	3.33	0.67
Overall value					**8.73**			**7.20**

Barometer			**Adafruit BME280 I2C**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value
Cost	0.2	dollars	14.95	7.32	2.20
Size	0.2	mm^2	2086.56	1.64	0.16
Accuracy	0.4	meters	1.00	2.50	1.00
Extra Requirements	0.2	components	0.00	10.00	2.00
Overall value					**5.36**

Table 28: Payload Barometer Decision Matrix

4.2.2.5 Solenoid Selection

Adafruit Small Push-Pull Solenoid, Adafruit Lock Style Solenoid, and Uxcell 4.5mm Push-Pull Solenoid were evaluated by their body length, mass, stroke length, and cost (Table 29). A small body length is desired due to the space constraint in payload, while a bigger stroke length is desired because it ensures a more secure lock of camera during flight. The 4.5mm Stroke Push-Pull Solenoid features a body length of 0.8 inch, a mass of 1.0 ounces, a stroke length of 0.2 inch, and a cost of $10.00. This solenoid was chosen because it provides the longest stroke length while having the shortest body length and lightest weight.

Solenoid			**Adafruit Small Push-Pull Solenoid**			**Adafruit Lock style Solenoid**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Body Length	0.3	inches	1.5	5.4	1.6	2.1	3.8	1.1
Mass	0.3	ounces	1.4	7.1	2.1	5.1	1.9	0.6
Stroke Length	0.2	inches	0.2	9.0	1.8	0.2	9.0	1.8
Cost	0.2	dollars	7.5	10.0	2.0	15.0	5.0	1.0
Overall value					**7.5**			**4.5**

Solenoid			**Uxcell 4.5mm Push-Pull Solenoid**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value
Body Length	0.3	inches	0.8	10.0	3.0
Mass	0.3	ounces	1.0	10.0	3.0
Stroke Length	0.2	inches	0.2	10.0	2.0

Cost	0.2	dollars	10.0	7.5	1.5
Overall value					**9.5**

Table 29: Payload Solenoid Decision Matrix

4.2.2.6 Stepper Motor Selection

28BYJ-48 Reduction gear, NEMA-17, and NEMA-8 motors were evaluated based on their volume, cost, mass, and torque (Table 30). The 28BYJ-48 Reduction Gear Stepper Motor features a volume of 1.1 in^3, cost of $8.50, mass of 1.3 oz, and a torque of 1.0 oz*in. This motor was chosen because of its low volume, cost, and mass. The low volume and mass are critical for fitting into the payload airframe.

Stepper Motor			**28BYJ-48 Reduction Gear**			**NEMA-17 Motor**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value	Mag.	Score	Value
Volume	0.3	In^3	1.1	6.5	1.9	5.1	1.3	0.4
Cost	0.3	dollars	8.5	10.0	3.0	14.0	6.1	1.8
Mass	0.2	ounces	1.3	10.0	2.0	13.8	0.9	0.2
Torque	0.2	oz*in	1.0	0.1	0.0	84.0	10.0	2.0
Overall value					**7.0**			**4.4**

Stepper Motor			**NEMA-8 Motor**		
Objective	Weighting Factor	Parameter	Mag.	Score	Value
Volume	0.3	In^3	0.7	10.0	3.0
Cost	0.3	dollars	20.3	4.2	1.3
Mass	0.2	ounces	2.1	6.2	1.2
Torque	0.2	oz*in	2.3	0.3	0.1
Overall value					**5.5**

Table 30: Payload Stepper Motor Decision Matrix

4.2.2.7 Radio Receiver Selection

An RTL-SDR (Software Defined Radio) dongle was chosen as a radio receiver. This radio dongle converts radio signals to digital signals and then send those signals via USB (Universal Serial Bus) ports to the Raspberry Pi to decode. This method of receiving radio was chosen because the integrated radio receivers that are commercially available do not operate in the range of 144.90 MHz to 145.10 MHz, which is the range NASA will be transmitting on. The receivers that are commercially available are too big to be used in this payload. As a result, an integrated SDR dongle was chosen as the method to receive APRS radio signal from NASA.

4.2.3 Selected Software Components

4.2.3.1 Serial Communication Protocol

The payload design makes use of an external IMU and barometer. These components are needed to get acceleration, orientation, and pressure data to the Raspberry Pi so a serial communication protocol will need to be implemented. The options available are limited as the IMU only supports the I2C and UART protocols and the barometer only supports the I2C and SPI protocols.

I2C was chosen in large part because both external components share the I2C protocol. This means that software and physical complexity is reduced since the same code and wires can be used to interface with both devices. The alternative being that the IMU would use the UART protocol, and the barometer would use the SPI protocol. The UART protocol relies on asynchronous communication meaning two different RX and TX data transmission lines would need to be used. In addition to the four different wires needed for the SPI protocol. Therefore, if I2C was not chosen there would need to be six wires connecting to the external components instead of two and both protocols would need to have separate software libraries written to control the sending and receiving of data. Due to these complications, I2C was chosen and will be used to communicate with both the IMU and barometer.

4.3 System Level Design

4.3.1 Payload Mechanical Design

The payload systems will be positioned 120° apart and linearly offset. Each payload system will have a camera housing that integrates a camera, camera mount, and the motors. The airframe has three rectangular cut-outs that each camera system will rotate out of. The camera mount sits almost flush with the cut-out airframe so that each camera can exit the airframe upon landing.

Each payload system includes: a camera with a case, a solenoid to hold the camera, camera case, and camera mount down, a stepper motor to rotate the system about the z-axis, and a spring hinge to rotate the system upward relative to the ground (Figure 44). The camera case is fastened to the camera to protect the lens from environmental damage. The solenoid tongue is extended into a cylindrical hole in an extrusion of the camera mount, and the system remains in the airframe cutout until the tongue is retracted and system released. The stepper motor is fastened to the spring hinge and spring mount with 8-32 fasteners so that when the solenoid tongue is retracted and spring hinge releases to its 90° position (Figure 53), the stepper motor can rotate the camera and camera mount about the z-axis.

The camera mount will be 3D printed so the top surface is almost flush with the original curvature of the airframe, so the camera mounts minimally affect aerodynamics. Upon landing, one solenoid tongue will retract, and that camera system will rotate 90° out of the airframe. A stepper motor will rotate the camera mount, and thus the camera, about the z-axis.

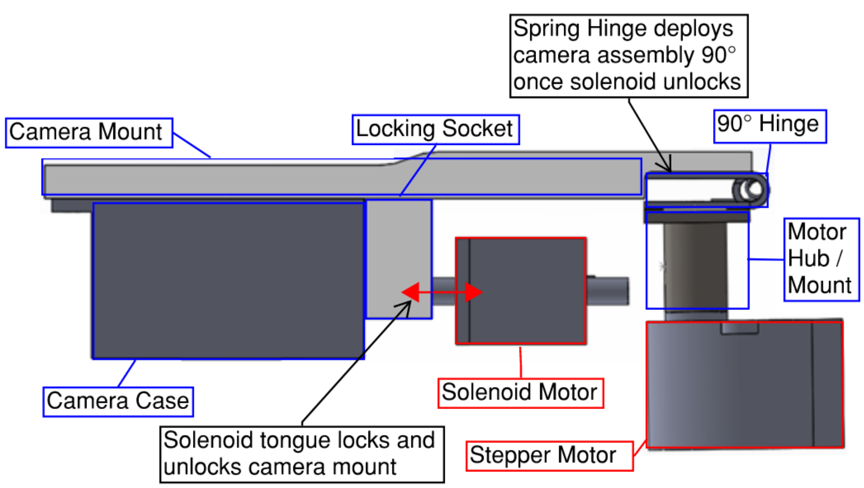

Figure 44: Camera System with Labeled Components

4.3.1.1 Camera Mount and System

The camera system includes a camera mounted to the PET-G camera mount (Figure 45, Figure 47) and is enclosed by a camera case (Figure 46) fastened to the inside of the airframe. Each camera system will be mounted on a 90° spring-loaded hinge with M4 fasteners, so that it lays almost flush with the exterior of the launch vehicle during flight and can rotate out of the airframe on the hinge once the vehicle has landed. The shape of the mount was designed to ensure there would be minimal effects on the aerodynamics while still functioning in a way that minimizes space and allows for rotation of the system. The spring-loaded hinge is secured closer to the forward end of the airframe so that the direction of the airflow will always push the hinge closed during flight. This is to protect the launch vehicle and ensure that mechanical failures will have no effect on the aerodynamics of the launch vehicle. The base of the mount has a small cutout for the hinge to be fastened to a flat rigid lip. The spring-loaded hinge is secured to the surface of the camera mount that is flush with the airframe with M6 fasteners. One side of the hinge is fastened underneath the contacting surface of the camera mount, and on the other side to a spring mount. The spring mount is fitted on the stepper motor that will rotate the camera, mount, locking lug, and spring system when prompted by RF commands (Figure 48, Figure 49).

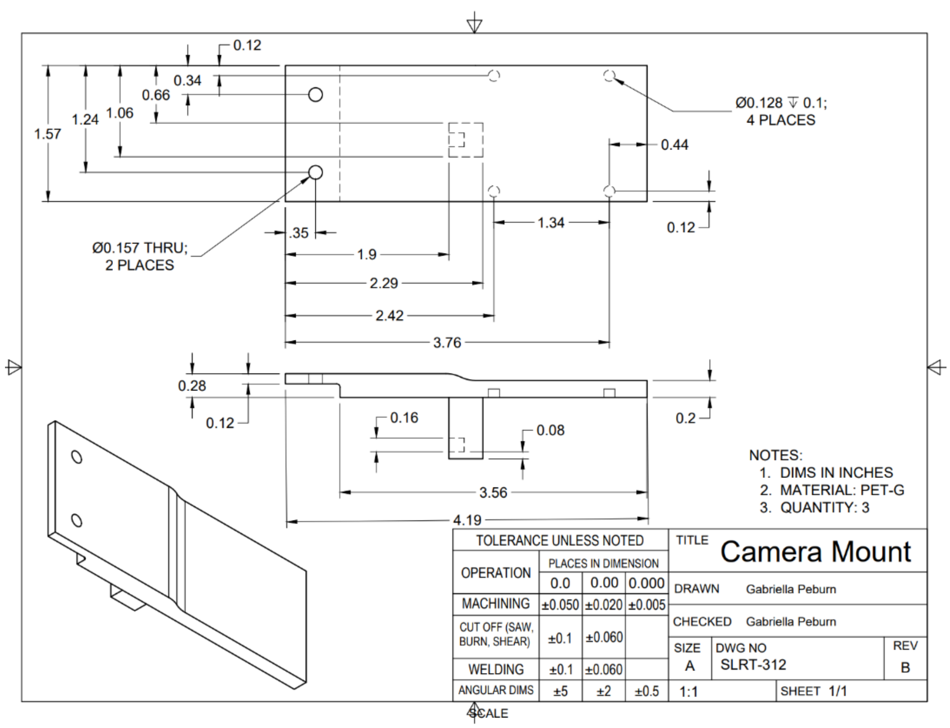

Figure 45: Camera Mount Drawing

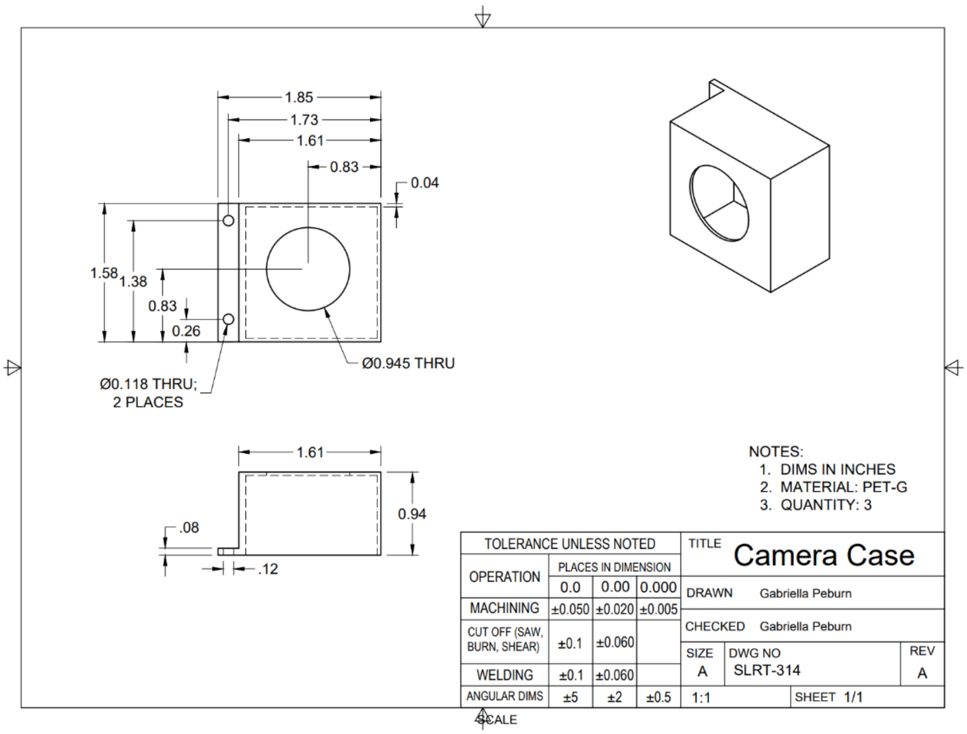

Figure 46: Camera Case

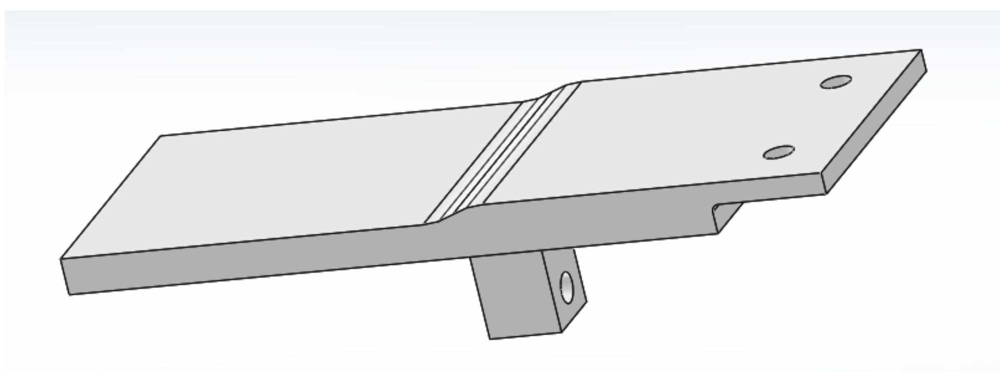

Figure 47: Camera Mount CAD

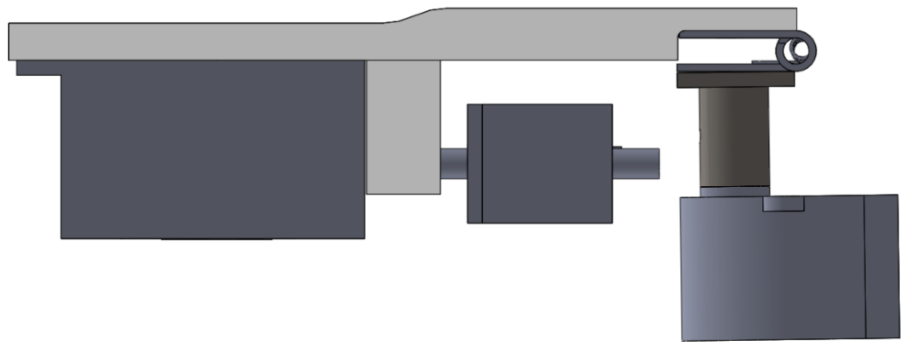

Figure 48: Payload Camera System Side View

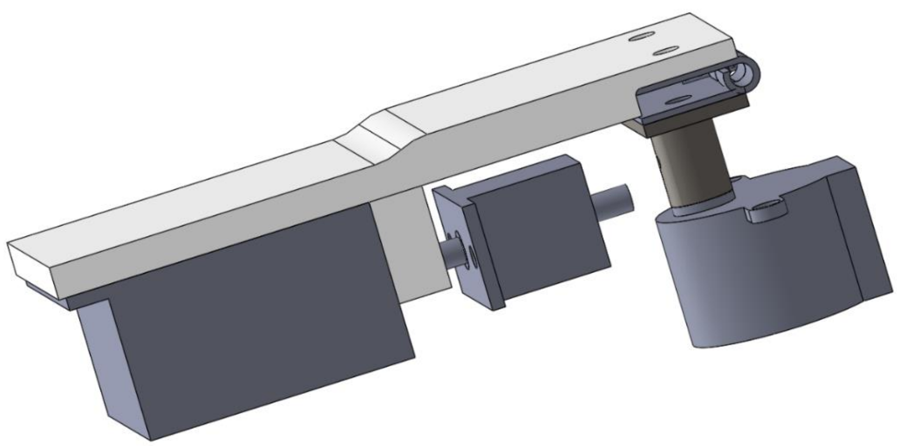

Figure 49: Payload Camera System Isometric View

The PET-G camera mount includes a locking lug extrusion inside the airframe next to the camera (Figure 50). The lug extrusion features a cylindrical cut-out for a mini solenoid tongue to fit inside. The camera system is held down by the tongue of the solenoid motor and can carry a maximum load of 7.94 oz, well more than the required load of 1 oz. It will retract itself out of the lug after the launch vehicle lands. The retraction of the tongue releases the hinge and springs the camera system out of the airframe. The solenoid tongue is extended by default, so the system is locked while powered off. The solenoid must be powered on to retract, so the camera systems cannot be extended mid-flight, providing mechanical safety for electronic failure.

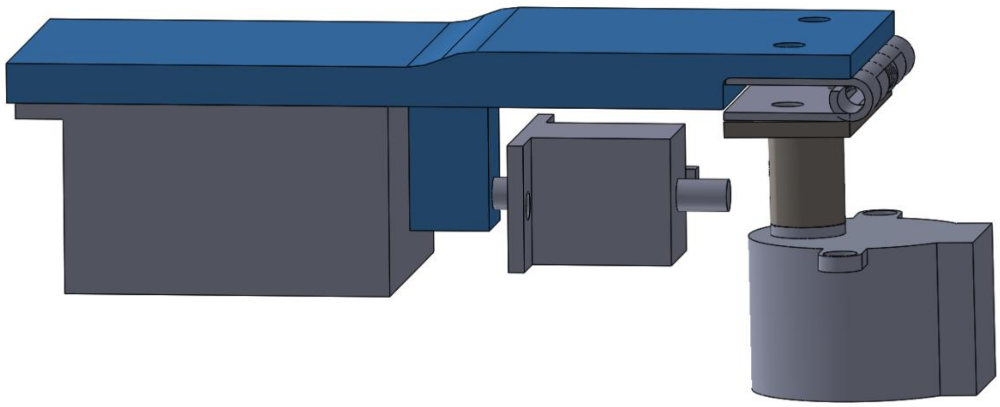

Figure 50: Camera Mount includes Locking Lug

4.3.1.2 Payload System

The payload system refers to everything inside the payload housing including the components in Figure 50. When the solenoid is retracted, the camera mount system will rotate up so the spring-loaded hinge will spring to its naturally released state out of the airframe (Figure 53, Figure 54). The spring-loaded hinge (Figure 51) is fastened to a spring mount and hub (Figure 52). The mount and hub are 3D printed with PET-G, and the torque is transmitted through one M6 threaded insert and set screw. The spring mount is secured to a stepper motor that rotates the camera system about the z-axis. The stepper motor aligned with the fin normal to the ground will receive commands for the payload challenge. An IMU accelerometer will determine which camera mount is rotated up, and only that camera will receive the commands and capture images. The corresponding stepper motor will rotate the entire camera system, including the hinge, at the surface of the airframe (Figure 55).

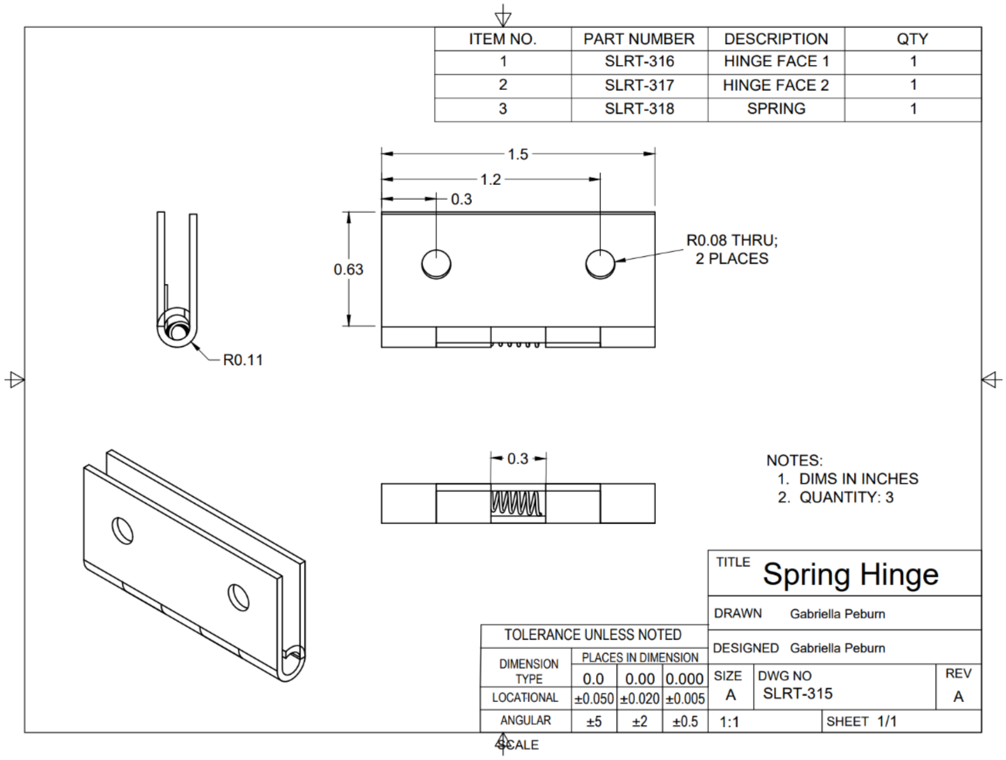

Figure 51: Spring-Loaded Hinge Drawing

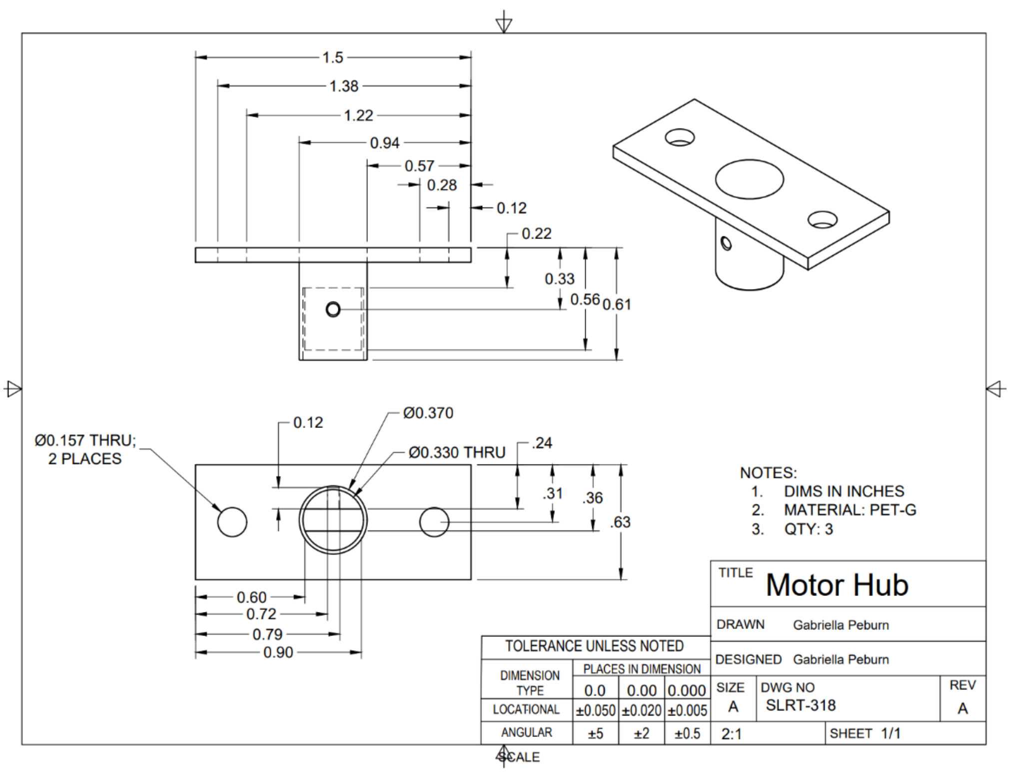

Figure 52: Stepper Motor Hub

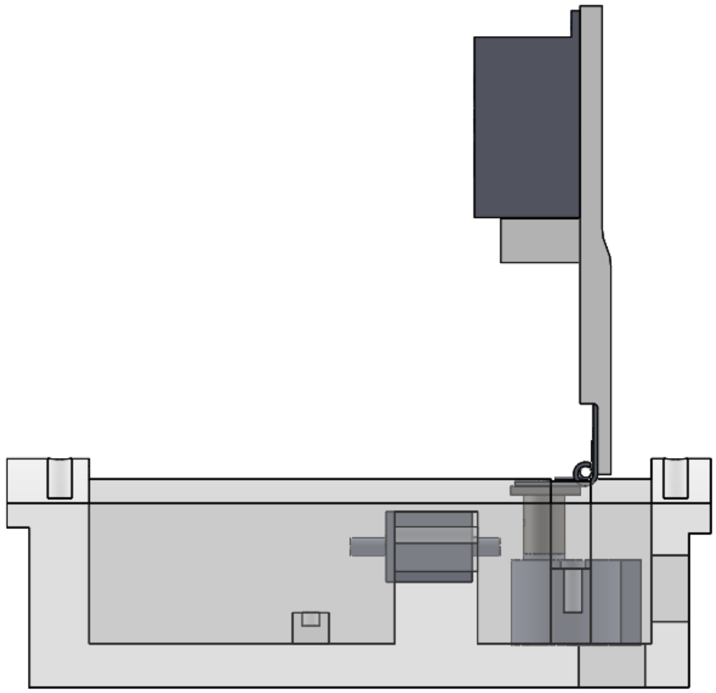

Figure 53: Payload System Rotated 90° Out of Airframe

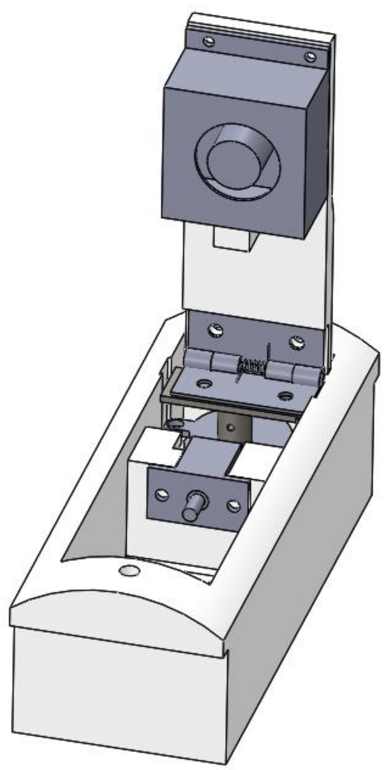

Figure 54: Payload System Rotated Isometric View

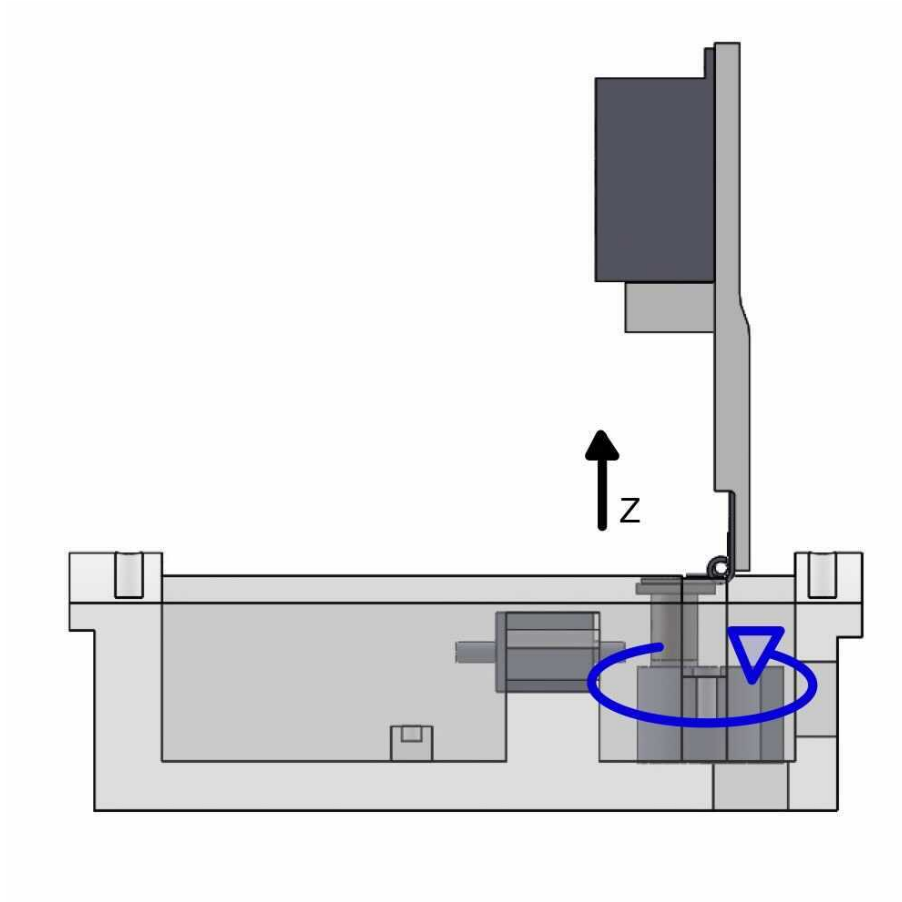

Figure 55: Direction of Rotation of Stepper for Camera System

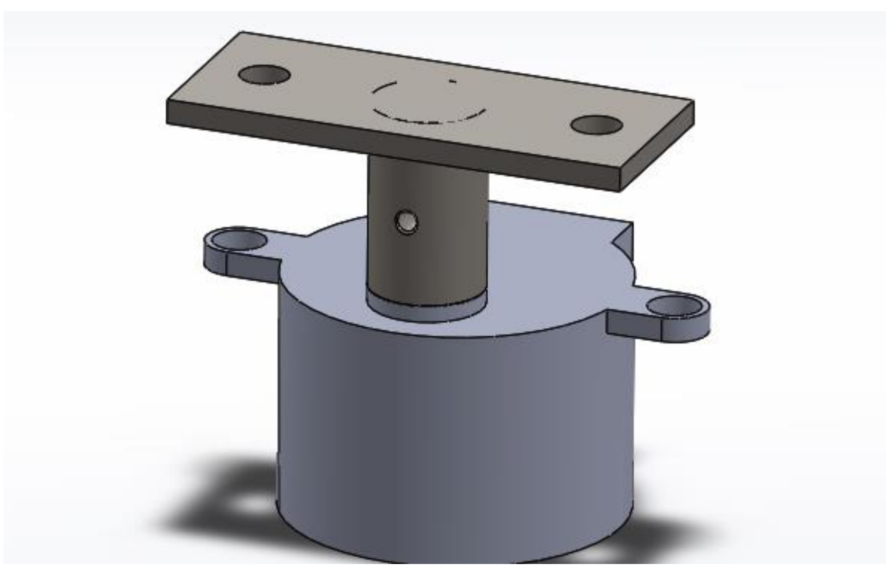

Figure 56: Motor Hub and Mount Assembly

4.3.1.3 Payload Housing

Each payload housing will be mounted inside the airframe with threaded inserts and 8-32 fasteners so that the outer surface of the mount is flush with the outer diameter of the airframe. The camera stepper motor is fastened under the airframe, and each of the three camera systems has its own housing (Figure 57). Each housing features two rectangular holes in its base for motor wires to pass through. The housing serves the purpose of securing the motors and fasteners while also protecting the launch vehicle from unwanted uncertainties like mechanical failure (Figure 58, Figure 59). The central payload housing of the three payload housings will include a designated fastening compartment for the radio. The 3D printed bottom surface of the radio compartment will be fastened to the walls of that compartment with M2.5 fasteners.

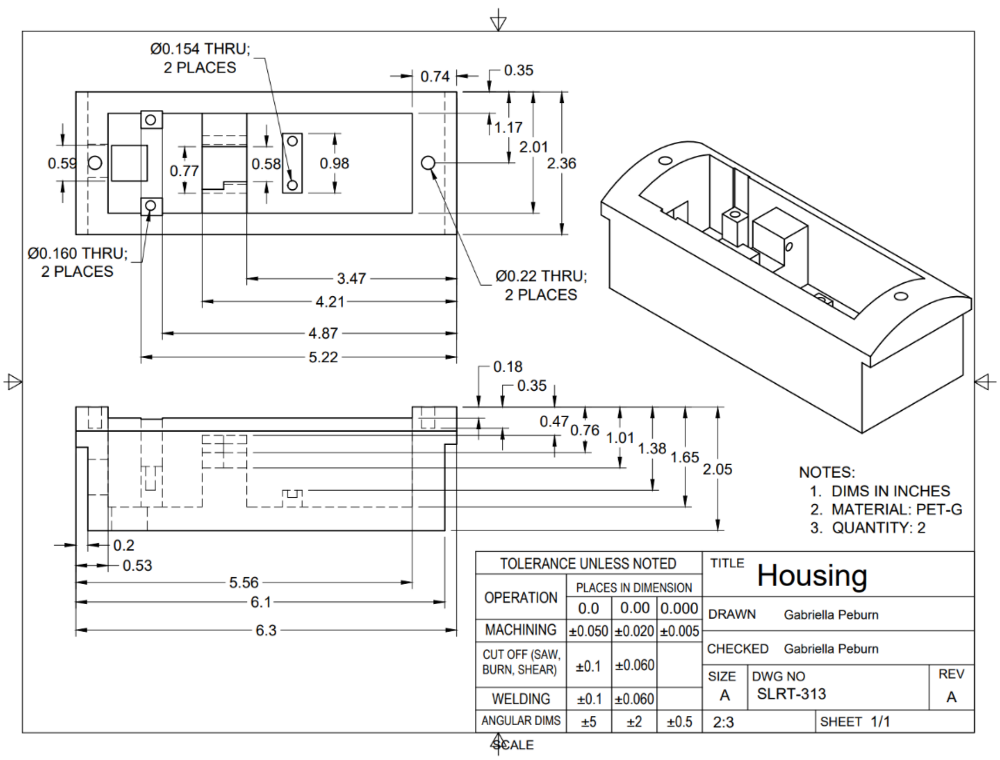

Figure 57: Payload Housing Drawing

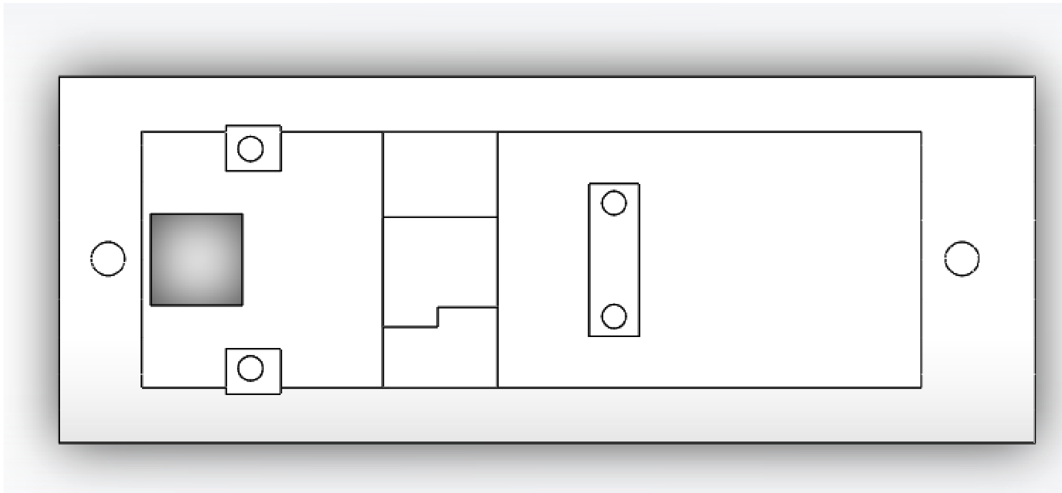

Figure 58: Payload Housing Top View

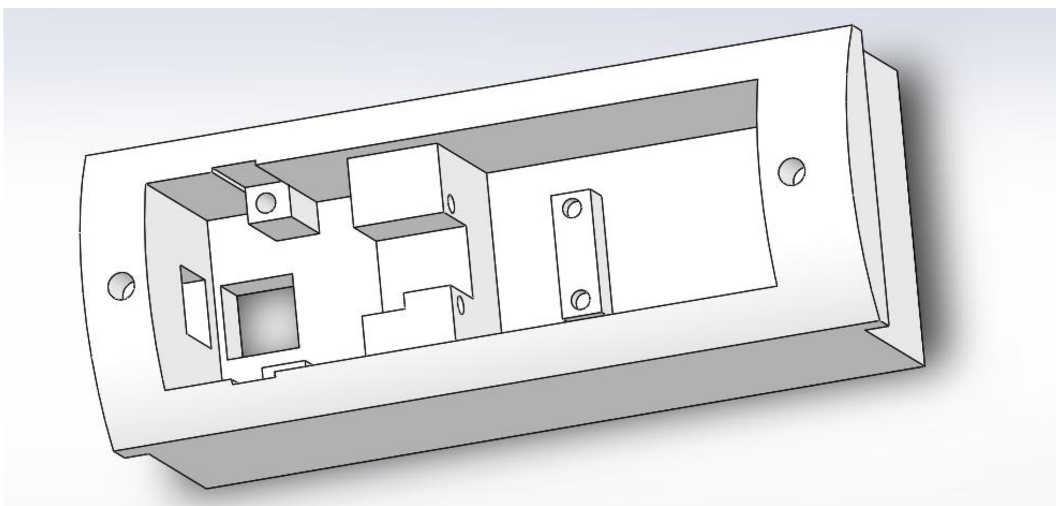

Figure 59: Payload Housing Isometric View

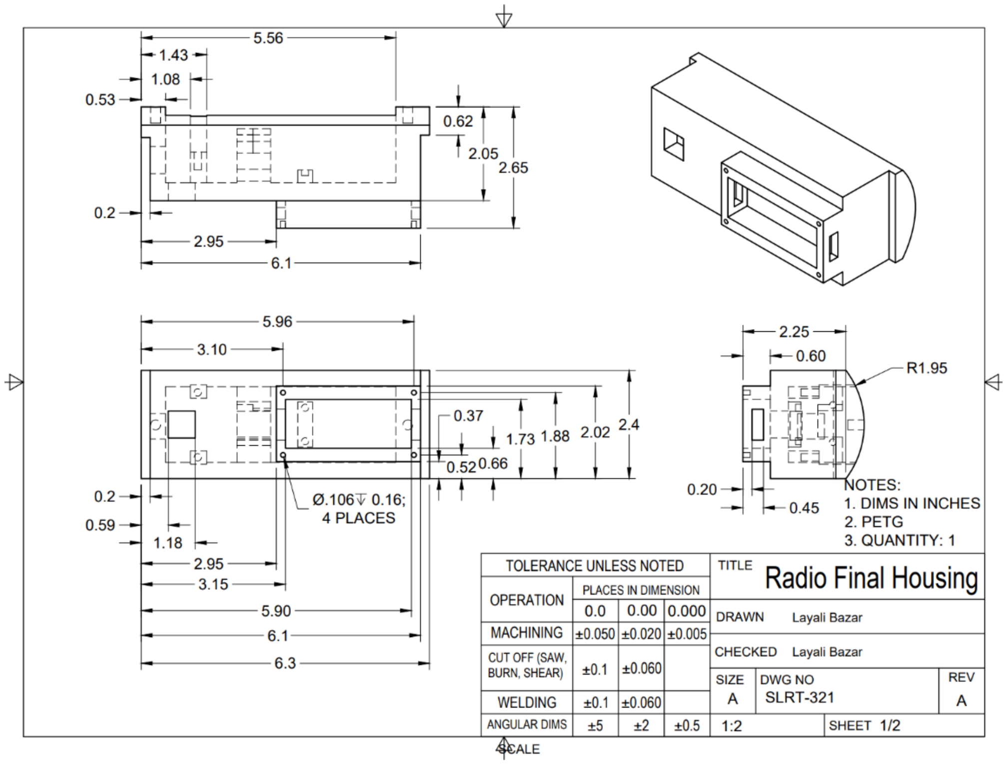

Figure 60: Radio Housing in Central Payload Housing

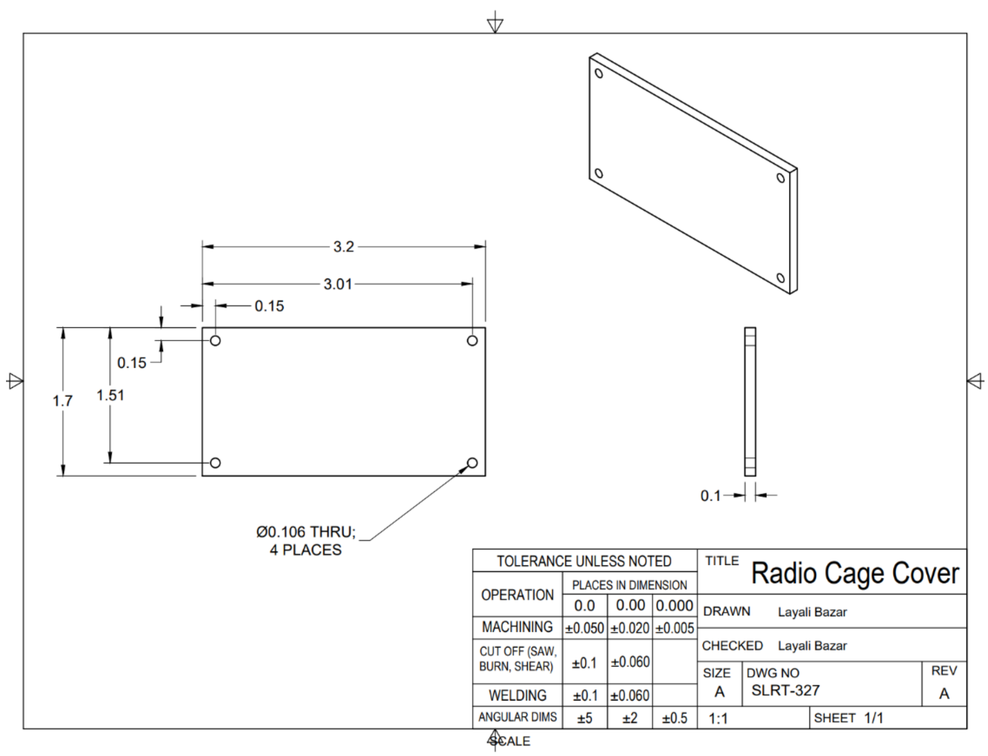

Figure 61: Radio Housing Cover

4.3.1.4 Electronics Housing

All the electronics are compiled on a 3D printed sled made of PET-G (Figure 62) and fit together inside the coupler (Figure 19, Figure 67). The batteries and battery covers will be housed in the coupler in their own retention housings (Figure 63, Figure 64). All these components are secured with M2.5 fasteners for easy assembly and testing. One payload housing will contain the radio housing (Figure 60). The batteries are contained in their own 3D printed cases fitted together with slots for the threaded rod (Figure 67, Figure 68, Figure 69).

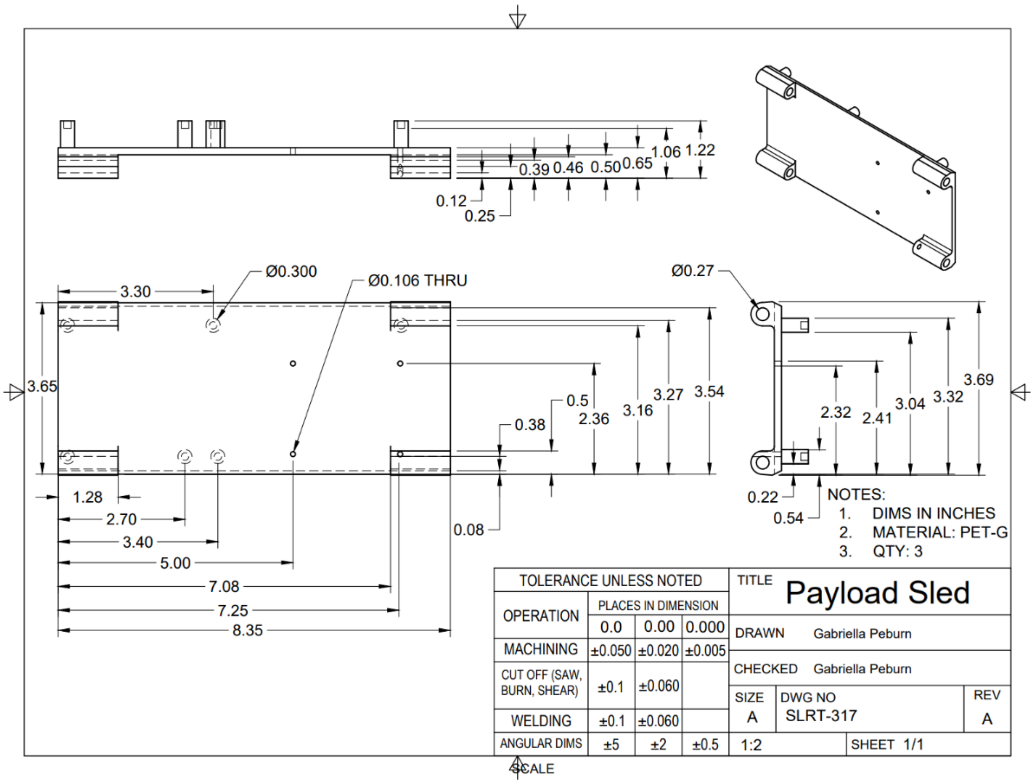

Figure 62: Payload Electronics Sled Drawing

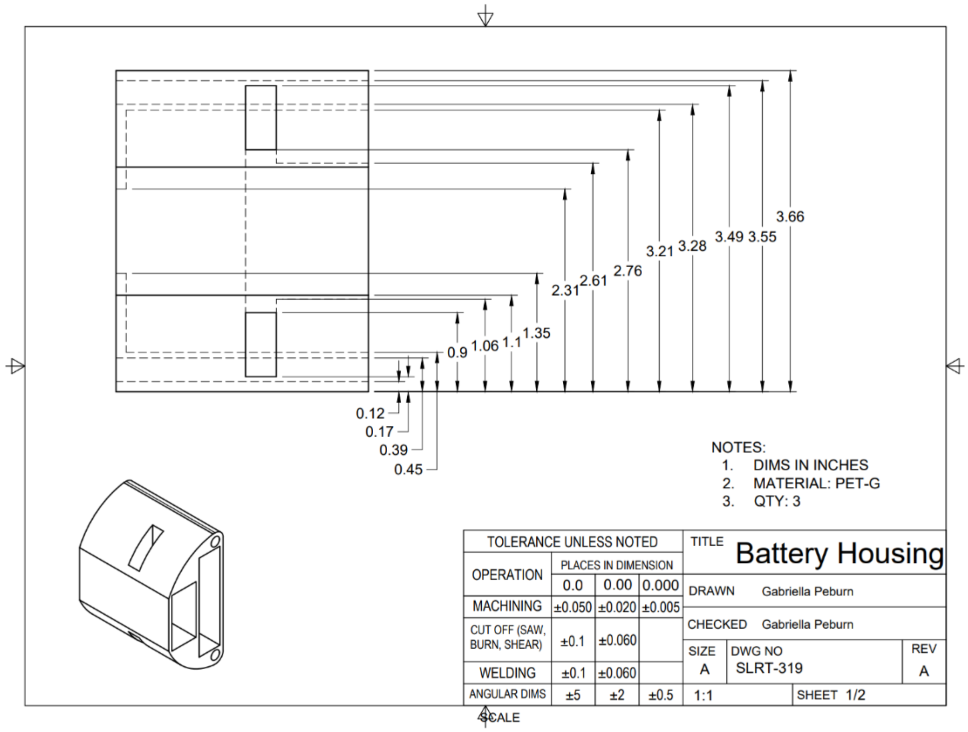

Figure 63: Battery Housing Drawing

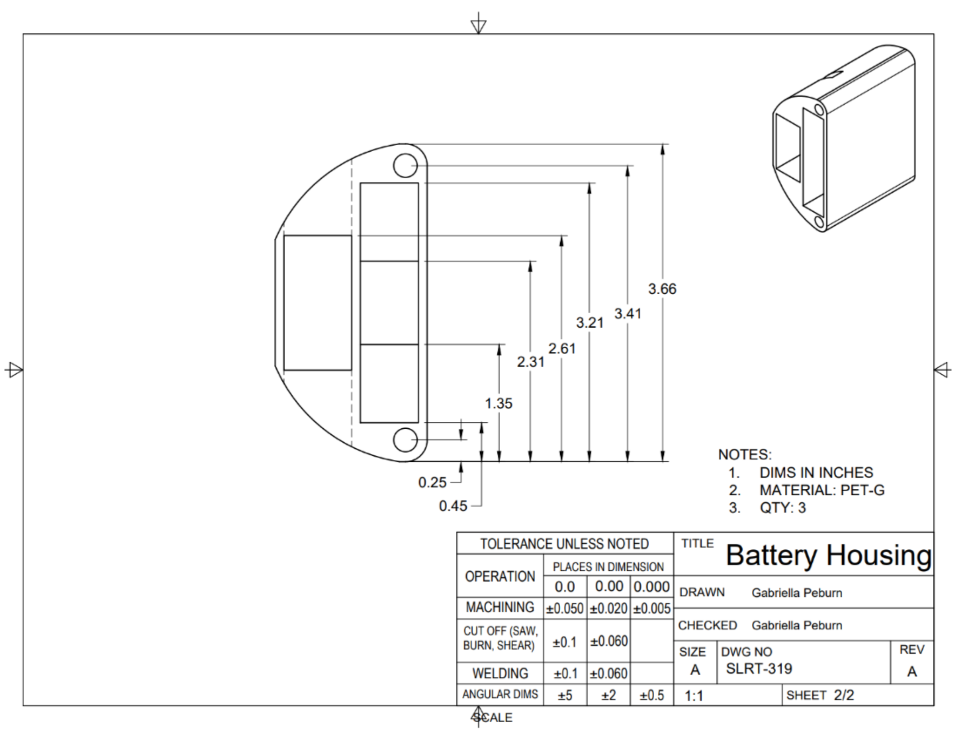

Figure 64: Battery Housing Drawing Sheet 2

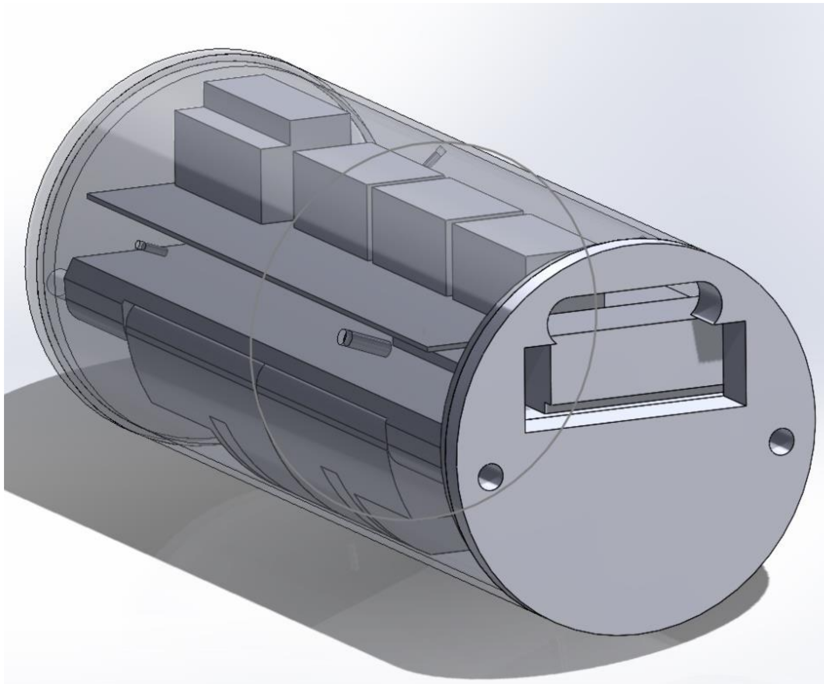

Figure 65: Isometric View of Payload Coupler Assembly

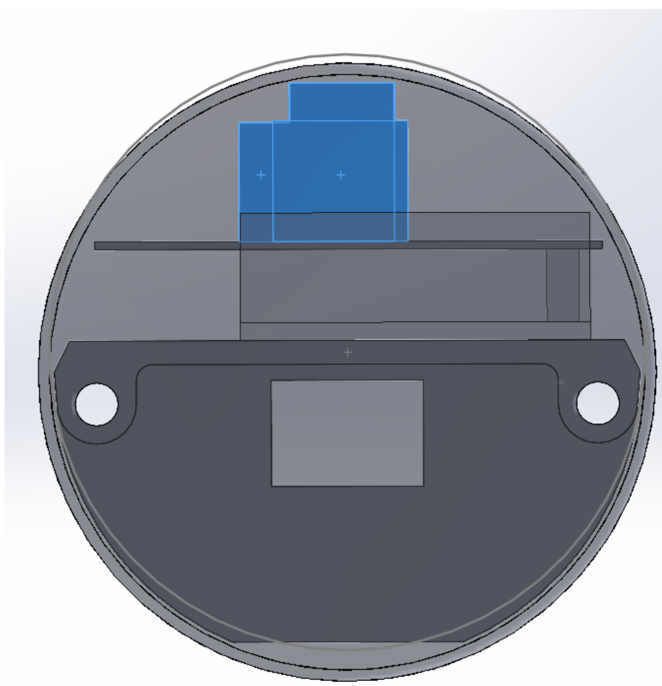

Figure 66: Side View of Payload Coupler Assembly

PARTS LIST			
ITEM	QTY	PART NUMBER	DESCRIPTION
1	1	SLRT-317	SLED
1	2	SLRT-319	BATTERY HOUSING

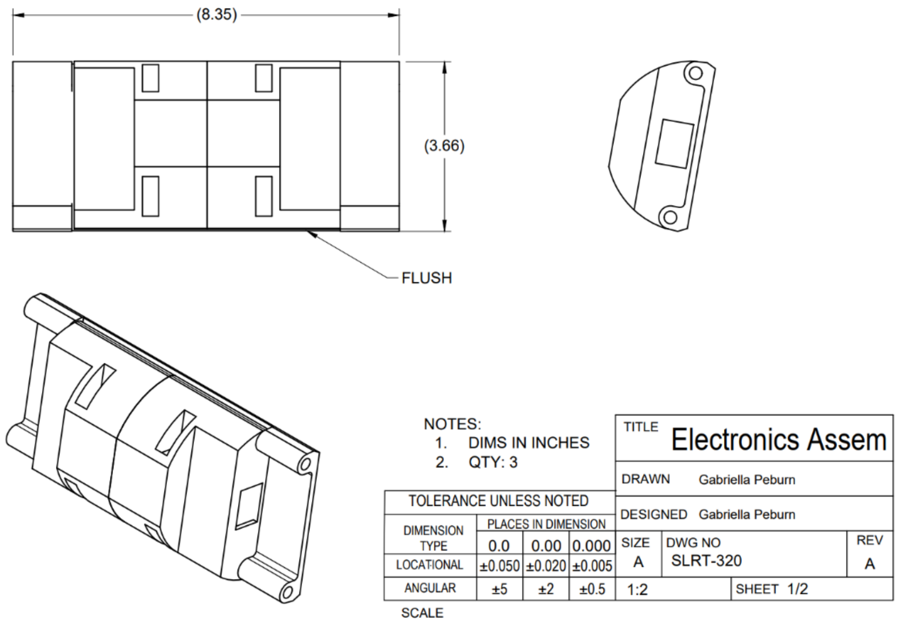

Figure 67: Payload Electronics Assembly Drawing

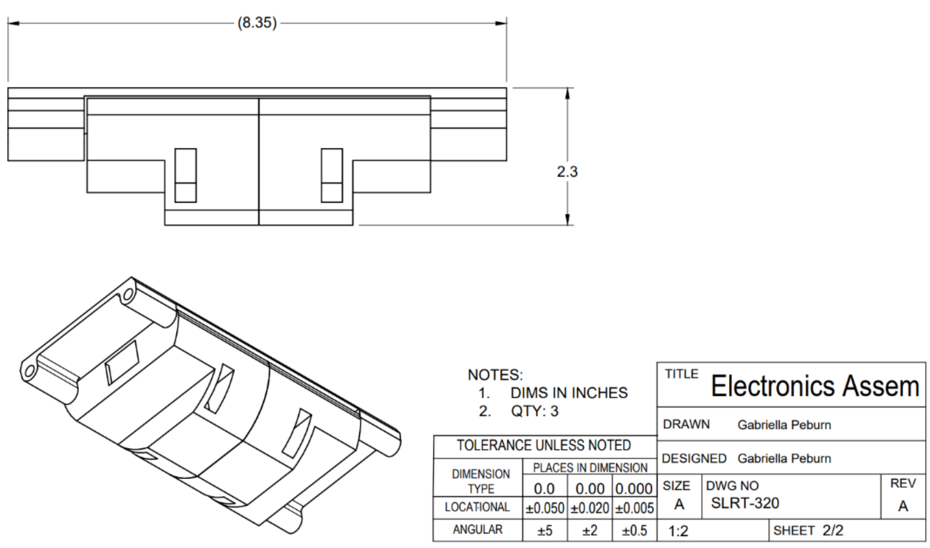

Figure 68: Payload Electronics Assembly Sheet 2

4.3.2 Payload Electronic Design

The payload uses a Raspberry Pi 4 Model B with 8GB of RAM (Random Access Memory) as the microprocessor. All components are connected to the microprocessors through either GPIO (General Purpose Input Output) pins or USB ports (Figure 69). The 5V Piezo is used as an output device. During assembly and testing of the payload, piezo will generate sequences specified by software to allow the payload team to know the state of payload system. Additional components connected to the microprocessor can be further separated into 3 subsystems: a radio subsystem, a sensor subsystem, and an actuator subsystem (Figure 69).

The radio subsystem consists of an RTL-SDR (Software Defined Radio) Dongle (Figure 69) to receive radio signal from NASA and a physical antenna to amplify the radio signal. The SDR dongle is an off-the-shelf part that converts radio signals to digital signals and send them to a microprocessor through a USB port.

The sensor subsystem consists of an BNO055 IMU, a BMP390 Barometer, 3 push buttons, and 3 cameras (Figure 69). The IMU is used to measure the orientation and acceleration of the rocket. This data will be computed to detect launch and landing. The barometer is used to measure the altitude of the rocket, and this data will be used as a redundency for the IMU data. A push button is used as feedback signals from each camera housing. When the camera is locked in place, the button will be pressed. Therefore, the Raspberry Pi will know the state of the camera, and Raspberry Pi will not rotate the camera about the z-axis if the camera is detected to be locked. Finally, the 3 USB cameras are connected to the Raspberry Pi via USB ports.

The actuator systems includes one solenoid, one stepper motors, and their controllers for each of the three systems. Everything in circle in Figure 69 are actuator subsystem. The solenoids are used to lock the camera in place. When the solenoid retracts due to an electrical current, the camera will spring out of the airframe due to the torque provided by the spring hinge. Each of the 3 solenoids are controlled by a circuit centered around a N-Channel MOSFET (Metal–Oxide–Semiconductor Field-Effect Transistor) gate. The stepper motors are used to rotate the camera about the z-axis accurately. Each of the stepper motors is controlled by an OTS ULN2003 stepper motor controller.

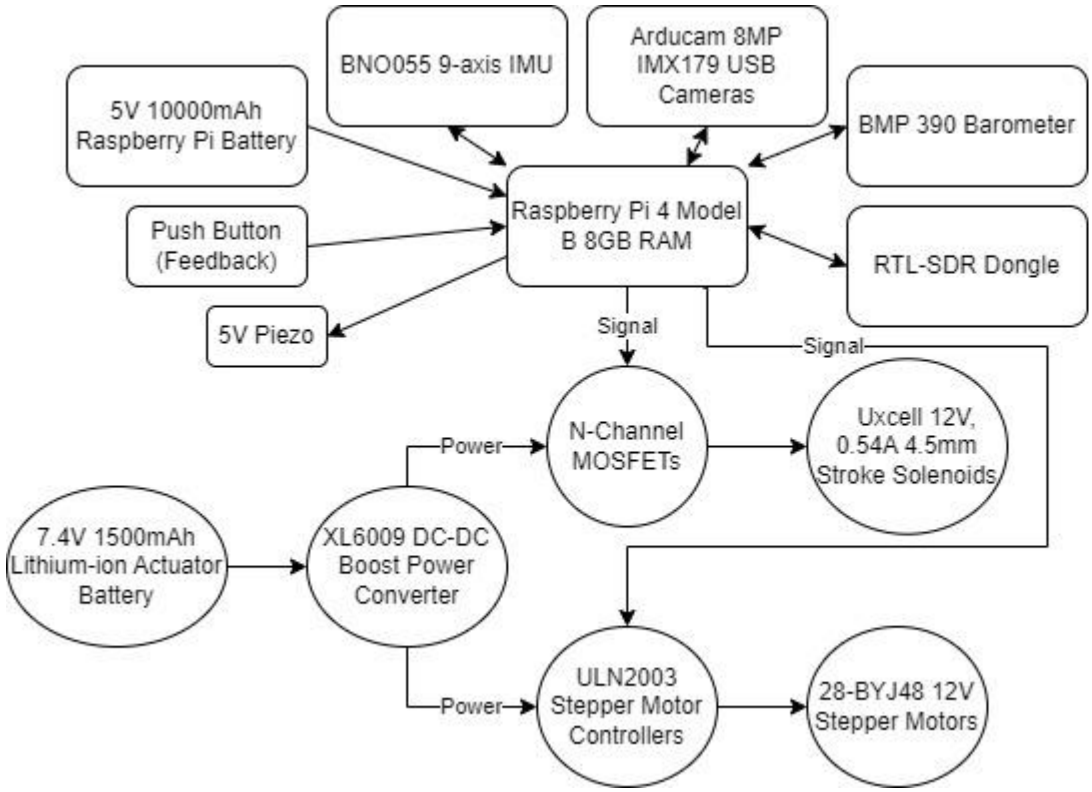

Figure 69: Payload Electronic Components Diagram

4.3.2.1 Power Supply

Figure 69 and Figure 70 shows that the payload has two separate power supplies. One power supply is for the Raspberry Pi, radio, and sensors; another power supply is for the actuators.

The power for Raspberry Pi, radio, and sensors are provided by a portable USB battery pack with 5V and capacity of 10000mAh. The portable charger is connected to Raspberry Pi through a USB Type-C port.

A separate 7.4V lithium-ion battery is used to provide power for the actuators. This battery has a capacity of 1500mAh. The XL6009 DC (Direct Current) to DC power converter is used to boost the voltage to 12V, which is the requirement of the motor.

Two separate power supply systems are utilized due to the different requirements of the two systems. The Raspberry Pi, radio, and sensors need to be on for a few hours when it’s waiting on the launch pad. Therefore, they will need a battery with big capacity to supply the power. The actuators, on the other hand, will only be on for a few seconds when the payload is operating. However, when actuators are operating, they draw at least 1A of current. If the Raspberry Pi and actuators share the same power supply, this additional load can cause the voltage to fluctuate and can potentially affect the operation of the Raspberry Pi. To ensure a stable power supply for the Raspberry Pi, a separate battery with smaller capacity is used to provide power for all the actuators on board. The batteries are housed in 3D printed covers made of PET-G (Figure 63).

4.3.2.2 Electronic Component Interface

The Raspberry Pi connects to the IMU, barometer, push buttons, piezo, and actuators through its GPIO pins (Figure 70). The cameras and radio dongles are connected to the Raspberry Pi through USB ports and are not shown in Figure 70. The IMU and barometer interface with Raspberry Pi through I2C communication pins. Each of the 3 push buttons are connected to a GPIO pin; the pins are used as input pins to receive signals from push buttons. All push buttons are grounded by a 10K Ohm pull-down resistor (Figure 70).

Each of the 3 solenoid control circuits are connected to a GPIO pin; these pins serve as output pins and send signals to control the N-Channel MOSFET gate, which in turn controls the solenoid. In each solenoid control circuit, a 10K Ohm pull-down resistor is used to ground the MOSFET gate and a 0.2K Ohm resistor are used to limit the current flowing through the GPIO pins. A diode is used to disspitate the electrical impulse generated by solenoid when the shaft extended back due to the internal spring (Figure 70). The piezo is controlled by the same circuit as solenoid controller, with the absence of diode.

Each of the 3 stepper motors are controlled by a ULN2003 stepper motor controller. Each stepper motor controller connects to 4 GPIO pins (Figure 70); those pins are used as output pins.

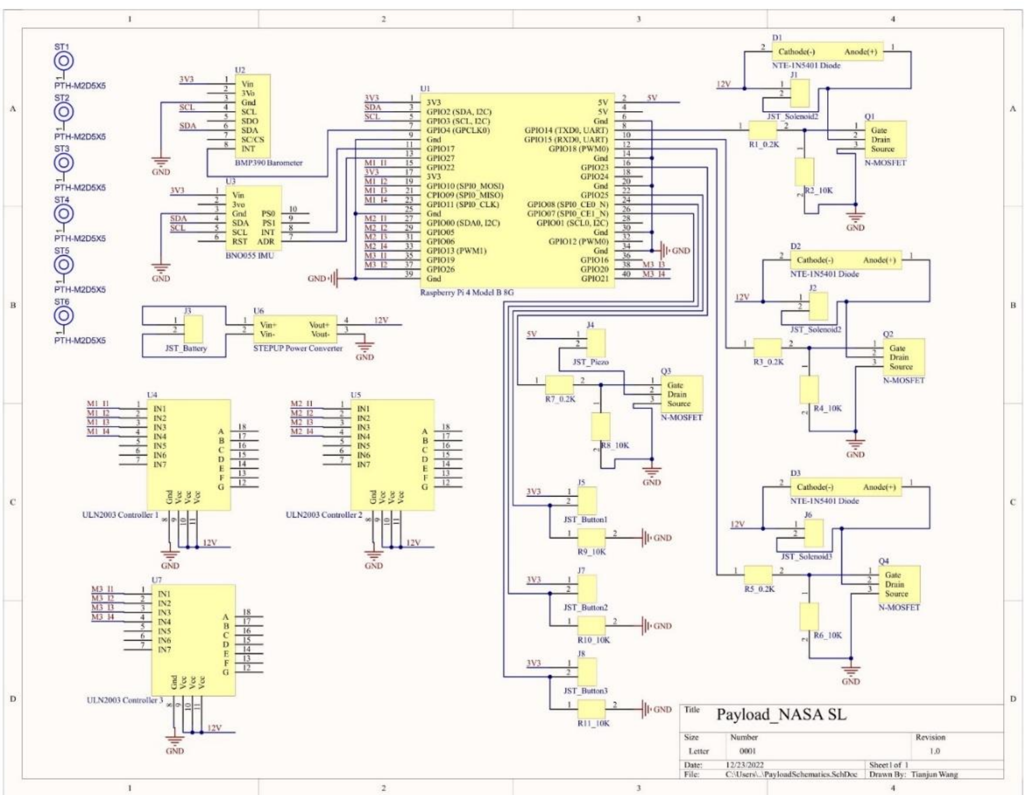

Figure 70: Schematics of Payload Electronics

All the connections shown in Figure 70 will be implemented by a custom PCB (Printed Circuit Board) designed by the team. The layout of the custom PCB is shown in Figure 71. The PCB has two layers. Blue shows the connections made in bottom layer, and red shows the connections made on top layer. The green texts are texts on the PCB that will help the team to identify the position of components during the assembly process. The blue mesh is the copper net that connects all the ground pins, and the red mesh is the copper net that connects all the 12V pins together. These are the pins that provide power to the actuators, and the copper net allows more current to flow through, thus increasing the robustness and reliability of the PCB.

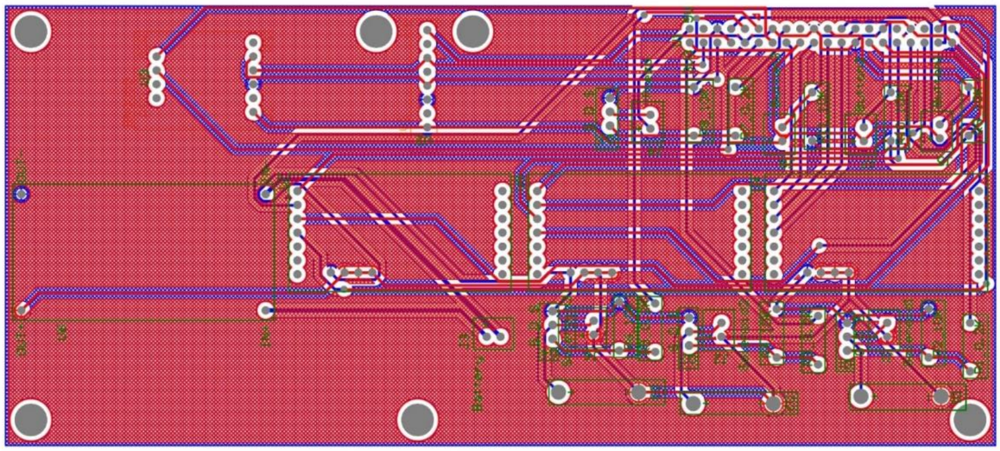

Figure 71: Payload Electronics PCB Layout

The PCB is 7.2 in by 3.2 in, and this was determined by the shape of the payload coupler. The XL6009 boost power converter and three stepper motor controllers are placed in the middle of the PCB because of the space constraint in the payload coupler. As shown in Figure 65 and Figure 66, the 4 parts can only fit in the rocket when they are placed in the middle of payload bay. The layout and shape of PCB is therefore dictated by this constraint.

4.3.3 Payload Software Design

The payload software design consists of four systems that will control all components of the payload to successfully meet all NASA and team goals. The software will be run on the Raspberry Pi 4 and will be enabled once the Raspberry Pi has been powered on. The main software program consists of a state system where only certain systems will be enabled depending on what state the payload is currently in. State transitions are triggered by the sensor system and are defined as strict conditions in the program to ensure that the payload state remains accurate. Figure 72 describes the state flowchart that the payload will follow once powered on.

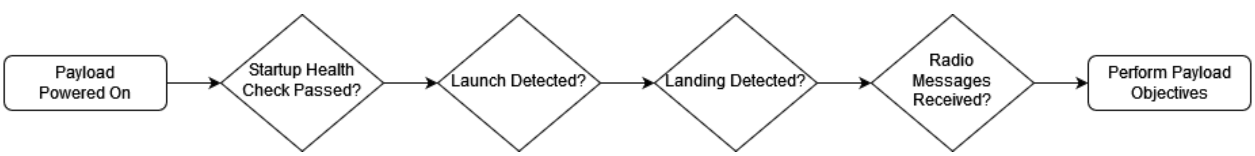

Figure 72: Software State Flowchart

4.3.3.1 Sensor System

The sensor system will be responsible for interfacing with the onboard IMU and barometer. The main goal of this system is to understand the current position and state of the payload. State tracking will be done by checking for sudden and significant changes in acceleration and pressure. This will be done as accurately as possible as the payload needs to know when the rocket launches and subsequently lands. The sequence of discrete steps the sensor system needs to do is illustrated in Figure 73.

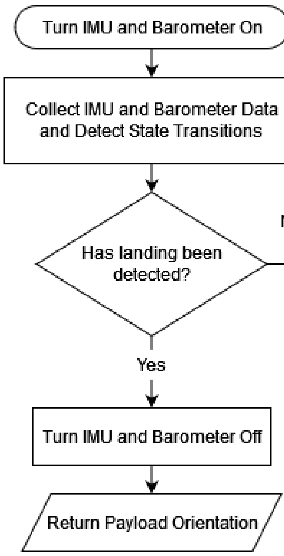

Figure 73: Sensor System Flowchart

Communication with the IMU and barometer will be facilitated via the I2C serial communication protocol. The benefit of this protocol is that only two pins on the Raspberry Pi need to be utilized as both devices can share the same connections. This is because I2C uses device addressing to communicate with external devices meaning that bus contention is avoided. Table 31 shows the preconfigured addresses that will be used to indicate a specific sensor for communication.

Sensor	Primary I2C Address	Secondary I2C Address
Adafruit BNO055 IMU	0x29	0x28
Adafruit BMP390 Barometer	0x76	0x77

Table 31: Primary and Secondary I2C Device Addresses

To get data from the IMU, the software system will need to send read requests via I2C with the target being specific registers that hold current acceleration sensor data. Table 32 lists the name, address, and purpose of each register that the sensor system will read from to collect all acceleration data.

Register Name	Register Address	Purpose
ACC_DATA_X_LSB	0x08	Lower 8 bits of acceleration in the X direction relative to the IMU
ACC_DATA_X_MSB	0x09	Upper 8 bits of acceleration in the X direction relative to the IMU
ACC_DATA_Y_LSB	0x0A	Lower 8 bits of acceleration in the Y direction relative to the IMU
ACC_DATA_Y_MSB	0x0B	Upper 8 bits of acceleration in the Y direction relative to the IMU
ACC_DATA_Z_LSB	0x0C	Lower 8 bits of acceleration in the Z direction relative to the IMU
ACC_DATA_Z_MSB	0x0D	Upper 8 bits of acceleration in the Z direction relative to the IMU

Table 32: IMU Acceleration Data Registers

Once all six registers are read from, the system will combine the values into three 16-bit signed integers corresponding to the X, Y, and Z acceleration component vectors. These three values, along with pressure data, will be continuously checked against specific constrains to see if the vehicle should transition to the next state.

These values will also be stored and made available for the other software systems as they require, such as for checking the orientation of the payload once it has landed. Specifically, the sensor system will need to determine which of the three camera assemblies is upright once the payload has landed and inform the other systems. To do this, the IMU will be situated in the payload such that the X component will be in line with the length of the rocket. This allows for the calculation of roll, the rotation around the X axis, via the two other component vectors. Since the only acceleration after the payload has landed that is acting on the IMU is the acceleration due to gravity, the upright camera assembly can be calculated based on the magnitude of either the Y or Z vectors. The result from this calculation will then be made available to the camera and motor systems when they need to decide which respective electrical component to activate.

The sensor system is also responsible for gathering pressure data from the barometer. Like the IMU, it requires sending I2C read requests to specific registers on the device. In this case, Table 33 shows the three registers needed to get pressure data.

Register Name	Register Address	Purpose
PRESS_XLSB_7_0	0x04	Lowest 8 bits of the 24-bit pressure data
PRESS_LSB_15_8	0x05	Middle 8 bits of the 24-bit pressure data
PRESS_MSB_23_16	0x06	Highest 8 bits of the 24-bit pressure data

Table 33: Barometer Pressure Registers

The number of registers is reduced since pressure data is just a singular numerical value compared to the three vectors for acceleration. However, the data is stored as a 24-bit value which provides a higher resolution than the 16-bit acceleration values which will be beneficial in ensuring that the payload accurately transitions to the next state only when it should.

4.3.3.2 Radio System

The radio system will be responsible for listening to the radio commands from NASA. The goal of this system is to receive which commands the payload should perform and send that information to the other systems. Two programs will be used in addition to the primary C++ payload software as illustrated in Figure 74.

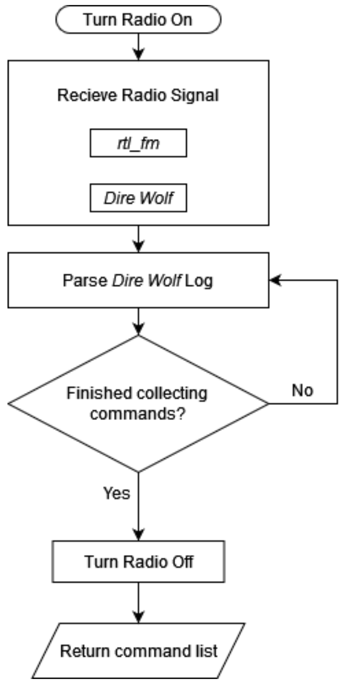

Figure 74: Radio System Flowchart

The first program is *rtl_fm* which is provided from the RTL-SDR software library that comes with the RTL-SDR receiver dongle. This program will be used to configure the RTL-SDR dongle so that it can listen to and receive radio commands. This includes tuning the radio to the frequency range of 144.90 MHz to 145.10 MHz as required by NASA. Table 34 describes the configuration values that will be used to properly setup the RTL-SDR dongle.

Program	Parameter	Value	Description
rtl_fm	-f	144.50M	The frequency the dongle will be tuned to
rtl_fm	-o	4	The output format of the "audio" from the radio

Table 34: rtl_fm Configuration Parameters and Values

The second program that will be used in conjunction with *rtl_fm* will be *Dire Wolf*. This program will be used to decode the APRS radio command messages. The input of this program will be the audio output of *rtl_fm* and the output will be a log file with the format 'yyyy-mm-dd.log'. The file name corresponds to the current date with the contents containing the fully decoded APRS message in plain text. Table 35 lists the different configuration values that allow for successful APRS decoding.

Program	Parameter	Value	Description
Dire Wolf	-l	.	The location of the log file output
Dire Wolf	-n	1	The number of audio channels
Dire Wolf	-r	24000	The rate at which to sample the incoming "audio"
Dire Wolf	-b	16	The number of bits to sample the audio

Table 35: Dire Wolf Configuration Parameters and Values

These two programs will be enabled via a system service on the Raspberry Pi once the sensor system detects that the payload has entered a landed state. The full command to enable the listening and decoding is shown in Figure 75.

rtl_fm -f 145.00M -o 4 - | direwolf -l . -n 1 -r 24000 -b 16 –

Figure 75: Linux Command to Initialize Radio Listening

Finally, the last part of the radio system will involve parsing the log file to detect if the message corresponds to the proper NASA callsign and contains the commands for the payload to execute. If it finds the proper message in the file, it will disable the system service and turn off the programs and send the list of commands to the camera and motor software systems for execution.

4.3.3.3 Motor System

The motor system will be responsible for controlling the solenoid and stepper motors within the payload. There will be two distinct subsystems, as illustrated in Figure 76, within the motor system. One will be responsible for retracting the solenoid of the proper camera housing once the payload has landed. The other will wait for incoming radio messages to rotate the camera either 60 degrees to the right or left. All motors will be controlled by either enabling or disabling specific GPIO pins on the Raspberry Pi.

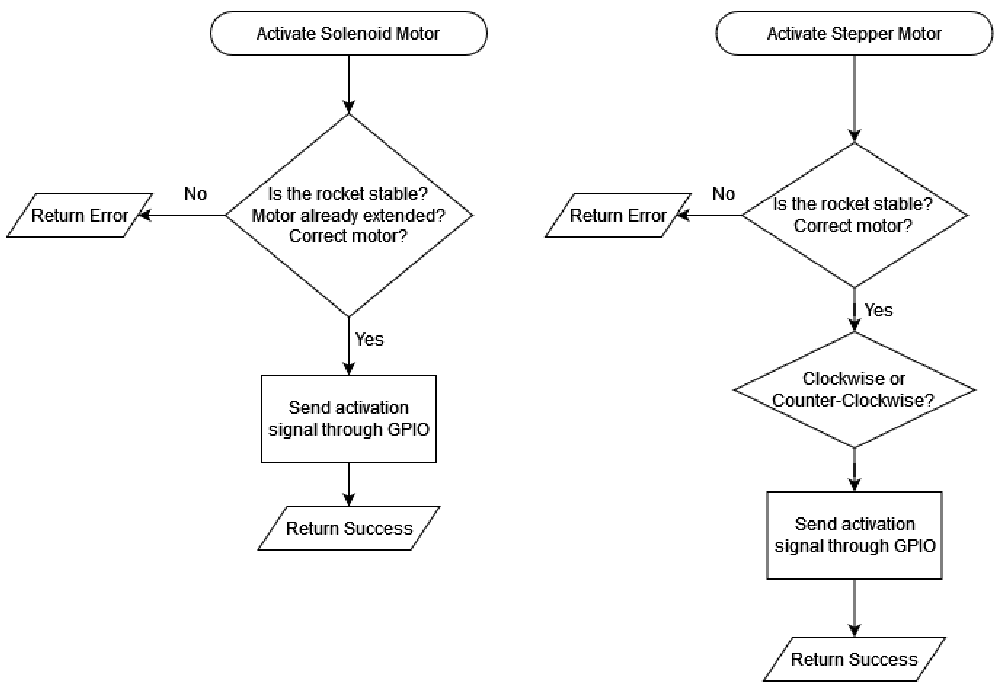

Figure 76: Motor System Flowchart

The first of the motor subsystems to be activated will be the solenoid. It will actuate once the payload has detected it has landed and the sensor system determines which of the three cameras should be extended. The activation is simple as it only requires setting the value of one GPIO pin on the Raspberry Pi. Once the correct solenoid has been actuated, there is nothing else for this system to do and it will remain idle for the rest of the mission.

The second motor system is responsible for controlling the stepper motors. As with the solenoid, this system will only control one of the motors depending on the orientation of the rocket as determined by the sensor system. The inputs for this motor controller will come from the radio system where a command of A1 will be responsible for turning the camera 60 degrees to the right and the B2 command will turn the camera 60 degrees to the left. To achieve this, there will be a preconfigured number of steps coded into the program. When the system receives a command from the radio system to active, it will perform those steps.

4.3.3.4 Camera System

The payload will be responsible for taking photos and applying filters and timestamps as required by the competition. To simplify this process, a software library will be used that provides the proper functionality to successfully achieve these tasks. OpenCV was selected as the primary library to be used for all image capture and post-processing as it provides the tools to meet and exceed all NASA and team requirements. Figure 77 illustrates the general structure of the camera system that OpenCV will provide support to ensure successful results.

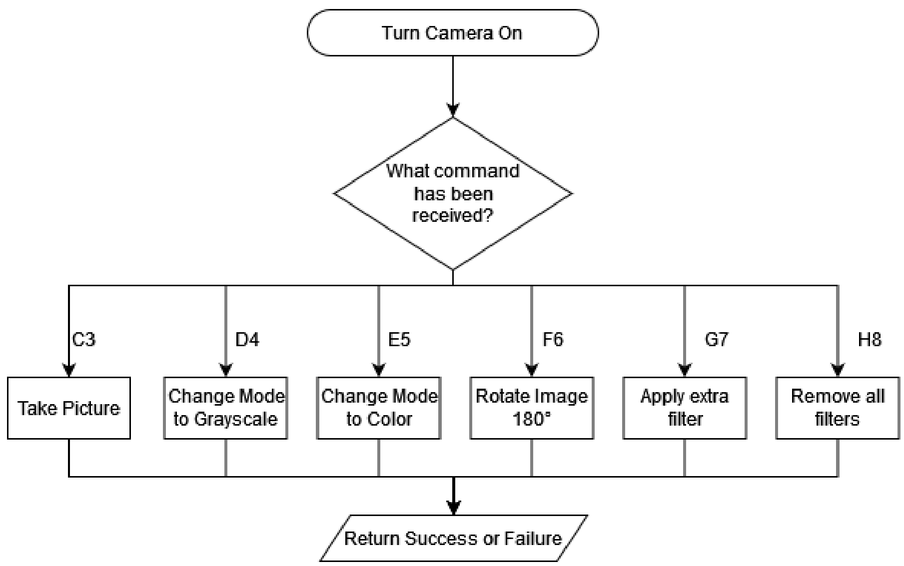

Figure 77: Camera System Flowchart

The first step needed is to be able to take a photo with the appropriate camera based on the orientation of the payload. OpenCV allows for this through the VideoCapture class which allows specifying an integer value to select the proper camera. Even though the class is called VideoCapture, it allows for taking a single frame capture which corresponds to an image. The camera system will pick between the three cameras indexed at either 0, 1, or 2 based on the orientation of the payload. When the camera system is enabled, it will request the orientation of the payload from the sensor system and use that value to pick the proper index.

As a result of this capture, the library returns a Mat object which will be used to apply all filters required by the competition and to add the timestamp. OpenCV contains an image processing module called Imgproc which will be used to convert and modify the image. The program will use a series functions provided by OpenCV such as cvtColor to make gray scaled images and a combination of getRotationMatrix2D and warpAffine to rotate the image. Finally, the G7 radio command asks for a custom filter to be applied to the image, which will be a blur effect that will be achieved via the OpenCV bilateralFilter function. All the raw images and their modifications will be saved separately to the Raspberry Pi’s SD card for inclusion in post mission reviews and reports.

4.3.4 Payload Integration

The three payload systems are fastened to the airframe by sliding each one inside and are only connected by the electronics wires. The wires are assembled first to the pi and batteries and the three systems are slid into the airframe. Each housing is fastened to the airframe with 8-32 fasteners and threaded heat set inserts. The payload coupler is secured to the aft with rivets and shear pins (Figure 78).

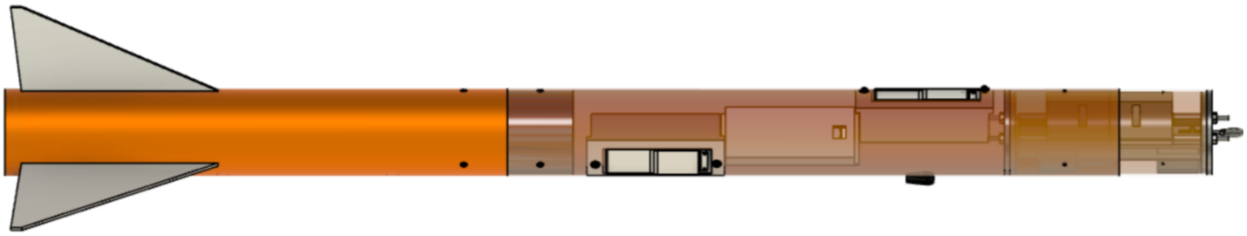

Figure 78: Payload Section in Airframe

4.3.5 Payload Mass Table

The payload assembly consists of a payload coupler assembly with most of electronics; a radio assembly with SDR radio dongle, antenna, cables, and adaptors; 2 camera housing assemblies (without radio) each with a camera, a solenoid, a stepper motor, a 3-D printed camera housing, a camera arm, and a camera cover; and 1 camera housing assembly (with radio) with all the parts mentioned with the addition of SDR radio dongle (Table 36).

Assembly	Components	Mass (oz.)
Payload Coupler Assembly	Raspberry Pi	1.73
	Motor Battery	3.53
	Pi Battery	4.94
	Stepper Motor Drivers	1.06
	IMU	0.11
	Barometer	0.09
	PCB	0.71
	Power Converter	0.55
	MOSFETs	0.22
	Diodes	0.11
	Wiring	0.85
	Electronics Sled	2.41
	Battery Housings	4.75
	Total	**21.04**
Radio Assembly	SDR Radio Dongle	1.09
	Antenna	0.32
	Cable	1.59
	Adaptor	0.15
	Total	**3.15**
	Camera	0.29

2 Camera Housing Assemblies (Without Radio)	Solenoid	1.34
	Stepper Motor	1.31
	Cables	1.90
	Push button	0.02
	Camera Mount	0.90
	Camera Housing	4.54
	Camera Cover	0.27
	Stepper Motor Hub	0.09
	1 Payload Housing Assembly	**10.67**
	2 Payload Housing Assembly	**21.33**
Payload Housing Assembly (With Radio)	Camera	0.29
	Solenoid	1.34
	Stepper Motor	1.31
	Cables	1.90
	Push button	0.02
	Camera Mount	0.90
	Camera Housing	5.46
	Camera Cover	0.27
	Stepper Motor Hub	0.09
	Total	**11.58**
	Full Payload Assembly	**57.10**

Table 36: Payload Component Masses

5. Safety

5.1 Safety Procedures

If any responsible lead is unable to attend a launch, their replacement must be approved by the Project Manager and the responsible lead. This change must also be communicated to the Safety Officer. If the individual tasked with performing the checklist verification is unable to attend a launch, their replacement must be approved by the Safety Officer and the Project Manager.

5.1.1 Avionics and Recovery Preparation

5.1.1.1 Relevant Personal Protective Equipment

None

5.1.1.2 Authority

Responsible Lead: Stephanie Baldwin (Avionics and Recovery Lead)

Checklist Verification: Luka Bjellos (Safety Officer)

5.1.1.3 Critical Testing Prior to Launch Date

Test LV-A-1: Altimeter Functionality Demonstration

Test LV-A-2: Altimeter Accuracy Test

Test LV-A-3: Avionics Battery Life Test

Test LV-A-5: Avionics Interference Demonstration

Test LV-A-6: GPS Accuracy Test

Test LV-A-7: GPS Range Test

Test LV-R-1, LV-R-2: Drogue, Main Parachute Drag Test

Test LV-R-4: Parachute Deployment Time Test

Test LV-R-5: Parachute Ejection Demonstration

Test LV-R-6: Parachute Unfolding Demonstration

5.1.1.4 Parachute Preparation Procedure

1. Attach swivel and quick link to parachute.
2. Put slip knot in recovery harness, about 1/3 of the length away from one of the ends.
3. Secure quick link on swivel to slip knot, wrench tight.
4. Attach other ends of recovery harness to their respective eyebolts, wrench tight.
5. Fold parachute from gore to gore, until all gores folded. All shroud lines should be aligned.
6. Do one Z-fold of the shroud lines inside parachute after folding from gore to gore is complete.
7. Do one Z-fold of parachute.
8. Fold the parcel in half.
9. Roll each side of the parcel as tight as possible.
10. Place in center of the parachute protector.
11. Z-fold recovery harness alongside parachute in protector.
12. Fold ends of parachute protector in and roll parachute and recovery harness inside protector.

Verify parachute properly folded; failure to do so may lead to hazard L.1.

	Responsible Lead	Checklist Verification
Drogue Parachute Folded	________________	____________________
Main Parachute Folded	________________	____________________

5.1.1.5 Avionics Bay Preparation Procedure

Check altimeter settings and verify deployment altitudes/delays

- ☐ Primary Drogue Deployment - At apogee
- ☐ Primary Main Deployment - At 600 ft
- ☐ Secondary Drogue Deployment - Delayed 1 s from apogee
- ☐ Secondary Main Deployment – At 550 feet

1. Plug in current mean altitude above sea level for the Entacore AIM to calibrate for ground level
2. Verify that batteries are plugged in in the correct orientation by powering on each altimeter with no charge plugged in. Expect 5-30 milliamps of current through the output terminals for each altimeter.
   - ☐ Primary Altimeter battery installed properly

- ☐ Secondary Altimeter battery installed properly

3. Ensure altimeters and batteries are secured to the avionics bay sled.
4. Pass threaded rods through the forward bulkhead and the avionics sled.

   Verify sled is oriented correctly, failure to do so will require disassembly of completed bay

   ☐ Verify correct orientation
5. Pass wires through the wire hole in the forward bulkhead and secure to one side of the terminal block for primary and secondary charges.
6. Pass wires for the aft bulkhead ejection charges through the avionics bay coupler section.
7. Secure the wires to their respective altimeter terminals and the terminal blocks on the outside of the aft bulkhead.
8. Nest the bulkheads into the avionics bay coupler, ensuring keylock switch is accessible through switch band.
9. Secure bulkheads with hex nuts, wrench tight.
10. Place clay around wire and threaded rod holes as needed to prevent ejection gases from entering and escaping through avionics bay.

	Responsible Lead	Checklist Verification
Avionics Bay Prepared	________________	____________________

5.1.2 Camera Preparation

5.1.2.1 Relevant Personal Protective Equipment

None

5.1.2.2 Authority

Responsible Lead: Gabriella Peburn (Payload Mechanical Lead)

Checklist Verification: Luka Bjellos (Safety Officer)

5.1.2.3 Critical Testing Prior to Launch Date

Test LV-L-1: Launch Rehearsal Demonstration

Test P-CF-2: Camera Functionality Demonstration

Test P-CF-5: Latch Release Demonstration

Test P-CF-6: Camera Rotation Demonstration

Test P-CF-9: Camera FOV Inspection

Test P-SF-4: Camera Angle Inspection

Test P-D-6: Camera Housing Impact Resistance Demonstration

5.1.2.4 Payload Camera Assembly Preparation Procedure

1. Install stepper motor in payload housing and feed wiring through housing wall.
2. Install solenoid motor in payload housing and feed wiring through housing wall.

3. Install push-button in payload housing and feed wiring through housing wall.
4. Install motor hub on stepper motors
5. Fasten spring hinge to motor hub.
6. Fasten camera mount to spring hinge.
7. Fasten camera to camera mount and feed wiring down camera mount, and into housing, then through housing wall.
8. Install camera case around camera.
9. Repeat steps 1-9 for all three payload assemblies.
10. Install SDR radio dongles in central payload housing.
11. Install payload housing radio cover.
12. Feed all electrical wiring from housings through payload airframe into payload bay.
13. Install payload housings into payload airframe by tightening fasteners with a Phillips head screwdriver.
    - ☐ Camera housing attached
14. Feed wiring from camera assemblies to the respective electronics in the forward payload coupler.
    - ☐ Payload wiring connected

Verify complete assembly with fully retained camera.

	Responsible Lead	Checklist Verification
Cameras Mounted	________________	____________________

5.1.3 Payload Preparation

5.1.3.1 Relevant Personal Protective Equipment

- Non-synthetic clothing to prevent static buildup

5.1.3.2 Authority

Responsible Person: Tianjun Wang (Payload Electronics Lead)

Gabriella Peburn (Payload Mechanics Lead)

Christopher Comes (Payload Software Lead)

Checklist Verification: Luka Bjellos (Safety Officer)

5.1.3.3 Critical Testing Prior to Launch Date

Tests P-CF-1 – P-CF-9: Payload Component Functionality Demonstrations

Test P-SI-4: Radio Integration Demonstration

Test P-D-4: Payload Acceleration Resilience Demonstration

Test P-D-5: Payload Impact Resistance Demonstration

Test P-SF-1: Payload Battery Life Test

Test P-SF-5: Power Loss Recovery Demonstration

Test P-SF-6: Landing Detection Demonstration

5.1.3.4 Payload Coupler Preparation and Final Assembly Procedure

1. Secure lithium-ion actuator battery and portable charger into battery housing, put threaded rods through the sled and battery housing to secure them.
2. Connect BNO055 IMU and BMP390 Barometer to the PCB and use hot glue to secure the connection
   - ☐ BNO055 IMU
   - ☐ BMP390 Barometer
3. Connect Raspberry Pi with PCB and use hot glue to secure the connection, and then secure them onto the electronic sled
4. Connect Stepper motors, Solenoids, push buttons, and piezo to corresponding terminals on the PCB
   - ☐ 3 Stepper Motors (ULN2003 Controller)
   - ☐ 3 Solenoids (JST-XH)
   - ☐ 3 Push Buttons (JST-XH)
   - ☐ 1 Piezo (JST-XH)
5. Connect USB cameras and SDR dongle to the corresponding USB ports on Raspberry Pi
   - ☐ 3 Cameras
   - ☐ 1 SDR Dongle
6. Connect actuator battery to PCB and portable charger to Raspberry Pi 4
   - ☐ Actuator battery to PCB (JST-XH)
   - ☐ Portable charger to Raspberry Pi (USB-C)
7. Insert the payload sled into the payload coupler and put nuts to secure bulkheads onto payload coupler
8. Insert the payload coupler into payload airframe and put the shear pin and rivets
9. Wait for software health check to run to check all connections are matched correctly and the SDR dongle is functioning properly – **return to step 2 and <u>disconnect power</u> if health check indicates errors**
   - ☐ Solenoids, Stepper motors, push buttons, and cameras are wired correctly
   - ☐ SDR dongle connection established
   - ☐ IMU and barometer connections established
10. Pack clay around each nut to seal the payload bay (Payload will be at launch detection state during this stage).

	Responsible Lead	Checklist Verification
Payload Bay Prepared and Sealed	____________________	______________________

5.1.4 Ejection Charge Preparation

5.1.4.1 Relevant Personal Protective Equipment

- Non-synthetic clothing to prevent static buildup
- Non-sparking spatula
- Non-sparking dish
- Non-sparking wooden dowel

5.1.4.2 Authority

Responsible Lead: Rylan Andrews (Testing Lead)

Checklist Verification: Luka Bjellos (Safety Officer)

5.1.4.3 Critical Testing Prior to Launch Date

Test LV-R-5: Parachute Ejection Demonstration

5.1.4.4 Ejection Charge Preparation Procedure

VERIFY CORRECT ENERGETIC AND AMOUNTS REQUIRED FOR ALL EJECTION CHARGES FROM CHARGE TESTS:

- ☐ Energetic: Black Powder
- ☐ Forward Avionics Bay Forward Bulkhead Primary Charge: 2.80 grams
- ☐ Forward Avionics Bay Forward Bulkhead Backup Charge: 3.50 grams
- ☐ Forward Avionics Bay Aft Bulkhead Primary Charge: 1.70 grams
- ☐ Forward Avionics Bay Aft Bulkhead Backup Charge: 2.10 grams

REMOVE ANY IGNITION SOURCES PRIOR TO PREPARING CHARGES. REMOVE ALL OTHER MATERIALS FROM WORK AREA. RELEASE ANY STATIC BUILDUP PRIOR TO HANDLING ENERGETIC. FAILURE TO DO SO MAY CAUSE HAZARD L.3.

Necessary Materials:

- ☐ Energetic (Above)
- ☐ 4 Premade E-Matches
- ☐ Fire-Resistant Insulation
- ☐ Electronic scale
- ☐ Non-sparking metal dish
- ☐ Non-sparking spatula
- ☐ Small wooden dowel
- ☐ Masking tape
- ☐ Writing Utensil

Ejection charges must be made one at a time to ensure correct amount of energetic is packed in each charge. During preparation, box for corresponding charge must be checked as work is performed.

1. Place small electronic scale on level surface.
2. Place metal dish onto scale and zero the scale.
   - ☐ Avionics Bay Forward Bulkhead Primary Charge: 2.80 grams
   - ☐ Avionics Bay Forward Bulkhead Backup Charge: 3.50 grams
   - ☐ Avionics Bay Aft Bulkhead Primary Charge: 1.70 grams
   - ☐ Avionics Bay Aft Bulkhead Backup Charge: 2.10 grams
3. Using a spatula, add energetic into metal dish until appropriate amount is measured.
4. Pour energetic into cardboard cylinder of premade e-match.
5. Insert fire-resistant insulation into cylinder and compress insulation using small wooden dowel.

- ☐ Insulation packed in Forward Bulkhead Primary Charge
- ☐ Insulation packed in Forward Bulkhead Backup Charge
- ☐ Insulation packed in Aft Bulkhead Primary Charge
- ☐ Insulation packed in Aft Bulkhead Backup Charge

6. Fold top of cylinder down and seal with masking tape.
7. Label charge with the corresponding energetic, amount, location, and primary/backup.
   - ☐ Avionics Bay Forward Bulkhead Primary Charge: 2.80 grams
   - ☐ Avionics Bay Forward Bulkhead Backup Charge: 3.50 grams
   - ☐ Avionics Bay Aft Bulkhead Primary Charge: 1.70 grams
   - ☐ Avionics Bay Aft Bulkhead Backup Charge: 2.10 grams
8. Initial below for each charge and repeat steps 1-7 for each required ejection charge.

	Responsible Lead	Checklist Verification
Forward Bulkhead Primary Charge	____________________	____________________
Forward Bulkhead Backup Charge	____________________	____________________
Aft Bulkhead Primary Charge	____________________	____________________
Aft Bulkhead Secondary Charge	____________________	____________________

5.1.5 Rocket Assembly Preparation

5.1.5.1 Relevant Personal Protective Equipment

- Non-synthetic clothing to prevent static buildup

5.1.5.2 Authority

Responsible Lead: Layali Bazar (Structures Lead)

Checklist Verification: Luka Bjellos (Safety Officer)

5.1.5.3 Critical Testing Prior to Launch Date

Test LV-L-1: Launch Rehearsal Demonstration

5.1.5.4 Rocket Assembly Procedure

Necessary Materials:

- ☐ Nosecone body
- ☐ Nosecone shoulder
- ☐ Forward airframe
- ☐ Assembled avionics bay
- ☐ Assembled payload bay
- ☐ Central airframe
- ☐ Aft assembly

- ☐ 6 shear pins
- ☐ 18 rivets
- ☐ Wrench
- ☐ Pliers
- ☐ Flathead screwdriver
- ☐ 4 quick links
- ☐ Insulation

1. Assemble nosecone section.
   - ☐ Secure the GPS inside the nosecone coupler using velcro
   - ☐ Use 3 rivets to connect nosecone body to the nosecone coupler
2. Assemble aft section.
   - ☐ Use 3 rivets to connect the forward payload coupler to the payload airframe
   - ☐ Use 3 rivets to connect the aft payload coupler to the payload airframe
3. Connect avionics bay to nosecone section
   - ☐ Secure the recovery harness to the forward avionics bulkhead using a quick link.
   - ☐ Feed the recovery harness and main parachute through the forward airframe
   - ☐ Secure the recovery harness to the nosecone using a quick link

   Ensure wires of ejection charge are accessible and held apart while the lead is not touching any surface.

   - ☐ Wire ejection charge to avionics bay
   - ☐ Use 3 rivets to connect the forward airframe to the avionics bay
   - ☐ Place insulation material in the forward airframe
   - ☐ Pack main parachute into the forward airframe
   - ☐ Use 3 shear pins to secure forward section to nosecone section
4. Connect avionics bay to aft section
   - ☐ Secure recovery harness to the aft avionics bulkhead using a quick link
   - ☐ Feed the recovery harness and drogue parachute through the central airframe

   Ensure wires of ejection charge are accessible and held apart while the lead is not touching any surface.

   - ☐ Wire ejection charge to avionics bay
   - ☐ Use 3 rivets to connect the avionics bay to the central airframe
   - ☐ Place ejection charge in central airframe
   - ☐ Place insulation material in central airframe
   - ☐ Pack drogue parachute into central airframe
   - ☐ Use 3 shear pins to connect central airframe to aft section

	Responsible Lead	Checklist Verification
Rocket Assembled	______________________	______________________

5.1.6 Motor Preparation

5.1.6.1 Relevant Personal Protective Equipment

- Non-synthetic clothing to prevent static buildup

5.1.6.2 Authority

Responsible Person: Jimmy Yawn (Team NAR/TRA Mentor)

Checklist Verification: Luka Bjellos (Safety Officer)

5.1.6.3 Critical Testing Prior to Launch Date

None

5.1.6.4 Motor Preparation Procedure

1. Grease O-rings and Threaded Sections.
2. Insert Propellant grains into Propellant Sleeve.
    a. Should be flush with end of side opposite to Aerotech logo.
3. Prepare the Delay grain.
    a. Press the inside of the delay grain cap for proper fit.
    b. Put the spacer into the delay grain cap.
    c. Slide the delay grain inside.
    d. Put the Delay Grain O-ring on the lip of the delay grain, avoiding getting grease on the grain.
4. Insert the delay grain into the forward closure.

    Verify delay grain inserted

    - ☐ Delay Grain Inserted
5. Insert Aft and Forward Seal Disks to the ends of the Propellant.
    a. Use the aluminum forward seal disk for bigger motors.
6. Insert Aft O-ring (Thick) onto the aft seal disk.
7. Insert Forward O-ring (Thin) onto forward seal disk.

    Verify seal disks and O-rings inserted

    - ☐ Aft Seal Disk and O-Ring Inserted
    - ☐ Forward Seal Disk and O-Ring Inserted
8. Screw on forward closure halfway.
9. Put nozzle on aft closure.
10. Screw on aft closure with nozzle in center circle halfway.
11. Screw in both closures until fully sealed.
12. Add a small diamond-shaped cut to the nozzle plug so igniter can fit through; put it on the nozzle.

IGNITER MUST NOT BE INSERTED INTO MOTOR UNTIL ON THE PAD

Responsible Lead　　　Checklist Verification

Motor Prepared (w/o Igniter) ____________________ ______________________

5.1.7 Setup on Launch Pad

5.1.7.1 Relevant Personal Protective Equipment

- Non-synthetic clothing to prevent static buildup

5.1.7.2 Authority

Responsible Individuals: Jimmy Yawn (Team NAR/TRA Mentor)

Stephanie Baldwin (Avionics & Recovery Lead)

Checklist Verification: Erik Dearmin (Project Manager)

5.1.7.3 Critical Testing Prior to Launch Date

None

5.1.7.4 Setup on Launch Pad Procedure

VERIFY ALL SETUP CHECKLISTS ARE COMPLETE BEFORE BEGINNING LAUNCH PAD SETUP:

- ☐ Ejection charge checklist
- ☐ Payload preparation checklist
- ☐ Avionics preparation checklist
- ☐ Motor preparation checklist
- ☐ Rocket preparation checklist

Follow directions from the Range Safety Officer at all times.

1. Verify correct launch rail
   - ☐ 15-15 rail
   - ☐ 12 ft
2. Load launch vehicle on launch rail
3. Arm altimeters using keylock switch
   - ☐ Altimeters armed
   - ☐ Continuity confirmed

	Responsible Leads		Checklist Verification
Vehicle on Pad	________________	________________	________________________

5.1.8 Igniter Installation

5.1.8.1 Relevant Personal Protective Equipment

- Non-synthetic clothing to prevent static buildup

5.1.8.2 Authority

Responsible Person: Jimmy Yawn (Team NAR/TRA Mentor)

Checklist Verification: Erik Dearmin (Project Manager)

5.1.8.3 Critical Testing Prior to Launch Date

None

5.1.8.4 Igniter Installation Procedure

VERIFY LAUNCH PAD SETUP CHECKLIST IS COMPLETE BEFORE IGNITER INSTALLATION

- ☐ Launch pad setup checklist complete

1. Insert igniter all the way into the motor until the igniter touches the end of the motor.

   Ensure that the wire does not lower throughout the following processes

2. Put the igniter through the small hole in the nozzle plug, and seat nozzle plug into nozzle.
3. Strike the alligator clips together to ensure they are not powered.
4. Using alligator clips, clip the wire on its tip (about 1/4 in of wire) and then wrap the remaining wire around the alligator clip

   Ensure alligator clips and igniter wires are not touching the metal launch rail stand

	Responsible Lead	Checklist Verification
Igniter Installed	____________________	______________________

5.1.9 Launch Procedure

5.1.9.1 Relevant Personal Protective Equipment

- Non-synthetic clothing to prevent static buildup

5.1.9.2 Authority

Responsible Lead: Erik Dearmin (Project Manager)

Checklist verification: Luka Bjellos (Safety Officer)

5.1.9.3 Critical Testing Prior to Launch Date

Test LV-L-1: Launch Rehearsal Demonstration

5.1.9.4 Launch Procedure

VERIFY IGNITER INSTALLATION CHECKLIST IS COMPLETE BEFORE BEGINNING LAUNCH CHECKLIST

- ☐ Igniter installation checklist completed

Follow directions from the Range Safety Officer at all times.

1. Assemble payload components
   - ☐ Payload checklist completed
2. Assemble avionics bay
   - ☐ Recovery checklist completed
3. Assemble launch vehicle
   - ☐ Rocket preparation checklist completed
4. Prepare motor
   - ☐ Motor preparation checklist completed
5. Insert assembled motor into motor tube of launch vehicle
   a. Motor inserted
   b. Thread motor retainer onto motor assembly

Line of motor must remain clear of all persons.

6. Find rocket center of gravity
   - ☐ Balance rocket horizontally until center of gravity is found
7. Check stability of launch vehicle
   - ☐ Stability is 2.0 or higher at rail exit
   - ☐ Thrust to weight ratio of 5:1 or higher
   - ☐ Velocity off rail of 52 fps or higher
8. Bring launch vehicle to launch pad
   - ☐ Launch pad checklist completed
   - ☐ Igniter installation checklist completed
9. Launch
   - ☐ Follow all Range Safety Officer instructions
   - ☐ Verify all personnel are at least 300 ft from launch pad
   - ☐ Verify pad area is cleared for 100 ft around launch pad
   - ☐ Verify windspeed is less than 20 mph
   - ☐ Verify airspace is clear
   - ☐ Verify continuity with igniter

	Responsible Lead	Checklist Verifications	
Launch Ready	________________	______________	________________

An immediate launch can occur once lead and verifications sign off, pending RSO final approval.

5.1.10 Troubleshooting

5.1.10.1 Relevant Personal Protective Equipment

- Non-synthetic clothing to prevent static buildup

5.1.10.2 Authority

Responsible Lead for Misalignment and Altimeters: Stephanie Baldwin (Avionics and Recovery Lead)

Responsible Person for Motor Issue: Jimmy Yawn (Team NAR/TRA Mentor)

Checklist Verification: Luka Bjellos (Safety Officer)

5.1.10.3 Critical Testing Prior to Launch Date

None

5.1.10.4 Troubleshooting Procedure

5.1.10.4.1 Avionics bay misalignment

1. Disassemble avionics bay
    - ☐ Avionics bay disassembled
2. Check for improper component placement
    - ☐ Avionics sled installed backwards or upside down
    - ☐ Wrong bulkheads used on avionics bay
    - ☐ Improper wiring of avionics components
3. Reassemble avionics bay
    - ☐ Avionics bay assembly procedure completed

	Responsible Lead	Checklist Verification
Avionics Bay Properly Aligned	______________	________________

5.1.10.4.2 Altimeters do not have continuity or do not turn on

1. Disassemble avionics bay
    - ☐ Avionics bay disassembled
2. Check all wiring for loose connections
    - ☐ Altimeter connections
    - ☐ Terminal connections
    - ☐ Battery connections
3. Reassemble avionics bay
    - ☐ Avionics bay assembly procedure completed

	Responsible Lead	Checklist Verification
Altimeter Troubleshooting Complete	______________	________________

5.1.10.4.3 Motor fails to ignite

1. Wait 60 seconds before approaching launch pad
2. Disarm altimeters
    - ☐ Altimeters disarmed
3. Remove old igniter
4. Install new igniter
    - ☐ Igniter installation procedure completed

Responsible Lead	Checklist Verification
______________	________________

Igniter Troubleshooting Complete ______________________ ___________________

5.1.11 Post-Flight Inspection

5.1.11.1 Relevant Personal Protective Equipment

☐ Gloves – determination will be made by Safety Officers if rocket's recovery location requires gloves

5.1.11.2 Authority

Responsible Leads: Erik Dearmin (Project Manager)

Checklist Verification: Luka Bjellos (Safety Officer)

5.1.11.3 Critical Testing Prior to Launch Date

None

5.1.11.4 Post-Flight Inspection Procedure

Identify and make team aware of any environmental hazards in recovery area prior to approaching rocket. Only leads may approach rocket for recovery and inspection.

☐ Area safe and only necessary personnel recovering rocket

1. Locate ejection charge wiring to verify all charges properly detonated
   Note: If undetonated, wait an additional minute before approaching vehicle
2. Listen for and take audio recording of altimeter beeps (for maximum altitude)
   ☐ Altitude recorded: _____
3. Turn off keylock switches
   ☐ Switches off
4. Inspect launch vehicle for external damage and proper parachute deployment (Take pictures)
   Observations: ____________________________________
5. Lift rocket from ground, 1 person holding each section, ensuring that no section is stuck, or shock cord tangled. **Caution: Do not hold aft airframe by motor retainer, this may result in burns**
6. Carefully move rocket back to prep area.
7. Remove D-Links.
8. Open avionics bay and payload bay.
9. Inspect bays and bulkheads for ejection debris.
   Observations: ____________________________________

	Responsible Lead	Checklist Verification
Post-flight inspection complete	_________________	___________________

5.2 Personnel Hazards Analysis

The personnel hazards analysis evaluates the level of risk to safety of team members and equipment and risk mitigation strategies. Each hazard will be assigned a risk score ranging from 1 to 100, with 1 representing the lowest possible severity and likelihood, and 100 representing the highest possible severity and likelihood. The risk score is calculated by multiplying the severity value 'S' of each risk (1-10) by its likelihood value 'L' (1-10) (Table 37, Table 38).

	Severity (S)	**Likelihood (L)**
1, 2	Little to no equipment damage/very minor or no injury	1-20% occurrence, very unlikely
3, 4	Minor equipment damage/minor injury	21-40% occurrence, unlikely
5, 6	Moderate equipment damage/mild injury	41-60% occurrence, uncertain likelihood
7, 8	Major equipment damage/mild to major injury	61-80% occurrence, likely
9, 10	Irreparable equipment damage/major injury or death of personnel	81-100% occurrence, very likely

Table 37: Risk Assessment Chart (RAC)

Likelihood

Severity

	1	2	3	4	5	6	7	8	9	10
1	1	2	3	4	5	6	7	8	9	10
2	2	4	6	8	10	12	14	16	18	20
3	3	6	9	12	15	18	21	24	27	30
4	4	8	12	16	20	24	28	32	36	40
5	5	10	15	20	25	30	35	40	45	50
6	6	12	18	24	30	36	42	48	54	60
7	7	14	21	28	35	42	49	56	63	70
8	8	16	24	32	40	48	56	64	72	80
9	9	18	27	36	45	54	63	72	81	90
10	10	20	30	40	50	60	70	80	90	100

Table 38: Risk Assessment Score Chart

5.2.1 Chemical Hazards

Chemical hazards are those posed by the team's chemical inventory. Potential occurrences for chemical hazards and their respective effects, risk score, and mitigation strategy were evaluated (Table 39).

Hazard	Cause	Effect	S	L	Score	Mitigation Strategy

All-purpose cement contact	Exposure to fumes during manufacturing	Dizziness	**5**	**1**	**5**	Use of PPE (KN-85 facemasks or better) per MSDS in well-ventilated, open areas. Keep a safe distance if not involved
All-purpose cement ignites	Liquid form exposed to heat source during ejection charge or bulkhead testing	Burns, inhalation of smoke	**8**	**2**	**16**	Keep in well-ventilated, cool. Keep away from heat sources (ignited black powder/charges during testing) and oxidizers (Cl_2, F_2, H_2O_2)
Cleaning agent contact with skin	Use of cleaning agents to clean workspace	Minor skin irritation	**1**	**4**	**4**	Use of long sleeves, long pants, and close-toed shoes. Keep a safe distance if not involved
Cleaning agent contact with eyes	Use of cleaning agents to clean workspace	Severe eye irritation, potential blindness without medical care	**9**	**2**	**18**	Use of safety glasses. Keep a safe distance if not involved
Epoxy contacts skin	Exposure to uncured epoxy during manufacturing	Skin irritation	**5**	**4**	**20**	Use of gloves, long sleeves, long pants, and close-toed shoes. Keep a safe distance if not involved
Epoxy contacts eyes	Residual epoxy left on gloves contacting eyes	Severe eye irritation, potential blindness without eye washing	**9**	**1**	**9**	Use of safety glasses. Keep a safe distance if not involved
Gelcoat compound ignites	Liquid form exposed to heat source during ejection charge or bulkhead testing	Burns ranging from 1st to 3rd degree (dependent on exposure), smoke inhalation	**8**	**2**	**16**	Keep in well-ventilated, cool storage. Keep away from heat sources (ignited black powder/charges during testing) and oxidizers (Cl_2, F_2, H_2O_2)
Gelcoat fume inhalation	Prolonged toxic fume exposure	Dizziness	**4**	**1**	**4**	Use of PPE (KN-95 facemasks or better) per MSDS in well-ventilated areas
Paint thinner contacts skin	Spills, accidental contact with bare skin during painting	Minor skin irritation	**7**	**1**	**7**	Use of long sleeves, long pants, and close-toed shoes. Keep a safe distance if not involved
Paint thinner contacts eyes	Splatter, accidental spray towards eyes during painting	Severe eye irritation, potential blindness without eye washing	**9**	**1**	**9**	Use of safety glasses. Keep a safe distance if not involved
Paint thinner ignites	Liquid form exposed to heat	Burns ranging from 1st to 3rd	**9**	**2**	**18**	Keep in well-ventilated, cool storage. Keep away from heat

	source during ejection charge or bulkhead testing	degree (dependent on exposure), smoke inhalation				sources (ignited black powder/charges during testing) and oxidizers (Cl_2, F_2, H_2O_2)
Spray paint can explodes	Compressed gas in paint can exposed to heat source (strong sunlight) while painting	Minor hearing damage, potential skin and eye lacerations, potential shrapnel	**9**	**1**	**9**	Use of PPE (safety glasses, gloves, long sleeves, long pants, close-toed shoes) per MSDS, keep spray paint can out of strong sunlight if painting outside
Spray paint contact with skin	Spills, accidental contact with bare skin during painting	Minor skin irritation	**2**	**5**	**10**	Use of PPE (gloves, long sleeves, long pants, close-toed shoes) per MSDS, spray down and away from persons in area
Spray paint contact with eyes	Splatter, accidental spray towards eyes during painting	Severe eye irritation, potential blindness without eye washing	**3**	**3**	**9**	Use of PPE (safety glasses) per MSDS, spray down and away from persons in area
Spray paint ignites	Paint in air exposed to heat source (ejection charge or bulkhead testing done indoors)	Burns ranging from 1st to 3rd degree (dependent on exposure), smoke inhalation	**9**	**1**	**9**	Use of PPE (safety glasses, gloves, long sleeves, long pants, close-toed shoes) per MSDS, avoid using uncured paint during ejection charge and bulkhead testing
Spray paint inhalation	Paint aerosols are inhaled	Dizziness, respiratory inhalation	**5**	**4**	**20**	Use of PPE per MSDS, use spray paint in ventilated areas

Table 39: Chemical Hazard Identification

5.2.2 Physical Hazards

Physical hazards are those posed by manufacturing processes, launch preparation, and launching (Table 40, Table 41, Table 42).

5.2.2.1 Manufacturing Hazards

Hazard	Cause	Effect	S	L	Score	Mitigation Strategy
Bandsaw blade contact	Hand close to blade when cutting fiberglass for airframe	Skin laceration	**8**	**2**	**16**	Keep hands out of blade path when cutting materials. Use proper PPE (short sleeves, long pants, close-toed shoes) per MSDS. Avoid use of gloves. Plan operation with operators' manual
Live drill contact	Hand under or too close to drill bit while drilling holes for airframe (e.g.	Skin laceration	**8**	**1**	**8**	Keep hands away from cutting tools when machine is powered. Use proper PPE (short sleeves, long pants, close-toed shoes) per MSDS. Avoid use of gloves.

	avionics bay keys)					Plan operation with operators' manual
Sharp tool, drill, or endmill contact	Grabbing tool with bare hands during milling or drilling	Skin laceration	**6**	**2**	**12**	Use a rag as a buffer when handling sharp tools
Vise Pinches	Hand or fingers placed inside the vice while vise is being tightened	Pinching or minor skin laceration	**5**	**2**	**10**	Keep hands away from machine when in use, keep observers away from machine when tightening vises. Never place hand between vises when tightening, always use a jig or parallels to hold workpiece
Hammer Injury	Hand or finger under hammer (eg. tightening vise, securing workpiece between vises in milling machine)	Pinching or contusion, possible skin laceration	**4**	**2**	**8**	Ensure that body parts are away from workpiece before swinging hammer, avoid swinging with excessive force
Waterjet stream contact	Finger, hand, or arm in waterjet stream during fin or centering ring manufacturing	Skin laceration, potential bone or nerve damage	**9**	**1**	**9**	Check waterjet for leaks, use of PPE (gloves, long sleeves) per MSDS, stand far from machine, never reach across the path of the waterjet stream. Plan operation with operators' manual
Waterjet boom injury	Standing too close to boom during fin or centering ring manufacturing	Minor blunt trauma to the head, potential concussion	**6**	**1**	**6**	Stand far from machine when in use. Plan operation with operators' manual
Machining noise exposure	Proximity to loud machines in workspace	Hearing damage	**4**	**1**	**4**	Use of either noise cancelling headphones, earmuffs, or earplugs
Metal chips contact	Touching metal chips with bare hands	Skin laceration	**5**	**1**	**5**	Wear long pants and close toed shoes, use compressed air or rag to remove excess chips. Use gloves only when working with sheet metal
Fiberglass debris contact	Hands touch fiberglass dust during airframe manufacturing	Skin irritation or skin laceration	**4**	**1**	**4**	Wear gloves, long sleeves, long pants, and close-toed shoes when handling fiberglass

Fiberglass inhalation	Exposure to fiberglass dust in air	Respiratory irritation, eventual cancer	**10**	**1**	**10**	Wear particulate respirators (KN-95 not sufficient) when machining and cutting fiberglass
Soldering injury	Exposure to hot solder during avionics or payload bay manufacturing	Skin burns, primarily 1st degree	**4**	**1**	**4**	Wear safety glasses, gloves, long sleeves, long pants, and close-toed shoes. Keep away if observing
Battery acid contact	Broken or punctured battery casing during avionics or payload bay manufacturing	Skin irritation	**2**	**1**	**2**	Handle and store batteries carefully, keep away from sharp objects

Table 40: Manufacturing Hazard Identification

5.2.2.2 Launch Preparation Hazards

Hazard	Cause	Effect	S	L	Score	Mitigation Strategy
Premature motor ignition	Ignition during motor loading from high heat, electrical spark, friction, or impact during assembly	Premature launch of vehicle, severe blunt trauma, or impaling	**9**	**1**	**9**	Keep the motor out of strong sunlight for extended times, keep away from live boards in avionics and payload bays, and handle powder gently when transporting. During loading, have experienced team member place propellant in the motor, and give them space to ensure proper loading
Black powder ignition	Exposure to high heat, electrical spark, friction, or impact during assembly	Blunt trauma or lacerations from flying debris	**8**	**1**	**8**	Keep powder out of strong sunlight for extended times, keep away from live boards in avionics and payload bays, and handle powder gently when transporting
Premature altimeter activation	Sudden pressure differential (dropping the rocket, exposing altimeter barometers to wind) or faulty assembly	Premature ignition of ejection charges, causing blunt trauma or flying debris	**7**	**1**	**7**	Ensure proper avionics bay assembly (which includes checking for a tight fit), avoid activating altimeters with sudden pressure changes, confirm proper key switch activation. Follow checklist for assembling rocket correctly
Live wiring contact	Improper payload and avionics bay assembly	Skin burns, primarily 1st degree	**5**	**2**	**10**	Wear gloves, long sleeves, long pants, and close-toed shoes when soldering. Avoid bare skin contact with any live systems

Table 41: Launch Preparation Hazard Identification

5.2.2.3 Launching Hazards

Hazard	Cause	Effect	S	L	Score	Mitigation Strategy
Ballistic launch vehicle	Improper assembly/risky design choices	Severe impact injury	9	1	9	Maintain proper stand-off distance. Consult NAR/TRA safe distance table and follow RSO instructions
	Launch vehicle fails to separate	Severe impact injury	9	1	9	Use redundant altimeters and ejection charges, ground test ejection charges before launching, maintain proper stand-off distance
Falling debris	Premature or faulty ejection charges, lack of parachute deployment or function	Impact injury or skin laceration	6	1	6	Use redundant altimeters and ejection charges, ground test ejection charges before launching, maintain proper stand-off distance. Consult NAR/TRA safe distance table and follow RSO instructions
	Recovery harness failure	Impact injury or skin laceration	6	1	6	Select a recovery harness that is sufficiently strong to withstand separation forces, inspect recovery harness for tears or fraying before use
	Eyebolt failure and subsequent detachment during ejection	Impact injury or skin laceration	6	1	6	Ensure eyebolts are properly fastened to PVC bulkheads with properly torqued steel nuts, and that sufficient epoxy is properly set on a sanded, clean surface
Launch site fire	Premature motor ignition	Burns ranging from 1st to 3rd degree (dependent on exposure), smoke inhalation	9	1	9	Keep vehicle out of strong sunlight for extended times, keep away from live boards in avionics and payload bays, and handle propellant gently when transporting
	Premature ejection charges	Burns ranging from 1st to 3rd degree (dependent on exposure), smoke inhalation	9	1	9	Keep vehicle out of strong sunlight for extended times, keep away from live boards in avionics and payload bays, and handle powder gently when transporting

Table 42: Launch Operations Hazard Identification

5.2.3 Biological Hazards

Biological hazards are threats posed by bacteria or other living organisms (Table 43).

Hazard	Cause	Effect	S	L	Score	Mitigation

Spread of COVID-19 and other illnesses	Not sanitizing hands frequently, refusal to stay home/follow CDC guidelines	Member gets COVID-19 or other illness	**5**	**3**	**15**	Provide accessible sources of sanitation, tell sick members to stay at home, and keep members updated on current CDC guidelines
Sun damage	Prolonged sun exposure	Skin irritation, sun burns	**2**	**4**	**8**	Provide sunscreen; wear long sleeves, long pants, hats, and sunglasses

Table 43: Biological Hazard Identification

5.3 Failure Mode and Effects Analysis

Failure mode and effects analysis (FMEA) is the process of reviewing as many components, assemblies, and subsystems as possible to identify potential failure modes in a system and their causes and effects. By identifying failure modes, corrective actions can also be applied to mitigate risk. FMEAs have been created for the following categories: Structures (Table 46), Payload (Table 47, Table 48, Table 49), Avionics and Recovery (Table 50) and Flight Dynamics (Table 51). Three ratings are given to each failure mode to quantify the significance of the failure: severity, likelihood, and detection. Severity (S) is rated from 1 to 10 where a rating of 1 means that the failure has no effect while a rating of 10 is a catastrophic failure. Likelihood (L) is rated from 1 to 10 where a rating of 1 means that the failure has little to no chance of occurring while a 10 indicated it is incredibly likely to occur. Detection (D) is rated from 1 to 10 where a 1 is a failure that has a high likelihood of detection while a 10 is a failure that has an extremely low likelihood of detection. A risk priority number (RPN, 1-1000) is calculated as the product of the ratings and will be used to inform the team where mitigation strategies are needed, and which risks should be mitigated first. Severity, Likelihood, Detection, and RPN will be visualized using the following charts (Table 44, Table 45).

	Severity (S)	**Likelihood (L)**	**Detection (D)**
1, 2	Little to no effect on flight/little to no equipment damage	1-20% occurrence, very unlikely	81-100% detection chance, very likely detection
3, 4	Slight effect on flight/minor equipment damage	21-40% occurrence, unlikely	61-80% detection chance, likely detection
5, 6	Moderate effect on flight/moderate equipment damage	41-60% occurrence, uncertain likelihood	41-60% detection chance, uncertain detection
7, 8	Major effect on flight/major equipment damage	61-80% occurrence, likely	21-40% detection chance, unlikely detection
9, 10	Complete vehicle loss/irreparable equipment damage	81-100% occurrence, very likely	1-20% detection chance, very unlikely detection

Table 44: Risk Priority Number Chart

RPN Score

1-100	101-200	201-300	301-400	401-500	501-600	601-700	701-800	801-900	901+

Table 45: RPN Score Chart

5.3.1 Structures

Component	Function	Failure Mode	Failure Cause	Failure Effects			S[1]	L[2]	D[3]	RPN[4]	Corrective Actions
				Local Effects	Next Higher Level	System Effects					
Bulkhead	Protect the bays from heat of ejection charges and keeps them separate from the rest of the launch vehicle.	Breaks	Manufacturing defects such as not complying to drawings which determine the size of the bulkheads.	Fails to sufficiently seal components such as electronics, parachutes, or camera housings.	Ejection charges fail to separate vehicle	Parachutes are not deployed and/or internal components are damaged	**9**	**2**	**2**	**36**	Inspect launch vehicle before and after each launch. Perform ejection testing.
Bulkhead	Protect the bays from heat of ejection charges and keeps them separate from the rest of the launch vehicle.	Epoxy fails	Improper application	Fails to sufficiently seal components such as electronics, parachutes, or camera housings.	Ejection charges fail to separate vehicle	Parachutes are not deployed and/or internal components are damaged	**9**	**2**	**2**	**36**	Follow proper procedures for applying epoxy.
Fins	Provide launch vehicle stability.	Epoxy fails	Improper application	Launch vehicle loses stability and flight orientation	Uncontrollable flight	Launch vehicle drifts, becomes uncontrollable, and creates a hazard	**8**	**2**	**5**	**80**	Follow proper procedures for applying epoxy.
Centering Rings	Supports and aligns motor tube.	Breaks	Manufacturing defect	Launch vehicle loses stability and flight orientation	Uncontrollable flight	Launch vehicle drifts, becomes uncontrollable, and creates a hazard	**10**	**2**	**5**	**100**	Inspect immediately after manufacturing and inspect launch vehicle before and after each launch.
Centering Rings	Supports and aligns motor tube.	Epoxy fails	Improper application	Launch vehicle loses stability and flight orientation	Uncontrollable flight	Launch vehicle drifts, becomes uncontrollable, and creates a hazard	**10**	**2**	**6**	**120**	Follow proper procedures for applying epoxy.
Airframe	Hold electronic components, hardware, recovery equipment, and payload.	Breaks	Manufacturing defect and/or poor transportation	Fails to contain internal components	Launch vehicle assembly fails	Launch vehicle is unrecoverable and a hazard to those nearby due to falling debris	**10**	**3**	**2**	**60**	Members must follow the procedure of transporting components and inspect the launch vehicle before and after launch.
Coupler	Contains vehicle avionics bay and payload electronics.	Breaks	Manufacturing defect and/or improper transportation	Fails to contain internal components	Launch vehicle assembly fails	Launch vehicle is unrecoverable	**10**	**3**	**2**	**60**	Inspected component for defects after manufacturing and before and after launch.
Shear Pins	Allows launch vehicle to remain connected until	Shears too early	Excessive pressure and force from	Airframe and couplers separate	Early separation and premature	Reduced altitude	**7**	**3**	**9**	**189**	Test ejection charges and ensure pins are

	separation events.		previous events during launch.		parachute deployment						housed correction so does not shear prematurely.
Shear Pins	Allows launch vehicle to remain connected until separation events.	Do not break	Insufficient strength of ejection charge during separation	Parachutes do not deploy	Rapid descent of launch vehicle with a large impact force	Launch vehicle is unrecoverable	**10**	**2**	**8**	**160**	Test ejection charges to find enough force necessary to shear the pins.
Motor	Produces thrust for flight of launch vehicle	Breaks	Improper Installment	Payload with sustain severe damage and mission will fail	Launch vehicle assembly fails	Launch vehicle is unrecoverable and pose a hazard to those nearby	**10**	**3**	**3**	**90**	Motors will be inspected before installment and installed by someone trained.

Table 46: Structures FMEA

5.3.2 Payloads

5.3.2.1 Payload Mechanical

Component	**Function**	**Failure Mode**	**Failure Cause**	**Failure Effects**			**S¹**	**L²**	**D³**	**RPN⁴**	**Corrective Actions**
				Local Effects	**Next Higher Level**	**System Effects**					
Camera mount and locking lug	Acts as airframe and holds camera and hinge system	Cracking and breaking	Forces upon landing impact	Camera housing is not stable	Camera housing breaks off	The payload is unable to view the surrounding areas.	**8**	**2**	**1**	**16**	Impact testing, high infill in 3D print
Spring-loaded Hinge	Allows camera to deploy out of the airframe.	Hinge breaks off from the mount	Hinge applies excessive torque for the rigid spring mount body	Hinge breaks	Camera housing breaks off	The payload is unable to view the surrounding areas.	**7**	**2**	**2**	**28**	Torque calculations, prototype testing
Solenoid Motor	Latches on to the camera mount hook and holds spring-loaded hinge system down to be flush with airframe	Torque failure	Not enough or too much torque for the stresses from the load from the hinge.	Motor fails	Payload does not rise out of the airframe	Payload Challenge fails	**7**	**3**	**1**	**21**	Motor torque calculations, prototype testing
Stepper Motor	Rotates camera system	Torque failure	Not enough or too much torque for the stresses from the camera mount load.	Motor fails	Cameras do not rotate	Payload Challenge fails	**7**	**3**	**1**	**21**	Motor torque calculations, prototype testing
Payload Housing	Retains Camera System	Fastener Failure	Forces upon landing impact	Housing comes unmounted	Unmounted housing damages other payload components	Payload Challenge Fails	**8**	**2**	**1**	**16**	Impact testing, high infill in 3D print

Table 47: Payload Mechanics FMEA

5.3.2.2 Payload Electronics

Component	**Function**	**Failure Mode**	**Failure Cause**	**Failure Effects**			**S¹**	**L²**	**D³**	**RPN⁴**	**Corrective Actions**
				Local Effects	**Next Higher Level**	**System Effects**					
Actuator Battery	Provide power to electrical	Short circuit	Unintended contact between	Battery experience thermal	Payload failure due to the loss of power	Payload Failure	**9**	**2**	**4**	**72**	Use PCB to reduce the number of jumper wires

	components of payload		positive and ground terminal	runaway and fails							used, make sure no metal is exposed and all wires are securely connected
Stepper Motor	Rotate the camera about the z-axis	Motor failure	Incorrect wiring	Motor is unresponsive to the Raspberry Pi	Camera cannot rotate around z-axis	Payload failure	**5**	**3**	**2**	**30**	Use connectors on the field to prevent incorrect wiring, conduct built-in default software health check that will rotate motors every time Raspberry Pi is powered on
Solenoid	Unlock the latch to allow camera spring up	Motor failure	Unexpected motor damage	Motor stops working	Camera cannot deploy	Payload failure and possible damage due to stepper motor turning inside tub	**6**	**3**	**2**	**36**	Built-in default software health check that will retract solenoid when Raspberry Pi is powered on
RTL-SDR Radio	Receive APRS command from NASA	Interference	Magnetic field created by motors and Raspberry Pi	Noisy signals	Raspberry Pi doesn't receive command	Payload failure	**2**	**4**	**4**	**32**	Place radio antenna away from Raspberry Pi and motors to avoid interference, and make sure software does not read the signals received when motors are on
IMU	Detect launch and landing as well as the orientation of the rocket	Excessive noise in the signal	Static noise in the electronics and irregular movement created during launch preparation	IMU gives incorrect reading	Raspberry Pi incorrectly detects landing and initiate payload	Payload failure and possible launch vehicle failure	**8**	**1**	**1**	**8**	Use software filter to get rid of noises, and use a barometer as a redundancy
Raspberry Pi	Process radio signals and controls cameras and motors	Unstable voltage supply	Transient voltage created by motors	Raspberry Pi reboots and enters launch detection mode	Raspberry Pi won't reach radio command listening loop, and payload will not receive command	Payload failure	**3**	**3**	**4**	**36**	Use separate power supply for motors to ensure a stable current and voltage for Raspberry Pi

Table 48: Payload Electronics FMEA

5.3.2.3 Payload Software

Component	Function	Failure Mode	Failure Cause	Failure Effects			S[1]	L[2]	D[3]	RPN[4]	Corrective Actions
				Local Effects	Next Higher Level	System Effects					
Raspberry Pi	Controls payload operations	Software error	Unaccounted edge case in software, causing system to crash	The flight computer reboots and attempts to restore to previous state	No data is saved and the device reboots to the initial condition	The payload is unable to complete the assigned objective	**3**	**3**	**4**	**36**	Incorporate failure modes into software design

RTL-SDR Radio	Receives incoming radio signals	Device fails to send accurate data to Pi	Device configured improperly or unable to send data to controller	No signals will be sent to the Raspberry Pi	The Raspberry Pi fails to change state with incoming commands	The payload is unable to receive commands and therefore fails to perform assigned operations	2	4	4	32	Reboot and reconnect the USB device and reconfigure the receive frequency
ArduCam Camera	Take photos of local environment	Camera fails to listen to controller and is unable to take photos	Software configuration error not allowing proper inaction between camera and Raspberry Pi	Unable to take photo of environment	NASA radio transmissions will go without a response	The payload requirement to take photos would not be met	2	4	4	32	Reboot and reconnect the USB device and reload photo software

Table 49: Payload software FMEA

5.3.3 Avionics and Recovery

Component	Function	Failure Mode	Failure Cause	Failure Effects			S[1]	L[2]	D[3]	RPN[4]	Corrective Actions
				Local Effects	Next Higher Level	System Effects					
Altimeter	Determine the altitude of the launch vehicle at all points throughout flight to accurately set off ejection charges and deploy parachutes at the proper altitudes.	Sudden power loss	In-flight motions disconnect altimeter and battery	The altimeter stops monitoring and recording the launch vehicle's altitude	Ejection charges are not set off parachutes do not deploy	The launch vehicle descends with no deployed parachutes, high chance of significant damage to the launch vehicle upon landing	10	1	6	60	Secure altimeters, their respective batteries, and wiring in the avionics bay.
Altimeter	Determine the height of the launch vehicle at all points throughout flight to accurately set off ejection charges and deploy parachutes at the proper altitudes.	Parachute deployment order reversed (main deployed at apogee)	Improper wiring of the altimeter (main ejection charge wiring attached to drogue terminal)	The main parachute is deployed at apogee instead of the drogue	The launch vehicle descends at a slow speed for the entire descent	The launch vehicle drifts much further than intended, resulting in a longer recovery and possible loss of the launch vehicle	6	2	2	24	Clear labeling of all wiring terminals and ejection charges, inspection of wiring prior to launch
Altimeter	Determine the height of the launch vehicle at all points throughout flight to accurately set off ejection	Detonation of ejection charges on the launch pad	Improper wiring resulting in reversal of polarity of the battery	Instant detonation of ejection charges when altimeters are armed	Sections of the launch vehicle separate on the ground, potentially falling on and injuring nearby students	Unable to launch, possible damage to launch vehicle and injury to students near the pad	10	2	2	40	Test wiring of the altimeters before attaching ejection charges

	charges and deploy parachutes at the proper altitudes.										
Ejection Charges	Provides force necessary to cause separation of launch vehicle sections	Ejection charges do not detonate or fail to provide sufficient force to cause separation	Altimeters fail to set off ejection charge, ejection charge too small to cause separation, igniter failure, wires between the altimeters and ejection charges become disconnected	Launch vehicle sections do not separate	Parachutes are not deployed at the correct altitude or at all	Possible ballistic descent of the launch vehicle, high chance of damage to launch vehicle upon landing	**10**	**1**	**3**	**30**	Conduct ejection testing (test ID LV-R-5 and LV-R-6) to determine sufficient charge size, perform corrective actions for altimeters prior to launch
Recovery Harness	Tether sections of the launch vehicle to each other and to the parachutes after separation	Tears or breaks in the recovery harness	Use of a recovery harness that is damaged or has insufficient strength to remain functional after parachute deployments, melting of recovery harness due to ejection gases	Two or more sections of the launch vehicle become untethered	Sections not tethered to a parachute descend rapidly	Possible damage to sections not tethered to parachutes	**7**	**1**	**3**	**21**	Select recovery harnesses with sufficient strength and heat resistance, inspect recovery harnesses prior to use
Main Parachute	Slows the launch vehicle to its final descent velocity	Tears or holes in parachute	Insufficient protection from sharp objects, failure or improper use of parachute protector	Parachute is less efficient at slowing the launch vehicle	The launch vehicle descends at a faster velocity than designed	Possible damage to launch vehicle upon landing	**5**	**1**	**3**	**15**	Inspect parachute prior to use and patch holes as necessary
Main Parachute	Slows the launch vehicle to its final descent velocity	Tangled lines	Improper folding of parachute	Parachute is unable to fully inflate	The launch vehicle descends at a faster velocity than designed	Possible damage to launch vehicle upon landing	**5**	**3**	**2**	**30**	Store carefully to prevent tangling lines during storage, inspect packed parachute before using a parachute protector or putting it in the launch vehicle, conduct parachute deployment testing (LV-R-4) to evaluate packing effectiveness
Parachute Protector	Shields parachute from ejection gases during deployment	Holes or tears in parachute protector	Insufficient protection from sharp objects, damage and wear from previous use	Parachutes are not protected from ejection gases	Holes and burns may be made in parachutes due to ejection gases	Launch vehicle descends faster than anticipated	**6**	**2**	**1**	**12**	Inspect parachute protectors prior to use
Drogue Parachute	Slows initial descent of the launch vehicle and is deployed at apogee	Tears or holes in parachute	Insufficient protection from sharp objects, failure of parachute protector	Drogue parachute is less efficient at slowing the launch vehicle	The descent speed of the launch vehicle isn't slowed prior to main parachute deployment	Possible damage to launch vehicle during main parachute deployment	**5**	**1**	**3**	**15**	Inspect drogue parachute prior to use, ensure that parachute protector is used correctly
Drogue Parachute	Slows initial descent of the launch	Tangled lines	Improper folding of parachute	Parachute is unable to fully inflate	The launch vehicle descends at	Possible damage to launch	**5**	**3**	**2**	**30**	Store carefully to prevent tangling lines during

	vehicle and is deployed at apogee				a faster velocity than designed	vehicle upon landing					storage, inspect packed parachute before using a parachute protector or putting it in the launch vehicle, conduct parachute deployment testing (LV-R-4) to evaluate packing effectiveness
Recovery Hardware (Eyebolt)	Connect recovery harness to payload and avionics bay bulkheads	Breaks during parachute deployment	Improper epoxying of hardware into bulkheads, insufficient strength of recovery hardware	Two or more sections of the launch vehicle become untethered	Sections not tethered to a parachute descend rapidly	Possible damage to sections not tethered to parachutes	**7**	**1**	**1**	**7**	Select recovery hardware strong enough to withstand deployment forces, ensure that eyebolt epoxy is applied correctly
Recovery Hardware (Quick Links)	Connect recovery harness to payload and avionics bay bulkheads, connect parachute swivel to recovery harness	Breaks or opens during parachute deployment	Insufficient torque used to close quick links, insufficient strength of quick links	Two or more sections of the launch vehicle become untethered, the launch vehicle become untethered from the parachute	Sections not tethered to a parachute descend rapidly	Possible damage to sections not tethered to parachutes	**9**	**3**	**1**	**27**	Select recovery hardware strong enough to withstand deployment forces, ensure that quick links are torqued correctly

Table 50: Avionics and recovery FMEA

5.3.4 Flight Dynamics

Component	**Function**	**Failure Mode**	**Failure Cause**	**Failure Effects**			**S**[1]	**L**[2]	**D**[3]	**RPN**[4]	**Corrective Actions**
				Local Effects	**Next Higher Level**	**System Effects**					
Propellant	Generates thrust to propel the rocket.	Propellant Failure	Grain Defects, Improper storage, Water Damage.	Improper propellant burn.	Abrupt changes in thrust.	The launch vehicle has unpredictable trajectory/flight, or the rocket doesn't take off. Additional risks of over pressuring.	**9**	**3**	**4**	**108**	Ensure the integrity of the propellant grains by visually checking for defects. Store motors in a Climate Regulated room, and handle with care.
Motor Hardware	Controls the mass flow rate of the propellant burn.	Hardware Deformation	Structural failure of the hardware.	The thrust exit area, exhaust pressure, and the mass flow rate change.	Abrupt changes in the thrust vector, and impulse.	The launch vehicle has an altered trajectory creating potential danger to bystanders	**9**	**3**	**5**	**135**	Ensure defects are not present on the hardware by visually checking for them. Always handle the hardware with care.
Motor Case (including the forward and aft closures)	Encloses the propellant grain and maintains pressure.	Case Deformation	Structural failure of the motor case including the forward or aft enclosures.	Internal pressure is not maintained, and propellant interacts with other components.	Motor assembly is damaged, and integrity of the motor and launch vehicle are compromised.	The launch vehicle is prone to having an unpredictable flight.	**7**	**2**	**8**	**112**	Ensure the integrity of the case by visually inspecting for defects. Always handle the motor case with care, and always have a protective cover over the casing until launch.
Motor Tube	Encloses the motor	Motor Tube is dislodged.	Structural failure of	The motor case is not	Risk of motor case forced	The launch vehicle is	**6**	**3**	**6**	**108**	Ensure the alignment of the tube by visually

	assembly in the correct position.		the motor tube.	held in the correct position.	through the launch vehicle, and misaligned thrust vector.	damaged, and the flight trajectory is altered.					inspecting for defects. Always handle the motor tube with care.
Motor Retainer	Retains the motor inside the rocket.	Motor Retainer cracks or breaks.	Structural failure of the motor retainer or unfastened screws.	The motor case assembly is not held in place.	Risk of motor case assembly forced through the launch vehicle.	The launch vehicle is damaged, and the motor's function is lost.	**7**	**4**	**7**	**196**	Ensure the motor is retained by tightening the screws.
Thrust Plate	Transfers the thrust from through the centering rings and to the airframe.	Thrust Plate cracks or breaks.	Structural failure of the thrust plate.	The integrity of the centering rings and the airframe are compromised.	Centering rings are damaged, and risk of damage to airframe.	The launch vehicle is damaged, and the flight trajectory is altered.	**5**	**4**	**7**	**140**	Ensure the thrust plate is properly fastened by visually inspecting for defects. Always handle the thrust plate with care.

Table 51: Flight dynamics FMEA

5.4 Environmental Concerns

Environmental hazards are those that the vehicle and environment impose on each other (Table 52) (Table 53). This also uses the same scoring system as the Personnel Hazards Analysis.

5.4.1 Effect of Environment on Launch Vehicle

Hazard	Cause	Effect	S	L	Score	Mitigation
Precipitation soaks launch vehicle	Dramatic humid conditions at launch site	Electronic Disruption and Energetics Leakage	**8**	**3**	**24**	Supply a canopy/large tent for prep area
Descent into body of water	Launch vehicle drifts out of range	Failed Recovery	**9**	**2**	**18**	Minimize drift with drogue, angle launch rail into wind
Launch vehicle lands in unknown terrain	Excessive drift	Unretrievable or Collision Damage	**8**	**3**	**24**	Check launch site for obstacles
Wind	Drift or trajectory change	Unretrievable	**5**	**2**	**10**	Angle launch rail into wind, scrub launch if wind reaches unsafe speed
Dryness	Brittle adhesive at epoxy joints	Fins, centering rings, and bulkheads crack or loosen	**7**	**3**	**21**	Manufacture with long curing time
High Temperatures	Electrical Components Overheat	Avionics and payload fail	**7**	**4**	**28**	Canopy at prep site
Debris entering airframe	Separated vehicle lands in dirt or sand	Debris could be lodged into payload mechanism, which could cause camera	**1**	**4**	**4**	Survey launch area and choose best launch rail based on windspeeds and corresponding drift calculations

		deployment to fail				

Table 52: Effects of Environment on Launch Vehicle

5.4.2 Effect of Launch Vehicle on Environment

Hazard	Cause	Effect	S	L	Score	Mitigation
Fire at launch prep site	Black powder spills and ignites	Fire damages private property	**8**	**3**	**24**	Ensure black powder is separated from electronics and any ignition sources during prep; store in ammo box until use
Fire at launch pad	Motor exhaust ignites surrounding grass	Fire damages private property	**3**	**6**	**18**	Remove dry grass from area directly around launch pad
Litter	Trash or components left on launch field	Pollution of local area	**2**	**6**	**12**	Inspect launch prep area before leaving to ensure no litter is left on the ground, and use garbage pack to collect litter
Launch vehicle/debris from launch vehicle	Parts of the launch vehicle separate completely and are lost, or the entire launch vehicle is lost	Pollution of local area	**10**	**2**	**20**	Have redundancies in the recovery system to prevent ballistic descent, conduct ground testing to ensure that recovery components are sufficiently strong
Chemical Leaks	Punctured batteries	Chemicals harm local plants and wildlife	5	2	**10**	Handle batteries with care and store them insecure areas away from sharp objects.

Table 53: Effects of Launch Vehicle on Environment

6. Project Plan

6.1 Launch Vehicle Testing

Testing information has been tabulated for quick reference. Each test has been assigned a test ID, which provides a method for referencing tests elsewhere in project documentation. The test ID is assigned based on the type of test it describes, outlined below (Table 54).

ID Abbreviation	Category
LV	Launch Vehicle Test
P	Payload Test
MS	Material Strength
MP	Material Property
SP	System Property
A	Avionics
R	Recovery
L	Launch Related
CF	Component Functionality
SI	System Integration
D	Durability
SF	System Functionality

Table 54: Test ID Abbreviations

6.1.1 Testing Rationale and Resultant Effects

The following section describes the testing plan that will prove the integrity of the launch vehicle design. It includes the objective and rationale for each test and predicts how the results of each test may influence design choices (Table 55).

Test ID	Title	Objective	Rationale	Resultant Effects
LV-MS-1	Airframe Material Compression Test	Measure compressive strength of airframe material.	Ensure that airframe material is capable of withstanding compressive forces during flight and landing. If the airframe does not have sufficient compressive strength, the launch vehicle could be destroyed during acceleration or sustain damage upon ground impact.	If the airframe material is found to have insufficient compressive strength, a new material will be selected.
LV-MS-2	Fin Material Bend Test	Measure flexural strength of fin material.	Ensure fin material is capable of withstanding forces from ground impact upon landing. If the fin material does not have sufficient flexural strength, the launch vehicle may sustain damage on landing and not be able to	If the fin material is found to have insufficient flexural strength, a new material will be selected.

			be flown for subsequent launches.	
LV-MS-3	Bulkhead Material Strength Test	Measure strength of bulkhead and eyebolt assembly.	Ensure bulkhead assembly is capable of withstanding force of parachute ejection. The bulkheads must be capable of withstanding large forces during separation events-- failure may result in the severing of vehicle sections from the recovery system.	If the bulkhead assembly is found to have insufficient strength, the materials used will be changed or the assembly will be reinforced.
LV-MS-4	Epoxy Strength Test	Measure shear strength of epoxy.	Ensure epoxy is capable of withstanding forces of vehicle operation. Epoxy secures several critical systems to the launch vehicle, including fins and recovery hardware, and must be able to resist the forces to which these attachments may be subjected.	If the epoxy is found to have insufficient shear strength, a different epoxy will be chosen, or mating surfaces will be joined with additional epoxy.
LV-MS-5	Recovery Harness Knot Efficiency Test	Determine the most efficient knot for use with recovery harness.	Ensure recovery harness attachments are capable of withstanding force of parachute ejection. Knots reduce the effective strength of rope by significant factors, so it is crucial that the reduction in strength is minimal for the knot that secures the recovery harness to the eyebolts on the launch vehicle.	The knot with the smallest reduction in strength will be used in the recovery system of the launch vehicle.
LV-MS-6	Recovery Harness Material Strength Test	Measure yield strength of recovery harness material.	Ensure recovery harness material is capable of withstanding force of parachute ejection.	If the recovery harness material is found to have insufficient strength, a new material will be chosen.

LV-MS-7	Airframe Material Impact Resistance Test	Measure the impact resistance of the airframe material.	Ensure airframe is capable of withstanding forces during vehicle recovery, during which time the vehicle sections will impact the ground with significant kinetic energy. The airframe material must be able to resist this impact with no damage.	If the airframe material is found to have insufficient impact resistance, a new material will be chosen, or the planned ground-hit velocity of the launch vehicle will be reduced.
LV-MS-8	Bulkhead Impact Resistance Test	Measure the impact resistance of the bulkhead material.	Ensure bulkheads are capable of withstanding forces during vehicle recovery. If damage is sustained during landing, the launch vehicle may not be able to be flown for subsequent launch attempts.	If the bulkhead is found to be unable to resist impacts without damage, the bulkhead will be made thicker, or a different material will be chosen.
LV-MS-9	Airframe Material Zippering Resistance Test	Determine the resistance of the airframe material to zippering.	Ensure airframe material can withstand the shearing force of the recovery harness decelerating the launch vehicle. Zippering in the airframe would render the launch vehicle unable to fly for subsequent launch attempts.	If airframe material is found to not be sufficiently resistant to zippering, the recovery harness will be modified, or a different material will be selected.
LV-MP-1	Epoxy Density Inspection	Measure the linear mass density of the epoxy fillets used in the launch vehicle.	Obtain density of epoxy for use in computer simulations. This will allow for more accurate estimations of launch vehicle properties and performance.	Epoxy density will be used to improve simulations.
LV-MP-2	Epoxy Environmental Exposure Test	Measure the resistance of the epoxy to adverse conditions.	Ensure epoxy does not weaken significantly from exposure to likely launch day conditions. The epoxy used in the launch vehicle may be exposed to high temperatures from the motor, and to the launch environment. It must retain strength through these conditions for multiple launch attempts.	If epoxy if found to be unable to resist the effects of launch conditions, a different epoxy will be selected.

LV-SP-1	Rotation and Rolling Analysis	Determine the rotation and rolling moment of the launch vehicle.	Identify and correct stability issues in launch vehicle.	Launch vehicle will be ballasted as needed.
LV-SP-2	Launch Vehicle Drag Analysis	Determine the potential for any adverse drag.	Identify and correct issues arising from excess or undesirable drag.	If dangerous drag is detected, adjust design to minimize the impact on the airframe exterior.
LV-SP-3	Center of Gravity Inspection	Determine the location of the center of gravity of the launch vehicle.	Ensure launch vehicle will fly in a stable manner. The simulated center of gravity may diverge from the actual center of gravity, so capturing this data improves simulations.	The actual center of gravity will be used to update computer simulations. Ballast may need to be added to ensure stability of the rocket depending on the results of the simulations.
LV-SP-4	Vehicle Vibration Test	Identify the resonant frequency of the launch vehicle.	Ensure launch vehicle can withstand vibrations during flight. Vibrations could compromise the structure of the launch vehicle, resulting in damage.	If dangerous vibrations are possible during launch, the rigidity of the rocket will be improved.
LV-SP-5	Fastening Hardware Vibration Test	Determine the resistance of the recovery hardware to vibration.	Ensure fastening hardware will remain secure during flight. Vibrations during launch can loosen hardware, causing sections of the airframe to be disconnected from the recovery system.	If hardware is found to be susceptible to loosening when subject to vibrations, anaerobic adhesives may be applied, and the installation torque will be inspected.
LV-A-1	Altimeter Functionality Demonstration	Determine whether altimeter detects changes in pressure.	Verify that altimeter responds to differing pressure. This test will ensure that the altimeters function.	If altimeter is found to be non-functional, a different altimeter will be selected.

LV-A-2	Altimeter Accuracy Test	Measure the accuracy of the altimeter in detecting a change in altitude.	Verify the accuracy of the altimeters used in the launch vehicle. Previous damage may have compromised the ability of the altimeters to report altitude data, which could result in ejection events occurring at the wrong altitude.	If altimeter is found to be inaccurate, a new altimeter will be selected.
LV-A-3	Avionics Battery Life Test	Measure the battery life of the avionics system.	Ensure launch vehicle can remain launch-ready for required period of time.	If battery life is found to be insufficient, a larger battery will be installed.
LV-A-4	Avionics Bulkhead Explosive Resistance Test	Determine the minimum size bulkhead needed to shield against ejection events.	Ensure avionics bay bulkheads are capable of withstanding ejection charges. The bulkheads shield sensitive electronics and must therefore be capable of withstanding the force of ejection charges detonating.	If bulkhead strength is found to be insufficient, the bulkhead will be made thicker, or a new material will be selected.
LV-A-5	Avionics Interference Demonstration	Identify conditions that will result in avionics failure due to interference.	Ensure avionics bay will not suffer interference from payload operation. Interference due to the GPS or payload could impair the ability of the altimeter to accurately trigger ejection events, resulting in parachute deployment at the wrong altitude, or failure to deploy at all.	If the avionics system is found to be susceptible to interference, additional shielding will be installed.
LV-A-6	GPS Accuracy Test	Determine the accuracy of the GPS.	The GPS is used to locate the launch vehicle in the event that line-of-sight tracking fails. As a result, the GPS must be able to accurately report the current location.	If the GPS is unable to accurately report location, a new GPS will be selected.

LV-A-7	GPS Range Test	Determine whether GPS is able to transmit over necessary range.	During recovery, the launch vehicle may drift over a significant distance. The GPS must be able to transmit the location of the launch vehicle over this distance.	If the GPS is unable to transmit over the required distance, a new GPS will be selected.
LV-R-1	Drogue Parachute Drag Test	Measure the coefficient of drag of the parachute.	Ensure parachute provides sufficient drag to reduce the launch vehicle descent rate to a speed at which the main parachute can be safely ejected. If the descent rate is too rapid, ejection of the main parachute could cause the airframe to zipper.	If the coefficient of drag of the parachute is found to be undesirable, a different parachute will be selected.
LV-R-2	Main Parachute Drag Test	Measure the coefficient of drag of the parachute.	Ensure parachute provides sufficient drag to reduce the ground hit velocity of the launch vehicle to a speed at which the vehicle will not be damaged.	If the coefficient of drag of the parachute is found to be undesirable, a different parachute will be selected.
LV-R-3	Parachute Preparation Test	Measure the packed dimensions of parachutes when folded using different methods.	Determine best way to ensure parachutes can be properly loaded into the launch vehicle. Limited space is available within the airframe, so the most space efficient packing method is desirable.	If parachute is found to be too large, a different packing method will be employed, or a different parachute will be selected.
LV-R-4	Parachute Deployment Time Test	Measure the time for the parachute to open upon deployment.	Ensure that parachute will quickly deploy upon ejection. Quickly has been defined as less time than it takes for the launch vehicle to fall 100 ft (disregarding air resistance).	If parachute is found to open too slowly, a different packing method will be employed.
LV-R-5	Parachute Ejection Demonstration	Verify that ejection charge is large enough to cause separation.	Ensure vehicle separates and parachute is ejected from airframe. If the ejection charge is not large enough, it may not cause separation or ejection of the parachute, which would result in a failure of the recovery system and probable damage	If the vehicle sections fail to separate or the parachutes fail to eject, the ejection charge will be made larger.

			to the vehicle upon ground hit.	
LV-R-6	Parachute Unfolding Demonstration	Determine whether packing method will result in tangling upon deployment.	Ensure parachute does not tangle upon deployment. Tangling of the shroud lines would reduce the efficiency of the parachute and cause a higher descent rate than is desirable.	If parachute tangles upon opening, a different folding method will be selected.
LV-L-1	Launch Rehearsal Demonstration	Measure time required to prepare launch vehicle for launch.	Ensure launch vehicle and payload can be prepared in allotted time, defined by the competition requirements as two hours.	If team is unable to prepare the launch vehicle in the allotted time, the assembly process will be streamlined and practiced further.
LV-L-2	Subscale Demonstration Launch	Verify successful function of launch vehicle concept.	Ensure subscale design can be successfully launched and recovered.	If subscale launch is unsuccessful, the cause of failure will be identified and corrected, and a new attempt will be made.
LV-L-3	Vehicle Demonstration Launch	Verify successful function of launch vehicle.	Ensure launch vehicle can be successfully launched and recovered.	If the vehicle demonstration launch is unsuccessful, the cause of failure will be identified and corrected, and a new attempt will be made.

Table 55: Launch Vehicle Testing Plan: Objectives, Rationales, and Resultant Effects

6.1.2 Testing Plan and Description

The following section details the measured variable and success criteria for each test, along with an overview of the methodology (Table 56).

Test ID	Title	Variable	Success Criteria	Overview of Procedure	Required Materials

LV-MS-1	Airframe Material Compression Test	Ultimate compressive strength of airframe material.	Airframe material is able to withstand forces similar to expected launch and landing conditions with no damage.	Use Instron UTM to experimentally determine maximum compressive strength of a section of airframe.	Instron UTM, airframe material sample
LV-MS-2	Fin Material Bend Test	Ultimate flexural strength of fin material.	Fin material is able to withstand forces similar to expected landing conditions with no damage.	Use Instron UTM to experimentally determine maximum flexural strength of a sample of fin material through a four-point flexural test.	Instron UTM, fin material sample
LV-MS-3	Bulkhead Material Strength Test	Ultimate tensile strength of bulkhead and eyebolt assembly.	Bulkhead material is able to withstand forces similar to conditions expected during ejection events.	Use Instron UTM to experimentally determine maximum tensile strength of bulkhead and eyebolt assembly.	Instron UTM, bulkhead material, eyebolt
LV-MS-4	Epoxy Strength Test	Ultimate shear strength of epoxy.	Epoxy is capable of withstanding forces similar to expected launch and landing conditions.	Use Instron UTM to experimentally determine maximum shear strength of epoxy through a lap shear test.	Instron UTM, epoxy, material sample
LV-MS-5	Recovery Harness Knot Efficiency Test	Ultimate tensile strength for various knots.	A knot is identified with a minimal reduction of recovery harness strength.	Use Instron UTM to experimentally determine the efficiency of several knots used to secure recovery harness to eyebolts.	Twine, test stand, bucket, weight, hanging scale, rope

LV-MS-6	Recovery Harness Material Strength Test	Yield strength of recover harness material.	Recovery harness material is capable of withstanding forces similar to conditions expected during ejection events.	Use Instron UTM to experimentally determine maximum yield strength of recovery harness material.	Instron UTM, recovery harness material sample
LV-MS-7	Airframe Material Impact Resistance Test	Impact resistance of airframe material.	Airframe material is able to withstand impact with no visible damage.	Drop section of airframe from a height that simulates flight recovery conditions.	Airframe material sample
LV-MS-8	Bulkhead Impact Resistance Test	Impact resistance of bulkhead material.	Bulkhead material is able to withstand impact with no visible damage.	Drop a bulkhead attached to a weighted airframe from a height that simulates flight recovery conditions.	Airframe material sample, bulkhead
LV-MS-9	Airframe Material Zippering Resistance Test	Shear strength of airframe material.	Airframe material is able to withstand 100 lbs of shear force applied by the recovery harness without zippering.	Apply brief force to airframe and recovery harness assembly to simulate ejection conditions.	Airframe material sample, recovery harness, hanging scale, vise, protective cloth
LV-MP-1	Epoxy Density Inspection	Linear mass density of epoxy fillets.	Epoxy density is determined.	Prepare a known volume of epoxy, and once cured, measure weight.	JB Weld steel and hardener epoxy, RocketPoxy steel and hardener epoxy, scale, stirring utensil and receptacle, flat test material, ruler

LV-MP-2	Epoxy Environmental Exposure Test	Ultimate shear strength of epoxy.	Epoxy does not demonstrate a reduction in strength of greater than 20% due to exposure to adverse conditions.	Prepare epoxy sample and, once cured, place outside for a minimum of two hours before testing shear strength.	Instron UTM, epoxy, material sample
LV-SP-1	Rotation and Rolling Analysis	Rotation and rolling moment.	Stable flight is predicted by simulations.	Simulate launch vehicle behavior in radial and axial direction.	MATLAB, OpenRocket
LV-SP-2	Launch Vehicle Drag Analysis	Drag of launch vehicle.	Drag due to payload is minimal.	Simulate launch vehicle aerodynamics to measure drag.	SolidWorks
LV-SP-3	Center of Gravity Inspection	Center of gravity of launch vehicle.	Center of gravity is identified.	Balance rocket along axis and record location.	Launch vehicle, rope
LV-SP-4	Vehicle Vibration Test	Resonant frequency of launch vehicle.	Vehicle does not demonstrate a resonant frequency that will result in structural damage.	Simulate vibrations vehicle is likely to experience during launch.	Launch vehicle, mallet, tuning device
LV-SP-5	Fastening Hardware Vibration Test	Fastener torque.	Hardware does not loosen when subjected to vibration.	Simulate vibrations D-links and nuts and bolts used in launch vehicle are likely to experience during launch.	D-links, nuts, bolts, DC motor, test stand
LV-A-1	Altimeter Functionality Demonstration	Altimeter response to pressure variation.	Altimeter is able to identify a change in air pressure.	Place altimeter in a jar that can be pressurized using a syringe. Pressurize and record change in volume of air to calculate change in pressure.	Altimeter, modified jar, syringe, tubing, ruler

LV-A-2	Altimeter Accuracy Test	Altitude reported by altimeter.	Altimeter is able to report altitude with a percent error of less than 5%.	Move altimeter to several known altitudes and record altimeter reading.	Altimeter, laptop, altimeter USB cables, measuring tape
LV-A-3	Avionics Battery Life Test	Battery voltage over time.	Battery voltage does not fall below 8.4V (the point at which the altimeter will turn off) for a duration of three hours.	Run avionics instrumentation on standby until battery is depleted, and record voltage at intervals during the three-hour period.	Stratologger altimeter, Entacore altimeter, laptop, altimeter USB cables, two 9V batteries
LV-A-4	Avionics Bulkhead Explosive Resistance Test	Thickness of avionics bulkhead.	Bulkhead is able to resist an ejection event with no visible damage to bulkhead or interior of avionics bay.	Fire ejection charge against bulkheads of varying thicknesses to determine necessary size of bulkhead for avionics bay.	Bulkheads, airframe, ejection charge, 9V battery, test stand
LV-A-5	Avionics Interference Demonstrati on	Avionics functionality.	Avionics electronics do not demonstrate a reduction in functionality when exposed to potential interference.	Perform altimeter accuracy test with avionics bay assembly in final configuration with operational payload positioned the same distance away as when installed in the launch vehicle.	Payload assembly, avionics bay assembly
LV-A-6	GPS Accuracy Test	GPS positional error.	GPS reports position with percent error of less than 5%.	Turn on GPS and take reading. Compare to known location coordinates.	GPS

LV-A-7	GPS Range Test	GPS transmission range.	GPS is capable of transmitting over a minimum distance of one mile.	Turn on GPS and take reading. Move GPS away from receiver in increments of 0.25 mi, until a distance of 1 mi is achieved. Ensure that receiver is able to receive location at each increment.	GPS, GPS receiver
LV-R-1	Drogue Parachute Drag Test	Coefficient of drag of parachute.	Measured coefficient of drag will result in acceptable launch vehicle descent rates according to simulations.	Drop weighted parachute from height to simulate vehicle recovery conditions.	Parachute, weight, drop site, camera, tape measure
LV-R-2	Main Parachute Drag Test	Coefficient of drag of parachute.	Measured coefficient of drag will result in acceptable launch vehicle descent rates according to simulations.	Drop weighted parachute from height to simulate vehicle recovery conditions.	Parachute, weight, drop site, camera, tape measure
LV-R-3	Parachute Preparation Test	Dimensions of packed parachutes	Packed dimensions of each method are determined.	Fold drogue and main parachute and install in launch vehicle.	Parachute, parachute protector, airframe section
LV-R-4	Parachute Deployment Time Test	Time to open upon deployment.	Parachute will open in less time than an object will fall in 100 ft, disregarding air resistance—this has been calculated to be 2.49 seconds following ejection.	Drop folded and weighted parachute from a height to simulate vehicle recovery conditions.	Parachute, parachute protector, weight, airframe section

LV-R-5	Parachute Ejection Demonstration	Ejection charge size.	Vehicle sections will separate, and parachute will be fully ejected from the airframe.	Prepare vehicle in launch configuration and manually fire ejection charges using a wire and 9V battery, for both main and drogue charges.	Launch vehicle, drogue and main parachute, parachute protectors, recovery harness, insulation material, ejection charges, wire, test stand, 9V battery
LV-R-6	Parachute Unfolding Demonstration	Tangled shroud lines or recovery harnesses.	Parachute will open without tangling of shroud lines or recovery harnesses.	Fold parachute and pull each end of the recovery harness, unfolding parachute.	Recovery harness, parachutes, parachute protectors
LV-L-1	Launch Rehearsal Demonstration	Assembly time.	Assembly of launch vehicle will be accomplished in less than two hours.	Prepare full-scale launch vehicle assembly and payload assembly in launch configuration (excepting motor).	Launch vehicle, payload, and associated components
LV-L-2	Subscale Demonstration Launch	Subscale launch vehicle performance.	Subscale launch vehicle will successfully launch and be recovered without damage to the vehicle.	Launch subscale vehicle.	Subscale model, launch rail, remote igniter
LV-L-3	Vehicle Demonstration Launch	Full-scale launch vehicle performance.	Launch vehicle will successfully launch and be recovered without damage to the vehicle.	Launch full-scale vehicle in final configuration.	Launch vehicle

Table 56: Launch Vehicle Testing Plan: Variables, Success Criteria, Methods, and Materials

6.1.3 Completed Launch Vehicle Testing

The following table tracks the completion status of each test; the results of completed tests are described in this section (Table 57).

Test ID	Title	Completion Status
LV-MS-1	Airframe Material Compression Test	Incomplete
LV-MS-2	Fin Material Bend Test	Incomplete
LV-MS-3	Bulkhead Material Strength Test	Incomplete
LV-MS-4	Epoxy Strength Test	Incomplete
LV-MS-5	Recovery Harness Knot Efficiency Test	Complete
LV-MS-6	Recovery Harness Material Strength Test	Incomplete
LV-MS-7	Airframe Material Impact Resistance Test	Complete
LV-MS-8	Bulkhead Impact Resistance Test	Incomplete
LV-MS-9	Airframe Material Zippering Resistance Test	Complete
LV-MP-1	Epoxy Density Inspection	Complete
LV-MP-2	Epoxy Environmental Exposure Test	Incomplete
LV-SP-1	Rotation and Rolling Analysis	Incomplete
LV-SP-2	Launch Vehicle Drag Analysis	Incomplete
LV-SP-3	Center of Gravity Inspection	Partial; subscale complete
LV-SP-4	Vehicle Vibration Test	Incomplete
LV-SP-5	Fastening Hardware Vibration Test	Incomplete
LV-A-1	Altimeter Functionality Demonstration	Incomplete
LV-A-2	Altimeter Accuracy Test	Complete
LV-A-3	Avionics Battery Life Test	Complete
LV-A-4	Avionics Bulkhead Explosive Resistance Test	Incomplete
LV-A-5	Avionics Interference Demonstration	Incomplete
LV-A-6	GPS Accuracy Test	Incomplete
LV-A-7	GPS Range Test	Incomplete
LV-R-1	Drogue Parachute Drag Test	Complete
LV-R-2	Main Parachute Drag Test	Complete
LV-R-3	Parachute Preparation Test	Complete
LV-R-4	Parachute Deployment Time Test	Complete
LV-R-5	Parachute Ejection Demonstration	Partial; subscale complete
LV-R-6	Parachute Unfolding Demonstration	Partial; subscale complete
LV-L-1	Launch Rehearsal Demonstration	Incomplete
LV-L-2	Subscale Demonstration Launch	Complete
LV-L-3	Vehicle Demonstration Launch	Incomplete

Table 57: Completed Launch Vehicle Tests

6.1.3.1 Recovery Harness Knot Efficiency Test

Test ID: LV-MS-5

Knots are employed to secure the recovery harness to the quick links and eyebolts that are then fastened to bulkheads in the launch vehicle. During ejection events, the recovery harness (and, by extension, the knots) are subjected to significant forces. Knots typically represent the weak point in a line, reducing the strength of the line by significant factors. This test will compare the strength of different knots that could be used to secure the recovery harness to the eyebolts in the launch vehicle.

Independent variable: Knot type.

Dependent variable: Knot strength.

Success criteria: A knot is identified with a minimal reduction of recovery harness strength.

Materials:

- Twine
- Test stand (cinder blocks and PVC pipes)
- Bucket
- Weight
- Hanging scale
- Rope

A test stand was constructed using cinder blocks and PVC pipes. The test stand included a crossbar around which the twine could be wrapped and was tall enough to suspend the bucket and weight above the ground below (Figure 79).

Figure 79: Knot Strength Test Stand

Twine was used since it has a significantly lower ultimate tensile strength than the recovery harness material. However, the comparative knot strengths should be similar for both materials. The twine was wrapped around the PVC crossbar on one end and the knot was tied around the handle of the bucket. This was done so that the stress would be concentrated on the knot, and the point of failure would be at the knot (Figure 80).

Figure 80: Failure at Knot Location

Weight was then loaded into the bucket until failure of the knot. Once failure occurred, the weight was measured using a hanging scale, also attached to the test stand (Figure 81).

Figure 81: Measurement of Weight at Tensile Failure

Research revealed three potential knot candidates: the figure-eight knot, the bowline, and the slip knot (Figure 82). The figure-eight and bowline both produce a fixed-size loop in the line, and the slip knot creates a variable size loop; all are useful for securing the recovery harness.

Figure 82: From Left to Right: The Figure-Eight Knot, the Bowline, and the Slip Knot

The results for each have been tabulated and graphed below (Figure 83, Figure 84). The figure-eight and slip knot demonstrated very similar strengths, whereas the bowline was found to be significantly weaker.

Knot Strength Comparison (lbs)			
Trial	**Figure-Eight**	**Bowline**	**Slip Knot**
1	58.67	36.39	53.58
2	46.26	39.82	45.32
3	38.54	51.76	44.70
Average	**47.82**	**42.66**	**47.87**

Figure 83: Tabulated Knot Strength Data

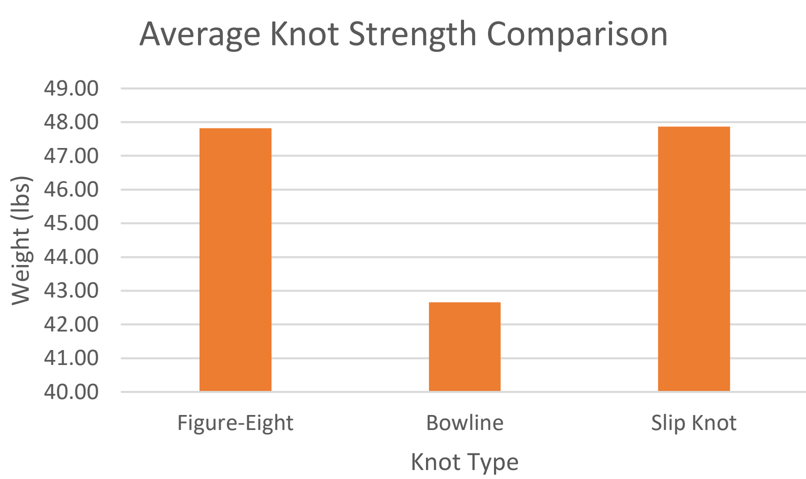

Figure 84: Graphed Knot Strength Data

6.1.3.2 Airframe Material Impact Resistance Test

Test ID: LV-MS-7

Recovery of the launch vehicle, specifically the landing, is a launch phase in which airframe components are subjected to potentially damaging forces upon impact with the ground at high velocities. Thus, it is necessary to evaluate different airframe material samples for their durability to impacts.

Independent variable: Airframe material.

Dependent variable: Airframe material impact resistance.

Success criteria: The test is deemed successful if the airframe material has no damage after landing.

Materials:

- G-12 fiberglass cylindrical sample
- Phenolic cylindrical sample
- Quantum tube cylindrical sample
- Blue tube cylindrical sample

Potential airframe materials must be exposed to high velocity collisions with the ground to ensure that no permanent damage will occur in flight scenarios. To test this, each potential airframe material was dropped from the top floor of a parking garage, an approximate height of 33 ft and 5 in, into a safe (clear of people), grass-covered area below.

Based on the height at which each airframe sample was dropped, an estimate of the impact velocity of the samples was calculated. Determining the drag for a cylindrical object tumbling in the air is prohibitively complex, but by disregarding air resistance it was estimated that each airframe sample impacted the ground with an approximate velocity of 46 ft/s (Equation 11). Since this is a significantly greater velocity than the expected ground hit velocity of the launch vehicle under typical launch conditions, it was assumed that these methods will yield an acceptably accurate demonstration of actual performance.

$$v = \sqrt{v_o + 2a\Delta x} \sim \sqrt{(0) + 2(32\ \text{ft/s}^2)(33\ \text{ft})} \sim 46\ \text{ft/s}$$

Equation 11: Ground Hit Velocity of Airframe Sections

Following each individual drop test, the airframe materials sample were collected and inspected thoroughly for signs of visible damage, including cracks, tears, or deformations. None of the airframe tubes showed any sign of damage after landing (Figure 85). Therefore, all airframe materials demonstrated sufficient impact resistance to meet the success criteria for this test.

Figure 85: Phenolic (top left), Quantum Tube (top right), Blue Tube (bottom left), and G-12 Fiberglass (bottom right) Airframes After Drop Test

6.1.3.3 Airframe Material Zippering Resistance Test

Test ID: LV-MS-9

During recovery events, the parachutes and recovery harness are forcibly ejected from the airframe and cause a rapid deceleration of the launch vehicle. As this occurs, the recovery harness can pull against the airframe, potentially tearing the airframe in a process known as ‘zippering’. This test will determine whether the airframe material will have sufficient strength to resist zippering during ejection events.

Independent variable: Airframe material.

Dependent variable: Airframe shear strength.

Success criteria: The test is deemed successful if the airframe resists 100 lbs of shear force exerted by the recovery harness material without any visible damage. This value has been selected as the airframe is not expected to be subject to a greater force than 100 lbs.

Materials:

- Airframe material sample – G12 Fiberglass (approximately 9 in long)
- Recovery harness material sample
- Digital hanging scale (capable of measuring at least 100 lbs of force)
- Vise (or other method of holding airframe material sample)
- Protective cloth

The airframe material sample was mounted in a vise, and the recovery harness material sample was placed through the cylindrical airframe (Figure 86). One end of the recovery harness was secured to a stationary object, and the other end was attached to the digital hanging scale. The recovery harness was then pulled away from the airframe with increasing force, until the scale read 100 lbs. At this point, the force was released, and the airframe was inspected for damage.

Figure 86: Airframe Material Zippering Resistance Test

No damage was identified on the airframe sample; thus, the test was determined to be a success, and no changes were determined to be necessary to the launch vehicle design.

6.1.3.4 Epoxy Density Inspection

Test ID: LV-MP-1

Epoxy is used on both the inside and outside of the rocket frame to secures certain components in place, such as the fins. Determining the density of the different epoxy brands used will improve the accuracy of computer simulations.

Independent variable: Epoxy.

Dependent variable: Linear mass density of epoxy.

Success criteria: Linear mass density of epoxy is determined.

Materials:

- JB Weld steel and hardener epoxy
- RocketPoxy steel and hardener epoxy
- Scale
- Stirring utensil & receptacle
- 4 pieces of flat test material (2 for each epoxy type)
- Ruler

Approximately 1 ounce of JB Weld epoxy was prepared by thoroughly mixing equal amounts by weight of steel and hardener in a small container. A similar sample of RocketPoxy epoxy was also prepared. The flat test material samples were weighed, then filleted together with epoxy at a right angle (Figure 87). At least 6 hours were allowed for the epoxy to set.

Figure 87: Epoxy Fillet

Once set, the material samples were weighed again. The linear density of the epoxy was calculated by subtracting the mass of the original material samples from the mass of the samples with epoxy and dividing by the length of the samples spanned by the epoxy fillet (Equation 12). The calculated linear mass density of the JB Weld epoxy was found to be 0.0748 oz/in, and the linear mass density of the RocketPoxy was found to be 0.0802 oz/in (Table 58).

$$\lambda = \frac{m_e}{L} = \frac{m_f - 2m}{L}$$

Equation 12: Linear Density of Epoxy Fillet

Description	Variable	Measured Value	
		JB-Weld	RocketPoxy
Mass of test material	m	2.753 oz	2.753 oz
Mass of epoxied test material	m_f	5.859 oz	5.877 oz
Mass of epoxy	m_e	0.353 oz	0.371 in
Length of material	L	4.719 in	4.625 in
Linear density	λ	0.0748 oz/in	0.0802 oz/in

Table 58: Epoxy Density Inspection Results

6.1.3.5 Altimeter Accuracy Inspection

Test ID: LV-A-2

The altimeters determine when the recovery events occur, and therefore must be able to accurately report the altitude of the launch vehicle. This inspection will ensure that the altimeters being used are sufficiently accurate.

Independent variable: Altimeter.

Dependent variable: Percent error with respect to change in altitude.

Success criteria: The inspection is deemed successful if the percent error of the altimeter reading is less than or equal to5%.

Materials:

- Altimeter
- Laptop
- Altimeter USB cables
- Measuring tape

This inspection was performed on a Stratologger (to be used for the primary recovery system), and four Entacore altimeters (one of which was to be used for the secondary recovery system). The goal was to ensure that the altimeters used in the launch vehicle reported the altitude with minimal error. The Entacore altimeters were calibrated to the local elevation before beginning. The Stratologger does not require this step, as it automatically performs this function. Then, altitude readings were recorded from each altimeter at a low height, using the altimeter software on the laptop. The altimeters were raised 33.417 ft, and a second reading was taken. This process was repeated for three trials.

The Entacore altimeter software is capable of displaying height, so these values were recorded. However, the Stratologger software only displays pressure; thus, the pressure readings were recorded, and the height change was calculated using the barometric formula (Equation 13) (Table 59). Percent error was calculated using the formula for experimental error, where the measured altitude change, Δh_m, is compared to the actual altitude change, Δh_a (Equation 14).

$$\Delta h = \frac{RT \ln(P_{HI}/P_{LO})}{-gM}$$

Equation 13: Modified Barometric Formula

Description	Variable	Value
Universal Gas Constant	R	8.949x10^4 lb·ft^2/(lb-mol·K·s^2)
Temperature	T	294.261 K
Upper Pressure	P_{HI}	- inHg
Lower Pressure	P_{LO}	- inHg
Gravitational Acceleration	G	32.147 ft/s^2
Molar Mass of Air	M	28.964 lb/lb-mol

Table 59: Barometric Formula Variables and Constants

$$\text{Percent Error} = \left|\frac{\Delta h_m - \Delta h_a}{\Delta h_a}\right| \times 100$$

Equation 14: Experimental Error Formula

The results show that the Stratologger altimeter and three of the four Entacore altimeters demonstrated sufficient accuracy, with errors under 5% (Table 60). However, one of the Entacore altimeters exceeded the error limit. As a result, this altimeter will not be used in the launch vehicle. However, the other four altimeters will be acceptable. Ultimately, the Stratologger and the Entacore (ID#: AIMUSB1342) were used in the subscale launch vehicle and will be for the full-scale vehicle.

Stratologger				
Trial	**Lower Altitude (inHg)**	**Upper Altitude (inHg)**	**Δh (ft)**	**Percent Error**
1	30.179	30.142	34.666	
2	30.179	30.142	34.666	
3	30.179	30.142	34.666	
Average			**34.666**	**3.60%**
Entacore (ID#: AIMUSB1349)				
Trial	**Lower Altitude (ft)**	**Upper Altitude (ft)**	**Δh (ft)**	**Percent Error**
1	171.916	204.396	32.480	
2	170.276	204.396	34.120	
3	171.260	203.740	32.480	
Average			**33.027**	**1.18%**
Entacore (ID#: AIMUSB1342)				
Trial	**Lower Altitude (ft)**	**Upper Altitude (ft)**	**Δh (ft)**	**Percent Error**
1	173.556	205.381	31.825	
2	171.916	205.709	33.793	
3	171.916	205.709	33.793	
Average			**33.137**	**0.84%**
Entacore (ID#: AIMUSB01338)				
Trial	**Lower Altitude (ft)**	**Upper Altitude (ft)**	**Δh (ft)**	**Percent Error**
1	160.433	193.241	32.808	
2	161.745	191.929	30.184	
3	161.089	193.570	32.481	
Average			**31.824**	**5.00%**
Entacore (ID#: AIMUSB01316)				
Trial	**Lower Altitude (ft)**	**Upper Altitude (ft)**	**Δh (ft)**	**Percent Error**
1	170.276	200.787	30.511	
2	171.260	200.131	28.871	
3	170.604	199.147	28.543	
Average			**29.308**	**14.02%**

Table 60: Altimeter Accuracy Inspection Results

6.1.3.6 Avionics Battery Life Test

Test ID: LV-A-3

Once the avionics aboard the launch vehicle are armed, a number of delays could result in the launch vehicle sitting idle on the launch pad for an extended period. Thus, the avionics battery must be capable of providing power to the electronics for this entire period, which has been defined as two hours.

Independent variable: Battery.

Dependent variable: Battery voltage.

Success criteria: The test is deemed successful if the battery voltage does not drop below 8.4V after the altimeters are allowed to idle for three hours.

Materials:

- Stratologger altimeter
- Entacore altimeter
- Laptop
- Altimeter USB cables
- Two 9V batteries

Both altimeters were configured as they would be during flight, albeit without the ejection charges installed. The altimeters were then monitored using the corresponding software through the laptop and USB cables. This allowed the battery voltage to be monitored. The voltage was recorded at various times during a three-hour period. This period was chosen, as it ensures a minimum factor of safety of 1.5 for the altimeter battery life.

The results showed that the 9V batteries used provide sufficient power for the two-hour minimum duration, with an additional factor of safety of 1.5 (Figure 88). The battery voltage did not drop below 8.4 volts for the entire three hours, indicating the potential for additional battery life beyond the tested time period. Notably, however, the Entacore demonstrated a higher rate of power consumption than the Stratologger.

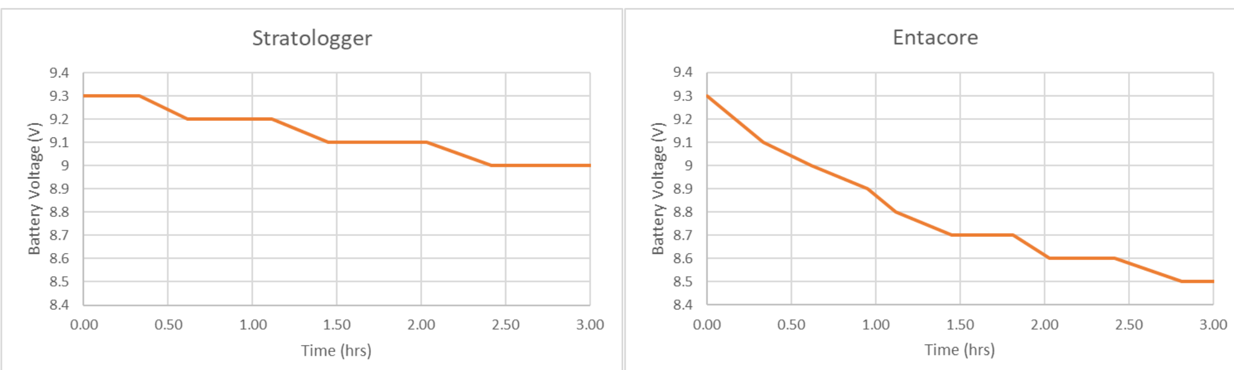

Figure 88: Battery Voltage

6.1.3.7 Drogue Parachute Drag Test

Test ID: LV-R-1

Determining the coefficient of drag for parachutes used in the launch vehicle will assist in improving simulations for descent rates. The drogue parachute must be able to reduce the descent rate of the launch vehicle to a point that the main parachute can be safely ejected. However, it must also permit the launch vehicle to descend quickly enough to limit drift due to wind.

Independent variable: Parachute.

Dependent variable: Coefficient of drag.

Success Criteria: Coefficient of drag is measured.

Materials:

- Parachute
- Weight
- Drop site
- Camera
- Tape measure

Weights were measured and attached to each parachute being tested (masses have been tabulated in Table 61 and Table 62). The amount of weight used was scaled roughly based on parachute size so that a reasonable descent rate was obtained in testing. Then the parachutes were dropped from an elevated location, approximately 60 ft above ground (Figure 89). A camera was positioned at a fixed distance above the ground (measured with the tape measure) and recorded a video of the parachute descent.

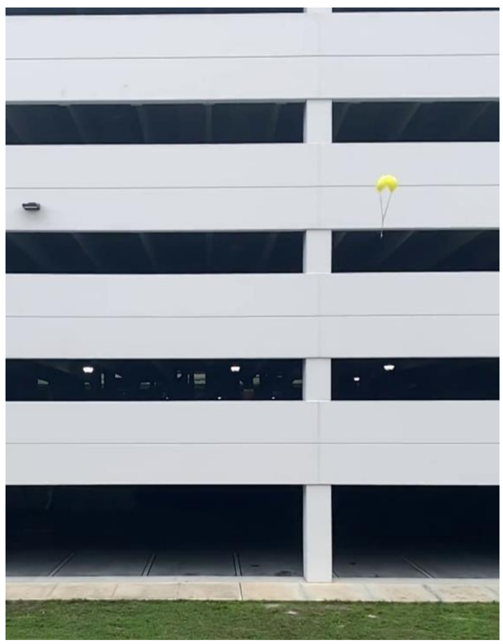

Figure 89: Drogue Parachute Drag Testing

Later, a script was written in Python to assign a frame number to each frame in the video, which allowed the number of frames between when the parachute passed level with the camera and when it hit the ground to be counted. Then, the elapsed time could be calculated from this data and the descent rate calculated. From here, the attached mass, diameter of the parachute, and the descent rate were used to calculate the coefficient of drag.

The data for both the subscale and full-scale drogue parachute has been tabulated for reference. When the values for the initial drogue parachute were inputted into simulations for the launch vehicle, the resulting descent rates were not ideal. Thus, a second parachute was selected and tested. The results for both parachutes have been included below (Table 61, Table 62).

Subscale Drogue - 1						
Trial	**Elapsed Time (s)**	**Distance (ft)**	**Descent Rate (ft/s)**	**Mass (oz)**	**Diameter (in)**	**Coefficient of Drag**
1	0.883	14.167	16.038	17.108	24	1.113
2	1.008	14.167	14.050	17.108	24	1.451
3	0.883	14.167	16.038	17.108	24	1.113
Average			**15.375**			**1.226**
Subscale Drogue – 2						
Trial	**Elapsed Time (s)**	**Distance (ft)**	**Descent Rate (ft/s)**	**Mass (oz)**	**Diameter (in)**	**Coefficient of Drag**
1	1.100	14.167	12.879	4.000	24	0.404
2	0.933	14.167	15.179	4.000	24	0.291
3	1.067	14.167	13.282	4.000	24	0.380
Average			**13.780**			**0.358**

Table 61: Subscale Drogue Coefficient of Drag

Full-Scale Drogue						
Trial	**Elapsed Time (s)**	**Distance (ft)**	**Descent Rate (ft/s)**	**Mass (oz)**	**Diameter (in)**	**Coefficient of Drag**
1	1.367	14.167	10.366	17.813	24	2.775
2	1.325	14.167	10.692	17.813	24	2.608
3	1.108	14.167	12.782	17.813	24	1.825
4	1.042	14.167	13.600	17.813	24	1.612
Average			**11.860**			**2.205**

Table 62: Full-Scale Drogue Coefficient of Drag

6.1.3.8 Main Parachute Drag Test

Test ID: LV-R-2

Determining the coefficient of drag for parachutes used in the launch vehicle will assist in improving simulations for descent rates. The main parachute must be able to reduce the descent rate of the launch vehicle to a point that upon landing, the launch vehicle will not sustain damage due to the impact.

Independent variable: Parachute.

Dependent variable: Coefficient of drag.

Success Criteria: Coefficient of drag is measured.

Materials:

- Parachute
- Weight
- Drop site
- Camera

- Tape measure

Weights were measured and attached to each parachute being tested (masses have been tabulated in Table 63 and Table 64). The amount of weight used was scaled roughly based on parachute size so that a reasonable descent rate was obtained in testing. Then the parachutes were dropped from an elevated location, approximately 60 ft above ground (Figure 90). A camera was positioned at a fixed distance above the ground (measured with the tape measure) and recorded a video of the parachute descent.

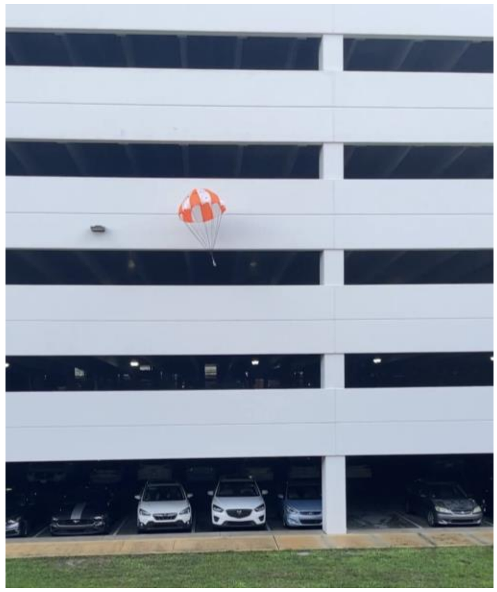

Figure 90: Main Parachute Drag Testing

Later, a script was written in Python to assign a frame number to each frame in the video, which allowed the number of frames between when the parachute passed level with the camera and when it hit the ground to be counted. Then, the elapsed time could be calculated from this data and the descent rate calculated. From here, the attached mass, diameter of the parachute, and the descent rate were used to calculate the coefficient of drag.

The data for both the subscale and full-scale main parachute has been tabulated for reference. Later inspection of the first parachute revealed significant damage from previous launches that had the potential to result in failure of the parachute during recovery. So, a second parachute was selected and tested. However, when simulations were performed using the size and determined coefficient of drag of this new parachute, the descent rates were found to be too high. Thus, a third parachute was selected and tested. It is this parachute that was ultimately used in the subscale launch vehicle. The results for all three parachutes have been included below (Table 63, Table 64).

Subscale Main - 1						
Trial	**Elapsed Time (s)**	**Distance (ft)**	**Descent Rate (ft/s)**	**Mass (oz)**	**Diameter (in)**	**Coefficient of Drag**
1	2.942	14.167	4.816	24.163	48	4.360
2	2.258	14.167	6.273	24.163	48	2.569
3	1.992	14.167	7.113	24.163	48	1.998

Average			6.067			2.976
Subscale Main - 2						
Trial	**Elapsed Time (s)**	**Distance (ft)**	**Descent Rate (ft/s)**	**Mass (oz)**	**Diameter (in)**	**Coefficient of Drag**
1	2.033	14.167	6.967	5.800	48	0.500
2	1.433	14.167	9.884	5.800	48	0.248
3	1.367	14.167	10.366	5.800	48	0.226
Average			**9.072**			**0.325**
Subscale Main - 3						
Trial	**Elapsed Time (s)**	**Distance (ft)**	**Descent Rate (ft/s)**	**Mass (oz)**	**Diameter (in)**	**Coefficient of Drag**
1	2.275	14.167	6.227	22.200	60	1.533
2	2.458	14.167	5.763	22.200	60	1.790
3	2.483	14.167	5.705	22.200	60	1.827
Average			**5.898**			**1.717**

Table 63: Subscale Main Coefficient of Drag

Full-Scale Main						
Trial	**Elapsed Time (s)**	**Distance (ft)**	**Descent Rate (ft/s)**	**Mass (oz)**	**Diameter (in)**	**Coefficient of Drag**
1	2.158	14.167	6.564	33.158	72	1.431
2	3.183	14.167	4.450	33.158	72	3.114
3	3.358	14.167	4.218	33.158	72	3.465
4	2.550	14.167	5.556	33.158	72	1.998
Average			**5.197**			**2.502**

Table 64: Full-Scale Main Coefficient of Drag

6.1.3.9 Parachute Preparation Test

Test ID: LV-R-3

Space is limited inside the launch vehicle, so parachutes must be folded in a space-efficient manner. This test explored several different folding methods, evaluating the space efficiency of each.

Independent variable: Folding method.

Dependent variable: Packed length of parachute (while maintaining ability to fit within airframe).

Materials:

- Parachute
- Parachute protector
- Airframe section

Each parachute was folded and placed within the parachute protector using each of the three folding methods, and the same method of wrapping within the parachute protector. The three methods are as follows:

Method 1: The parachute is folded in half and laid flat. Grab the parachute at the edge where one of the shroud lines is attached and fold the parachute over to the next shroud line. Repeat this process, until the parachute has been folded into a long triangle with all the shroud lines extending from one point of the triangle. Z-fold the shroud lines and center them on the folded parachute. Fold the long ends of the triangular parachute over the shroud lines. Finally, Z-fold the parachute and wrap with a parachute protector (Figure 91).

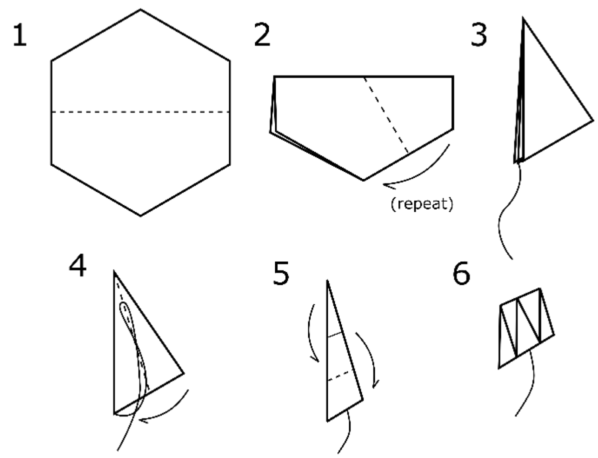

Figure 91: Method 1 Folding Steps

Method 2: This method is based on suggestions from Rocketman Parachutes. This method is initially intended for parabolic parachutes but can be adapted for other geometries. Fold the parachute in half and lay it on a flat surface so that the shroud lines line up. Then, fold the parachute in half, once again ensuring the shroud lines are aligned. Now, tri-fold the long sides of the parachute inward and fold the parachute in half long-ways. Zigzag the shroud lines and lay them on top of the parachute. Finally, fold the parachute in half again and pack in the parachute protector (Figure 92).

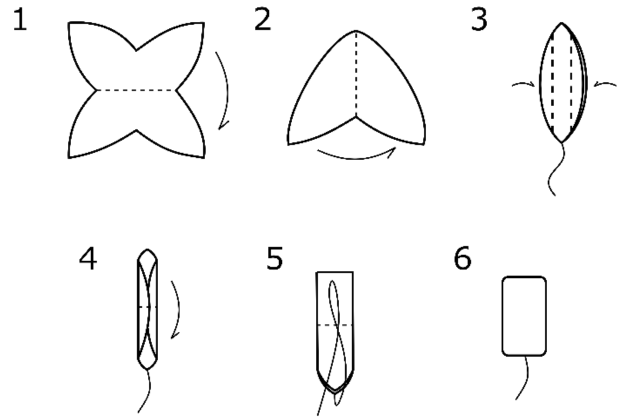

Figure 92: Method 2 Folding Steps

Method 3: This method is based on suggestions from Queensland Rocketry Society. Open the parachute completely and lay it flat. Fold the parachute in half, lining up the shroud lines. Fold the left and right sides downward toward the center, lining up the shroud lines once again. At this point, the parachute should be triangular with all the shroud lines lined up and extending from the bottom center of the triangle. Zigzag the lines and lay them on the parachute before tri-folding the long edges of the triangle over the shroud lines. Finish with a z-fold and wrap the parachute in a parachute protector (Figure 93).

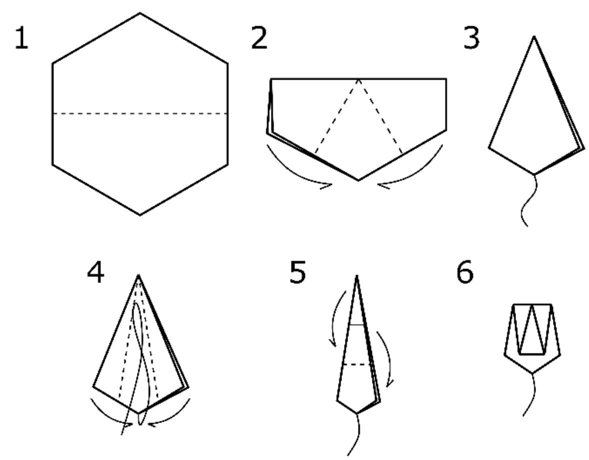

Figure 93: Method 3 Folding Steps

After preparation, the parachute was placed into an airframe section to ensure that the folding method resulted in packed dimensions that fit within diameter of the launch vehicle (a 3 in diameter section was used for the subscale parachutes, and a 4 in diameter section was used for the full-scale parachutes). The length of the packed parachute was then measured. The lengths have been tabulated below (Table 65). Note that there was no most efficient packing method identified; instead, different packing methods seem to be better for different parachute geometries. Therefore, packing method 1 will be further studied for use with the subscale parachutes, and packing method 2 will be explored for use in the full-scale launch vehicle.

Length of Packed Parachute (in)			
Packing Method	**1**	**2**	**3**
Subscale Drogue	2.13	5.50	4.75
Subscale Main	8.88	10.75	10.31
Full-scale Drogue	7.00	5.90	7.25
Full-scale Main	11.63	9.25	14.25

Table 65: Lengths of Packed Parachutes

6.1.3.10 Parachute Deployment Time Test

Test ID: LV-R-4

The time for the parachute to deploy is important to slow the launch vehicle at the appropriate time. Ejecting the parachute at the correct altitude/time means little if the parachute does not open quickly after ejection. Since the altimeter will be set to eject the main parachute at 100 ft above the lower limit, 'quickly' has been defined to be the time for an object to fall 100 ft (disregarding air resistance), calculated to be 2.49 s. To determine which packing methods would accomplish this goal, the time for deployment for the three packing methods was measured.

Independent variable: Folding method.

Dependent variable: Time for parachute to open.

Success criteria: The parachute must open in under 2.49 seconds (determined from the time for an object in free-fall to descend 100 ft, disregarding air resistance).

Materials:

- Parachute
- Parachute protector
- Weight
- Airframe section

A weight was attached to the parachute, and then the parachute was folded and packed into the section of airframe with the parachute protector. The parachute was then released from a drop location and recorded as it descended. The time required for the parachute to open and fill with air was determined from the video. This process was then repeated for three trials, and for each folding method. Refer to the description of test LV-R-3, Parachute Preparation Test, for a description of these folding methods. The results, which have been tabulated, demonstrate that all three folding methods meet the success criteria (Table 66).

Folding Method 1	
Trial	**Time to Deploy (s)**
1	1.933
2	1.792
3	2.450
Average	**2.058**

Folding Method 2	
Trial	**Time to Deploy (s)**
1	1.175
2	1.546
3	1.879
Average	**1.533**
Folding Method 3	
Trial	**Time to Deploy (s)**
1	1.679
2	2.088
3	2.363
Average	**2.043**

Table 66: Deployment Times for Folding Methods

6.2 Payload Testing

6.2.1 Testing Rationale and Resultant Effects

The following section describes the testing plan that will prove the integrity of the payload design. It includes the objective and rationale for each test and predicts how the results of each test may influence design choices (Table 67).

Test ID	Title	Objective	Rationale	Resultant Effects
P-CF-1	Raspberry Pi Functionality Demonstration	Run software package on Raspberry Pi.	Ensure Raspberry Pi can run necessary commands. Identify any operating system or hardware incompatibilities.	If payload software is unable to run on Raspberry Pi, issues will be investigated and fixed.
P-CF-2	Camera Functionality Demonstration	Capture and save photograph using Raspberry Pi and camera.	Ensure camera is responsive to commands.	Issues with camera responsiveness will be investigated and the software modified.
P-CF-3	Radio Reception Test	Receive and record an APRS transmission with the SDR dongle and the Raspberry Pi. Determine the effective range.	Ensure payload can successfully receive transmissions over the necessary range.	If reception is unsuccessful, cause will be investigated. Repositioning the antenna or reconfiguring the software may be necessary.

P-CF-4	Image Manipulation Software Demonstration	Perform prescribed image manipulations on a sample image.	Ensure payload software can execute required image manipulation commands.	If modifications are not successful, the cause of the problem will be identified, and the software will be updated.
P-CF-5	Latch Release Demonstration	Direct solenoid to release camera latch using software.	Ensure solenoid is responsive to software commands and has sufficient power to release latch.	If the solenoid does not possess sufficient strength to release the latch, a different solenoid will be selected, or the design of the latch will be adjusted.
P-CF-6	Camera Rotation Demonstration	Rotate camera through 360 degrees.	Ensure that camera can be rotated through software commands.	If the camera is incapable of rotating through the full range of motion, the design may be adjusted, or software updated.
P-CF-7	Camera Deployment Demonstration	Deploy the correct (vertically oriented) camera assembly.	Ensure that camera can be successfully deployed from the payload housing.	If the wrong camera is deployed, the IMU will be investigated for errors, and may be replaced.
P-CF-8	IMU Accuracy Test	Determine accuracy of IMU (inertial measurement unit).	Ensure IMU yields accurate readings for use by payload systems.	If the IMU is unable to accurately report data, a different IMU will be selected.
P-CF-9	Camera FOV Inspection	Determine the Field of View (FOV) of the camera.	Verify that the field of view of the camera is between 100 deg and 180 deg.	If the camera has an FOV outside the delineated range, a new camera will be selected.
P-SI-1	Raspberry Pi Camera Integration Demonstration	Detect cameras from Raspberry Pi once installed.	Ensure that Raspberry Pi can interface with the cameras.	If the Raspberry Pi is unable to communicate with the cameras, the software will be updated to fix issues.
P-SI-2	Raspberry Pi Motor Integration Demonstration	Detect motors from Raspberry Pi once installed.	Ensure that Raspberry Pi can successfully control payload hardware.	If the Raspberry Pi is unable to control the motors, the physical connections will be inspected, and the software tested for issues.

P-SI-3	Raspberry Pi Radio Integration Demonstration	Detect SDR dongle from Raspberry Pi once installed.	Ensure that radio can receive transmissions and relay them to the Raspberry Pi.	If the SDR dongle does not work, the software used will be inspected for issues and any issues found will be corrected.
P-SI-4	Radio Integration Demonstration	Raise camera, capture and manipulate images using APRS commands.	Ensure that radio does not interfere with payload function.	If the radio interferes with electronic function, the electronics will be shielded against interference. If the payload still fails to function, software and hardware components will be inspected for issues.
P-SI-5	Control Circuit Resistance Inspection	Determine resistance of solenoid control circuit.	Ensure motors are supplied with proper voltage for operation.	If the resistance of the control circuit is too high for proper motor function, a different control circuit will be used, or the supplied voltage will be increased.
P-D-1	Ejection Demonstration	Verify sealing of payload bay during ejection events.	Ensure payload is undamaged by ejection events.	If payload bulkhead does not shield payload from ejection events, the sealing will be inspected, and the bulkhead may be replaced or modified.
P-D-2	Electronics Vibration Test	Determine ability of payload to operate after exposure to vibrations.	Ensure that payload will not be damaged during flight.	If the electronics do not function after exposure to vibration, the mounting method will be made more rigid, or vibration-damping materials will be incorporated.
P-D-3	Payload Housing Compressive Strength Test	Measure compressive strength of payload housing.	Ensure that payload assembly can withstand forces present during flight.	If payload housing is found to have insufficient compressive strength, a new material will be selected or the design will be modified to increase strength.

P-D-4	Payload Acceleration Resilience Demonstration	Determine ability of payload to operate after exposure to significant acceleration.	Ensure that payload will retain functionality after flight.	If payload does not function following exposure to acceleration, the electronic attachment points and mounting system will be reinforced.
P-D-5	Payload Impact Resistance Demonstration	Measure resistance to impact of payload.	Ensure that payload can withstand forces present during recovery of the launch vehicle.	If payload does not function following impact, the electronic attachment points and mounting system will be reinforced.
P-D-6	Camera Housing Impact Resistance Demonstration	Measure resistance to impact of camera housings.	Ensure that the method of attaching the camera housings to the airframe is strong enough to resist landing forces.	If camera housing sustains damage during test, attachment points will be reinforced to improve strength.
P-SF-1	Payload Battery Life Test	Measure battery life of payload.	Ensure payload can operate on standby for two-hour minimum requirement.	If the capacity of the batteries used in the payload is found to be insufficient, a larger battery will be selected.
P-SF-2	Overheating Test	Measure temperature of payload during operation.	Ensure payload will not overheat when subject to launch day conditions.	If the payload is found to overheat during normal operation, heatsinks will be incorporated into the payload design.
P-SF-3	Payload Performance Test	Measure time for payload to complete commands.	Ensure payload can accomplish all goals in the necessary time frame.	If the payload is unable to complete tasks in the required time span, the software will be optimized for greater performance.
P-SF-4	Camera Angle Inspection	Measure the angle from vertical at which the camera rests after landing.	Ensure camera is oriented nearly straight upright.	If the camera angle is found to be too great, the design will be adjusted to level the camera.
P-SF-5	Power Loss Recovery Demonstration	Observe behavior of payload when battery is briefly disconnected.	Ensure payload can recover from temporary power loss.	If the payload is unable to recover from power loss, the cause of failure will be investigated and corrected.

P-SF-6	Landing Detection Demonstration	Determine whether payload is capable of detecting landing.	Ensure payload can determine when to commence mission processes.	If payload is unable to detect landing, the software will be tested for issues and updated, or a new IMU will be installed.
P-L-1	Payload Demonstration Launch	Demonstrate successful function of all payload objectives.	Ensure payload is able to operate after flight.	Issues during payload demonstration flight will be identified and corrected as needed.

Table 67: Payload Testing Plan: Objectives, Rationales, and Resultant Effects

6.2.2 Testing Plan and Description

The following section details the measured variable and success criteria for each test, along with an overview of the methodology (Table 68).

Test ID	Title	Variable	Success Criteria	Overview of Procedure	Required Materials
P-CF-1	Raspberry Pi Functionality Demonstration	Functionality of program.	Software runs with full functionality and no error messages.	Connect keyboard, mouse, and monitor to Raspberry Pi and confirm that it is fully functional.	Raspberry Pi, laptop
P-CF-2	Camera Functionality Demonstration	Responsiveness of cameras to software commands.	Cameras perform all commands as instructed.	Instruct camera to take a photo.	Camera, Raspberry Pi, software
P-CF-3	Radio Reception Test	Functionality of radio.	Radio receives APRS transmission and is correctly decoded by the Raspberry Pi.	Transmit radio instructions to payload receiver and determine if instructions were received.	Radio, Raspberry Pi
P-CF-4	Image Manipulation Software Demonstration	Manipulation of images.	Each manipulation function performs as expected.	Manipulate images in required ways (rotate, apply filters, timestamp) using software.	Payload software

P-CF-5	Latch Release Demonstration	Strength of solenoid.	Solenoid is successfully able to release latch.	Instruct solenoid to release the latch securing the camera assembly inside the payload.	Latch assembly, Raspberry Pi, software
P-CF-6	Camera Rotation Demonstration	Angle of possible rotation.	Camera is able to rotate a minimum of 360 degrees.	Instruct motor to rotate camera around z-axis through software.	Camera assembly, Raspberry Pi, software
P-CF-7	Camera Deployment Demonstration	Functionality of camera subsystem.	Correct camera is unlatched and extended vertically.	Initiate camera deployment through software commands.	Payload assembly
P-CF-8	IMU Accuracy Test	Percent error of IMU readings.	IMU reports readings within 15% of actual, after drift is corrected.	Record IMU readings for several acceleration scenarios and compare to correct values.	IMU
P-CF-9	Camera FOV Inspection	Camera FOV.	Camera FOV is between 100 deg and 180 deg.	Determine left and right limits of camera view. Construct lines leading from these points to the location of the camera sensor. Measure the angle between these lines.	Twine, protractor, camera.
P-SI-1	Raspberry Pi Camera Integration Demonstration	Functionality of cameras when connected to Raspberry Pi.	Raspberry Pi detects cameras.	Perform multiple commands on Raspberry Pi to take photos using cameras.	Cameras, Raspberry Pi
P-SI-2	Raspberry Pi Motor Integration Demonstration	Functionality of motors when connected to Raspberry Pi.	Raspberry Pi detects motors.	Perform multiple commands on Raspberry Pi to manipulate all motors contained in payload.	Motors, Raspberry Pi

P-SI-3	Raspberry Pi Radio Integration Demonstration	Functionality of radio when connected to Raspberry Pi.	Raspberry Pi detects radio and is able to receive data.	Transmit signal to radio and verify that Raspberry Pi received the signal.	Radio, Raspberry Pi
P-SI-4	Radio Integration Demonstration	Functionality of motors and cameras when operated with radio commands.	Radio is able to receive APRS signals and perform payload functions, including rotating cameras and taking photographs.	Perform payload functions while radio signal is being transmitted.	Payload, radio transmitter
P-SI-5	Control Circuit Resistance Inspection	Control circuit resistance.	Control circuit resistance is determined.	Test circuit voltage and current with and without the control circuit and compare the equivalent resistances of each circuit configuration.	Battery, 10 kΩ resistor, wires, breadboard, control circuit, multimeter.
P-D-1	Ejection Demonstration	Pressure of payload compartment.	No pressure change detected in payload bay.	Assemble rocket in launch configuration with payload and ignite ejection charge.	Launch vehicle, payload assembly
P-D-2	Electronics Vibration Test	Functionality of payload.	Payload functionality is unchanged after exposure to vibration.	Subject payload to vibrations similar to those experienced during launch and verify that payload is undamaged.	Payload assembly
P-D-3	Payload Housing Compressive Strength Test	Ultimate compressive strength of payload airframe.	Payload housing is able to withstand forces similar to those expected during launch and recovery.	Use Instron UTM to experimentally determine maximum compressive strength of payload housing.	Instron UTM, Airframe section altered to simulate payload assembly

P-D-4	Payload Acceleration Resilience Demonstration	Functionality of payload.	Payload functionality is unchanged after exposure to acceleration.	Subject payload to G-forces similar to those experienced during launch and verify that payload still functions.	Payload assembly, rope
P-D-5	Payload Impact Resistance Demonstration	Impact resistance of payload assembly.	No visible damage is present after impact.	Drop payload section from height to simulate recovery conditions.	Payload assembly, parachute
P-D-6	Camera Housing Impact Resistance Demonstration	Impact resistance of camera housing.	No visible damage is present after impact.	Drop camera housing installed in sample airframe section from height to simulate recovery conditions.	Camera housing prototype, payload airframe sample
P-SF-1	Payload Battery Life Test	Voltage of payload battery.	Voltage of battery does not drop below minimum threshold during a three-hour period.	Operate payload on standby until battery depletion occurs.	Payload assembly
P-SF-2	Overheating Test	Temperature of payload during operation.	Temperature does not exceed 85C during three-hour testing period.	Operate payload for a minimum of two hours outside while monitoring internal temperature.	Payload assembly
P-SF-3	Payload Performance Test	Time to complete tasks.	Payload is able to accomplish tasks in under two minutes after reception of radio command.	Command payload to execute several commands, and time how long it takes to accomplish commands.	Payload assembly, software, radio
P-SF-4	Camera Angle Inspection	Angle from vertical of camera upon deployment.	Camera is oriented less than 10 degrees from vertical.	Simulate recovery scenarios and determine the angle at which the camera rests following landing.	Payload assembly, launch vehicle aft section.

P-SF-5	Power Loss Recovery Demonstration	Functionality of payload.	Payload is able to recovery functionality in less than two minutes after power is reconnected.	Temporarily disconnect power from the payload. Verify that once power is reconnected, payload is able to restart.	Payload assembly
P-SF-6	Landing Detection Demonstration	Functionality of IMU.	Payload is able to detect landing.	Simulate landing conditions and determine whether payload detected landing.	Payload assembly
P-L-1	Payload Demonstration Launch	Functionality of payload.	Payload demonstrates full functionality.	Launch payload in final configuration in full-scale launch vehicle.	Launch vehicle, payload

Table 68: Payload Testing Plan: Variables, Success Criteria, Methods, and Materials

6.2.3 Completed Payload Testing

The following table tracks the completion status of each test; the results of completed tests are described in this section (Table 69).

Test ID	Title	Completion Status
P-CF-1	Raspberry Pi Functionality Demonstration	Incomplete
P-CF-2	Camera Functionality Demonstration	Incomplete
P-CF-3	Radio Reception Test	Incomplete
P-CF-4	Image Manipulation Software Demonstration	Incomplete
P-CF-5	Latch Release Demonstration	Incomplete
P-CF-6	Camera Rotation Demonstration	Incomplete
P-CF-7	Camera Deployment Demonstration	Incomplete
P-CF-8	IMU Accuracy Test	Incomplete
P-CF-9	Camera FOV Inspection	Incomplete
P-SI-1	Raspberry Pi Camera Integration Demonstration	Incomplete
P-SI-2	Raspberry Pi Motor Integration Demonstration	Incomplete
P-SI-3	Raspberry Pi Radio Integration Demonstration	Incomplete
P-SI-4	Radio Integration Demonstration	Incomplete
P-SI-5	Control Circuit Resistance Inspection	Complete
P-D-1	Ejection Demonstration	Incomplete
P-D-2	Electronics Vibration Test	Incomplete
P-D-3	Payload Housing Compressive Strength Test	Incomplete
P-D-4	Payload Acceleration Resilience Demonstration	Incomplete
P-D-5	Payload Impact Resistance Demonstration	Incomplete
P-D-6	Camera Housing Impact Resistance Demonstration	Complete

P-SF-1	Payload Battery Life Test	Incomplete
P-SF-2	Overheating Test	Incomplete
P-SF-3	Payload Performance Inspection	Incomplete
P-SF-4	Camera Angle Inspection	Incomplete
P-SF-5	Power Loss Recovery Demonstration	Incomplete
P-SF-6	Landing Detection Demonstration	Incomplete
P-L-1	Payload Demonstration Launch	Incomplete

Table 69: Completed Payload Testing

6.2.3.1 Control Circuit Resistance Inspection

Test ID: P-SI-5

A control circuit is used to actuate the solenoids used in the latch system of the camera retention system by way of software on the Raspberry Pi. Otherwise, the only way to control the solenoid is physically connecting and disconnecting the solenoid from the battery. It is critical that the addition of this control circuit does not significantly affect the voltage and current supplied to the solenoid, or the solenoid may not function properly. This test determines the effect of the control circuit on the equivalent resistance of the solenoid motor circuit.

Independent variable: Control method.

Dependent variable: Change in equivalent resistance.

Success criteria: Change in resistance does not negatively affect solenoid function.

Materials:

- Battery
- 10 kΩ resistor
- Wires
- Breadboard
- Control circuit
- Multimeter

The motor circuit was configured with and without the control circuit (Figure 94). The corresponding circuit diagrams are also included (Figure 95).

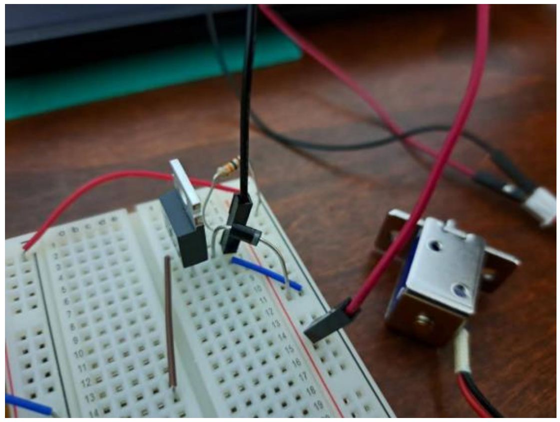

Figure 94: Control Circuit Installed on Breadboard

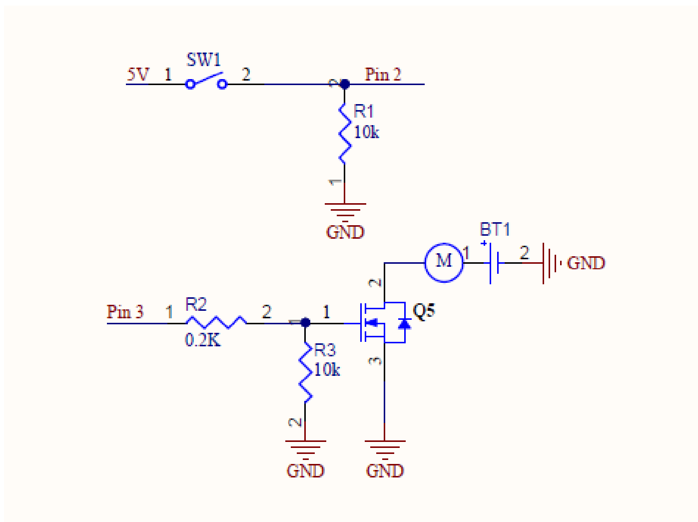

Figure 95: Circuit Diagrams with and without Control Circuit

The total voltage was then measured, as was the current. From these two values, the equivalent resistance of the circuit was calculated, and compared (Equation 15). The data has been tabulated; the results show that the control circuit has a minimal resistance of 1.49 Ω and is therefore unlikely to negatively affect the solenoid's function (Table 70).

$$R = V/I$$

Equation 15: Ohm's Law

	With Control Circuit	Without Control Circuit	Difference
Voltage (V)	8.28	8.26	
Current (A)	0.33	0.35	
Resistance (Ω)	25.09	23.60	1.49

Table 70: Control Circuit Resistance Inspection Results

6.2.3.2 Camera Housing Impact Resistance Test

Test ID: P-D-6

The camera housings are secured to the payload airframe using screws and threaded inserts. These attachments present a potential failure point, particularly during landing when the payload undergoes an intense impact. As a result, the design of the payload was impact tested.

Independent variable: Camera housing design.

Dependent variable: Resistance to impact.

Success criteria: No visible damage present following the test.

Materials:

- Camera housing prototype
- Sample of airframe material with slot

The camera housing was installed in the airframe sample, simulating the configuration in the launch vehicle. The assembly was then dropped from a height of approximately 25 ft and inspected for damage. This process was repeated for three trials. On the first two trials, no damage was detected; however, on the third, the camera housing failed. The forward and aft wall sheared, and the forward threaded insert was torn from the housing (Figure 96). In response to this failure, the 3D printed forward and aft walls were designed to be thicker to prevent shear failures.

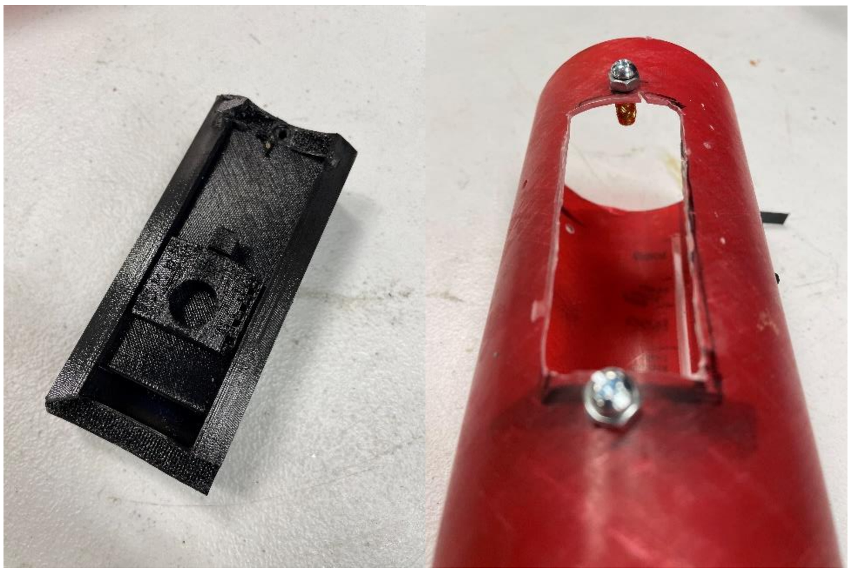

Figure 96: Camera Housing Failure

6.3 Completed Subscale Testing

This section details the testing performed specifically for the subscale launch vehicle. The following tests will have to be repeated for the full-scale launch vehicle since the results may change due to differences between the vehicles.

6.3.1 Subscale Center of Gravity Inspection

Test ID: LV-SP-3

While simulations were used to obtain an estimate of the location of the center of gravity of the launch vehicle, this location typically differs in the as-built launch vehicle. Therefore, this test is conducted to improve the results of simulations that rely on the location of the center of gravity. If the center of gravity diverges significantly from simulated estimates, the vehicle may need to be ballasted.

Independent variable: Launch vehicle ballast.

Dependent variable: Center of gravity.

Success criteria: Center of gravity is determined and does not diverge significantly from simulations.

Materials:

- Launch vehicle in launch configuration
- Rope

The fully prepared launch vehicle was wrapped around the center using the rope. The launch vehicle was then suspended from the rope. If the launch vehicle tended to fall forward, the rope would be moved forward, and conversely if the vehicle fell backward. The location of the rope was adjusted until the launch vehicle balanced, at which point this location was marked and measured. The center of gravity for the subscale launch vehicle was found to be 56.1 in from the aft, slightly forward from the simulated estimate of 52.5 in.

6.3.2 Subscale Parachute Ejection Demonstration

Test ID: LV-R-5

Ejection events are critical during the recovery stage of flight. Failure for the launch vehicle to separate or for the parachute to eject are likely to result in damage to the launch vehicle as it impacts the ground with excessive kinetic energy. Thus, ejection demonstrations are performed to determine the size of black powder charge necessary for ejection events.

Independent variable: Ejection charge masses.

Dependent variable: Separation.

Success criteria: Airframe separates, and parachute is ejected.

Materials:

- Launch vehicle
- Parachutes and parachute protectors
- Recovery harness
- Insulation material
- E-matches
- 3FFF Black Powder

- Wires
- Altimeters
- Laptop
- Test stand

Initial charge sizes were calculated using the ideal gas law and the size of the airframe. The charges were then prepared by loading the requisite amount of black powder into the e-matches and securing with insulation material and masking tape. The ejection charges were tested beginning with the primary charges, which were installed in the terminal blocks of the avionics bay. Extension wires were fixed to the opposite terminal and fed out of the avionics bay to a safe distance away. Here, the wires could be connected to the altimeters.

The parachutes, recovery harnesses, and insulation material were then installed in the launch vehicle as it would be for the launch. The launch vehicle was also ballasted and configured as it would be on launch day. At this point, the launch vehicle was placed on the test stands and connected to the altimeters. After clearing the area to ensure the safety of those involved in the test, the charges were activated using the altimeter software on the laptop. The drogue was ejected first, followed by the main in order to simulate the actual series of ejection events.

In the first trial, the central and payload airframe containing the drogue parachute successfully separated, but the parachute did not eject (Figure 97).

Figure 97: Failed Drogue Ejection

However, the main parachute did successfully eject, with separation of the nosecone and forward airframe (Figure 98).

Figure 98: Successful Main Ejection

The size of the primary drogue ejection charge was increased in increments of 0.15 g until a successful separation and ejection occurred (Figure 99).

Figure 99: Successful Drogue Ejection

Secondary charges were tested as well, and all were successful. The final ejection charge sizes have been tabulated (Table 71).

Final Ejection Charge Masses	
Parachute	**Mass (g)**
Drogue - Primary	1.15
Main - Primary	1.39
Drogue - Secondary	1.45
Main - Secondary	1.74

Table 71: Ejection Charge Masses

6.3.3 Subscale Parachute Unfolding Demonstration

Test ID: LV-R-6

Parachutes must be packed with a method that allows the parachute to unfold without tangling when ejected from the launch vehicle. This includes both the shroud lines that are a part of the parachute, and the recovery harness, which is packed with the parachute.

Independent variable: Folding method.

Dependent variable: Tangled shroud lines and recovery harnesses.

Success criteria: Parachute unfolds without tangling.

Materials:

- Parachutes
- Parachute protectors
- Recovery harness

The parachutes, parachute protectors, and recovery harnesses were folded outside of the launch vehicle using the same method that would be employed for launch. One team member held the folded parachute while two others pulled on each end of the recovery harness, simulating a separation event (Figure 100). The parachute was then inspected for tangling. This process was repeated for both the drogue and the main parachute. In both instances, no tangling was identified, and the test was deemed successful.

Figure 100: Parachute Unfolding Demonstration Set-Up

6.3.4 Subscale Demonstration Launch

Test ID: LV-L-2

The subscale demonstration launch serves as an integration test, incorporating multiple subsystems operating simultaneously. It also presents an opportunity to test the design concept for the payload system.

Independent variable: Launch vehicle.

Dependent variable: Subscale launch vehicle flight performance.

Success criteria: Launch vehicle will successfully launch and be recovered with no damage.

Materials:

- Fully assembled subscale launch vehicle
- Launch rail
- Remote igniter

The launch vehicle was prepared by installing the avionics bay, the ejection charges, and parachutes. The team opted to install prototype camera housings, but not include the payload electronics or motors. Instead, the payload bay was ballasted to simulate the weight of the actual payload. The motor was then installed, and the launch vehicle was mounted to the launch rail. Before launch, the altimeters were armed.

The launch was conducted successfully, achieving an apogee of 3,557 ft, and the vehicle was successfully recovered. Upon inspection, no damage was found on the launch vehicle, marking the flight as successful (Figure 101, Figure 102). Additionally, the payload concept was shown to be effective—one of the camera housings landed vertically (Figure 103). Some tangling did occur with the drogue parachute (Figure 104). It is suspected that the issue is the folding method used, so a new folding technique will be explored.

Figure 101: Central Airframe Immediately Following Landing

Figure 102: Main Parachute Immediately Following Landing

Figure 103: Aft Section (Containing Payload) Immediately Following Landing

Figure 104: Drogue Parachute Immediately Following Landing

6.4 Requirements Compliance

6.4.1 Competition Requirements

6.4.1.1 General Requirements

The team has detailed a compliance plan based on general requirements (Table 72).

Requirement	Verification Type	Compliance Plan	Compliance Status
1.1. Students on the team will do 100% of the project, including design, construction, written reports, presentations, and flight preparation except for assembling the motors and handling black powder or any variant of ejection charges, or preparing and installing electric matches (to be done by the team's mentor). Teams will submit new work. Excessive use of past work will merit penalties.	Inspection	The design, construction, and flight preparations for the entirety of the launch vehicle and payload will be completed by students on the team. All reports and presentations will be completed by the students. New work will be submitted and will not be plagiarized or recycled.	Partial verification; all work so far has solely been completed by students. This includes the design, reports, presentations, construction of the subscale launch vehicle, and flight preparation aside from motor and black powder preparation.
1.2. The team will provide and maintain a project plan to include, but not limited to the following items: project milestones, budget and community support, checklists, personnel assignments, STEM engagement events, and risks and mitigations.	Inspection	The team will create a project plan that includes an outline of project milestones, a budget and funding plan, community support, checklists, personnel assignments, STEM engagement events, risks, and mitigations. This document will be updated as needed and included in each new report.	Verified; The project plan is included in section 6.5 of this report.
1.3. The team shall identify all team members who plan to attend Launch Week activities by the Critical Design Review (CDR). Team members will include: 1.3.1. Students actively engaged in the project throughout the entire year.	Inspection	All team members who attend Launch Week activities will be members who have actively engaged in the project throughout the entire year, a selected mentor, and up to two adult educators. A list of these members will be created prior to CDR.	Verified; The list of members attending Launch Week activities is included in section 7.1 of this report.

1.3.2. One mentor (see requirement 1.13). 1.3.3. No more than two adult educators.			
1.4. Teams shall engage a minimum of 250 participants in Educational Direct Engagement STEM activities in order to be eligible for STEM Engagement scoring and awards. These activities can be conducted in-person or virtually. To satisfy this requirement, all events shall occur between project acceptance and the FRR due date. A template of the STEM Engagement Activity Report can be found on pages 39–42.	Inspection	The team will engage with at least 250 participants using direct educational engagement STEM activities. These activities will either occur in person or online between project acceptance and the FRR due date. This requirement will be documented using the STEM Engagement Activity Report. It will be satisfied through partnerships with Alachua County Public Schools and Hands on Gainesville that provide the team the opportunity to engage with local elementary and middle school students.	Partially verified; the team has engaged with 250 out of 250 participants though direct educational engagements. The team will submit the STEM Engagement Activity reports for these events by the FRR deadline.
1.5. The team will establish and maintain a social media presence to inform the public about team activities.	Inspection	The team has established an active social media presence and will continue to maintain it to inform the public about team activities on Instagram (@SwampLaunch), Twitter(@SwampLaunch), Facebook (@Swamp Launch Rocket Team), and LinkedIn (@Swamp Launch Rocket Team).	Verified; the team has an active social media presence on various platforms.

1.6. Teams will email all deliverables to the NASA project management team by the deadline specified in the handbook for each milestone. In the event that a deliverable is too large to attach to an email, inclusion of a link to download the file will be sufficient. Late submissions of PDR, CDR, FRR milestone documents shall be accepted up to 72 hours after the submission deadline. Late submissions shall incur an overall penalty. No PDR, CDR, FRR milestone documents shall be accepted beyond the 72-hour window. Teams that fail to submit the PDR, CDR, FRR milestone documents shall be eliminated from the project.	Inspection	The team will email all deliverables by the deadlines written in the handbook. Should the deliverable be too larger to an email, the team will include a link to download a file. In the event that the team's submissions are late, the team will only have 72 hours after the deadline to send the deliverables with an overall penalty. No deliverables will be accepted past the 72-hour window.	Verified; the team's Project Manager, Erik Dearmin, has emailed all deliverables by the deadlines in the handbook, and will continue to do so.
1.7 Teams who do not satisfactorily complete each milestone review (PDR, CDR, FRR) shall be provided action items needed to be completed following their review and shall be required to address action items in a delta review session. After the delta session the NASA management panel shall meet to determine the teams' status in the program and the team shall be notified shortly thereafter.	Inspection	If the team does not complete each milestone to a satisfactory degree, the team will be provided action items to complete after the review and must address the action items in a delta review session. NASA management will determine the team's status in the program after the delta session and notify the team.	Partial verification; the team has satisfactorily completed the milestone reviews thus far but will complete any future action items should any arise from future milestone reviews.

1.8. All deliverables shall be in PDF format.	Inspection	All deliverables will be submitted in PDF format.	Partial verification; All deliverables thus far have been submitted as PDFs. All future deliverables will also be submitted as PDFs.
1.9. In every report, teams will provide a table of contents including major sections and their respective sub-sections.	Inspection	The team will provide a table of contents in each report that includes all major sections and sub-sections.	Partial verification; PDR included a table of contents of all major sections and subsections. Future reports will continue to do so.
1.10. In every report, the team will include the page number at the bottom of the page.	Inspection	Page numbers will be included at the bottom of each page in every report.	Partial verification; PDR included page numbers in the bottom right-hand corner of each page. Future reports will continue to do so.
1.11. The team will provide any computer equipment necessary to perform a video teleconference with the review panel. This includes, but is not limited to, a computer system, video camera, speaker telephone, and a sufficient Internet connection. Cellular phones should be used for speakerphone capability only as a last resort.	Inspection	The team will provide all equipment for teleconference review panels, including video camera, microphone, computer, and internet connection.	Partial verification; the team provided all video equipment for the PDR milestone and will continue to do so for future milestone reviews.
1.12. All teams attending Launch Week will be required to use the launch pads provided by Student Launch's launch services provider. No custom pads will be permitted at the NASA Launch Complex. At launch, 8-foot 1010 rails	Inspection and Demonstration	The team will use the provided launch pads and 8-foot 1010 rails, or the 12-foot 1515 rails canted 5 to 10 degrees away from the crowd.	Unverified; the full-scale launch vehicle has not launched at this time. However, the team plans on using the 12 ft 1515 launch rail canted 5 degrees away from the crowd.

and 12-foot 1515 rails will be provided. The launch rails will be canted 5 to 10 degrees away from the crowd on Launch Day. The exact cant will depend on Launch Day wind conditions.			
1.13. Each team shall identify a "mentor." A mentor is defined as an adult who is included as a team member, who will be supporting the team (or multiple teams) throughout the project year, and may or may not be affiliated with the school, institution, or organization. The mentor shall maintain a current certification, and be in good standing, through the National Association of Rocketry (NAR) or Tripoli Rocketry Association (TRA) for the motor impulse of the launch vehicle and must have flown and successfully recovered (using electronic, staged recovery) a minimum of 2 flights in this or a higher impulse class, prior to PDR. The mentor is designated as the individual owner of the rocket for liability purposes and must travel with the team to Launch Week. One travel stipend will be provided per mentor regardless of the number of teams he or she supports. The stipend will only be provided if	Inspection	The team's mentor is Jimmy Yawn, a Level 3 certified NAR member. The team's mentor maintains a current certification, is in good standing with the NAR, and has flown and recovered the required number of flights. Mr. Yawn is the designated owner of the rocket and will be traveling with the team to Launch Week.	Verified; the team has identified Jimmy Yawn as the team's mentor. The team has maintained consistent contact with Mr. Yawn throughout the competition so far and will continue to do so.

the team passes FRR and the team and mentor attend Launch Week in April.			
1.14. Teams will track and report the number of hours spent working on each milestone.	Inspection	The team will record all progress and the number of hours spent on each milestone will be reported.	Verified; the number of hours spent working is included in section 1.3 of the reports so far and will continue to be provided in future milestone review reports.

Table 72: General Requirements

6.4.1.2 Vehicle Requirements

The team has detailed a compliance plan based on vehicle requirements (Table 73).

Requirement	**Verification Type**	**Compliance Plan**	**Compliance Status**
2.1. The vehicle will deliver the payload to an apogee altitude between 4,000 and 6,000 feet above ground level (AGL). Teams flying below 3,500 feet or above 6,500 feet on their competition launch will receive zero altitude points towards their overall project score and will not be eligible for the Altitude Award.	Analysis	An OpenRocket simulation will be used to estimate the apogee of the launch vehicle, and design adjustments will be made to ensure this estimate is between 4,000 and 6,000 ft.	Verified; the estimated apogee is 4780 ft.
2.2. Teams shall declare their target altitude goal at the PDR milestone. The declared target	Analysis	A target altitude goal was declared at the PDR milestone, based	Verified; the submitted target apogee is 4600 ft.

altitude will be used to determine the team's altitude score.		on an OpenRocket simulation.	
2.3. The launch vehicle will be designed to be recoverable and reusable. Reusable is defined as being able to launch again on the same day without repairs or modifications.	Demonstration	The launch vehicle will be designed such that it will not be damaged during launch or recovery, therefore being reusable.	Partially verified; test LV-L-2, Subscale Demonstration Launch, was successful, and the vehicle was recoverable and reusable. Test LV-L-3, Vehicle Demonstration Launch, will be used to confirm that the full-scale launch vehicle is reusable.
2.4. The launch vehicle will have a maximum of four (4) independent sections. An independent section is defined as a section that is either tethered to the main vehicle or is recovered separately from the main vehicle using its own parachute. 2.4.1. Coupler/airframe shoulders which are located at in-flight separation points will be at least 2 airframe diameters in length. (One body diameter of surface contact with each airframe section). 2.4.2. Nosecone shoulders which are located at in-flight separation points will be at least ½ body diameter in length.	Inspection	The launch vehicle will be designed with fewer than four independent sections, and with couplers that are a minimum of two airframe diameters in length. The nosecone shoulder will not be shorter than a half body diameter in length.	Verified; the design includes three independent sections, and all couplers at separation points are a minimum of two airframe diameters in length. The nosecone shoulder is a minimum of 1/2 airframe diameter in length.
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.	Demonstration	The team will perform a rehearsal assembly to practice assembling the launch vehicle. This rehearsal is a part of Test LV-L-1, Launch Rehearsal Demonstration.	Unverified; test LV-L-1, Launch Rehearsal Demonstration, has not been completed.

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 on-board components, although the capability to withstand longer delays is highly encouraged.	Test	Test LV-A-3, Avionics Battery Life Test, will measure the battery life of the avionics subsystem. Test P-SF-1, Payload Battery Life Test, will measure the battery life of the payload subsystems. In both cases, the test will ensure that the subsystems meet the minimum requirement of two hours, with an additional factor of safety.	Partially verified; test LV-A-3, Avionics Battery Life Test, has been completed, and shown that the avionics battery life exceeds three hours, providing a factor of safety of 1.5. Test P-SF-1, Payload Battery Life Test, has not been completed.
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.	Inspection	The launch vehicle will employ a motor compatible with a standard 12-volt direct current firing system.	Verified; the launch vehicle will employ an Aerotech L1090, which is compatible with a 12-volt firing 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).	Inspection	The launch vehicle will employ a motor that does not require external circuitry to initiate launch. Nor will the payload or avionics require external circuitry.	Verified; the launch vehicle will employ an Aerotech L1090, which does not require external circuitry. The components used in the avionics and payload subsystems do not require external circuitry either.
2.9. Each team shall use commercially available e-matches or igniters. Hand-dipped igniters shall not be permitted.	Inspection	The recovery system will be designed to employ commercially available e-matches.	Verified; the recovery system uses commercially available e-matches for ejection charges.

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). 2.10.1. Final motor choices will be declared by the Critical Design Review (CDR) milestone. 2.10.2. Any motor change after CDR shall be approved by the NASA Range Safety Officer (RSO). Changes for the sole purpose of altitude adjustment will not be approved. A penalty against the team's overall score will be incurred when a motor change is made after the CDR milestone, regardless of the reason.	Inspection and Analysis	The team will select an approved motor and declare this selection by the Critical Design Review deadline. In the event that a motor change is necessary following the CDR deadline, the approval of the NASA Range Safety Officer will be acquired before a change is made.	Verified; the launch vehicle will employ an Aerotech L1090, which is an approved and certified motor. This motor has been chosen through OpenRocket simulations.
2.11. The launch vehicle will be limited to a single motor propulsion system.	Analysis	The team will design the launch vehicle with the single-motor propulsion system as a design constraint.	Verified; the OpenRocket simulations have been designed with a single motor propulsion system.
2.12. The total impulse provided by a college or University launch vehicle will not exceed 5,120 Newton-seconds (L-class).	Analysis	The team will select a motor with a total impulse less than 5,120 N-s.	Verified; the team has selected an Aerotech L1090 motor, based on OpenRocket simulations. This motor has a total impulse of 2,671 N-s.

2.13. Pressure vessels on the vehicle will be approved by the RSO and will meet the following criteria: 2.13.1. The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) will be 4:1 with supporting design documentation included in all milestone reviews. 2.13.2. Each pressure vessel will include a pressure relief valve that sees the full pressure of the tank and is capable of withstanding the maximum pressure and flow rate of the tank. 2.13.3. The full pedigree of the tank will be described, including the application for which the tank was designed and the history of the tank. This will include the number of pressure cycles put on the tank, the dates of pressurization/depressurization, and the name of the person or entity administering each pressure event.	Inspection	The team will not include pressure vessels on the launch vehicle.	Verified; the launch vehicle does not contain any pressure vessels.
2.14. The launch vehicle will have a minimum static stability margin of 2.0 at the point of rail exit. Rail exit is defined at the point where the forward rail button loses contact with the rail.	Analysis	OpenRocket simulations will be employed to predict the minimum static stability margin at rail exit. The design of the vehicle will be adjusted so that this prediction exceeds 2.0.	Verified; the static stability margin at rail exit is predicted to be 2.13.
2.15. The launch vehicle will have a minimum thrust to weight ratio of 5.0 : 1.0.	Analysis	OpenRocket simulations will be employed to calculate the thrust to weight ratio of the launch vehicle. The design of the vehicle will be adjusted so that this	Verified; the thrust to weight ratio of the launch vehicle has been calculated to be 8.53:1.

		value exceeds a ratio of 5:1.	
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 aerodynamic effect on the rocket's stability.	Analysis	The camera housings will not significantly affect the launch vehicle's stability. Additionally, test LV-L-2, Subscale Demonstration Launch, and test LV-SP-2, Launch Vehicle Drag Analysis, will be performed to ensure there is minimal aerodynamic effect.	Partially verified; test LV-L-2, Subscale Demonstration Launch, was successfully completed. No aerodynamic issues were identified. Test LV-SP-2, Launch Vehicle Drag Analysis, has not been completed.
2.17. The launch vehicle will accelerate to a minimum velocity of 52 fps at rail exit.	Analysis	OpenRocket simulations will be employed to predict the velocity of the launch vehicle at rail exit. The design of the vehicle will be adjusted so that this value exceeds 52 ft/s.	Verified; the predicted rail exit velocity is 87 ft/s.
2.18. All teams will successfully launch and recover a subscale model of their rocket prior to CDR. Success of the subscale is at the sole discretion of the NASA review panel. The subscale flight may be conducted at any time between proposal award and the CDR submission deadline. Subscale flight data shall be reported in the CDR report and presentation at the CDR milestone. Subscales are required to use a minimum motor impulse class of E (Mid Power motor).	Demonstration	Test LV-L-2, Subscale Demonstration Launch, will be performed to demonstrate the ability of a subscale model to perform a successful flight. The model will employ a minimum motor impulse class of E.	Verified; test LV-L-2, Subscale Demonstration Launch, was determined to be successful. The model employed an Aerotech J415.

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.	Analysis	The subscale model will be designed to be physically similar to the full-scale launch vehicle. The full-scale launch vehicle will not be used; a separate launch vehicle will be designed and manufactured.	Verified; the subscale model is a 75% scale model of the full-scale launch vehicle. It has been designed to have similar flight characteristics to the full-scale design.
2.18.2. The subscale model will carry an altimeter capable of recording the model's apogee altitude.	Inspection	Two altimeters--a primary and secondary altimeter--will be installed on the subscale launch vehicle.	Verified; the subscale launch vehicle contained a Stratologger altimeter and and Entacore AIM altimeter during flight.
2.18.3. The subscale rocket shall be a newly constructed rocket, designed and built specifically for this year's project.	Inspection	The subscale launch vehicle will be designed and manufactured specifically for this year's competition.	Verified; manufacturing of the launch vehicle occurred between November 14th, 2022, and November 29th, 2022.
2.18.4. Proof of a successful flight shall be supplied in the CDR report. 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 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.	Demonstration	The team will provide altimeter data from the launch vehicle, or a video of the flight. Photographs of the launch vehicle in the landed configuration will also be included in the CDR report.	Verified; test LV-L-2, Subscale Demonstration Launch, was successful. The required information can be found in section (section) of this report.
2.18.5. The subscale rocket shall not exceed 75% of the dimensions (length and diameter) of your designed full-scale rocket. For example, if	Analysis	The subscale launch vehicle will be designed to not exceed 75% of the	Verified; the subscale launch vehicle dimensions do not exceed 75% of the

your full-scale rocket is a 4" diameter 100" length rocket your subscale shall not exceed 3" diameter and 75" in length.		dimensions of the full-scale design.	dimensions of the full-scale design.
2.19. All teams will complete demonstration flights as outlined below.	See below.	See below.	See below.
2.19.1. Vehicle Demonstration Flight—All teams will successfully launch and recover their full-scale rocket prior to FRR in its final flight configuration. The rocket flown shall be the same rocket to be flown for their competition launch. The purpose of the Vehicle Demonstration Flight is to validate the launch vehicle's stability, structural integrity, recovery systems, and the team's ability to prepare the launch vehicle for flight. A successful flight is defined as a launch in which all hardware is functioning properly (i.e. drogue chute at apogee, main chute at the intended lower altitude, functioning tracking devices, etc.). The following criteria shall be met during the full-scale demonstration flight:	Demonstration and Test	The team will perform test LV-L-3, Vehicle Demonstration Launch, in which the full-scale launch vehicle will be flown. The performance of the launch vehicle, including stability, structural integrity, and recovery systems will be monitored. Prior to launch, test LV-A-2, Altimeter Accuracy Test, and test LV-M-1, Airframe Material Compression Test, will be performed to ensure the recovery system is functional and the vehicle has sufficient structural integrity. Test LV-L-1, Launch Rehearsal Demonstration, will be performed to ensure the team can prepare the launch vehicle for flight.	Unverified; test LV-L-3, Vehicle Demonstration Launch, and test LV-L-1, Launch Rehearsal Demonstration, have not been completed.
2.19.1.1. The vehicle and recovery system will have functioned as designed.	Demonstration	The team will perform test LV-L-3, Vehicle Demonstration Launch, in which the full-scale launch vehicle will be flown. The performance of the recovery systems will be monitored.	Unverified; test LV-L-3, Vehicle Demonstration Launch, has not been completed.

2.19.1.2. The full-scale rocket shall be a newly constructed rocket, designed and built specifically for this year's project.	Inspection	The team will design and construct a launch vehicle specifically for this year's competition.	Partially verified; the launch vehicle has been designed specifically for this year's competition but has not yet been constructed.
2.19.1.3. The payload does not have to be flown during the full-scale Vehicle Demonstration Flight. The following requirements still apply: 2.19.1.3.1. If the payload is not flown, mass simulators will be used to simulate the payload mass. 2.19.1.3.2. The mass simulators will be located in the same approximate location on the rocket as the missing payload mass.	Demonstration	If the payload is not flown during the vehicle demonstration launch, the launch vehicle will be appropriately ballasted.	Unverified; test LV-L-3, Vehicle Demonstration Launch, has not been completed.
2.19.1.4. If the payload changes the external surfaces of the rocket (such as camera housings or external probes) or manages the total energy of the vehicle, those systems will be active during the full-scale Vehicle Demonstration Flight.	Demonstration	If the full payload is not flown during the vehicle demonstration launch, a partial payload that includes prototype camera housings will be installed.	Unverified; test LV-L-3, Vehicle Demonstration Launch, has not been completed.
2.19.1.5. Teams shall fly the competition launch motor for the Vehicle Demonstration Flight. The team may request a waiver for the use of an alternative motor in advance if the home launch field cannot support the full impulse of the competition launch motor or in other extenuating circumstances.	Demonstration	The team will launch the full-scale launch vehicle using the competition launch motor.	Unverified; test LV-L-3, Vehicle Demonstration Launch, has not been completed.
2.19.1.6. The vehicle shall be flown in its fully ballasted configuration during the full-scale test flight. Fully ballasted refers to the maximum amount of ballast that will be flown during the competition launch flight. Additional ballast may	Demonstration	The team will fly the launch vehicle with the maximum amount of ballast that will be flown during the competition launch flight.	Unverified; test LV-L-3, Vehicle Demonstration Launch, has not been completed.

not be added without a re-flight of the full-scale launch vehicle.			
2.19.1.7. After successfully completing the full-scale demonstration flight, the launch vehicle or any of its components will not be modified without the concurrence of the NASA Range Safety Officer (RSO).	Inspection	The team will not modify the launch vehicle following successful completion of the full-scale demonstration flight.	Unverified; test LV-L-3, Vehicle Demonstration Launch, has not been completed.
2.19.1.8. Proof of a successful flight shall be supplied in the FRR report. 2.19.1.8.1. Altimeter flight profile data output with accompanying altitude and velocity versus time plots is required to meet this requirement. Altimeter flight profile graph(s) that are not complete (liftoff through landing) shall not be accepted. 2.19.1.8.2. Quality pictures of the as landed configuration of all sections of the launch vehicle shall be included in the FRR report. This includes but not limited to nosecone, recovery system, airframe, and booster.	Demonstration	The team will provide altimeter data from the launch vehicle, or a video of the flight. Photographs of the launch vehicle in the landed configuration will also be included in the FRR report.	Unverified; test LV-L-3, Vehicle Demonstration Launch, has not been completed.
2.19.1.9. Vehicle Demonstration flights shall be completed by the FRR submission deadline. No exceptions will be made. If the Student Launch office determines that a Vehicle Demonstration Re-flight is necessary, then an extension may be granted. THIS EXTENSION IS ONLY VALID FOR RE-FLIGHTS, NOT FIRST TIME FLIGHTS. Teams completing a required re-flight shall submit	Demonstration	The vehicle demonstration flight will be completed prior to the FRR submission deadline. If a re-flight is required, an FRR Addendum will be submitted by the associated deadline.	Unverified; test LV-L-3, Vehicle Demonstration Launch, has not been completed. It is tentatively scheduled for February 18, 2023.

an FRR Addendum by the FRR Addendum deadline.			
2.19.2. Payload Demonstration Flight—All teams will successfully launch and recover their full-scale rocket containing the completed payload prior to the Payload Demonstration Flight deadline. The rocket flown shall be the same rocket to be flown as their competition launch. The purpose of the Payload Demonstration Flight is to prove the launch vehicle’s ability to safely retain the constructed payload during flight and to show that all aspects of the payload perform as designed. A successful flight is defined as a launch in which the rocket experiences stable ascent and the payload is fully retained until it is deployed (if applicable) as designed. The following criteria shall be met during the Payload Demonstration Flight:	Demonstration	The team will perform test P-L-1, Payload Demonstration Launch, to demonstrate the launch vehicle's ability to safely retain the payload and the payload's ability to accomplish mission objectives.	Unverified; test P-L-1, Payload Demonstration Launch, has not been completed.
2.19.2.1. The payload shall be fully retained until the intended point of deployment (if applicable), all retention mechanisms shall function as designed, and the retention mechanism shall not sustain damage requiring repair.	Demonstration and Test	The payload has been designed to remain within the launch vehicle for the duration of the mission. After landing, the payload will extend a camera out from the airframe. To ensure this does not occur until after landing, tests P-D-4, Payload Acceleration Resilience Test, P-CF-	Unverified; tests P-D-4, Payload Acceleration Resilience Test, P-CF-8, IMU Accuracy Test, and P-CF-5, Latch Release Test, have not been completed.

		8, IMU Accuracy Test, and P-CF-5, Latch Release Test, will assess the ability of the relevant subsystems to perform these tasks.	
2.19.2.2. The payload flown shall be the final, active version.	Demonstration	The final, active version of the payload will be installed on the launch vehicle for the Payload Demonstration Launch.	Unverified; test P-L-1, Payload Demonstration Launch, has not been completed.
2.19.2.3. If the above criteria are met during the original Vehicle Demonstration Flight, occurring prior to the FRR deadline and the information is included in the FRR package, the additional flight and FRR Addendum are not required.	Demonstration	If the payload is flown during the Vehicle Demonstration Launch, a separate Payload Demonstration Launch will not be performed.	Unverified; test P-L-1, Payload Demonstration Launch, has not been completed.
2.19.2.4. Payload Demonstration Flights shall be completed by the FRR Addendum deadline. NO EXTENSIONS WILL BE GRANTED.	Demonstration	The Payload Demonstration Launch will be performed before the FRR Addendum deadline.	Unverified; test P-L-1, Payload Demonstration Launch, has not been completed. However, it has been tentatively scheduled for March 18, 2023.
2.20. An FRR Addendum will be required for any team completing a Payload Demonstration Flight or NASA-required Vehicle Demonstration Re-flight after the submission of the FRR Report. 2.20.1. Teams required to complete a Vehicle Demonstration Re-Flight and failing to submit the FRR Addendum by the deadline will not be permitted to fly a final competition launch. 2.20.2. Teams who successfully	Demonstration	The team will submit an FRR Addendum for any Payload Demonstration Launch completed after the deadline for the FRR Report. If necessary, a Vehicle Demonstration Re-Flight will also be completed prior to the deadline for the FRR Addendum. The team understands that if either the Payload Demonstration Flight	Unverified; neither test LV-L-3, Vehicle Demonstration Launch, or test P-L-1, Payload Demonstration Launch, have been completed. However, they have been tentatively scheduled for February 18, 2023, and March 18, 2023.

complete a Vehicle Demonstration Flight but fail to qualify the payload by satisfactorily completing the Payload Demonstration Flight requirement will not be permitted to fly a final competition launch. 2.20.3. Teams who complete a Payload Demonstration Flight which is not fully successful may petition the NASA RSO for permission to fly the payload at launch week. Permission will not be granted if the RSO or the Review Panel have any safety concerns.		or Vehicle Demonstration Flights are not completed by their associated deadlines, the team will not be permitted to launch at launch week.	
2.21. The team's name and Launch Day contact information shall be in or on the rocket airframe as well as in or on any section of the vehicle that separates during flight and is not tethered to the main airframe. This information shall be included in a manner that allows the information to be retrieved without the need to open or separate the vehicle.	Inspection	The team will include the team name and Launch Day contact information on the launch vehicle airframe. No sections of the launch vehicle will detach.	Unverified; the launch vehicle has not been constructed.
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.	Inspection and Demonstration	The design will sufficiently protect Lithium Polymer batteries from ground impacts, and will be brightly colored and clear marked as a fire hazard. Test P-D-5, Payload Impact Resistance Demonstration, will also ensure that the payload is resistant to impacts.	Unverified; the payload has not been constructed, and test P-D-5, Payload Impact Resistance Demonstration, has not been completed.
2.23. Vehicle Prohibitions	See below.	See below.	See below.

2.23.1. The launch vehicle will not utilize forward firing motors.	Inspection	The launch vehicle will contain a single, rear firing motor.	Verified; the launch vehicle contains a single Aerotech L1090 rear-firing motor.
2.23.2. The launch vehicle will not utilize motors that expel titanium sponges (Sparky, Skidmark, MetalStorm, etc.)	Inspection	The launch vehicle will not contain a motor that expels titanium sponges.	Verified; the launch vehicle contains an Aerotech L1090 motor that does not expel titanium sponges.
2.23.3. The launch vehicle will not utilize hybrid motors.	Inspection	The launch vehicle will not utilize a hybrid motor.	Verified; the launch vehicle employs an Aerotech L1090 reloadable motor.
2.23.4. The launch vehicle will not utilize a cluster of motors.	Inspection	The launch vehicle will not employ a cluster of motors.	Verified; the launch vehicle will use a single Aerotech L1090 motor.
2.23.5. The launch vehicle will not utilize friction fitting for motors.	Inspection	The launch vehicle will not employ friction fitting for motors.	Verified; the motor will be installed and secured using a motor retention system, in the form of a thrust plate.
2.23.6. The launch vehicle will not exceed Mach 1 at any point during flight.	Analysis	OpenRocket simulations will be performed to ensure the estimated maximum speed of the launch vehicle will not exceed Mach 1.	Verified; the estimated maximum speed of the launch vehicle is 646 ft/s, which is equivalent to Mach 0.57.
2.23.7. Vehicle ballast will not exceed 10% of the total unballasted weight of the rocket as it would sit on the pad (i.e. a rocket with an unballasted weight of 40 lbs. on the pad may contain a maximum of 4 lbs. of ballast).	Inspection	The total amount of ballast installed on the vehicle will not exceed 10% of the unballasted vehicle.	Unverified; the launch vehicle has not been constructed. The vehicle is estimated to weigh 26.3 lbs, so the total vehicle ballast will not exceed 2.63 lbs.
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).	Inspection	Transmitters used prior to landing will not exceed 250 mW of power.	Verified; the transmitter used, a Big Red Bee 900 GPS, employs a 250 mW transmitter.
2.23.9. Transmitters will not create excessive interference. Teams will utilize unique frequencies, hand-shake/passcode systems, or other means to mitigate	Inspection	The transmitters used in the launch vehicle will be operated in a mode that minimizes bandwidth use.	Verified; the Big Red Bee 900 GPS will be operated in the $BRBTX Summary mode, which limits transmission rates

interference caused to or received from other teams.			to minimize use of the RF bandwidth.
2.23.10. Excessive and/or dense metal will not be utilized in the construction of the vehicle. Use of light-weight metal will be permitted but limited to the amount necessary to ensure structural integrity of the airframe under the expected operating stresses.	Inspection	Use of metal in the launch vehicle will be minimized, and only used in areas of need.	Verified; the launch vehicle has been designed to include metal in only the nose cone tip, bulkhead assemblies (in the form of eyebolts and threaded rods), and in the motor.

Table 73: NASA Vehicle Requirements

6.4.1.3 Recovery Requirements

The team has detailed a compliance plan based on recovery requirements (Table 74).

Requirement	Verification Type	Compliance Plan	Compliance Status
3.1. The full-scale launch vehicle will stage the deployment of its recovery devices, where a drogue parachute is deployed at apogee, and a main parachute is deployed at a lower altitude. Tumble or streamer recovery from apogee to main parachute deployment is also permissible, provided that kinetic energy during drogue stage descent is reasonable, as deemed by the RSO.	Demonstration	The recovery system will be designed as a dual-deploy system, involving a drogue parachute deployed at apogee and a main parachute deployed at a lower altitude. Test LV-L-3, Vehicle Demonstration Launch, will demonstrate this functionality.	Partially verified; the launch vehicle has been designed with a dual-deploy recovery system that fulfills the requirement. However, test LV-L-3, Vehicle Demonstration Launch, has not been completed.
3.1.1. The main parachute shall be deployed no lower than 500 feet.	Demonstration	The recovery system will be designed such that the main parachute will be deployed at 600 ft, with a secondary ejection charge set to fire at 550 ft. Thus, both the primary and secondary altimeters will be able to fulfill this requirement. Test LV-L-3, Vehicle Demonstration Launch, will verify this functionality.	Partially verified; the launch vehicle has been designed a recovery system that fulfills the requirement. However, test LV-L-3, Vehicle Demonstration Launch, has not been completed.

3.1.2. The apogee event may contain a delay of no more than 2 seconds.	Demonstration	The recovery system will be designed so that the drogue parachute is ejected at apogee, or within two seconds of apogee. Test LV-L-3, Vehicle Demonstration Launch, will verify this functionality.	Partially verified; the launch vehicle has been designed such that the primary ejection charge fires at apogee with a secondary ejection charge firing one second later. However, test LV-L-3, Vehicle Demonstration Launch, has not been completed.
3.1.3. Motor ejection is not a permissible form of primary or secondary deployment.	Inspection	The recovery system will not be designed to include a motor ejection event.	Verified; the launch vehicle has not been designed to include a motor ejection event.
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.	Demonstration	The team will perform test LV-R-5, Parachute Ejection Demonstration, for all electronically initiated ejection events prior to flight.	Partially verified; test LV-R-5, Parachute Ejection Demonstration, has been completed for the subscale launch vehicle, but not for the full-scale launch vehicle.
3.3. Each independent section of the launch vehicle will have a maximum kinetic energy of 75 ft-lbf at landing. Teams whose heaviest section of their launch vehicle, as verified by vehicle demonstration flight data, stays under 65 ft-lbf will be awarded bonus points.	Analysis and Demonstration	The team will perform simulations to estimate the maximum kinetic energy of each vehicle section at landing and design the recovery system to ensure these values fall below 75 ft-lbf. Test LV-L-3, Vehicle Demonstration Launch, will confirm that the actual maximum kinetic energy of each section remains below 75 ft-lbf.	Partially verified; the estimate for the maximum kinetic energy at ground-hit for the most massive section is 66.1 ft-lbf. However, this value has not been verified as test LV-L-3, Vehicle Demonstration Launch, has not been completed.
3.4. The recovery system will contain redundant, commercially available barometric altimeters that are specifically designed for	Inspection	The recovery system will employ two different commercially available barometric altimeters	Verified; the recovery system employs a PerfectFlite Stratologger CF

initiation of rocketry recovery events. The term “altimeters” includes both simple altimeters and more sophisticated flight computers.		designed for high power rocketry.	altimeter and an Entacore AIM USB altimeter.
3.5. Each altimeter will have a dedicated power supply, and all recovery electronics will be powered by commercially available batteries.	Inspection	Each altimeter will be wired separately, using an independent, commercially available battery as a power supply.	Verified; the recovery system employs two Duracell 9V batteries, one for each altimeter.
3.6. Each altimeter will be armed by a dedicated mechanical arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad.	Inspection	Each altimeter will be armed with a dedicated mechanical arming switch accessible from the exterior of the launch vehicle when in flight configuration on the launch pad.	Verified; the launch vehicle will include two key-lock switches located on the avionics bay switch band, so that they are accessible from the launch vehicle exterior.
3.7. Each arming switch will be capable of being locked in the ON position for launch (i.e. cannot be disarmed due to flight forces).	Inspection	The mechanical arming switches used will be capable of being locked in the ON position.	Verified; the launch vehicle will employ key-lock switches that are only able to be actuated using a key.
3.8. The recovery system, GPS and altimeters, electrical circuits will be completely independent of any payload electrical circuits.	Inspection	The recovery system will be completely isolated from the payload electrical circuits.	Verified; the GPS, altimeters, and payload electronics will be located in separate, completely isolated bays.
3.9. Removable shear pins will be used for both the main parachute compartment and the drogue parachute compartment.	Inspection	Shear pins will be used to secure the airframe sections containing the main parachute and drogue parachute.	Verified; shear pins will be used to fasten the airframe sections containing the main and drogue parachutes.
3.10. The recovery area will be limited to a 2,500 ft. radius from the launch pads.	Analysis	OpenRocket simulations will be performed to ensure the expected recovery area is limited to a radius of 2,500 ft from the launch pad.	Verified; the maximum expected drift is 2,466 ft.

3.11. Descent time of the launch vehicle will be limited to 90 seconds (apogee to touch down). Teams whose launch vehicle descent, as verified by vehicle demonstration flight data, stays under 80 seconds will be awarded bonus points.	Analysis	OpenRocket simulations will be performed to ensure the expected descent time of the launch vehicle remains under 90 s.	Verified; the expected total descent time is 85.4 s.
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. Avionic interference test 3.12.1. Any rocket section or payload component, which lands untethered to the launch vehicle, will contain an active electronic GPS tracking device. Avionic interference test 3.12.2. The electronic GPS tracking device(s) will be fully functional during the official competition launch.	Inspection and Test	A GPS tracking device capable of transmitting the location of the launch vehicle will be installed in the nosecone shoulder of the vehicle. No section will land untethered from the launch vehicle. Tests LV-A-6, GPS Accuracy Test, and LV-A-7, GPS Range Test, will assess the ability of the GPS to transmit accurate location data.	Partially verified; the design of the launch vehicle includes a Big Red Bee 900 GPS that will be located in the nosecone shoulder of the launch vehicle. However, tests LV-A-6, GPS Accuracy Test, and LV-A-7, GPS Range Test, have not been completed.
3.13. The recovery system electronics will not be adversely affected by any other on-board electronic devices during flight (from launch until landing). 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. 3.13.2. The recovery system electronics will be shielded from all onboard transmitting devices to avoid inadvertent excitation of the recovery system electronics. 3.13.3. The recovery system	Inspection and Demonstration	The altimeters used in the recovery system will be located in a dedicated avionics bay that is shielded from other on-board electronics. Additionally, test LV-A-5, Avionics Interference Demonstration, will be performed to verify the efficacy of the shielding.	Partially verified; the launch vehicle design includes a shielded avionics bay that will include only the altimeters. However, test LV-A-5, Avionics Interference Demonstration, has not been completed.

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. 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.			

Table 74: NASA Recovery Requirements

6.4.1.4 Payload Requirements

The team has detailed a compliance plan based on payload requirements (Table 75).

Requirement	Verification Type	Compliance Plan	Compliance Status
4.1. College/University Division—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. The method(s)/design(s) utilized to complete the payload mission shall be at the team's discretion and shall be permitted so long as the designs are deemed safe, obey FAA and legal requirements, and adhere to the intent of the challenge. An additional experiment (limit of 1) is allowed, and may be flown, but will not contribute to scoring. If the team chooses to fly an additional experiment, they will provide the appropriate documentation in all design reports so the experiment may be reviewed for flight safety.	Inspection and Demonstration	The payload will be designed so that it can autonomously receive RF commands and perform corresponding actions with an on-board camera system. This functionality will be confirmed with test P-SI-4, Radio Integration Demonstration. FAA and legal compliance will be ensured via oversight from a designated Safety Officer. If additional experiments are flown, appropriate documentation will be provided in all design reports.	Partially verified; the payload has been designed to accomplish the outlined objectives. However, test P-SI-4, Radio Integration Demonstration, has not been completed. Monitoring of FAA and legal compliance is ongoing. No additional experiments will be flown on the launch vehicle.
4.2. Radio Frequency Command (RAFCO) Mission Requirements	See below	See below	See below

4.2.1. Launch Vehicle shall contain an automated camera system capable of swiveling 360 deg to take images of the entire surrounding area of the launch vehicle.	Demonstration	Test P-CF-6, Camera Rotation Demonstration, will confirm the ability of the camera assembly to rotate 360 deg around the z-axis.	Unverified; test P-CF-6, Camera Rotation Demonstration, has not been completed.
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 planetary surface oriented down.	Demonstration and Inspection	Test P-SF-4, Camera Angle Inspection, will confirm that the camera is oriented on the z-axis. Test P-CF-6, Camera Rotation Demonstration, will verify that the camera assembly is capable of rotating around this axis.	Unverified; tests P-SF-4, Camera Angle Inspection, and P-CF-6, Camera Rotation Demonstration, have not been completed.
4.2.1.2. The camera shall have a FOV of at least 100º and a maximum FOV of 180º.	Inspection	Test P-CF-9, Camera FOV Inspection, will be performed to confirm the field of view (FOV) of the camera.	Unverified; test P-CF-9, Camera FOV Inspection, has not been completed.
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.	Demonstration	Test P-CF-4, Image Manipulation Software Demonstration, will confirm the ability of the payload software to apply a time stamp the images. These images will be submitted to NASA in PLAR.	Unverified; test P-CF-4, Image Manipulation Software Demonstration, has not been completed.
4.2.1.4. The camera system shall execute the string of transmitted commands quickly, with a maximum of 30 seconds between photos taken.	Test	Test P-SF-3, Payload Performance Test, will measure the time required for the payload to execute the required commands. It will be ensured that the maximum time between photos being taken is 30 s.	Unverified; test P-SF-3, Payload Performance Test, has not been completed.

4.2.2. NASA Student Launch Management Team shall transmit a RF sequence that shall contain a radio call sign followed by a sequence of tasks to be completed. The list of potential commands to be given on launch day along with their radio transcriptions which shall be sent in a RF message using APRS transmission in no particular order are: A1—Turn camera 60º to the right B2—Turn camera 60º to the left C3—Take picture D4—Change camera mode from color to grayscale E5—Change camera mode back from grayscale to color F6—Rotate image 180º (upside down). G7—Special effects filter (Apply any filter or image distortion you want and state what filter or distortion was used). H8—Remove all filters. 4.2.2.1. An example transmission sequence could look something like, "XX4XXX C3 A1 D4 C3 F6 C3 F6 B2 B2 C3." Note the call sign that NASA will use shall be distributed to teams at a later time.	Demonstration	Test P-SF-4, Radio Integration Demonstration, will be performed to ensure the payload can receive and act upon transmitted APRS commands.	Unverified; test P-SF-4, Radio Integration Demonstration, has not been completed.

4.2.3. The NASA Student Launch Management Panel shall transmit the RAFCO using APRS. 4.2.3.1. NASA will use dedicated frequencies to transmit the message. NASA will operate on the 2-Meter amateur radio band between the frequencies of 144.90 MHz and 145.10 MHz. No team shall be permitted to transmit on any frequency in this range. The specific frequency used will be shared with teams during Launch Week. NASA reserves the right to modify the transmission frequency as deemed necessary. 4.2.3.2. The NASA Management Team shall transmit the RAFCO every 2 minutes. 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.	Inspection and Demonstration	The payload will be configured to receive RF commands in the format of APRS transmissions in the 2-meter amateur radio band. Test P-SF-4, Radio Integration Demonstration, will confirm this ability. The team will not transmit on this frequency during launch. The payload will not activate until after landing detection. Test P-SF-6, Landing Detection Demonstration, will verify this functionality.	Unverified; tests P-SF-4, Radio Integration Demonstration, and P-SF-6, Landing Detection Demonstration, have not been completed.
4.2.4. The payload shall not be jettisoned.	Inspection	The payload will be designed to be permanently tethered to the launch vehicle.	Verified; the payload has been designed to be permanently tethered to the launch vehicle.
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.	Inspection	The time-stamped photos produced during the payload function will not be transmitted; the payload will not transmit any information. These photos will be included in PLAR.	Unverified.
4.3. General Payload Requirements	See below	See below	See below
4.3.1. Black Powder and/or similar energetics are only permitted for deployment of in-flight recovery systems. Energetics shall not be permitted for any surface operations.	Inspection	Black powder and other energetics will not be employed by the payload system.	Verified; the payload design does not incorporate any energetics, including black powder.

4.3.2. Teams shall abide by all FAA and NAR rules and regulations.	Inspection	A Safety Officer will be nominated to monitor compliance with FAA and NAR rules and regulations.	Ongoing verification; a Safety Officer has been nominated to monitor compliance.
4.3.3. Any secondary payload experiment element that is jettisoned during the recovery phase will receive real-time RSO permission prior to initiating the jettison event, unless exempted from the requirement the CDR milestone by NASA. 4.3.4. Unmanned aircraft system (UAS) payloads, if designed to be deployed during descent, will be tethered to the vehicle with a remotely controlled release mechanism until the RSO has given permission to release the UAS. 4.3.5. Teams flying UASs will abide by all applicable FAA regulations, including the FAA's Special Rule for Model Aircraft (Public Law 112–95 Section 336; see https://www.faa.gov/uas/faqs). 4.3.6. Any UAS weighing more than .55 lbs. shall be registered with the FAA and the registration number marked on the vehicle.	Inspection	No secondary payload experiment will be included on the launch vehicle. Nor will an unmanned aircraft system (UAS) be employed by the payload system or included in the launch vehicle.	Verified; no secondary payload experiment has been included in the launch vehicle, nor will a UAS be flown on the launch vehicle.

Table 75: NASA Payload Requirements

6.4.1.5 Safety Requirements

The team has detailed a compliance plan based on safety requirements (Table 76).

Requirement	Verification Type	Compliance Plan	Compliance Status
5.1. Each team will use a launch and safety checklist. The final checklists will be included in the FRR report and used during the Launch Readiness Review (LRR) and any Launch Day operations.	Inspection	A launch and safety checklist will be created for use in all full-scale launches. The final checklists will be included in the FRR report.	Unverified; the team will include the final checklists in the FRR report to be utilized during LRR and on Launch Day.

5.2. Each team shall identify a student safety officer who will be responsible for all items in section 5.3.	Inspection	Luka Bjellos will serve as the team's safety officer and will be responsible for all items in section 5.3.	Verified; Luka Bjellos was identified as the team's safety officer in the team's proposal.
5.3. The role and responsibilities of the safety officer will include, but are not limited to: 5.3.1. Monitor team activities with an emphasis on safety during: 5.3.1.1. Design of vehicle and payload 5.3.1.2. Construction of vehicle and payload components 5.3.1.3. Assembly of vehicle and payload 5.3.1.4. Ground testing of vehicle and payload 5.3.1.5. Subscale launch test(s) 5.3.1.6. Full-scale launch test(s) 5.3.1.7. Competition Launch 5.3.1.8. Recovery activities 5.3.1.9. STEM Engagement Activities	Inspection	The safety officer will be present for and monitor the design, manufacturing, and assembly of the payload and its components to ensure that the proper safety precautions are being taken. Additionally, the safety officer will monitor all ground testing of the vehicle and payload and attend all subscale and full-scale launches, including the competition launch. The safety officer will also monitor recovery activities and educational engagement activities for which there may be safety hazards.	Verified; the safety officer has followed all guidelines so far and will continue to do so.
5.3.2. Implement procedures developed by the team for construction, assembly, launch, and recovery activities.	Inspection	The safety officer will be present for all activities for which safety hazards may be present and ensure that safety precautions are being followed.	Partially verified; the safety officer has reviewed all procedures so far and will continue to do so in future activities.
5.3.3. Manage and maintain current revisions of the team's hazard analyses, failure modes analyses, procedures, and MSDS/chemical inventory data.	Inspection	The safety officer will maintain and update the team's hazard analyses, failure modes analyses, procedures, and MSDS/chemical inventory data.	Verified; the safety officer has maintained current revisions of the team's safety procedures and analyses in section 5.1 of this report.

5.3.4. Assist in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	Inspection	The safety officer will contribute to the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	Verified; the safety officer has written about hazard analyses, failure mode analyses, and procedures in section 5 in the report.
5.5. Teams will abide by all rules set forth by the FAA.	Inspection	The team will abide by all rules set by the FAA. Project plans will be examined to ensure that they abide by these rules.	Verified; the team's plans abide by all rules set by the FAA.

Table 76: NASA Safety Requirements

6.4.2 Team Derived Requirements

The team has developed requirements specific to the development of the launch vehicle, recovery system, and payload.

6.4.2.1 Vehicle Requirements

The team has developed team-derived requirements specific to the design of the launch vehicle (Table 77).

Requirement	Verification	Justification and Compliance Plan	Compliance Status
1.1 The airframe section containing the payload will retain sufficient structural strength to endure forces of flight and recovery.	Test	For safe flight, the airframe must not experience structural failure during launch, and must remain undamaged for multiple launches. An airframe compressive strength test (P-D-3) will determine the compressive strength of the modified airframe.	Partially verified; the subscale launch vehicle had a payload airframe with rectangular cutouts similar to the ones in the full-scale design. The subscale launch vehicle retained sufficient structural strength throughout flight and landing. The compressive strength test has not yet been performed.

Table 77: Team Vehicle Requirements

6.4.2.2 Recovery Requirements

The team has developed team-derived requirements specific to the design of the recovery system (Table 78).

Requirement	Verification	Justification and Compliance Plan	Compliance Status
2.1 The drogue parachute will have a descent rate of 81.8 ft/s.	Analysis and Test	This descent rate ensures that the launch vehicle will not be damaged upon landing. The parachute drag test (LV-R-1) and simulations will ensure that the parachute will induce enough drag.	Partially verified; test LV-R-1, Drogue Parachute Drag Test, has been completed, but test LV-L-3, Vehicle Demonstration Launch, has not been performed to confirm the estimated descent rates.
2.2 The main parachute will have a descent rate of 17.5 ft/s.	Analysis and Test	This descent rate ensures that the launch vehicle will not be damaged upon landing. The parachute drag test (LV-R-2) will ensure that the parachute will induce enough drag.	Partially verified; test LV-R-2, Main Parachute Drag Test, has been completed, but test LV-L-3, Vehicle Demonstration Launch, has not been performed to confirm the estimated descent rates.
2.3 The recovery harnesses will be at least 2.5 times the length of the total length of the airframe.	Inspection	The recovery harnesses must be sufficiently long enough to ensure the independent sections of the launch vehicle do not collide during descent. The Avionics and Recovery lead will measure the length of the recovery harness to verify that it is correct.	Partially verified; the recovery harness met the requirement for the subscale launch vehicle. However, the full-scale launch vehicle has not yet been constructed.
2.4 The secondary drogue ejection charge will activate 1 second after the primary ejection charge.	Test	The secondary ejection charge must activate quickly enough to ensure drogue deployment occurs at a safe	Verified; test LV-A-2, Altimeter Accuracy Test,

		velocity; however, over pressurization may occur if charge activates too quickly. The altimeter accuracy test (LV-A-2) will be performed to ensure the altimeter activates at the correct time.	has been completed.
2.5 The secondary main ejection charge will activate 50 feet lower than the primary ejection charge.	Test	The secondary ejection charge must activate quickly enough to ensure drogue deployment occurs at a safe velocity; however, over pressurization may occur if charge activates too quickly. The altimeter accuracy test (LV-A-2) will be performed to ensure the altimeter activates at the correct altitude.	Verified; test LV-A-2, Altimeter Accuracy Test, has been completed.
2.6 The secondary ejection charges will be 25% larger by weight than the primary ejection charges.	Inspection	Secondary charges are a necessary redundancy to ensure separation if the primary charge fails. As a result, they are slightly more powerful than primary charges. Ejection charges will be prepared with the proper amount of black powder using a scale, and the Avionics and Recovery lead will verify the quantity of black powder used.	Partially verified; the secondary ejection charges on the subscale launch vehicle were 25% larger than the primary charges. However, the ejection charges for the full-scale launch vehicles have not yet been prepared.
2.7 The time for the parachutes to open and fully inflate after ejection will not exceed 2.49 s, or the time for the vehicle to free-fall 100 ft.	Test	It is important for the parachutes used in the launch vehicle to open fully and quickly in order to slow the descent rate of the vehicle to a safe velocity. The parachute deployment time test (LV-R-4) will be conducted to determine which folding methods will meet this goal.	Verified; test LV-R-4, Parachute Deployment Time Test, has been completed.

Table 78: Team Recovery Requirements

6.4.2.3 Payload Requirements

The team has developed team-derived requirements specific to the design of the payload (Table 79).

Requirement	Verification	Justification and Compliance Plan	Compliance Status
3.1 The horizon line will be within 10 degrees of parallel with the bottom of the image.	Inspection	This requirement ensures that the camera is perpendicular to the ground and is able to take level images of the surroundings. The team will simulate representative recovery scenarios to gather image data and ensure that the horizon line is parallel to the bottom border of the image.	Unverified; test P-SF-4, Camera Angle Inspection, has not been completed.
3.2 Latch release mechanism will have sufficient strength to release camera arm.	Demonstration	The latch release must be strong enough so the correct camera can rotate into the upright position upon landing. The latch release demonstration (P-CF-5) will be conducted to ensure latch release is reliably able to release the camera arm.	Unverified; test P-CF-5, Latch Release Demonstration, has not been completed.
3.3 Payload will be able to recover from a brief power loss event.	Demonstration	The payload must be able to recover from power loss in order to accomplish the goal of the competition. The power loss recovery demonstration (P-SF-5) will be conducted to ensure that payload can reliably recover from power loss events.	Unverified; test P-SF-5, Power Loss Recovery Demonstration, has not been completed.
3.4 Payload will be able to remain idle for a minimum of two hours, with sufficient battery power for operation at the conclusion of the two hours.	Test	Since the launch vehicle may remain on the launch pad for up to two hours before launch, the payload must be able to remain idle for at least two hours and still be able to complete the mission. The payload battery life test (P-SF-1) will be performed to ensure the payload can retain sufficient battery power after significant delays.	Unverified; test P-SF-1, Payload Battery Life Test, has not been completed.
3.5 Payload electronics will be able to resist acceleration forces during launch.	Demonstration	Electrical and structural connections within the payload must be able to	Unverified; test P-D-4, Payload Acceleration

		withstand the forces of launch in order to carry out the mission objective. The payload acceleration resilience demonstration (P-D-4) will be performed to ensure the payload is able to withstand accelerations similar to those experienced during launch.	Resilience Demonstration, has not been completed.
3.6 Payload must be able to detect landing.	Demonstration	Payload must be able to detect landing so camera can be deployed, and other mission objectives started. The landing detection test (P-SF-6) will be performed to ensure payload can detect landing.	Unverified; test P-SF-6, Landing Detection Demonstration, has not been completed.
3.7 Payload radio system must be able to receive transmissions from a 5W handheld transmitter at a range of 2500 ft.	Test	The payload must be able to receive RF commands after landing, which will occur in a 2500 ft radius of launch. The radio reception test (P-CF-3) will be performed to verify this functionality.	Unverified; test P-CF-3, Radio Reception Test, has not been completed.
3.8 Payload must be able to complete received commands in less than two minutes after reception of commands.	Inspection	The payload must be able to complete the transmitted operations quickly following reception. Two minutes has been estimated to be an appropriate window for the completion of all commands. The payload performance inspection (P-SF-3) will determine how long the payload will take to perform each operation.	Unverified; test P-CF-3, Payload Performance Inspection, has not been completed.

Table 79: Team Payload Requirements

6.5 Budgeting and Timeline

6.5.1 Funding

This project will be primarily funded by the University of Florida's Mechanical and Aerospace Engineering Department along with the support of our sponsors, who provide both monetary and material donations. The team is sponsored by Blue Origin, Aerojet Rocketdyne, and Hands-On Gainesville. The team is actively seeking more corporate sponsorships by developing a sponsorship plan and reaching out to potential sponsors. To accomplish this, the team has developed a corporate sponsorship package that will be sent to potential sponsors. The current sponsorships range from $250 to $2,000, with additional material donations. Between the funding received from the University of Florida's Mechanical and Aerospace Engineering Department and corporate sponsorships, the team will have approximately $9,300.00 to utilize for the project.

Additional funding is raised by the team through working with local businesses to create fundraisers to benefit the business and the organization. Funding will first be received by Dr. Niemi, our faculty advisor, and will then be allocated to our group. Additionally, the team has started an alumni program to stay connected with dedicated members who have graduated, encouraging them to stay involved and support the future of the group.

6.5.2 Budget

The team's total expected budget for the 2022-2023 year is $6800.00 (Figure 105, Table 80). This budget is derived from total vehicle component costs of the subscale and full-scale vehicles (Table 81, Table 82), as well as general and testing costs (Table 83, Table 84). Testing costs are anything associated with testing campaigns for both vehicles. General costs are any costs that pertain to the development of both vehicles, excluding testing. This budget allows for a $2,500.00 contingency in the event of necessary changes, damage repair, or further testing.

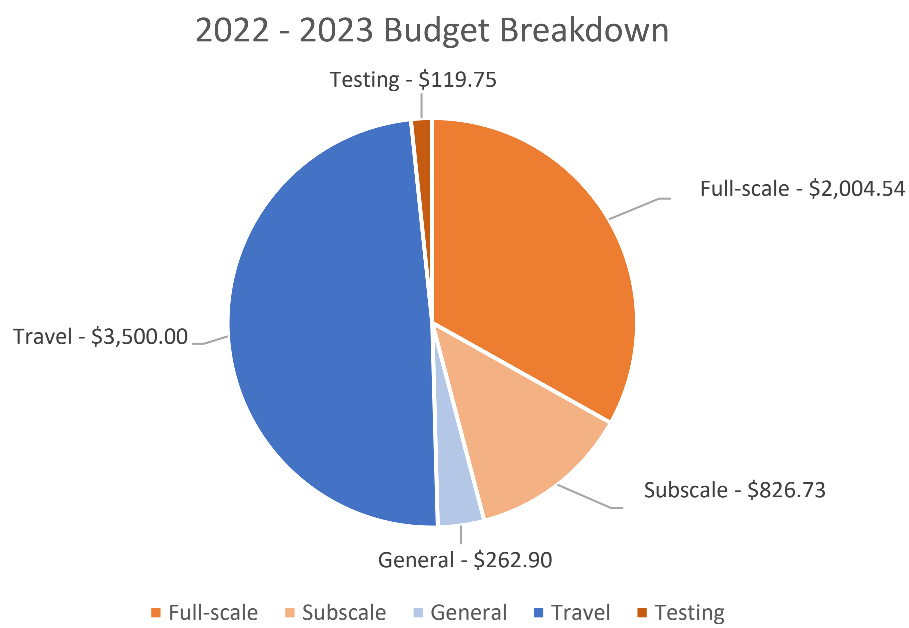

Figure 105: 2022-2023 Budget Breakdown

Cost Category	Total Cost (USD)
Full-scale	2004.54
Travel	3500.00
General	262.90
Subscale	826.73
Testing	165.54
Total	**$6759.71**

Table 80: Budget Breakdown by Category

Item Description	Vendor	Subteam	Price	Unit	Quantity	Total
Fiberglass Airframe G12-3.0	Wildman	Structures	$90.24	EA.	2	$180.48
Fiberglass Coupler G12-3.0	Wildman	Structures	$2.54	IN.	24	$60.69
Nosecone	Wildman	Structures	$64.90	EA.	1	$64.90
Fiberglass Airframe G12-2.1	Wildman	Structures	$31.68	EA.	1	$31.68
Rail Buttons	Apogee	Structures	$11.73	EA.	2	$23.46
Aerotech J415W-14A	Apogee	Flight Dyn.	$131.60	EA.	1	$131.60
Motor Hardware	Apogee	Flight Dyn.	$211.08	EA.	1	$211.08
3'' Type II PVC Round Stock	McMaster	Structures	$50.47	EA.	1	$50.47
Eyebolts	McMaster	Recovery	$4.90	EA.	8	$39.20
Terminal Blocks	Amazon	Recovery	$13.99	EA.	1	$13.99
Keylock Switch	Amazon	Recovery	$9.59	EA.	2	$19.18
G10 Fiberglass	Inventory	Structures	$0.00	EA.	1	$0.00
1/2 in Plywood	Inventory	Structures	$0.00	EA.	1	$0.00
					Total	**$826.73**

Table 81: Subscale Itemized Budget

Item Description	Vendor	Subteam	Price	Unit	Quantity	Total
USB Camera	Amazon	Payload Elec.	$60.99	EA.	3	$182.97
Solenoid motor (4.5mm)	Amazon	Payload Elec.	$9.81	EA.	4	$39.24
Stepper motor	Walmart	Payload Elec.	$7.83	EA.	4	$31.32
Radio Dongle (transceiver)	Ebay	Payload Elec.	$30.99	EA.	1	$30.99
Radio antenna	Amazon	Payload Elec.	$13.99	EA.	1	$13.99
Battery bank (raspberry pi)	Amazon	Payload Elec.	$13.99	EA.	1	$13.99
USB extension cable	Amazon	Payload Elec.	$5.84	EA.	1	$5.84
Radio adaptor kits	Amazon	Payload Elec.	$6.29	EA.	1	$6.29
USB-C Power cable 3.3ft	Amazon	Payload Elec.	$7.99	EA.	1	$7.99
DC-DC stepup converter	Amazon	Payload Elec.	$15.00	EA.	1	$15.00
PCB	JLC PCB	Payload Elec.	$30.00	EA.	1	$30.00
STEMMA QT 4-pin Female	Adafruit	Payload Elec.	$0.95	EA.	2	$1.90
STEMMA QT 4-pin Male	Adafruit	Payload Elec.	$0.95	EA.	2	$1.90
Adafruit BMP390	Adafruit	Payload Elec.	$10.95	EA.	1	$10.95
N-Channel MOSFET Gates	Digikey	Payload Elec.	$1.99	EA.	6	$11.94
JST SM Connector (20pc)	Walmart	Payload Elec.	$9.07	EA.	1	$9.07
Diodes	Digikey	Payload Elec.	$0.14	EA.	20	$2.80
Raspberry Pi	Inventory	Payload Elec.	$0.00	EA.	1	$0.00
Adafruit BNO055 IMU	Inventory	Payload Elec.	$0.00	EA.	1	$0.00
Li-Po Battery (Actuators)	Inventory	Payload Elec.	$0.00	EA.	1	$0.00
MPL3115A2 Barometer	Adafruit	Payload Elec.	$9.95	EA.	1	$9.95
JST XH connector	Amazon	Payload Elec.	$8.59	EA.	1	$8.59
2.54mm Pitch Header	Amazon	Payload Elec.	$9.99	EA.	1	$9.99
Push Button	Adafruit	Payload Elec.	$0.95	EA.	6	$5.70
Heat Shrink Insulation	Amazon	Payload Elec.	$6.99	EA.	1	$6.99
20 Gauge Wire	Amazon	Payload Elec.	$14.94	EA.	1	$14.94

5V Piezo	Adafruit	Payload Elec.	$0.95	EA.	3	$2.85
Nosecone	Wildman	Structures	$75.90	EA.	1	$75.90
Airframe	Wildman	Structures	$102.47	EA.	3	$307.41
Couplers	Wildman	Structures	$2.86	IN.	30	$85.80
Motor Tube	Wildman	Structures	$45.12	EA.	2	$90.24
Bulkheads	McMaster	Structures	$81.12	EA.	1	$81.12
Rail Buttons	Apogee	Structures	$11.73	EA.	3	$35.19
Eyebolts	McMaster	Recovery	$4.90	EA.	4	$19.60
D links	McMaster	Recovery	$2.78	EA.	2	$5.56
Terminal Blocks	Amazon	Recovery	$9.59	EA.	1	$9.59
Payload Hinge	Lowes	Payload Mech.	$2.99	EA.	3	$8.97
Aerotech L1090-W	Csrocketry	Flight Dyn.	$269.99	EA.	3	$809.97
Motor Retainer	Inventory	Flight Dyn.	$0.00	EA.	1	$0.00
Forward Closure	Inventory	Flight Dyn.	$0.00	EA.	1	$0.00
Aft Closure	Inventory	Flight Dyn.	$0.00	EA.	1	$0.00
Thrust Plate	Inventory	Flight Dyn.	$0.00	EA.	1	$0.00
Motor Casing	Inventory	Flight Dyn.	$0.00	EA.	1	$0.00
					Total	**$2004.54**

Table 82: Full-Scale Itemized Budget

Item Description	Vendor	Subteam	Price	Unit	Quantity	Total
PETG 3D Printer Filament	PRUSA	Payload	29.99	EA.	3	$89.97
Rivets	Apogee	Structures	$0.74	EA.	50	$37.05
Shear Pins	McMaster	Structures	$0.09	EA.	100	$8.74
RocketPoxy	Apogee	Structures	$67.14	EA.	1	$67.14
JB Weld	Amazon	Structures	$20.00	EA.	3	$60.00
					Total	**$262.90**

Table 83: General Itemized Budget

Item Description	Vendor	Subteam	Price	Unit	Quantity	Total
E-matches	Wildman	Testing	$17.95	EA.	5	$89.75
Baofeng UV-5R Radio	Amazon	Testing	$30.00	EA.	1	$30.00
					Total	**$119.75**

Table 84: Testing Itemized Budget

6.5.3 Timeline

The development of the project will be based around a scheduled timeline (Figure 106). The schedule has been developed to meet the deadlines set by NASA, and the team's own internal deadlines. Further schedules have been developed for specific phases of the project such as vehicle and payload development and manufacturing. Additionally, status checklists were developed for internal project tracking of milestones and important activities (Table 85, Table 86).

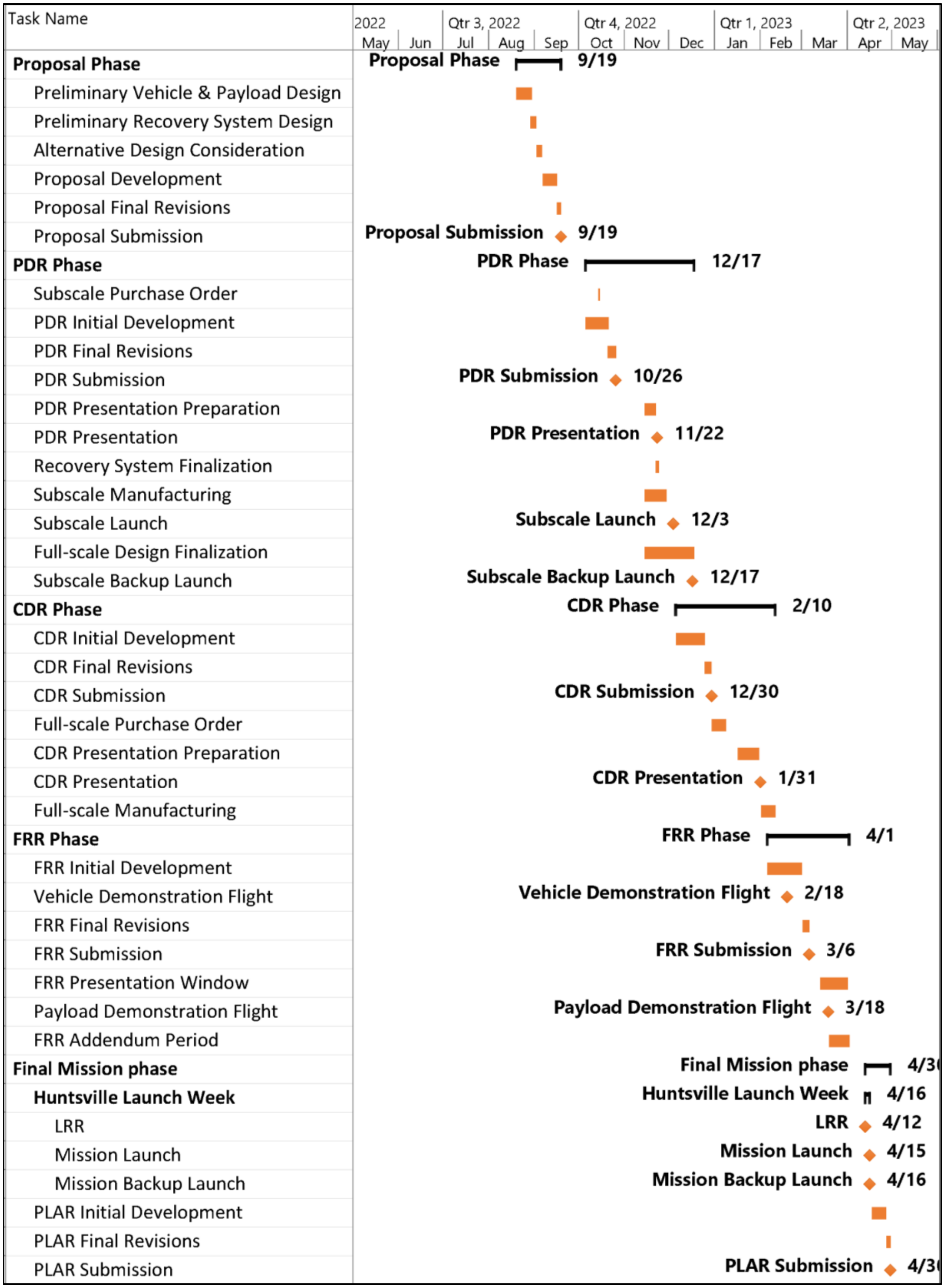

Figure 106: Full Project Schedule

Milestone	Date	Status
Proposal Submission	9/19/2022	Complete
PDR Submission	10/25/2022	Complete
PDR Presentation	11/22/2022	Complete
Subscale Launch	12/3/2022	Complete
Subscale Backup Launch	12/17/2022	Not Required
CDR Submission	1/9/2023	Complete
CDR Presentation	1/31/2023	Incomplete
Vehicle Demonstration Flight	2/18/2023	Incomplete
FRR Submission	3/6/2023	Incomplete
FRR Presentation	3/13-3/31/2023	Incomplete
Payload Demonstration Flight	3/18/2023	Incomplete
FRR Addendum Submission	4/2/2023	Incomplete
LRR at launch site	4/14/2023	Incomplete
Mission Launch	4/15/2023	Incomplete
Backup Mission Launch	4/16/2023	Incomplete
PLAR Submission	4/30/2023	Incomplete

Table 85: Project Milestone Status

Activity	Date	Status
Preliminary Vehicle Design	8/25 – 8/29/22	Complete
Preliminary Payload Design	8/19 – 8/29/22	Complete
Preliminary Recovery Design	8/25 – 8/29/22	Complete
Proposal Development	9/6 – 9/15/22	Complete
Proposal Revisions	9/16 – 9/18/22	Complete
Subscale Design	9/20 – 11/10/22	Complete
PDR Development	10/5 – 10/20/22	Complete
PDR Revisions	10/21 – 10/25/22	Complete
Subscale Manufacturing	11/14 – 11/28/22	Complete
Subscale Testing	11/29 – 12/1/22	Complete
Final Full-scale Design	11/11 – 12/20/22	Complete
CDR Development	12/5 – 12/24/22	Complete
CDR Revisions	12/25 – 12/30/22	Complete
Full-scale Manufacturing	2/1 – 2/11/23	Incomplete
Full-scale Testing	2/12 – 2/16/23	Incomplete
FRR Development	2/5 – 2/28/23	Incomplete
FRR Revisions	2/29 – 3/5/23	Incomplete
PLAR Development	4/17 – 4/26/23	Incomplete
PLAR Revisions	4/27 – 4/30/23	Incomplete

Table 86: Project Activity Status

6.5.3.1 Vehicle Timeline

Further schedules have been developed for vehicle development and manufacturing in more detail (Figure 107, Figure 108).

Figure 107: Structures Schedule

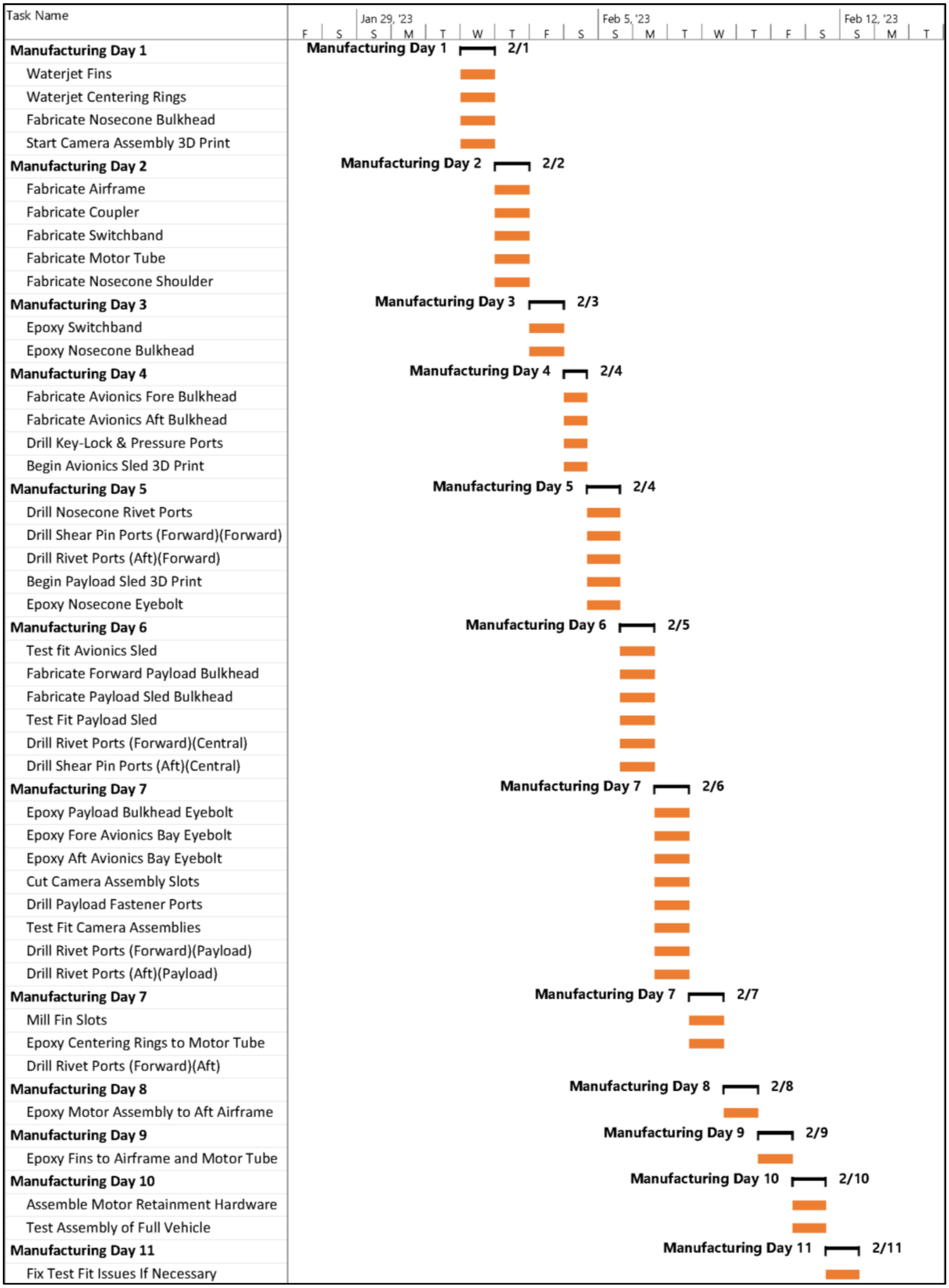

Figure 108: Full-Scale Manufacturing Schedule

6.5.3.2 Payload Timeline

Additional schedules have been developed for payload development and manufacturing in more detail (Figure 109, Figure 110).

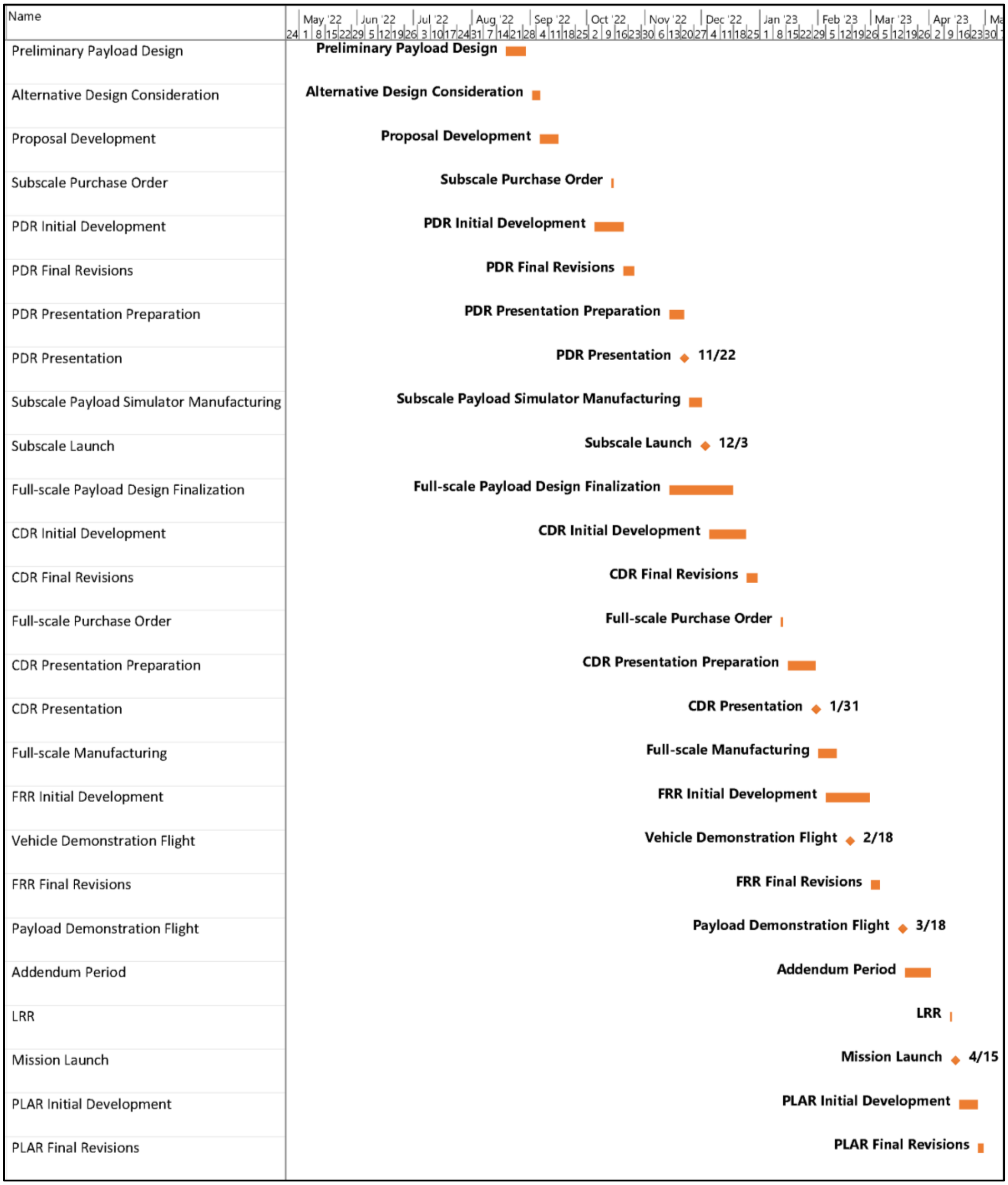

Figure 109: Payload Schedule

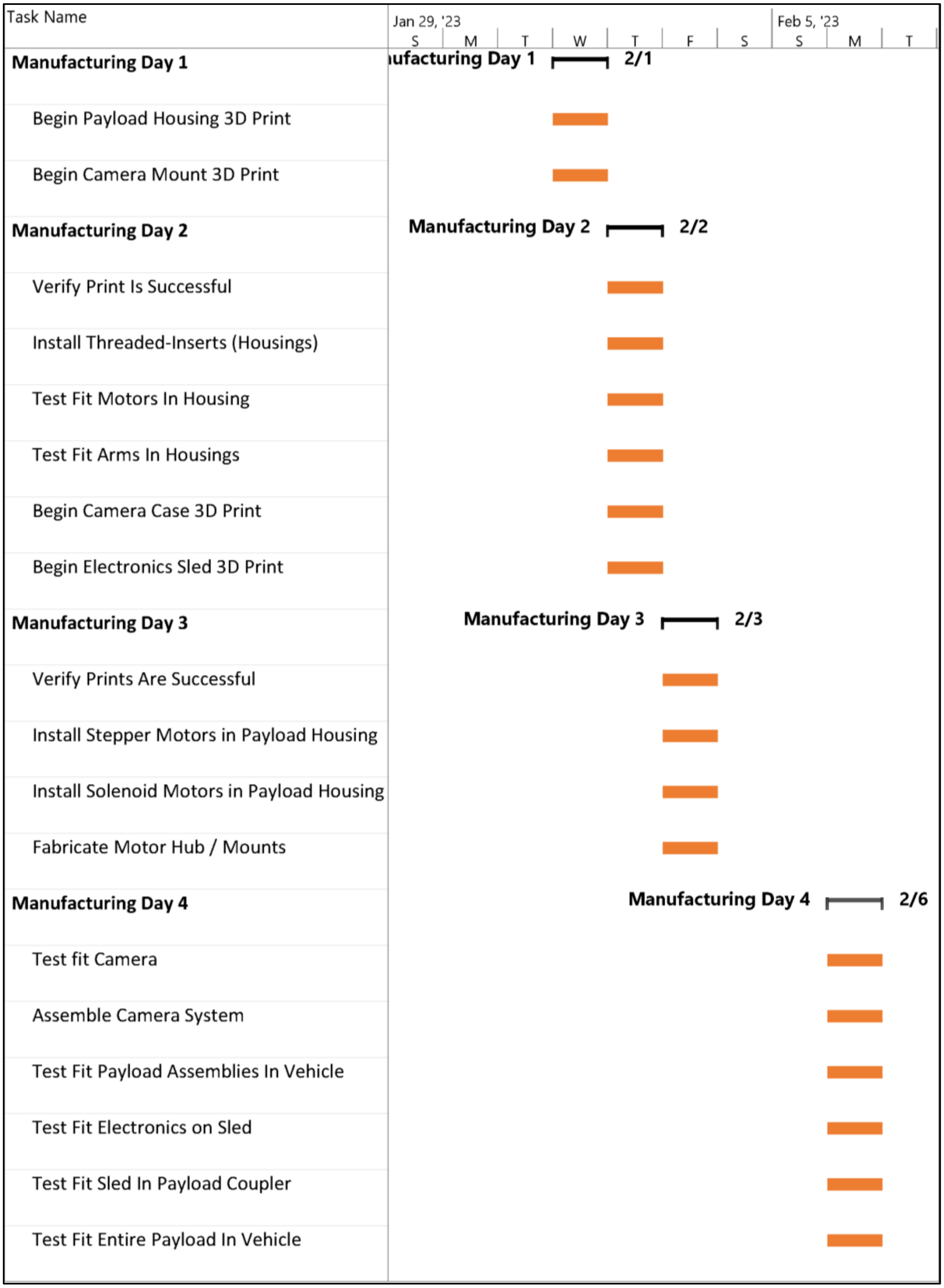

Figure 110: Payload Manufacturing Schedule

7. Launch Plans

7.1 Preliminary Launch Roster

As required, a preliminary launch roster was developed for the final mission launch in Huntsville, AL on April 15, 2023 (Table 87).

Subteam	Role	Name
Management	Project Manager	Erik Dearmin
Management	Chief Engineer	Maggie Wielatz
Management	L2 Certified Member	Megan Wnek
Management	Team President	Mikaela De Gracia
Management	Team Vice President	Brida Gibbons
Management	Member	Collin Larke
Mentor	Mentor	Jimmy Yawn
Safety	Safety Officer	Luka Bjellos
Structures	Lead	Layali Bazar
Avionics & Recovery	Lead	Stephanie Baldwin
Flight Dynamics	Lead	Mary Rowe
Flight Dynamics	Member	Krusha Patel
Payload Mechanics	Lead	Gabriella Peburn
Payload Electronics	Lead	Tianjun Wang
Payload Software	Lead	Christopher Comes
Testing	Lead	Rylan Andrews
Educational Engagement	Lead	Olivia Scarpo

Table 87: Preliminary Launch Roster

8. Conclusion

Based on the Swamp Launch Rocket Team's work on the design identified in this critical design review, the team is confident in the ability of the vehicle and the payload to meet the requirements set externally by NASA, and internally by the team.