NASA Student Launch

2016-2017 Preliminary Design Review

November 4, 2016

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1 Summary of PDR Report

1.1 Team Summary

School Name: University of Louisville
Organization: River City Rocketry
Location: J.B. Speed School of Engineering
132 Eastern Parkway
Louisville, KY 40292
Project Title: River City Rocketry 2016-2017
Name of Mentor: Daryl Hankes
TRA Number: 130
Certification Level: 3

1.2 Launch Vehicle Summary

Using OpenRocket to model the flight characteristics of the launch vehicle, the vehicle parameters were established. These characteristics are defined below in Table 1.

Length	138
Diameter	6
Mass (lbs)	45.9
Motor Choice	AeroTech L2200-G
Recovery System	Cruciform Drogue/Toroidal Main

Table 1: Launch vehicle parameters.

The diameter of the launch vehicle was chosen to be 6 inches to allow adequate room for all payloads and recovery hardware. A length of 138 inches was determined to provide adequate space for all recovery systems, payload containment, and mission electronics. With width and length defined, the weight of the launch vehicle was determined to be 45.9 lbs. In order to safely launch the vehicle and provide a margin of error for mass assumptions of various components, a AeroTech L2200-G solid ammonium perchlorate motor was chosen.

1.3 Payload Summary

The Experimental Payload challenge selected for is the Target Detection and Upright Landing. A coupler of the rocket will separate as an independent section and deploy into a multirotor. This payload will contain an onboard camera system capable of identifying and differentiating between the three randomly placed targets placed within a 300 ft radius of the launch rail. The multirotor will navigate to a position where the camera will be able to view and differentiate between the targets and land afterward safely on the ground.

All SOW requirements are referenced directly in this document. A full table of these requirements and their links can be found in Appendix II: Statement of Work Requirements.

2 Changes Made Since Proposal

2.1 Changes to Vehicle

2.1.1 Vehicle Design Changes

Change	Justification
The motor has been changed from an Aerotech 1420 Redline to an Aerotech L2200 Mojave Green.	The predicted overall weight of the launch vehicle has increased since proposal.
The geometry of the centering rings has been changed to account for the change in motor selection.	In order to maintain a minimum factor of safety of 2.0 throughout each centering ring, the size of the weight reduction slots was adjusted to account for the higher maximum thrust.

2.1.2 Recovery Design Changes

Change	Justification
Use of toroidal parachute for main instead of vortex ring.	1. Vortex ring prohibitively complex with ARRD double-staging recovery bay configuration. 2. Vortex ring prone to catastrophic failure modes.
Use of ARRD for double-staged recovery deployment instead of tender descender.	ARRD much more reliable for deployments and has much higher factor of safety due to 2,000lb load limit.
Vehicle separating into two main sections at apogee	Separation of booster mandatory at some point during recovery: 1. Separation after deployment bay main induces dangerously high terminal velocity under deployment bay drogue. 2. Booster separation under deployment bay drogue is impossible. This would create unacceptably extreme kinetic energies and opening forces.

2.1.3 Variable Drag System (VDS) Changes

Change	Justification
The main controller has been changed from a Raspberry Pi to a Teensy 3.6.	The Teensy has numerous advantages in terms of simplicity and performance.
Setpoint path equation (SPP) updated to piecewise function.	SPP will adjust for loss in braking power during latter part of ascent.
VDS structural components have been optimized to minimize weight.	The VDS will have a smaller overall weight, thus reducing the overall weight of the launch vehicle.
VDS actuation device has been changed from a AndyMark NeveRest 60 DC motor to a AndyMark NeveRest 40 DC motor.	The VDS will be able to actuate its drag blades faster.

Table 2: VDS Changes since proposal.

2.2 Changes to Payload

Change	Justification
The Deployment mechanism has changed from a metallic arm with a torque flange to a deployment parachute that will allow the multirotor to initialize under.	The risk associated with the metallic torque flange deployment arm were reevaluated as being too high. A deployment parachute system was implemented as a more optimal alternative.
The redundant recovery coupler tube has changed from being located below the upper bulkplate of the payload to being mounted to the upper bulkplate	This change was implemented in order to save space within the body of the payload.

2.3 Changes Made to Project Plan

2.3.1 Schedule Changes

Change	Justification
Addition of two full-scale test launches in March.	Extra launches provide the opportunity for additional verifications if needed.
Adding a full-scale recovery flight test.	Full verification of recovery procedure, especially the validation of the opening force on the deployment bay main parachute.
Accelerated design finish date of the payload moved to December 3rd, 2016.	Allows for immediate manufacturing for testing of Multirotor Recovery System (MRRS).

Table 3: Major project plan changes since proposal.

2.3.2 Budget Changes

Table 4 below indicates the fluctuations in our budget since proposal and where that change occurred. The green boxes indicate money that is being saved by an overestimation in component cost during proposal, whereas the red box indicates areas that ended up being more expensive than previously predicted.

Changes since Proposal			
Category	**Proposal Budget**	**Revised PDR Budget**	**Fluctuation**
Variable Drag System	Addition to Budget	$888.33	$0.00
Full-Scale Vehicle	$3,953.07	$3,834.16	-$118.91
Sub-Scale Vehicle	$733.24	$733.24	$0.00
Recvoery	$1,379.14	$1,744.99	$365.85
Payload	$2,124.00	$1,696.37	-$427.63
Educational Engagement	$2,027.76	$1,877.03	-$150.73
Travel Expenses	$5,750.00	$4,118.40	-$1,631.60
Promotional Materials	$2,187.50	$2,187.50	$0.00

Table 4: RCR budget changes since proposal.

3 Safety

Kevin Compton is the safety officer for River City Rocketry and will be monitoring the team throughout the 2016-2017 season. He is responsible for the overall safety of the team, students, and the public, throughout the duration of all team activities, as well as assuring compliance with all laws and regulations. The following requirements that the Safety Officer is responsible for are outlined below in Table 5.

Responsibilities	Verification
Provide a written team safety manual that includes hazards, safety plans and procedures, PPE requirements, MSDS sheets, operator manuals, FAA laws, and NAR and TRA regulations.	The safety officer developed a team safety manual that will be located on our website at http://www.rivercityrocketry.org/documents/
Confirm that all team members have read and comply with all regulations set forth by the team safety manual.	Each member of the team has signed the safety manual by the Preliminary Design Review.
Identify safety violations and take appropriate action to mitigate the hazard.	The team will provide risk mitigation tables for each technical sub-system, environmental hazard that effects the launch vehicle, and the launch vehicle effecting the environment.
Establish and brief the team on a safety plan for various environments, materials used, and testing.	The safety officer will be required to sign off each test document and ensure the test engineer knows the proper handling of materials, safety equipment needed, and emergency procedures for each test.
Establish a risk matrix that determines the risk level of each hazard based off of the probability of the occurrence and the severity of the event. Ensure that this type of analysis is done for each possible hazard.	The team developed a risk matrix showcased below in Table 8 that critiques our technical designs with severe scrutiny.
Oversee testing being performed to ensure that risks are mitigated.	The safety officer must sign off on each testing procedure before it is implemented.
Remain active in the design, construction, testing and flight of the rocket in order to quickly identify any new potential safety hazards and to ensure the team complies with the team safety plan.	The safety officer will assist in developing a prep-check list before launch day, weekly presentation updates, and risk re-evaluations.
Enforce proper use of Personal Protective Equipment (PPE) during construction, ground tests, and test flights of the rocket.	The safety manual has proper safety techniques for construction and ground testing that each team member must sign.
Make MSDS sheets and operator manuals available and easily accessible to the team at all times.	All MSDS sheets are in the safety manual that will be updated yearly as the team uses new materials.

Provide plan for proper purchase, storing, transporting, and use of all energetic devices.	All energetic devices will be transported by a vehicle and kept inside a clearly identified explosive box.
Ensure compliance with all local, state, and federal laws. Ensure compliance with all NAR and TRA regulations	The field that is selected for any particular launch will ensure the waiver is called in and all weather conditions are acceptable under NAR and TRA regulations.
Ensure the safety of all participants in educational outreach activities, providing PPE as necessary.	The team's safety manual covers proper outreach safety that all members review before signing.

Table 5: Safety Officer Responsibilities and verifications of responsibilities.

Kevin is requiring that all team members to read and sign the 2016-2017 River City Rocketry Safety Manual to ensure each member understands the proper safety precautions throughout the season. The safety manual is posted on our website and readily available for reference at any given time.

3.1 Hazard Analysis

Risk Assessment Matrix

Throughout the season the team will be examining each human interaction, environment, rocket system, and component, hazards have been identified and will continue to be brought to the team's attention. As each hazard is brought up it is assigned a risk level through the risk assessment matrix shown below in Table 8, which evaluates the severity of the hazard and the probability that the hazard will occur.

A severity value between 1 and 4 has been assigned to each hazard with a value of 1 being the most severe. In order to determine the severity of each hazard, the outcome of the mishap was compared to an established set of criteria based on the severity of personal injury, environmental impact, and damage to the rocket and/or equipment. The criteria is outlined below in Table 6.

Severity		
Description	**Value**	**Criteria**
Catastrophic	1	Could result in death, significant irreversible environmental effects, complete mission failure, monetary loss of $5k or more.
Critical	2	Could result in severe injuries, significant reversible environmental effects, partial mission failure, monetary loss of $500 or more but less than $5k.
Marginal	3	Could result in minor injuries, moderate environmental effects, complete failure of non-mission critical system, monetary loss of $100 or more but less than $500.

Negligible	4	Could result in insignificant injuries, minor environmental effects, partial failure of non-mission critical system, and monetary loss of less than $100.

Table 6: Severity value criteria.

A probability value between 1 and 5 has been assigned to each hazard with a value of 1 being most likely. The probability value was determined for each hazard based on an estimated percentage chance that the mishap will occur given the following:

- All personnel involved have undergone proper training on the equipment being used or processes being performed.
- All personnel have read and acknowledged that they have a clear understanding of all rules and regulations set forth by the latest version of the safety manual.
- Personal Protective Equipment (PPE) is used as indicated by the safety lab manual and MSDS.
- All procedures were correctly followed during construction of the rocket, testing, pre-launch preparations, and the launch.
- All components were thoroughly inspected for damage or fatigue prior to any test or launch.

The criteria set for the probability value is shown below in Table 7.

Probability		
Description	**Value**	**Criteria**
Almost Certain	1	Greater than a 90% chance that the mishap will occur.
Likely	2	Between 50% and 90% chance that the mishap will occur.
Moderate	3	Between 25% and 50% chance that the mishap will occur.
Unlikely	4	Between 1% and 25% chance that the mishap will occur.
Improbable	5	Less than a 1% chance that mishap will occur.

Table 7: Probability value criteria.

Through the combination of the severity table and probability table the risk assessment matrix was developed as shown below in Table 8. The matrix identifies each combination of severity vs. probability to result in a high, moderate, or low risk. The team's goal is to have every hazard to a low risk assessment by competition week. If a risk is not a low risk then action is to be taken either in the redesign, updated safety restrictions, or an update of requirements to reduce the overall risk of any particular hazard.

Risk Assessment Matrix				
Probability Value	**Severity Value**			
	Catastrophic-(1)	Critical-(2)	Marginal-(3)	Negligible-(4)
Almost Certain- (1)	2-High	3-High	4-High	5-Moderate
Likely-(2)	3-High	4-High	5-Moderate	6-Moderate
Moderate-(3)	4-High	5-Moderate	6-Moderate	7-Low
Unlikely-(4)	5-Moderate	6-Moderate	7-Low	8-Low
Improbable-(5)	6-Moderate	7-Low	8-Low	9-Low

Table 8: Risk assessment matrix.

Preliminary risk assessment tables have already been developed and are outlined in each individual technical design sub-system. Realization of these hazards brings attention to the possible failure mechanisms during the design or construction phase. To mitigate failure the team will address these hazards during design phase.

With some preliminary hazard detection accomplished, there are still some risks that are unacceptably high and won't be reduced until full-scale testing can drop the overall risk value. Justification and mitigation techniques are listed in the assessment tables for each hazard regarding why it is as low as it is. This may include analysis, safety precautions, and/or testing. In the event that any physical tests have been completed, the test report will be referenced in the assessment tables.

Lab and Machine Shop Risk Assessment

Construction and manufacturing of parts for the rocket will be performed in both on-campus and off-campus labs. The hazards assessed in Table 98 are risks present from working with machinery, tools, and chemicals in the lab.

VDS Actuation Risk Assessment

The hazards outlined in this section discuss the risks associated during testing and flight of the variable drag system. The VDS interfaces with the main structure of the vehicle, with potential risks in tools, manufacturing, and installment. This can be found in Table 25.

Payload Landing Risk Assessment

The hazards outlined in this section discuss the risks associated with the payload, which includes the upper half of the nose cone, landing upright. Since the payload separates from the vehicle it will encounter environmental hazards. This can be found in Table 93.

Payload Multirotor Risk Assessment

The hazards outlined in this section will discuss the risks associated with the deployment of the payload from the vehicle. The payload deployment interfaces with multiple systems, making it prone to hazards. This can be found in Table 94.

Recovery Risk Assessment

The hazards outlined below are risks associated with the recovery. Since there are two recovery systems onboard, many of the failure modes and results will apply to all of the systems but will be stated only once for conciseness.

Payload Redundant Recovery Risk Assessment

The hazards outlined in this risk assessment is associated with the redundant recovery that monitors the state of the payload pre-deployment and during flight. This assessment is strictly dealt with the electronical side that is monitoring and watches a pre-determined set of criteria that will deploy a backup parachute if any of the criteria were to be made true. Please refer to the recovery risk assessment for the deployment of the backup parachute. This can be found in Table 95.

Vehicle Assembly Risk Assessment

The hazards outlined in Table 40 are risks that could potentially be encountered throughout the assembly phase and during launch preparation.

Environmental Hazards to Rocket Risk Assessment

The hazards outlined in Table 99 are risks from the environment that could affect the rocket or a component of the rocket. Several of these hazards resulted in a moderate risk level and will remain that way for the remainder of the season. These hazards are the exception for needing to achieve a low risk level. This is because several of these hazards are out of the team's control, such as the weather. In the case that environmental hazards present themselves on launch day, putting the team at a moderate risk, the launch will be delayed until a low risk level can be achieved. The hazards that the team can control will be mitigated to attain a low risk level.

Hazards to Environment Risk Assessment

The hazards outlined in Table 100 are risks that construction, testing or launching of the rocket can pose to the environment.

Launch Procedures

The safety officer is responsible for writing, maintaining, and ensuring the use of up to date launch procedures as indicated in Appendix I. These are critical to ensure the safety of personnel, spectators, equipment and the environment. Checklists are to be used for any test launch and preparation leading to a launch.

The safety officer is responsible for writing and maintaining the pre-launch check list which is responsible for preparing the team before any sub scale or full scale launch test. This ensures that the team double checks all components and mitigates the risks of component failure and chance of an unsuccessful launch.

In order for River City Rocketry to be launch ready, the safety officer developed a safety preparation items list outlined in Table 9 below to showcase a general overview of what is needed to prepare before the team has a sub-scale or full-scale test launch.

	Variable Drag System	**Launch Vehicle**	**Recovery**	**Payload**

Validation Checks	All sensors have been tested are deemed flight ready by the VDS electrical lead, Ben Stringer	Black powder ejection tests will be performed before each test launch to ensure proper separation between sections and result in a 3 for 3 success rate on all first time separations.	Verify packing method of all parachutes by performing ground tests especially the booster main and payload main deployment bags.	Propulsion system verification under RC control to validate functionality of integrated electronics.
	The VDS housing is seeded correctly in the VDS bay and is deemed flight ready by the VDS mechanical lead, Justin Jonson	Full-scale and sub-scale assembly tests will be performed before each test flight to check fit all components as well as run a 2 hour electronics test.	Black powder deployment tests of all parachutes must result in a 3 for 3 success rate and clear the associated launch vehicle bays.	Deploying verification testing of the drone coming out of the launch vehicle as well as payload backup recovery tests. All deployments and recovery tests must result in a 3 for 3 success rate to be considered launch ready.
				Autonomous test flight, avoidance maneuver, and identification of targets associated with an upright landing.
Packing Criteria	Gather all tools that are necessary for each sub-system and only use these selected tools in assembly checks for both sub-scale and full-scale launch vehicle. to ensure all tools			
	Gather crucial extra components that can be used to fix each sub-system at the launch field if a component were to fail.			
	Run through a component sign off check list of each item and tool necessary for launch day. One of the captains and that sub-system lead must sign off on their check list to be considered launch ready.			
Launch Field	Weather conditions are not cloudy and the cloud ceiling is above the waiver altitude.			
	Wind Speeds are at or below 20mph.			
	Launch Field has appropriate launch rail and ignition system to fly the team's launch vehicle.			

Table 9: Safety preparation items needed before launch tests.

The safety officer is responsible for writing, maintaining, and ensuring the use of up to date launch procedures. These are critical to ensure the safety of personnel, spectators, equipment and the environment. The launch procedure checklist are to be used for any launch and preparation leading to launch while following the procedural thought process outlined in the previous figures.

The checklists are broken up into checklists for each subsystem for pre-launch day as well as launch day. This allows the team to keep organized and ensures a quick and efficient launch prep on launch day. Each subsystem checklist must be 100% complete and be signed by a representative of that subsystem and reviewed by one of the two captains. Checklists are then collected by the safety officer and the overall final assembly checklist can be started. After completion of the final assembly, all sub-team leads, captains and the safety officer must approve the rocket as being a go for launch. The "at the launch pad" checklist is then completed and personnel are assigned tasks of tracking each section of the rocket during recovery.

Each subsystem checklist includes the following features to ensure that assemblers are prepared, safe, and recognize all existing hazards:

Each checklist thoroughly written in order to set the team up for a safe and successful launch. Each subsystem checklist includes the following features to ensure that assemblers are prepared, safe, and recognize all existing hazards:

- Required equipment list
- Required hardware
- Required PPE
- ⚠**CAUTION** – label to identify where PPE must be used.
- **⚠WARNING** - label to signify importance of procedure by clearly identifying a potential failure and the result if not completed correctly.
- **⚠DANGER** - label to signal the use of explosives and indicates specific steps that should be taken to ensure safety.

3.2 NAR/TRA Procedures

3.2.1 NAR Safety Code

The table below describes each component of the NAR High Power Rocket Safety Code, effective August 2012, and how the team will comply with each component. This table has also been included in the team safety manual that all team members are required to review and acknowledge compliance.

NAR Code	Compliance
1. **Certification**. I will only fly high power rockets or possess high power rocket motors that are within the scope of my user certification and required licensing.	Only Darryl, the team mentor, and certified team members are permitted to handle the rocket motors.
2. **Materials.** I will use only lightweight materials such as paper, wood, rubber, plastic, fiberglass, or when necessary ductile metal, for the construction of my rocket.	The Mechanical Engineering team will be responsible for selecting the appropriate materials for construction of the rocket.

3. **Motors.** I will use only certified, commercially made rocket motors, and will not tamper with these motors or use them for any purposes except those recommended by the manufacturer. I will not allow smoking, open flames, nor heat sources within 25 feet of these motors.	Motors will be purchased through commercially rocket motor vendors such as Aerotech, Cesaroni, and Loki will only be handled by certified members of the team who are responsible for understanding how to properly store and handle the motors. Additionally, there is a portion on motor safety in the team lab manual that the entire team is responsible for understanding.
4. **Ignition System.** I will launch my rockets with an electrical launch system, and with electrical motor igniters that are installed in the motor only after my rocket is at the launch pad or in a designated prepping area. My launch system will have a safety interlock that is in series with the launch switch that is not installed until my rocket is ready for launch, and will use a launch switch that returns to the "off" position when released. The function of onboard energetics and firing circuits will be inhibited except when my rocket is in the launching position.	All launches will be at NAR/TRA certified events. The Range Safety Officer will have the final say over any safety issues.
5. **Misfires.** If my rocket does not launch when I press the button of my electrical launch system, I will remove the launcher's safety interlock or disconnect its batter and will wait 60 seconds after the last launch attempt before allowing anyone to approach the rocket.	The team will comply with this rule and any additional precautions that the Range Safety Officer makes on launch day.
6. **Launch Safety.** I will use a 5-second countdown before launch. I will ensure that a means is available to warn participants and spectators in the event of a problem. I will ensure that no person is closer to the launch pad than allowed by the accompanying Minimum Distance Table. When arming onboard energetics and firing circuits I will ensure that no person is at the pad except safety personnel and those required for arming and disarming operations. I will check the stability of my rocket before flight and will not fly it if it cannot be determined to be stable. When conducting a simultaneous launch of more than one high power rocket I will observe the additional requirements of NFPA 1127.	The team will comply with this rule and any determination the Range Safety Officer makes on launch day.

7. **Launcher.** I will launch my rocket from a stable device that provides rigid guidance until the rocket has attained a speed that ensures a stable flight, and that is pointed to within 20 degrees of vertical. If the wind speed exceeds 5 miles per hour I will use a launcher length that permits the rocket to attain a safe velocity before separation from the launcher. I will use a blast deflector to prevent the motor's exhaust from hitting the ground. I will ensure that dry grass is cleared around each launch pad in accordance with the accompanying Minimum Distance table, and will increase this distance by a factor of 1.5 and clear that area of all combustible material if the rocket motor being launched uses titanium sponge in the propellant.	The team will comply with this rule by launching out of the rails provided by NAR at competition.
8. **Flight Safety.** I will not launch my rocket at targets, into clouds, near airplanes, nor on trajectories that take it directly over the heads of spectators or beyond the boundaries of the launch site, and will not put any flammable or explosive payload in my rocket. I will not launch my rockets if wind speeds exceed 20 miles per hour. I will comply with Federal Aviation Administration airspace regulations when flying, and will ensure that my rocket will not exceed any applicable altitude limit in effect at that launch site.	The team will comply with this rule and any determination the Range Safety Officer makes on launch day.
9. **Launch Site.** I will launch my rocket outdoors, in an open area where trees, power lines, occupied buildings, and persons not involved in the launch do not present a hazard and that is at least as large on its smallest dimension as one-half of the maximum altitude to which rockets are allowed to be flown at that site or 1500 feet, whichever is greater, or 1000 feet for rockets with a combined total impulse of less than 160 N-sec, a total liftoff weight of less than 1500 grams and a maximum expected altitude of less than 610 meters (2000 feet).	All team launches will be at NAR/TRA certified events. The Range Safety Officer will have the final say over any rocketry safety issues.
10. **Launcher Location.** My launcher will be 1500 feet from any occupied building or from any public highway on which traffic flow exceeds 10 vehicles per hour, not including traffic flow related to the launch. It will also be	The team will comply with this rule and any determination the Range safety Officer makes on launch day.

no closer than the appropriate Minimum Personnel Distance from the accompanying table from any boundary of the launch site.	
11. **Recovery System.** I will use a recovery system such as a parachute in my rocket so that all parts of my rocket return safely and undamaged and can be flown again, and I will use only flame-resistant or fireproof recovery system wadding in my rocket.	The Recovery team will be responsible for designing and constructing a safe recovery system for the rocket. A safety checklist will be used on launch day to ensure that all critical steps in preparing and packing the recovery system and all necessary components into the rocket are completed.
12. **Recovery Safety.** I will not attempt to recover my rocket from power lines, tall trees, or other dangerous places, fly it under conditions where it is likely to recover in spectator areas or outside the launch site, nor attempt to catch it as it approaches the ground.	The team will comply with this rule and any determination the Range Safety Officer makes on launch day.

Table 10: NAR safety code compliance.

3.3 Team Safety

A team safety meeting will be held prior to any construction, tests, or launches in order to ensure that every team member is fully aware of all team safety regulations as detailed in the team safety manual. Each team member is required to review and acknowledge the safety manual. As revisions are made and released, team members are responsible for remaining up to date with team safety regulations. The team safety manual covers the following topics:

- Lab workshop safety
- Material safety
- Personal Protective Equipment regulations
- Launch safety procedures
- Educational engagement safety
- MSDS sheets
- Lab specific rules

Should a violation to the contract occur, the violator will be revoked of his or her eligibility to access any lab and will be prohibited from attending launches until the safety officer reinstates said member. The violator must review and reconfirm compliance with the safety rules prior to regaining eligibility.

Prior to each launch, a briefing will be held to review potential hazards and accident avoidance strategies. In order to prevent an accident, a thorough safety checklist will be created and will be reviewed on launch day. Once all subsystem checklists are completed, a final checklist must be completed and final approval granted by the safety officer and captain. The safety officer has the right to call off a launch at any time if Kevin determines anything to be unsafe or at a high risk level.

3.4 Local/State/Federal Law Compliance

The team has reviewed and acknowledged regulations regarding unmanned rocket launches and motor handling. Federal Aviation Regulations 14 CFR, Subchapter F, Part 101, Subpart C, Code of Federal Regulation 27 Part 55: Commerce in Explosives; and fire prevention, and NFPA 1127 "Code for High Power Rocket Motors." The previous law compliance is in affect for the following top 4 launch sites listed below in

Field Location	Status	Team Objective
Elizabethtown, Kentucky	1) Pending on waiver approval up to 7,000 ft 2) Less than an hour of travel 3) Moderate field size	1) Ideal field for test flights (possible main launch field) 2) Ideal for travel 3) Ideal for 0-20mph
Bowling Green, Kentucky	1) Pending on waiver approval up to 6,000 ft 2) Less than two hours of travel 3) Moderate field size	1) Ideal field for test flights (possible main/backup launch field) 2) Moderate for travel 3) Ideal for 0-20mph
Manchester, Tennessee	1) Operational to 10,000 ft 2) Only available part of the fall and spring semesters 3) Over 3 hours away 4) Large field size	1) Ideal field for test flights 2) Moderately inconvenient due to travel 3) Ideal for 0-20mph
Memphis, Tennessee	1) Operational to 5,000 ft 2) Available almost every weekend 3) Over 5 hours of travel 4) Small field size	1) To utilize this field as a backup field 2) Not ideal for launches due to travel 3) Ideal for 7mph winds or lower

Table 11: Top launch sites for River City Rocketry.

3.5 Motor Safety

Darryl Hankes, the team mentor, who has obtained his Level 3 TRA certification, will be responsible for acquiring, storing, and handling the teams rocket motors at all times. Team members that have attained a minimum of a Level 2 certification are also permitted to assist in this responsibility. By having obtained a Level 2 certification, the individual has demonstrated that he or she understands the safety guidelines regarding motors. Any certified member of the team that handles or stores the team's motors is responsible for following the appropriate measures. The motors for both test and competition launches will be transported by car to the launch site.

3.6 Safety Compliance Agreement

The University of Louisville River City Rocketry team understands and will abide by the following safety regulations declared by NASA. The following rules will be included in the team safety contract that all team members are required to sign in order to participate in any builds or launches with the team.

1. Range safety inspections of each rocket before it is flown. Each team shall comply with the determination of the safety inspection or may be removed from the program.
2. The Range Safety Officer has the final say on all rocket safety issues. Therefore, the Range Safety Officer has the right to deny the launch of any rocket for safety reasons.
3. Any team that does not comply with the safety requirements will not be allowed to launch their rocket.

4 Technical Design: Variable Drag System

In past years River City Rocketry has utilized a ballast system to achieve its target apogee altitude. While a ballast system is simple, it is subject to variability in motor impulse, rail friction, and weather conditions. As a result, ballasted vehicles often cannot achieve a level of precision in their apogee altitudes greater than ±167 ft (51 m).[1] In order to improve the consistency with which the team can achieve its target apogee, River City Rocketry has begun the development of the Variable Drag System (VDS).

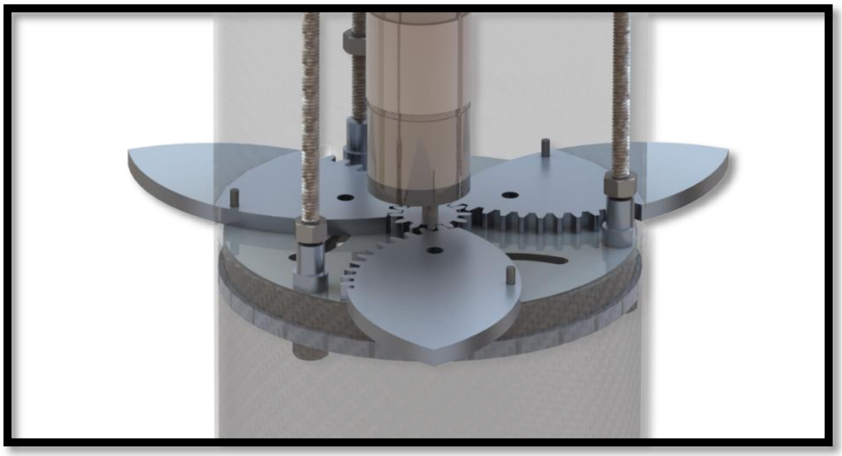


Figure 1: Variable Drag System (VDS) rendering (airframe transparent).

The VDS is set to replace the ballast system as the system responsible for determining the vehicle's apogee altitude and will be able to achieve a target apogee with ±10m accuracy. This will be achieved by dynamically changing the drag force of the vehicle during the coast phase, allowing the VDS to compensate for the variations in burn phase flight characteristics. The VDS varies the drag force on the vehicle by projecting three flat blades into the airstream surrounding the rocket. With the flat faces of the blades perpendicular to the airstream, the VDS is able to increase the projected area of the vehicle by a factor of 1.28 and the coefficient of drag by an estimated factor of 1.35.

Design Overview

The current VDS design uses three aluminum drag blades, actuated by a central gear. The central gear is driven by a single DC electric motor and can be actuated precisely to any position in its range with feedback provided by an encoder. The DC motor is controlled by a main electronic controller which is responsible for both actuation and the reading of sensor data. It uses this sensor data to determine the state of the vehicle and control the flight of the vehicle. This design was

[1] 95% confidence interval based on 27 samples from the NSL 2015-2016 competition flights

arrived at based on results from a system level trade study where multiple alternatives were evaluated.

4.1 System Level Trade Study

Many alternative designs were considered during the development of the VDS. The functionality of the drag control system is critical to the success of the launch vehicle, therefore, extensive research was carried out to ensure the best solution was pursued. Below is a brief introduction into each system that was considered for implementation.

4.1.1.1 *Tri-Aileron Drag System*

One design considered for the drag inducing system for the launch vehicle was a set of tri-ailerons that actuated against the air flow. The tri-ailerons would be connected to a linear actuator via three struts. This design was desirable due to its large increase in projected area. The whole assembly would have been housed in a carbon fiber enclosure and secured to the launch vehicle via epoxy. Each aileron would have hinged about a clevis and pin assembly that would have fastened to the enclosure. An assembly of the tri-aileron drag system is shown below in Figure 2.

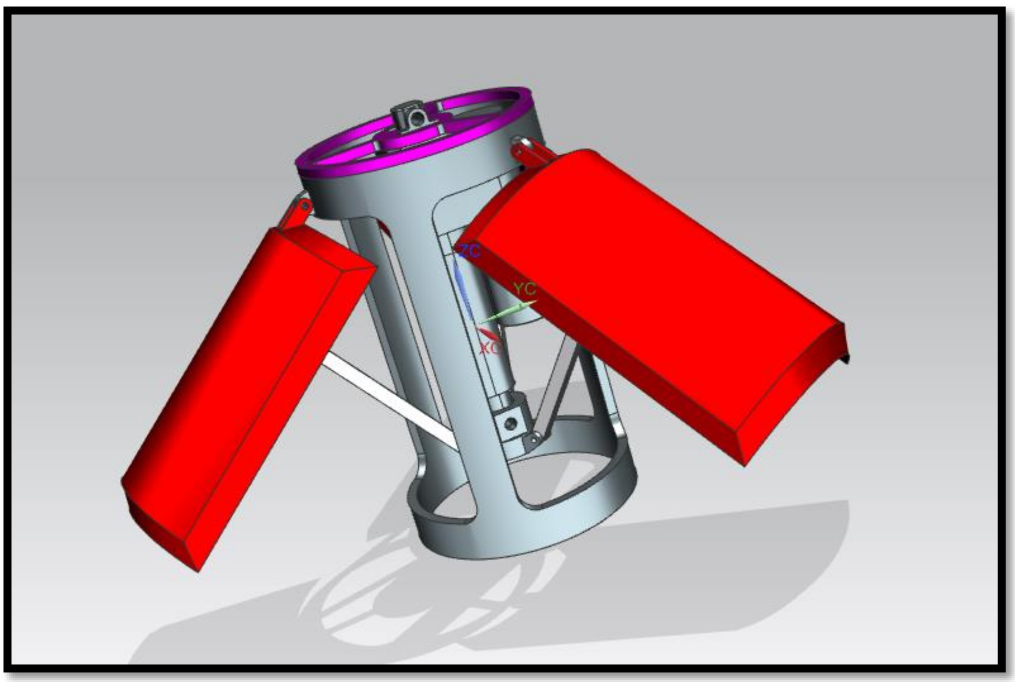


Figure 2: Tri-Aileron Drag System.

4.1.1.2 *Three Bladed VDS*

While the tri-aileron drag system was ideal in terms of increased projected area, it lacked the speed of actuation and control that was desired to get a precise target apogee altitude. The team decided to design the VDS so that the drag inducing blades actuate perpendicular to the airflow instead of against the airflow. A majority of previous air braking designs[2] have had actuating joints that work against the incident airflow. Through the perpendicular actuation of this design, the overall volume is minimized and the motor does not have to directly counteract the drag force. Minimizing the mass of the VDS allows the overall deceleration of the launch vehicle from the VDS to increase. By making the VDS as compact as possible, the overall launch vehicle length is reduced, thus making the launch vehicle more efficient. The entire VDS, including the electronics, is able to fit inside a single 6in. by 12in. carbon fiber coupler, which allows the VDS to be inserted and removed from the launch vehicle.

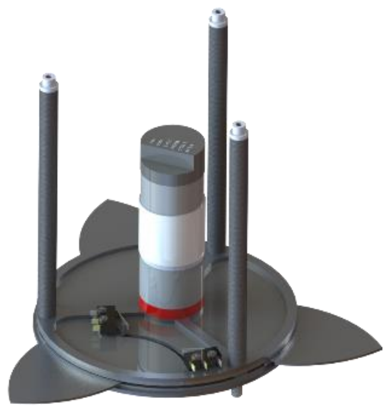


Figure 3: Three bladed VDS rendering.

4.1.1.3 *Six Bladed VDS*

The three bladed VDS was found to have the desired speed of actuation and projected in order to precisely reach a target altitude. However, concern was brought up about the braking power of a three bladed system. Specifically, concern about if the three bladed VDS would be able to induce enough drag to lower the apogee altitude of the launch vehicle from 5,600 feet to 5,280 feet. Before analysis was performed, a six bladed VDS was designed to combat this issue. A rendering of the six bladed VDS is shown below in Figure 4.

[2] Refers to such designs as Project Aquila, USU Chimaera Project, and Project Artemis

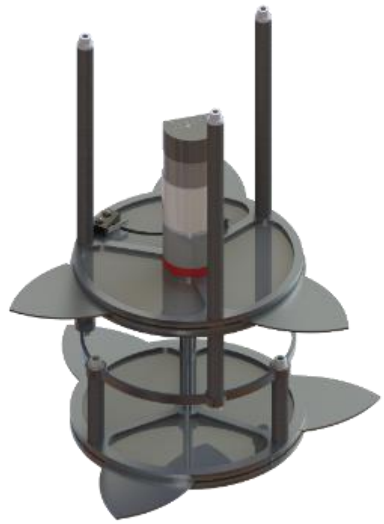


Figure 4: Six bladed VDS rendering.

The six bladed VDS utilized the same mechanical actuation of the three bladed VDS, however, increases the projected area by a factor of 2. The D shaft of the DC motor would have been connected to a coupling that extended the D shaft another 6 inches to the central gear of the slave blade system. Carbon fiber tubing would have attached the primary blade system to an aluminum offsetting centering ring that would have offset the blades of the slave blade system from the blades of the primary system. All six blades would have been driven by the same motor. However, the overall mass of the system was increased by a significant factor when compared to the three bladed VDS.

A Kepner-Tregoe Trade Study was created to compare all alternative designs considered for implementation as the drag control system. A Kepner-Tregoe Trade Study comparing all three designs is shown below in Table 12.

Drag Control System							
Options:		Three Blade Variable Drag System		Six Blade Variable Drag System		Three Drag Aileron System	
Mandatory Requirements							
Located aft of the center of gravity of the launch vehicle		Yes		Yes		Yes	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Actuation Speed (0-10)	20.00%	10	2	9	1.8	4	0.8
Projected Area (0-10)	20.00%	4	0.8	8	1.6	10	2
Continuous Actuation (0-10)	20.00%	10	2	10	2	0	0
System Simplicity (0-10)	5.00%	8	0.4	7	0.35	4	0.2
Laminar Fin Air Flow (0-10)	10.00%	8	0.8	6	0.6	5	0.5
Manufacturability (0-10)	5.00%	9	0.45	7	0.35	9	0.45
Price (0-10)	5.00%	9	0.45	8	0.4	6	0.3
Mass (0-10)	15.00%	9	1.35	7	1.05	7	1.05
Total Score		**8.25**		**8.15**		**5.3**	

Table 12: VDS system level trade study table.

As seen from the Kepner-Tregoe trade study above, the three blade VDS is the optimal choice for the drag control system for the launch vehicle. Due to its speed of actuation, mass, and system simplicity, the three bladed VDS is best suited to quickly and precisely adjust the drag of the launch vehicle.

4.2 Derivation of Requirements

Requirement 1.1 from the SOW requires that the vehicle be delivered to 5,280 ft. (1609 m) AGL. In order to fulfill requirement 1.1, the team has derived several requirements that define the design parameters of the VDS system. The high-level functional requirements for the VDS system under SOW requirement 1.1 are shown below in Figure 5.

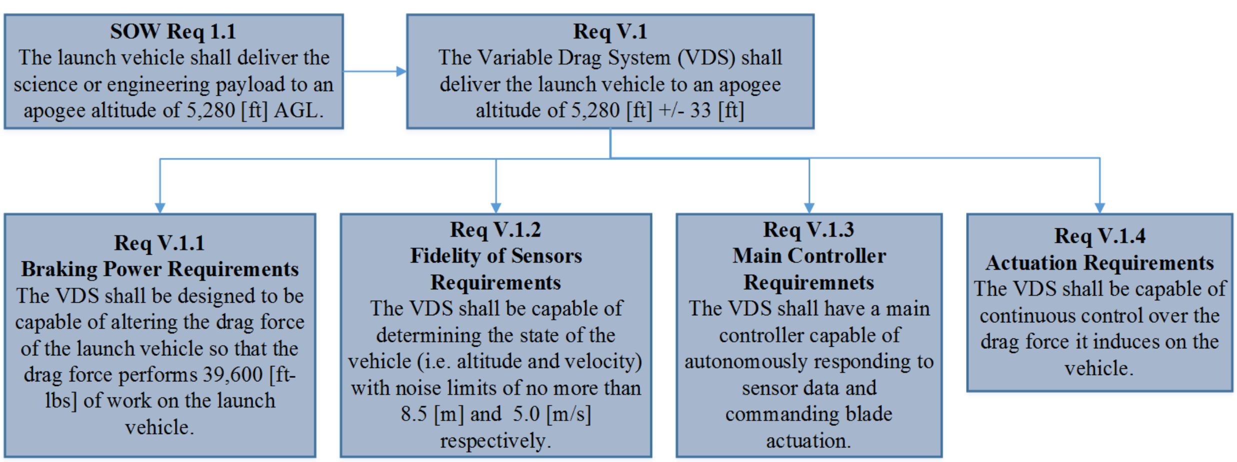


Figure 5: VDS functional requirements flowdown.

These high-level functional requirements outline what the VDS must accomplish in order to fulfill its parent requirement, SOW requirement 1.1. These requirements define a system that is capable of providing an acceptable amount of drag force, capable of determining the amount of drag it needs to induce, and capable of inducing drag in a controlled fashion. These requirements and their methods of verification are shown below in Table 13.

Requirement Number	Requirement	Method of Verification
1.1	The vehicle shall deliver the science or engineering payload to an apogee altitude of 5,280 ft. (1609 m).	Test Flight altimeters will record the apogee of the vehicle during both test launches and the competition launch.
V.1	The variable drag system shall be implemented in the vehicle to deliver the vehicle to an apogee of 5,280 ft. $\pm$ 33 ft. (1609 m $\pm$10 m)	Test Flight altimeters will record the apogee of the vehicle during both test launches and the competition launch.
V.1.1	The VDS shall be designed to be capable of altering the drag force of the launch vehicle so that the drag force performs 39,600 [ft-lbs] of work on the launch vehicle.	Test Test launches will verify the amount work the VDS induces on the launch vehicle.
V.1.2	The VDS shall be capable of determining the state of the vehicle (i.e. altitude and velocity) with noise limits of no more than $\pm$ 8.5 m and $\pm$ 5.0 m/s respectively.	Test This requirement will be verified in December flight testing. The recorded flight data will be analyzed to determine if the state noise is acceptable.
V.1.3	The VDS shall have a main controller capable of autonomously responding to sensor data and commanding blade actuation.	Inspection The main controller datasheet will be inspected to determine if it is capable of sensor input, computation, and pin I/O.
V.1.4	The VDS shall be capable of continuous control over the drag force it induces on the vehicle.	Inspection The VDS will utilize an actuating mechanism that can extend and retract the drag blades at any point during its actuation.

Table 13: VDS functional requirements.

4.2.1 Acceptable Braking Power Requirements Derivations

In order for the VDS to perform its mission, it must be capable of reducing the apogee of the vehicle from 5,600 feet to the target apogee. In order to do this with the given mass of the vehicle at 45 lbs, the VDS must be capable of performing no less 39,600 ft-lbs of work. The requirements in this section all contribute to the VDS's ability alter the drag force so that the drag performs this amount of work on the launch vehicle.

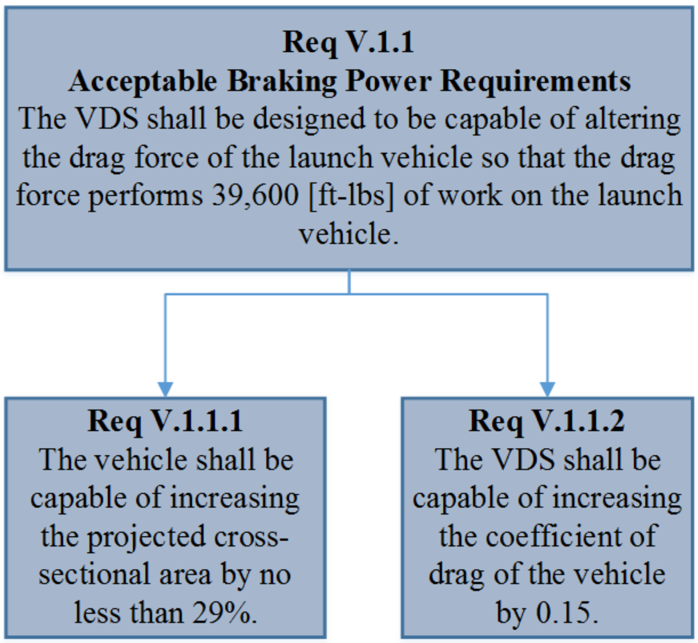


Figure 6: Braking power requirements flowdown.

Requirement Number	Requirement	Method of Verification
V.1.1	The VDS shall be designed to be capable of altering the drag force of the launch vehicle so that the drag force performs 39,600 ft-lbs of work on the launch vehicle.	Test Multiple test launches will be conducted to determine the amount of work the VDS is capable of performing.
V.1.1.1	The vehicle shall be capable of increasing the projected cross-sectional area by no less than 29%.	Inspection This requirement was verified by utilizing CAD programs to determine the increase in projected area of the launch vehicle after actuation.
V.1.1.2	The vehicle shall be capable of increasing the coefficient of drag of vehicle by no less than 0.15	Test This requirement will be verified during December flight testing and wind tunnel testing.

Table 14: VDS braking power requirements.

Derivation of Requirement V.1.1

39,600 ft-lbs was determined by analyzing the change in desired apogee altitude of the launch vehicle with VDS involvement and without VDS involvement. The launch vehicle will be able to achieve an apogee altitude of 5,600 feet, so the VDS will have to lower the apogee altitude of the launch vehicle by 320 feet to an altitude 5,280 feet at a minimum in order for the mission to be considered a success. The VDS will be required to lower the apogee altitude of the launch vehicle by 880 feet, is the minimum required change in apogee altitude by a factor of 2.75. With a factor of 2.75, the VDS will be able to drop the apogee altitude of the launch vehicle from 5,600 feet to 4,720. By having the ability to drop the apogee altitude much lower than 5,280 feet, the VDS has greater control over the apogee altitude as well as minimizes the risk of undershooting 5,280 feet solely due to launch weather conditions. The work done on the launch vehicle by drag that is induced by the VDS, W, was calculated using

$$W = \Delta mgh \quad (1)$$

where m is the overall mass of the launch vehicle, g is the gravitational constant, and h is the apogee altitude of the launch vehicle. The work was determined to be 39,600 ft-lbs.

Derivation of Requirement V.1.1.1

A prototype of the VDS was tested and proved to be effective in inducing enough drag on the launch vehicle to reach a specific altitude. The VDS prototype increased the projected area of the launch vehicle by 29%, therefore in order to improve upon the prototype and gain more control over the apogee altitude of the launch vehicle, a minimum increase in projected area of the launch vehicle due to the VDS was set at 29%.

Derivation of Requirement V.1.1.2

By conducting summer prototype test launches, the preliminary VDS design was determined to alter the coefficient of drag of the launch vehicle by approximately 0.15. With a change of coefficient of drag of 0.15, the VDS was able to induce enough drag to achieve a specific target altitude.

4.2.2 Fidelity of Sensors Requirements Derivations

In order for the VDS to perform its mission, the system must be capable of determining the state of the vehicle (i.e. altitude and velocity). These performance requirements ensure that the VDS is able to determine the state of the vehicle with high fidelity. These requirements and their methods of verification are shown below in Figure 7 and Table 15.

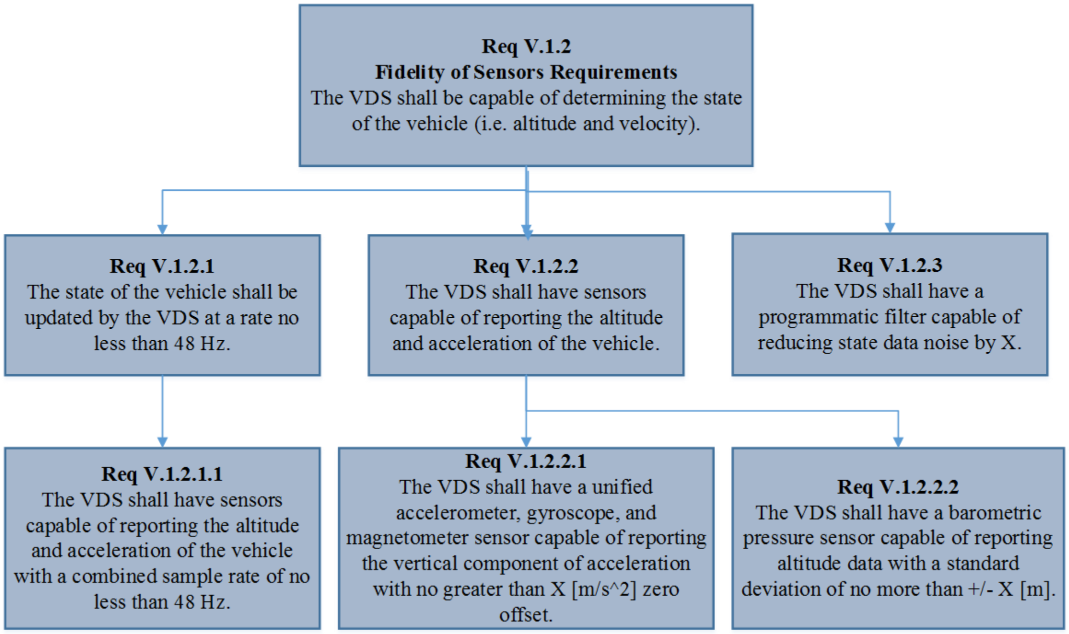


Figure 7: Sensor fidelity requirement flowdown.

Requirement Number	Requirement	Method of Verification
V.1.2	The VDS shall be capable of determining the state of the vehicle (i.e. altitude and velocity) with noise limits of no more than $\pm$ 8.5 m and $\pm$ 5.0 m/s respectively.	<u>Test</u> This requirement will be verified in December flight testing. The recorded flight data will be analyzed to determine if the state noise is acceptable.
V.1.2.1	The state of the vehicle shall be updated by the VDS at a rate no less than 48 Hz.	<u>Demonstration</u> This requirement will be verified preflight by running the flight software and demonstrating the VDS state update frequency.
V.1.2.1.1	The VDS shall have sensors capable of reporting the altitude and acceleration of the vehicle with a combined sample rate of no less than 48 Hz.	<u>Demonstration</u> This requirement will be verified preflight by running the flight software and demonstrating the VDS state update frequency.
V.1.2.2	The VDS shall have sensors capable of reporting the altitude and acceleration of the vehicle.	<u>Inspection</u> The datasheets of the chosen sensors will be inspected to ensure that they are capable of reporting altitude and acceleration.
V.1.2.2.1	The VDS shall sample the vehicle's altitude over 0.5211 sec $\pm$ 0.05 sec for the purpose of differentiating to find velocity.	<u>Inspection</u> The VDS software will be demonstrated in a ground test to sample the last 0.52 sec $\pm$ 0.05 sec

		for the purpose of differentiating to find velocity.
V.1.2.2.2	The VDS shall have a barometric pressure sensor capable of reporting altitude data with a noise limit of no more than $\pm$ 4.1 m.	Demonstration This requirement will be verified preflight by running the flight software and demonstrating the VDS altitude noise limit.
V.1.2.3	The VDS shall have a programmatic filter capable of reducing velocity data noise by no less than 25% as an added safety measure.	Test This requirement will be verified in December flight testing. The recorded flight data will be analyzed to determine if the improvement in state noise is acceptable.

Table 15: Sensor fidelity requirements.

Derivation of Requirement V.1.2

In order for the VDS to deliver the vehicle to 1 mile $\pm$ 33 ft. it must be capable of determining the vehicle's state with precision. The measures of precision in this case are noise limits, defined as

$$noise\ limits = 1.66 * std(noise)$$

The noise limits for the state of the vehicle will need to be no greater than $\pm$ 8.5 m and $\pm$ 5.0 m/s for altitude and velocity respectively. These requirement parameters have been derived through evaluating failure modes in simulation. Simulation was chosen as the method of derivation due to the unique effects of state noise on this specific system.

The simulation tested 225 cases in which the altitude noise limit was varied in the range 0-70 and the velocity data noise limit was varied within the range 0-12. Noise limits were varied by applying uniformly distributed noise to the state signals feeding into the blocks that simulate the VDS electronics. All of these cases were evaluated to be either passed or failed based on the $\pm$ 10 m criteria outlined in requirement V.1. The results of the 225 cases can be seen below in Figure 8.

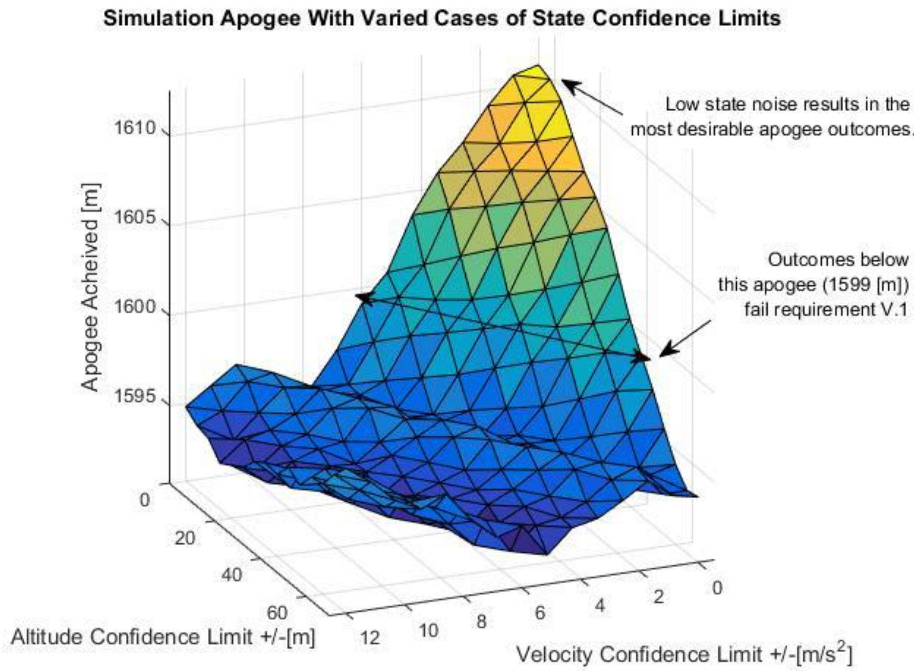


Figure 8: Simulation apogee with varied cases of state noise.

From the plot it can be seen that optimal performance occurs when there is zero state noise (yellow). Performance of the VDS begins to degrade as the uniformly distributed noise is increased and lower altitudes are achieved.

The failure modes of this system are taken to be all cases of state noise limit combinations that result in an apogee that was not within 10m of 1609 m. These cases were eliminated as potential requirement thresholds for state signal noise limits. All cases that are solutions to the intersection of the above plot and the plane apogee (z) =1599m are acceptable requirement thresholds with the exception of the zero intercepts which will be unachievable in practice. For this reason, the requirement threshold was taken to be the most nominal case among all the solutions where the velocity noise limit is $\pm 5.0\,\frac{m}{s^2}$ and the altitude noise limit is $\pm\, 8.5\,\frac{m}{s^2}$.

Derivation of Requirement V.1.2.1

The state of the vehicle shall be updated by the VDS at a sample rate no less than 48 Hz. This number was derived from the Nyquist-Shannon sampling theorem that states that a sufficient sample rate is one that samples the given signal (the vehicle's state) with a frequency twice that of the highest frequency present in the signal. The highest frequency present in the state of the vehicle being any velocity fluctuations due to the actuation of drag blades into the airstream. The minimum required sampling rate is therefore twice that of the blade actuation frequency.

The blade actuation frequency was taken to be the frequency at which the drag blades are capable of inducing measurable drag on the vehicle: (i.e the frequency at which the drag blades can oscillate to 1/8 extension and back). This frequency, which is expected to be 12 Hz, was then multiplied by two to achieve the minimum required sample rate[3]. An additional factor of safety of two was then applied to this number to achieve a minimum sample frequency of 48 Hz.

Derivation of Requirement V.1.2.1.1

The VDS shall have sensors capable of updating the state of the vehicle at a rate no less than 48 Hz. In order for the state of the vehicle to be updated at a rate that satisfies requirement V.1.2.1, the VDS must have sensors capable of sampling the vehicle state at a rate no less than 48 Hz. This requirement excludes interpolation methods. This requirement also allows for the combination of sensors to satisfy this requirement. The sensors do not need to satisfy this requirement individually.

Derivation of Requirement V.1.2.2

The VDS shall have sensors capable of reporting the altitude and acceleration of the vehicle. Despite that the VDS requires altitude and velocity in order to determine the drag needed, trade studies have shown that a sensor scheme comprised of altitude and acceleration based sensors is ideal.

Derivation of Requirements V.1.2.2.1 and V.1.2.2.2

The VDS shall sample the vehicle's altitude over 0.5211 [sec] $\pm$ 0.05 [sec] for the purpose of differentiating to find velocity and the VDS shall have a barometric pressure sensor capable of reporting altitude data with a noise limit of no more than $\pm$ 4.1 m. These requirements combined satisfy the requirement on the velocity noise limit. Because velocity is primarily found by differentiating altitude readings, noise on the velocity is primarily due to noise on the altitude signal. This noise can be mitigated by differentiating further into the past. To find the amount of time to differentiate into the past and the amount of noise acceptable on altitude signal, 400 cases were simulated in which these two variables were varied over a range. The results of these simulations can be seen below in Figure 9.

[3] The expected frequency of 12 Hz is an extrapolation based on tests with a DC motor similar to the proposed DC motor. The Neverrest 60 gear motor has been demonstrated to be capable of fully deploying the drag blades within 1 second. The Neverrest 40 gear motor is expected to be faster by a factor of the two motor's gear ratios.

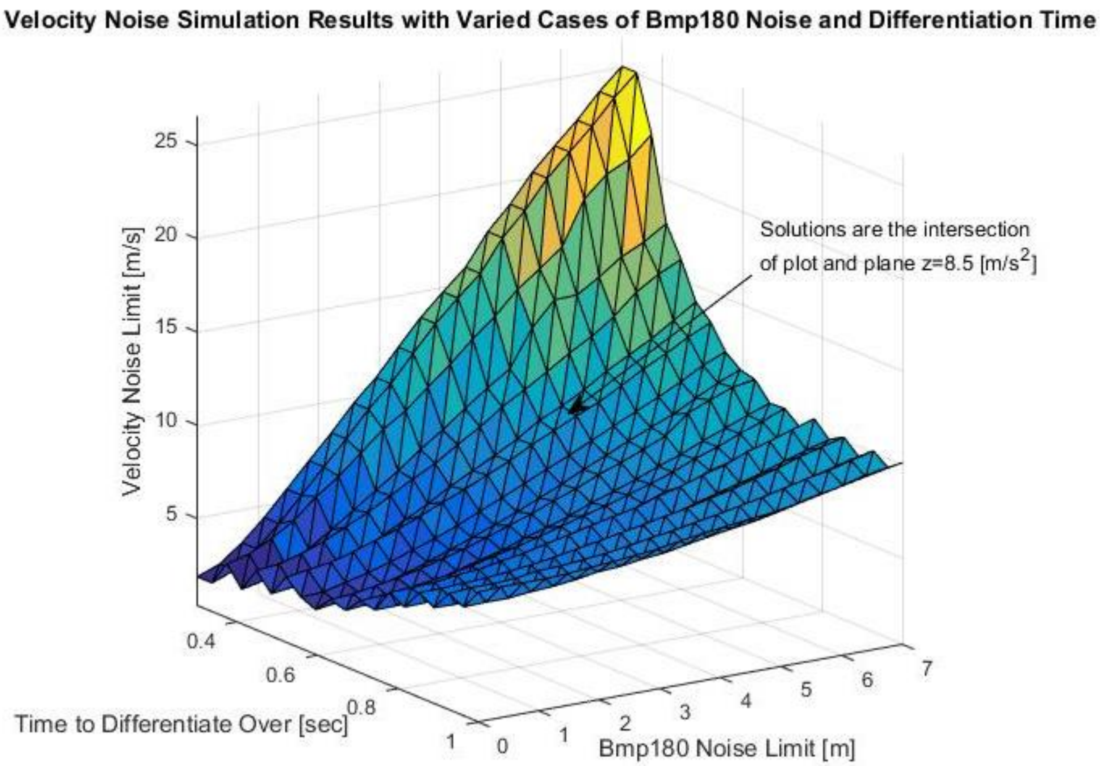


Figure 9:Velocity noise simulation results with varied cases of Bmp180 noise and differentiation time.

The above plot shows that cases of differentiation time and Bmp180 noise limits that result in velocity noise limits below 8.5 $[\frac{m}{s^2}]$ fail requirement V.1.2. The values for this requirement were found by solving for all cases that intersect with the z=8.5 plane. From these solutions, a nominal case was chosen resulting in a Bmp180 noise limit no greater than $\pm$ 4.1 [m] and a differentiation time of 0.5211 [sec].

Derivation of Requirement V.1.2.3

The VDS shall have a programmatic filter capable of reducing velocity data noise by no less than 25% as an added safety measure. To ensure a factor of safety on the state noise, particularly the velocity, a filter will be added that provides an added factor of safety of 1.25.

4.2.3 Main Controller Requirements Derivations

In order for the VDS to achieve its mission it must be capable autonomously responding to sensor data and commanding drag blade actuation. To facilitate this, the VDS will have a main controller that has robust computing power and sufficient I/O ports. The power requirements for this main controller are also included in this section. The main controller performance requirements and their methods of verification are outlined below in Figure 10 and Table 16.

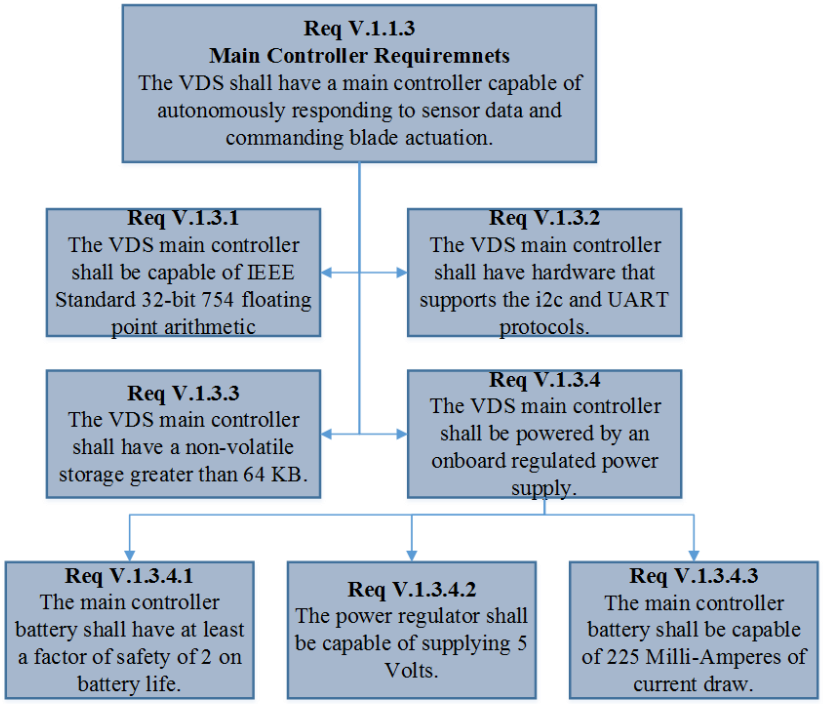


Figure 10: VDS main controller requirements flowdown.

Requirement Number	Requirement	Method of Verification
V.1.3	The VDS shall have a main controller capable of autonomously responding to sensor data and commanding blade actuation.	<u>Inspection</u> The main controller datasheet will be inspected to determine if it is capable of sensor input, computation, and pin I/O.
V.1.3.1	The VDS main controller shall be capable of IEEE Standard 754 32-bit floating point arithmetic.	<u>Inspection</u> The main controller datasheet will be inspected to determine if it is capable of IEEE Standard 754 32-bit floating point arithmetic.
V.1.3.2	The VDS main controller shall have hardware that supports the i2c and UART protocols.	<u>Inspection</u> The main controller datasheet will be inspected to determine whether or not it is capable of i2c and UART communication.
V.1.3.3	The VDS main controller shall have a non-volatile storage greater than 64 kB.	<u>Inspection</u> The main controller datasheet will be inspected to determine whether or not it has non-volatile storage greater than 64 kB.
V.1.3.4	The VDS main controller shall be powered by an onboard regulated power supply.	<u>Demonstration</u> The main controller will perform system operation with a regulated external power

		supply before scheduled test flights. (see Project Plan section)
V.1.3.4.1	The main controller battery shall have at least a factor of safety of 2 on battery life.	Demonstration The battery life safety margin will be verified by monitoring the battery charge during extensive operation.
V.1.3.4.2	The power regulator shall be capable of supplying 5 Volts.	Demonstration The power regulator will be demonstrated by outputting 5V from an external supply input.
V.1.3.4.3	The main controller battery shall be capable of 225 Milli-Amperes of current draw.	Demonstration The current draw will be quantified under full system operation. The recorded current shall be within the safety margin

Table 16: VDS main controller requirements.

Derivation of Requirement V.1.3.1

The VDS main controller must be capable of performing IEEE Standard 754 32-bit floating point arithmetic because of the need for a high level of fidelity in real-time calculations. Integer values were deemed unacceptable due to their inherent quantization error. The pseudo-floating point used on the Arduino platform was deemed unacceptable due to its propensity for error as well.

Derivation of Requirement V.1.3.2

In order for the VDS to communicate with the necessary sensors it must be capable of the communication protocols i2c and UART. These sensors communicate exclusively in these protocols.

Derivation of Requirement V.1.3.3

The VDS main controller must have non-volatile storage greater than 64 kB due to its need for program space. A prototype of the software was shown to occupy 24 kB of program memory and it is expected that an additional 15 kB of code and libraries will be incorporated into the VDS software package by CDR. Given this projection, 64 kB will be an acceptable amount of program storage space with an additional margin for expansion if necessary.

Derivation of Requirement V.1.3.4.1

The power supply of the VDS Electronics needs to sustain operation time within a factor of two-safety margin. The operation time considers static power consumption while waiting on the launch pad and active power consumption during the flight of the rocket. A factor of safety of two ensures the microcontroller will have sufficient power through the entire launch process (see Electrical Design section).

Derivation of Requirement V.1.3.4.2

An external battery will sustain the operation of the VDS Electronics. The Teensy 3.6 operates on an external supply between 3.6 and 6.0 [V]. The power regulator is required to provide a 5V source that will power the control electronics.

Derivation of Requirement V.1.3.4.3

The Teensy 3.6 system is calculated to operate on a 225 [mA] current draw safety factor with all components connected. The operating current was derived from the consumption of each major chip connected to the Teensy 3.6 microcontroller (see Electrical Design section). The power regulator will be able to accommodate for the safety factor current draw to prevent damage or component failure.

4.2.4 Actuation Requirements Derivations

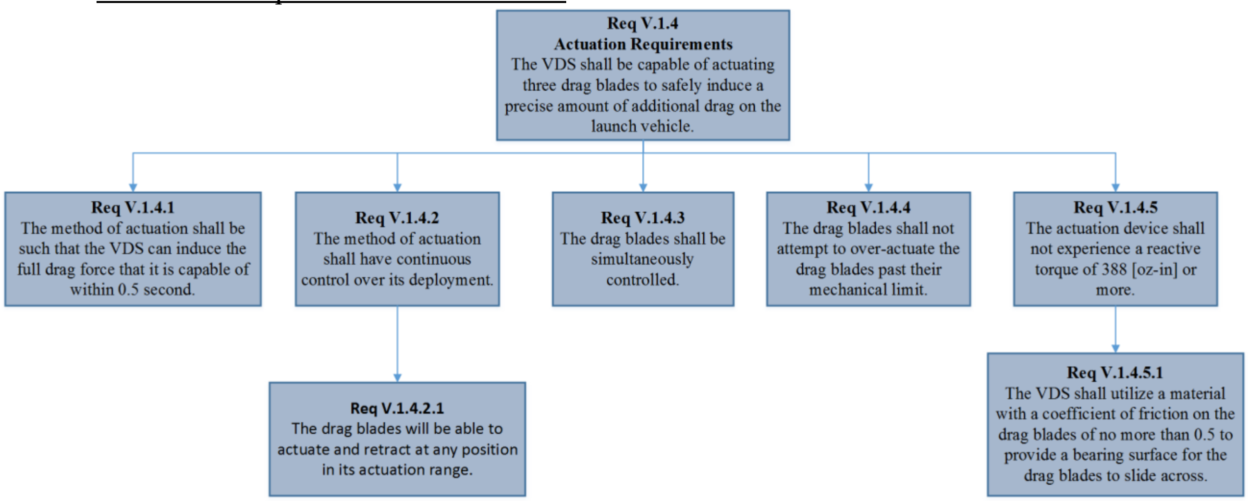


Figure 11: VDS actuation requirements flowdown.

Requirement Number	Requirement	Method of Verification
V.1.4	The VDS shall be capable of actuating three drag blades to safely induce a precise amount of additional drag on the launch vehicle.	Test The VDS will be tested through multiple test launches to verify the safety and effectiveness of the system.
V.1.4.1	The method of actuation shall be such that the VDS can induce the full drag force that it is capable of within half of a second.	Analysis The actuation device will be chosen so that the actuation time of the drag blades will be minimized.
V.1.4.5	The DC motor shall not experience a reactive torque of 388 [oz-in] or more.	Analysis The drag blades actuate perpendicular to the air flow during flight, thus reducing the torque that the motor directly has to contract in order to actuate the drag blades. The friction force will be calculated to ensure proper motor selection.
V.1.4.5.1	The VDS shall utilize a material with a coefficient of friction on the drag blades of no more than 0.5 to provide a bearing surface	Demonstration Delrin Acetal Resin was chosen for the bearing surface due to its coefficient of

	for the drag blades to slide across.	friction of 0.3 on aluminum as well as its stiffness.
V.1.4.2	The method of actuation shall have continuous control over its deployment.	<u>Demonstration</u> The VDS will utilize a gear system to actuate the drag blades, thus giving the motor the ability to quickly actuate or retract the drag blades to drastically adjust the drag force at any point during the ascent of the flight.
V.1.4.2.1	The drag blades shall be able to retract and actuate based on DC motor feedback.	<u>Demonstration</u> The VDS encoder will be demonstrated to report the position of the drag blades.
V.1.4.3	The drag shall be simultaneously controlled.	<u>Inspection</u> The drag blades shall actuate via the meshing between radial gear teeth located on the drag blades and the central gear to reduce the quantity of moving parts as well as simplify the system.
V.1.4.4	The drag blades shall not attempt to over-actuate the drag blades past their mechanical limit.	<u>Demonstration</u> Limit switches shall communicate complete actuation and retraction of the drag blades to the control system.

Table 17: VDS actuation requirements.

Derivation of Requirement V.1.4

In order for the launch vehicle to achieve an apogee altitude of 5280 [ft] ± 33[ft], the drag blades of the VDS need to be able to actuate and retract quickly and safely. By having the ability to precisely adjust the drag that the launch vehicle experiences during the ascent of the flight, the launch vehicle will able to achieve a specific apogee altitude with a high level of accuracy.

Derivation of Requirement V.1.4.1

Due to the level of precision in apogee altitude that the VDS is designed to achieve, the drag blades need to be able to actuate quickly. The VDS is designed to completely deploy the drag blades in less than half of a second. The quick actuation of the drag blades allows the VDS to take full advantage of the control scheme, which is able to real-time record the motion of the launch vehicle during the ascent of the flight.

Derivation of Requirement V.1.4.5

In order to choose the optimal DC motor, the maximum reaction torque was calculating by performing analysis on the dynamic loading of the VDS. In order to calculate the maximum torque required to actuate the drag blades in the VDS, the drag force that each blade experiences was determined using

$$D = \frac{C_d A \rho v^2}{2} \tag{1}$$

where C_d is the coefficient of drag, A is the projected area, v is the velocity, and ρ is the air density. After the maximum drag force is calculated, it is substituted into equation

$$f_k = D\mu \tag{2}$$

in order to calculate the friction force between each drag blade and the top Delrin plate. f_k is the friction force and μ is the coefficient of friction between aluminum and Delrin. After the friction force is computed, it is then substituted into

$$\tau = f_k r \tag{3}$$

in order to calculate the torque required for the motor to actuate one drag blade where T is torque and r is the distance from the centroid of the friction force to the contact point on the teeth of the servo gear. Using equations (1) through (3), the maximum torque the motor will have to overcome to actuate the drag blades with a factor of safety of 2 and a gear inefficiency of 70% is 358 oz-in. The DC motor is required to have a minimum stall torque of 358 [oz-in] to completely actuate the drag blades through a central gear.

Derivation of Requirement V.1.4.5.1
By choosing a material with a low coefficient of friction on aluminum, the speed of actuation of the VDS can be maximized. Delrin Acetal Resin was chosen for the bearing surface of the drag blades because of its low coefficient of friction on aluminum, which is rated at 0.3.

Derivation of Requirement V.1.4.2
To achieve a high level in precision in apogee altitude of the launch vehicle, the drag blades of the VDS need to have the ability to continuously adjust the drag of the launch vehicle.

Derivation of Requirement V.1.4.2.1
Motor feedback is required to achieve continuous control over the blade position. The control systems will use motor feedback to ensure precise actuation position of the drag blades.

Derivation of Requirement V.1.4.3
To reduce the risk of failure, the VDS was designed to be a robust system with as few moving parts are possible. To facilitate this, the VDS was designed to control all three drag blades simultaneously through a single gear which is controlled by a DC motor.

Derivation of Requirement V.1.4.4
To ensure that the VDS functions properly, redundant feedback was added to prevent over-torqueing of the motor through actuating or retracting past the drag blades mechanical limit. Limit switches that indicates when the drag blades are completely actuated and retracted were added to provide additional feedback to the control system.

4.3 Design

4.3.1 Electrical Design

Since proposal, alternative solutions have been formally evaluated in the form of trade-studies. These trade-studies have contributed to several reconsiderations such as the choice of main controller and the choice of power regulator. Additionally, several trade-studies have also confirmed many proposed design decisions such as the sensor scheme. The resulting design decisions have resulted in a new iteration of VDS electronics with the following main components:

- Teensy 3.6 Microcontroller
- LM7805 Linear Regulator
- BTN7960B Half Bridge Motor Drive (x2)
- BNO055 9-axis Absolute Orientation Sensor
- BMP180 Barometric Pressure Sensor

The components above provide the VDS accurate sensor readings and regulated circuit power. The hardware of the VDS Electronics will be on custom designed Printed Circuit Boards (PCBs). The whole system consists of two separate PCBs. The first PCB contains system sensors, the motor controlling circuit, and the Teensy 3.6 Microcontroller. The second PCB will contain the power conversion circuit, accessible system signals, circuit protection, power switches, power indication, and battery terminal connectors. The PCBs will stack on top of each other to minimize space.

4.3.1.1 *Main Controller*

The Teensy 3.6 main controller provides the VDS with sufficient data processing and storage capabilities under low power conditions. The Teensy uses a 32 bit 180 MHz ARM Cortex-M4 processor and floating-point unit that is capable of performing data computations from the sensors of the system. The Teensy has both volatile and non-volatile memory installed to capture sensor values. The Teensy contains 62 input and output pins that are capable of providing Pulse Width Modulation (PWM), I2C communication, and UART serial communication. The input and output will be used to control motor actuation and data collection.

The VDS Main Controller was chosen with two overarching categories: precise data handling and overall system integration. Four different controllers were on a basis of these categories. Table 18 below is a Kepner-Tregue trade study comparing the Teensy 3.6, Raspberry Pi, FPGA, and Arduino Pro Mini options:

VDS Main Controller									
Options:		Teensy 3.6		Raspberry Pi		FPGA		Arduino Pro Mini	
Mandatory Requirements									
Flash Memory > 64 Kb		Yes		Yes		Yes		No	
i2c and UART Capabilities		Yes		Yes		Yes		Yes	
Real Floating Point		Yes		Yes		Yes		No	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Development Environment Convenience (0-10)	0.05	8	0.4	7	0.35	2	0.1	8	0.4
General Performance (0-10)	0.15	6	0.9	4	0.6	10	1.5	1	0.15

Available Documentation (0-10)	0.2	7	1.4	8	1.6	2	0.4	8	1.6
Affordability	0.1	8	0.8	7	0.7	2	0.2	8	0.8
No Additional Hardware for Data Storage	0.1	10	1	10	1	0	0	0	0
Simplicity	0.19	7	1.33	4	0.76	2	0.38	8	1.52
Ease of Implementation	0.21		0	3	0.63	3	0.63	8	1.68
Total Score			5.83		5.64		3.21		6.15

Table 18: VDS main controller trade study table.

The results show that the Teensy 3.6 is the most suitable controller. The Teensy 3.6 satisfies the mandatory system requirements and provides additional system benefits. These benefits include low power consumption and ease of integration. The Teensy will operate under a much lower power consumption compared to the opposing Raspberry Pi. This will increase battery life and eliminate the need for a high-constrained power regulator system. The Teensy 3.6 also provides an ease of integration compared to the FPGA alternative. The Teensy 3.6 is compatible with the familiar Arduino programming environment. This will allow a smooth transition from the old Arduino-based design to the new Teensy-based design.

Outlook

The new Teensy-based design will be functionally verified by operating the VDS with the initial arduino script. This will quantify the basic operation of the new Teensy 3.6 microcontroller.

4.3.1.2 *Power Design*

The power source of the VDS electronics is responsible for supporting the sensors, control circuits, microcontroller and the DC motor. The limitations when selecting the right power source include system operation time and component consumption. The run time of the electronics will insure continuous operation during the full launch process. The current consumption of the system determines the battery to use. Table 19 below shows the resulting current consumption for the main components of the VDS:

Device	Current Consumption [mA]
Pressure Sensor	0.0105
Half Bridge	0.15
Bno055	12.3
Teensy 3.6 processor	99.85
Total Current	112.4605
With factor of safety = 2	224.921

Table 19: VDS maximum current consumption.

The total current directly relates to the runtime of the system. The factor of safety ensures the VDS Electronics sufficient time to operate continuously. The microcontroller will provide the necessary current to the system. The Teensy 3.6 microcontroller operates with the range of 3.6 volts to 6 volts. Table 20 shows a comparison between potential batteries that can sustain the current draw:

Current (mA) w/ Factor of Safety = 2	Battery Rating (mAh)	Operation Time (Hours)	Cost (Dollars)
224.921	180	0.80	2.9
	350	1.56	4.5
	500	2.22	5.55
	800	3.56	6.99
	1000	4.45	7.5
	1300	5.78	9.9

Table 20: VDS battery comparisons.

The battery comparison table shows that factor of safety of two can be affordably achieved. A regulated supply will output a steady voltage to power the Teensy 3.6. The table below is a Kepner-Tregoe trade study showing the options considered for the battery power regulator:

VDS Power Regulator							
Options:		Buck Converter		Boost Converter		Linear Regulator	
Mandatory Requirements							
Output Current Limit > .25A		Yes		Yes		Yes	
Able to Convert 5V		Yes		Yes		Yes	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Development Convenience (0-10)	0.1	8	0.8	8	0.8	10	0.1
Electrical Safety Margin (0-10)	0.45	9	4.05	9	4.05	9	4.05
Power Efficiency (0-10)	0.15	9	1.35	8	1.2	6	0.9
Affordability (0-10)	0.3	3	0.9	1.5	0.45	10	3
Total Score		0.327944573		0.300230947		0.37182448	

Table 21: VDS power regulator Kepner-Tregoe table.

The trade study shows that the linear regulator is the best option for the VDS Electronics. The current draw and voltage output requirements can be achieved with all options. The power efficiency is related to the heat dissipation of the regulator and battery lifetime. The linear regulator score on power efficiency is acceptable due to the overall low current draw of the system. The low current draw results in negligible effects in heat and battery lifetime. The other characteristics to consider are electrical safety margin and affordability. All regulator options provide a wide margin of safety in terms of current and voltage ratings. The linear regulator is rated for an output current of 1.5 [A] and an input voltage of 7 – 25 [V]. This is within system ratings. The cost of the regulator is the deciding factor. The linear regulator costs $0.95 while the buck and boost converters cost more than $10 for a single chip.

Outlook
The external power regulation will be tested by powering the new Teensy 3.6 system load. This test will quantify circuit characteristics.

4.3.1.3 *Wiring and Harnessing*

The VDS Electronics is contained on two Printed Circuit Boards (PCBs). The bottom PCB contains the control electronics. This control PCB will drive the motor, collect data through sensors, and house the Teensy 3.6 microcontroller. The top PCB is the panel board. The panel PCB will regulate battery power, contain circuit protection, and provide external component signals. The block diagram below shows the layout of the VDS Electronics:

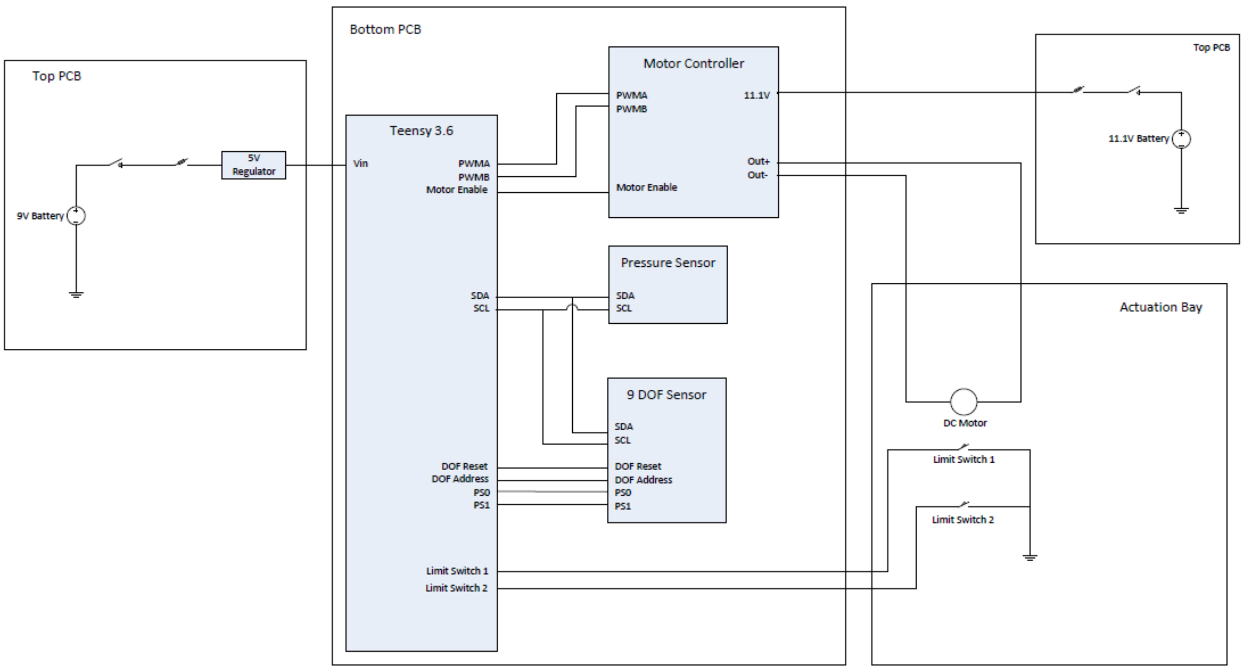


Figure 12: Block diagram of VDS Electronics.

The control and panel PCBs are contained in a separated bay from the DC motor and the actuation blades. The separate bays provide isolation from the actuating blades affecting pressure sensor readings. The panel PCB will receive battery power through two connectors and contain power switches and fuses for each supply. The power is routed down to the bottom control PCB to the necessary components. The motor and limit switch signals will propagate from the control PCB to the panel PCB through a detachable connector to the actuation bay. A connector attached to the separating bulk plate will provide separation and modularity to the system connections.

Outlook
The PCB design will be reviewed thoroughly before the manufacturing stage. The connectors and circuit components will then be completely assembled. The assembled harnessing will be tested by operating the VDS. This test will verify the PCB design and manufacturing process.

4.3.1.4 *Sensors*

Several sensor schemes were considered for the VDS electronics. Trade studies were conducted to evaluate the strengths and weaknesses of each scheme. The Kepner-Tregoe trade-off table can be seen below in Table 22.

VDS Sensor Schemes							
Options:		Altitude & Acceleration Bmp180 & Bno055		Altitude & Velocity Bmp180 & Pitot Tube		Altitude Only Bmp180 Only	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Available Documentation (0-10)	0.2	9	1.8	3	0.6	9	1.8
Fidelity of Data	0.4	7	2.8	6	2.4	3	1.2
Affordability	0.19	8	1.52	4	0.76	9	1.71
Ease of Implementation	0.21	6	1.26	1	0.21	8	1.68
Total Score			7.38		3.97		6.39

Table 22: VDS sensor schemes trade study table.

Altitude Only

The option of only including an altitude based sensor was originally considered very heavily for its simplicity. Several issues with this method were discovered during early development flight tests. The core issue discovered was this method's susceptibility to the magnification of error through differentiation so it scored poorly in the fidelity of data category of the trade-study. Mitigating this issue through the averaging of samples also introduced lag in the readings, further lowering its data fidelity score.

Altitude and Velocity

The option of including altitude and velocity was researched as a plausible sensor scheme for the VDS. Pitot tubes are the sensor scheme in which aircraft read their velocity. The pitot tube in conjunction with the Bmp180 would give the static and dynamic pressure readings necessary to calculate the launch vehicle's velocity through Mach numbers from NACA plots. The difficulty of mechanical and electrical implementation coupled with the cost of pitot tubes caused this sensor scheme to score low on the study table.

Altitude and Acceleration

The option of including sensors that read both altitude and acceleration was chosen as the VDS sensor scheme due to its high data fidelity and low cost. This method mitigates error due to differentiation by taking more samples in its least-squares velocity algorithm. This method can afford to do this because it makes up for the lag introduced by higher sampling by also integrating acceleration.

4.3.2 Control Theory

The control theory used in the design of the VDS covers all the applicable equations need to model the coast phase of a launch and the control scheme.

4.3.2.1 *Applicable Equations*

There are several important equations that model the behavior of the vehicle during the coast phase. These equations are used to design the VDS, used to simulate its behavior, and have been verified experimentally.

The VDS model equations are derived from the coast phase deceleration equation.

$$\mathrm{a} = -\mathrm{g} - \mathrm{c}\mathrm{v}^2$$

Where a is the vertical component of acceleration, g is the acceleration due to gravity, and v is the vertical component of velocity. The constant c represents the vehicle's unique drag characteristics.

$$c = \frac{C_d \rho A}{2m}$$

Where A is the cross-sectional area of the vehicle, C_d is the coefficient of drag of the vehicle, and m is the mass of the vehicle after burn. ρ, the density of air, , is taken to be a constant 1.225 $\frac{kg}{m^3}$ despite that it changes with altitude. These changes were taken to be negligible and ignored for the purpose of computational efficiency.

Other forms can be derived from the coast phase deceleration equation such as the velocity WRT height form. This form is shown below.

$$\mathrm{v(h)} = -\mathrm{e}^{-hc}\sqrt{\frac{\mathrm{g}}{\mathrm{c}}\mathrm{e}^{2K_2 c} - \mathrm{e}^{-2hc}}$$

4.3.2.2 *Control Scheme*

The control scheme is the autonomous decision-making process that the VDS performs during flight to achieve its goal of an exact apogee altitude. It does this by continually comparing its real-time vertical velocity to a predetermined ideal flight path and correcting for any deviations. This ideal flight path, or 'set point path', leads the rocket to a velocity of 0 m/s at the target altitude AGL of 1609.34 [m] (1 mile).

The Setpoint Path

The setpoint path (SPP) is an equation of velocity as a function of altitude, $\mathrm{v_{spp}(h)}$. It is derived from the coast phase deceleration equation and has an altitude axis (h) intercept equal to 1609 [m] (1 mile). The SPP is given as

$$v_{spp} = \begin{cases} -\mathrm{e}^{-\mathrm{h}\bar{c}}\sqrt{\frac{\mathrm{g}}{\bar{c}}\mathrm{e}^{2K_2\bar{c}} - \mathrm{e}^{-2\mathrm{h}\bar{c}}} \left[\frac{m}{s}\right] & , v < 125 \left[\frac{m}{s}\right] \\ -\mathrm{e}^{-\mathrm{h}c_{min}}\sqrt{\frac{\mathrm{g}}{c_{min}}\mathrm{e}^{2K_2 c_{min}} - \mathrm{e}^{-2\mathrm{h}c_{min}}} \left[\frac{m}{s}\right] & , v \geq 125 \left[\frac{m}{s}\right] \\ 0 \left[\frac{m}{s}\right] & , \mathrm{h} > 1609\ [m] \end{cases}$$

where $\bar{c}$ is the average drag characteristics constant given by

$$\bar{c} = \frac{\rho(A_r + A_{r+b})(C_r + C_{r+b})}{8m}$$

where A_r is the cross-sectional area of the vehicle, A_{r+b} is the cross sectional area of the rocket and brakes, C_r is the coefficient of drag of the vehicle, and C_{r+b} is the coefficient of drag of the rocket and brakes.

The minimum drag characteristics constant, c_{min}, is given by

$$c_{min} = \frac{\rho A_r C_r}{2m}$$

Calculating the SPP results in a plot shown below in Figure 13.

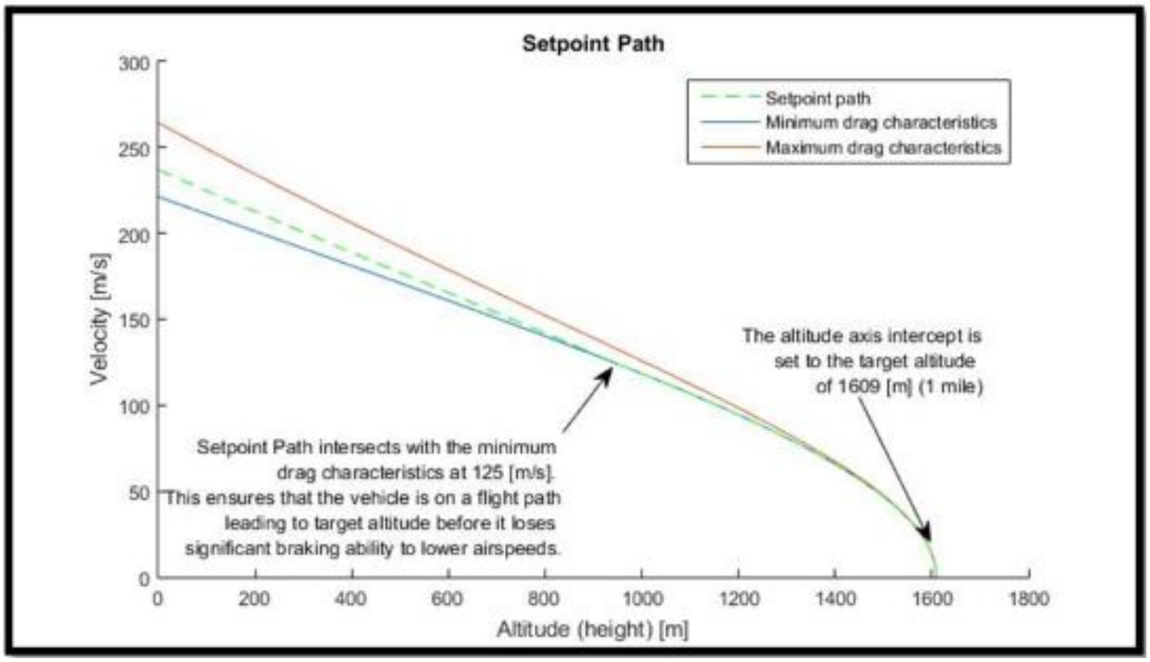


Figure 13: Setpoint path.

The piecewise SPP can be seen above in its three parts. The first part, where v>125, is calculated using an average drag characteristic constant to facilitate a smooth transition to the minimum drag characteristic path. The second part follows the minimum drag characteristic path to the target altitude. The third part where vspp = 0 for h>1609 ensures that the VDS will deploy the brakes if it passes its target altitude.

This piecewise SPP has been developed as an alternative to an originally not piecewise SPP that simply had average drag characteristics throughout. This SPP was found in both simulation and test launches to lead to an achieved apogee that was consistently higher than the target. This is likely due to the fact that the VDS loses braking ability as it slows down. This is remedied in the new SPP that puts the vehicle on a path to the target altitude at 125 m/s, before it loses significant braking ability.

4.3.3 Mechanical Design

In order for the VDS to be the most mechanically efficient system possible, several factors were taken into consideration for the mechanical design of the VDS:

1. Volume
2. Actuation speed
3. Mass
4. Drag area

The VDS prototype and current assembly of the VDS is shown below in Figure 14 and Figure 15.

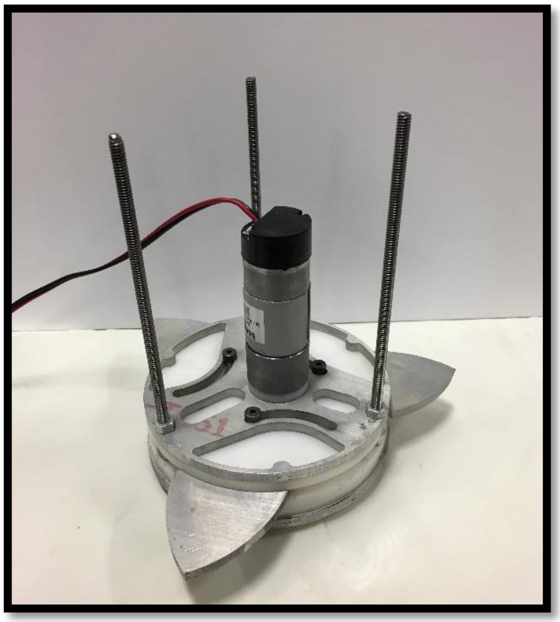


Figure 14: VDS prototype.

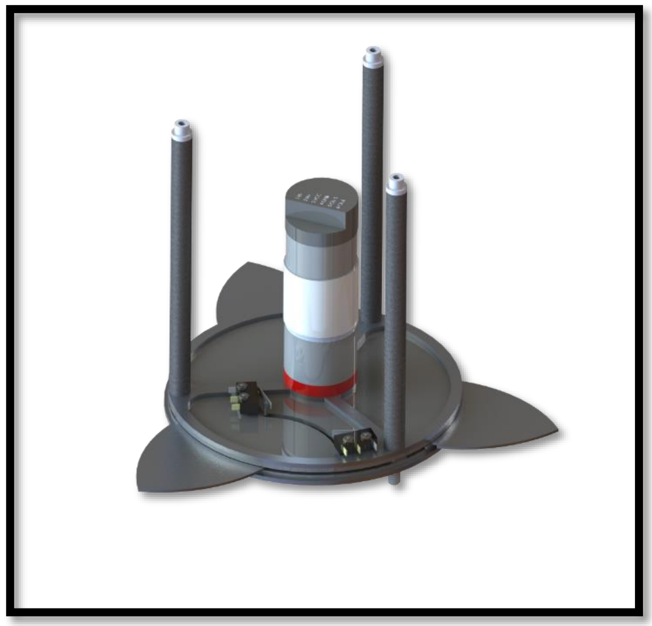


Figure 15: Rendering of current configuration of VDS.

4.3.3.1 *Actuation*

In order to optimize the actuation speed of the VDS, the drag blades radially extend perpendicular to the rocket body. The load of the drag force exerted on the drag blades is transferred to the support plates of the VDS, rather than directly on the motor. Actuating the drag blades perpendicular to the drag force reduces the torque the motor will have to output to actuate the drag blades, which in turn allows the drag blades to extend faster. The three drag blades are controlled simultaneously by a central gear, which is attached to the motor via D shaft and set screw. The control of all three drag blades by a central gear reduces the risk of mechanical failure. Each drag blade contains a set of radial gear teeth that mesh with the central motor gear. Involute gear teeth were chosen for the central gear and drag blades due to their reliability and efficiency. The meshing between the central gear and drag blades can be seen below in Figure 16.

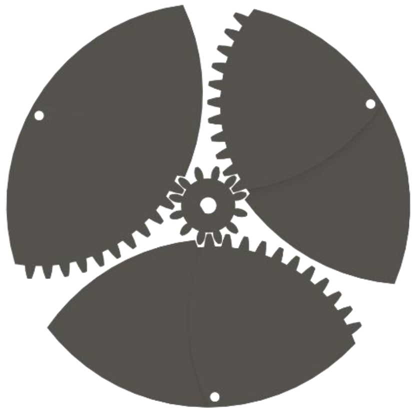


Figure 16: Gear Meshing of Drag Blades.

Each drag blade pivots around a $^{1}/_{8}$" Dowel Pin. After full actuation, approximately half of the drag blade is exposed to the exterior of the launch vehicle, and half of the drag blade is located within the VDS assembly. This configuration ensures the maximum amount of area each drag blade extends outside of the airframe of the launch vehicle, while allowing the central motor gear to simultaneously control the actuation of each drag blade. Controlling each drag blade through a central motor gear simplifies the mechanical design and control system of the VDS. Actuation of the blades can be seen in Figure 17 and Figure 18.

Figure 17: VDS top view with no actuation.

Figure 18: VDS top view with full actuation.

The drag blades are manufactured from $^{1}/_{8}$" 6061-T6 aluminum using a Maxiem 450 Water Jet. The drag blades will be manufactured from 6061-T6 aluminum due to its rigidity. The area of the drag blade that is exposed to the air flow is reduced down to a thickness of 0.06 inch. A rendering of a drag blade can be seen below in Figure 19.

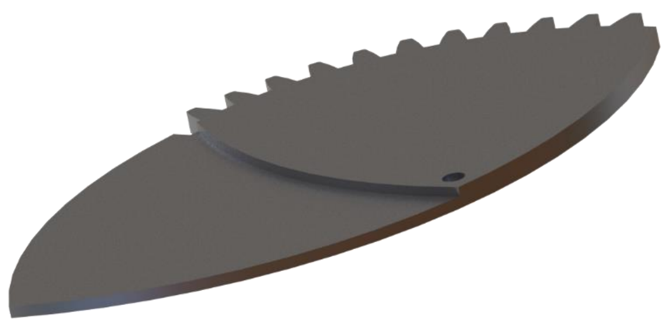


Figure 19: Drag blade rendering.

The drag blades sit between two $^{1}/_{8}$" Delrin acetal resin plates, which will be laser cut. Delrin was chosen for the drag blades to slide across due to its low coefficient of friction with aluminum, which is rated at 0.3. Placing the drag blades between a two plates with a material with a low coefficient of friction allows for a precise actuation of the drag blades, while also allowing a slick surface for the drag blades to slide across when compared to aluminum, which is approximately 1.05. An additional Delrin plate was placed below the bottom Delrin plate to hold the dowel pins in place and add support for the drag blades. Three custom machined aluminum spacers are placed between the Delrin plates to ensure proper alignment of all of the plates of the assembly and prevent overtightening of plates on the drag blades to minimize the friction force of the Delrin plates on the drag blades during actuation.

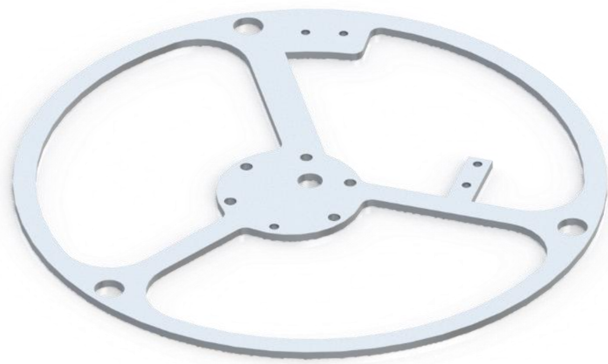


Figure 20: Top Support plate.

A prototype of the VDS was manufactured and extensively tested via four full scale test launches. Due to correct tolerances between the gear teeth of the drag blades and the gear teeth of the central gear, the VDS actuated without any anomalies. Gear binding was not found to be an issue.

4.3.3.2 *Motor Selection*

Several different motors were considered for the actuation device of the VDS. The three motors considered for implementation in the VDS were a servo motor, a DC motor, and a stepper motor. A Kepner-Tregoe Trade Study comparing all three motors is shown below in Table 23: Kepner-Tregoe Trade Study of VDS actuation device.

Actuation Device							
Options:		Servo motor		DC motor		Stepper motor	
Mandatory Requirements							
Able to overcome 388 [oz-in] of torque.		No		Yes		Yes	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
RPM (0-10)	30.00%	2	0.6	6	1.8	10	3
Stall Torque (0-10)	20.00%	3	0.6	7	1.4	10	2
Size (0-10)	20.00%	9	1.8	6	1.2	1	0.2
Price (0-10)	5.00%	4	0.2	6	0.3	8	0.4
Mass (0-10)	25.00%	10	2.5	6	1.5	2	0.5
Total Score		**5.7**		**6.2**		**6.1**	

Table 23: Kepner-Tregoe Trade Study of VDS actuation device.

While the stepper motor would result in the fastest actuation, it would also increase the overall mass of the VDS by the largest margin. As shown above, the DC motor is the ideal device for actuation of the VDS due to its combination of mass, actuation speed, and torque output. The

AndyMark Neverest 40 DC motor was selected to actuate the VDS. The technical specifications can be seen below in Table 24.

Gearbox Output Power	14W
Stall Torque	350 oz-in
No-Load Speed	160 rpm
Weight	0.75 lb

Table 24: AndyMark Neverest 40 DC motor technical specifications.

4.3.3.3 *Analysis*

By analyzing past failure modes of other teams' air braking designs, the team decided to design a robust system that would be able to withstand all of the drag forces during flight. The thickness of the drag blades was optimized to minimize mass while maintaining an acceptable factor of safety. Through experimental test launches with a prototype of the VDS, it was determined that maximum drag force exerted on the drag blades was approximately 20 pounds. Due to changes in maximum velocity in the launch vehicle and uncertainties in the calculation of drag, each drag blade was designed to be able withstand the full drag force that was experimentally determined with a factor of two. A Finite Element Analysis was performed on the drag blade to ensure that the structural integrity of the drag blade design. A minimum factor of safety of 2.2 within each drag blade was determined from the FEA simulations when subjected to conservative estimates of the load each drag blades would experience. Was The results from the FEA simulation is shown below in Figure 21.

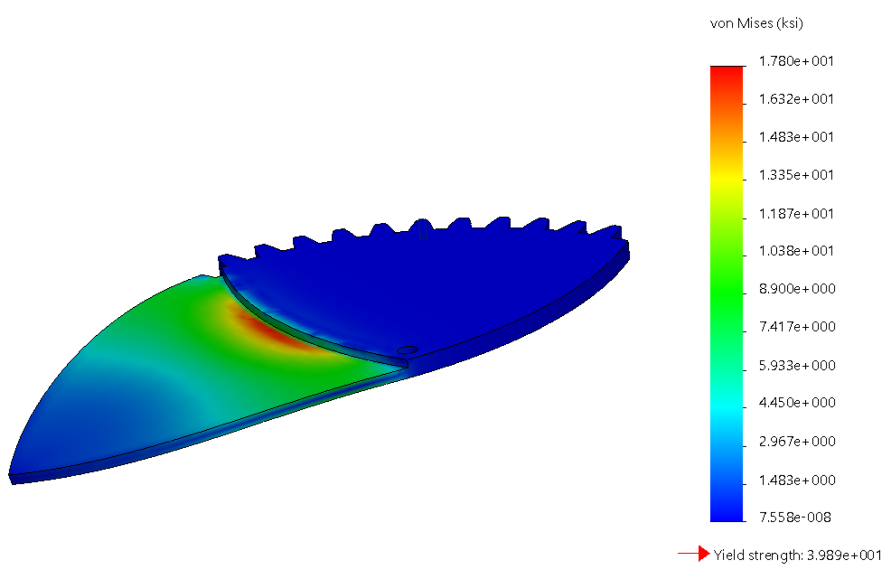


Figure 21: Drag blade FEA stress plot.

The structural integrity of the gear teeth was also verified through FEA simulations. Under worst case scenario, the teeth on the central gear and drag blades will be required to withstand the forces exerted under the stall torque induced by the DC motor. A minimum factor of safety of 1.9 within the central gear was determined from the FEA simulation. The stress plot from the FEA simulation of the central gear teeth when subjected to the stall torque of the motor is shown below in Figure 22.

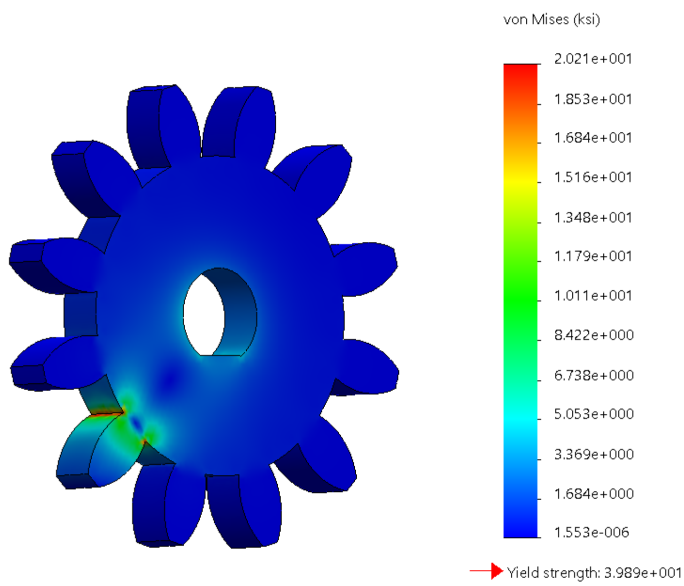


Figure 22: Central gear teeth FEA stress plot.

An FEA simulation was performed on the entire VDS assembly to ensure the minimum factor of safety throughout the assembly was greater than two. With expected loads applied to each drag blade, the minimum factor of safety throughout the VDS assembly was determined to be 3.4 from the FEA simulation. A stress plot from the FEA simulation of the VDS assembly is shown below in Figure 23.

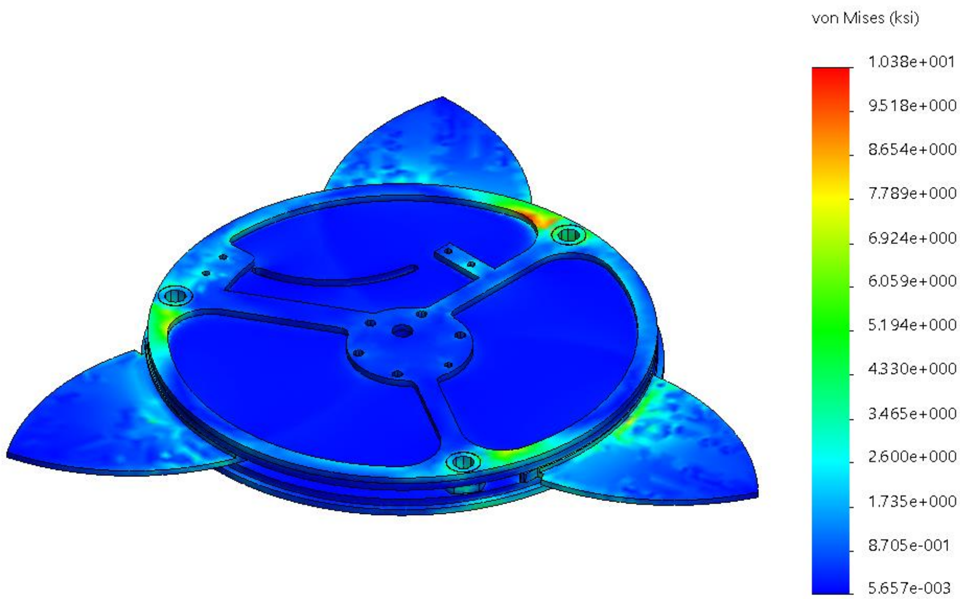


Figure 23: VDS assembly FEA stress plot.

4.4 VDS Timeline Overview

The VDS project plan is outlined below in Figure 24.

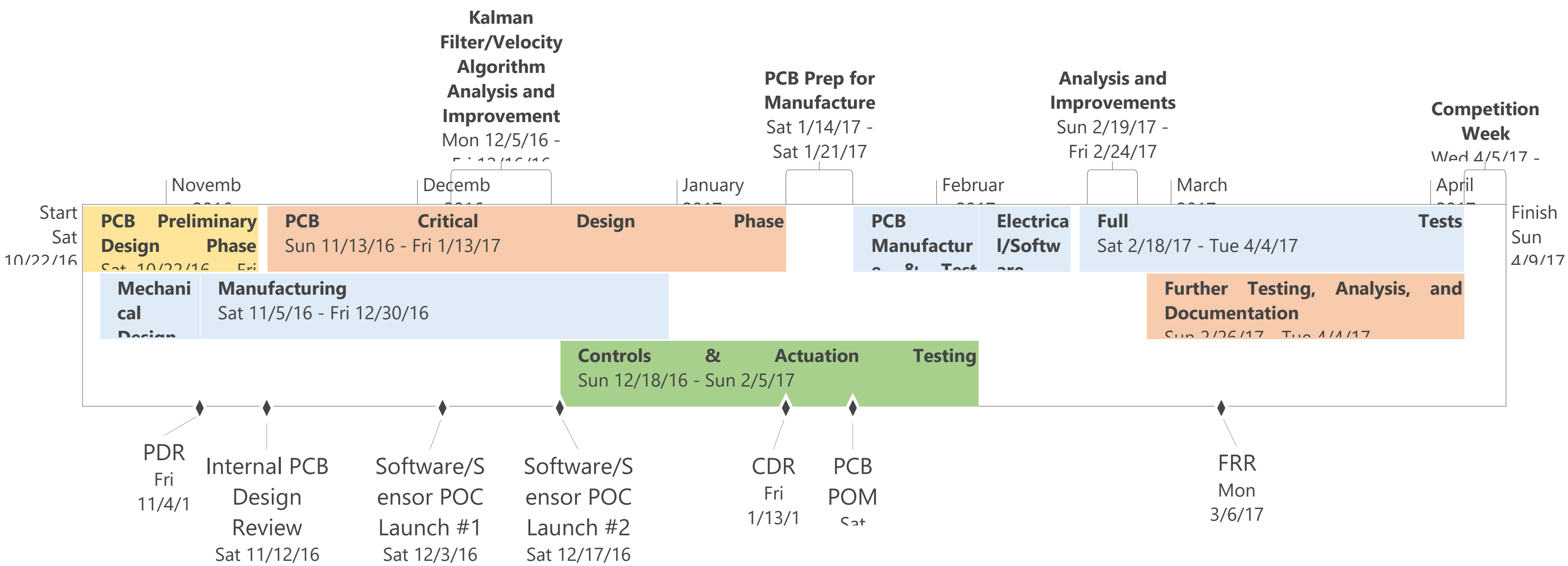


Figure 24: VDS project plan timeline.

This timeline includes the design and manufacture schedules for the VDS PCBs, the test schedule for the software, and the schedule for the design and machining of mechanical components. The timeline also includes important milestones such as software proof of concept (POC) test flights and full scale test flights.

December Test Launches

The software/sensor POC launches in December are an important step in verifying many of the VDS requirements. These two launches will test the sensors, the software that communicates with them, the algorithms for interpreting the sensor data, Kalman filter, and the data storage. This list will cover roughly half of the verifications needed for the VDS.

Full Scale Test Launches

The VDS will undergo comprehensive testing beginning in February. These tests will be on the competition vehicle with the final mass and final motor. These launches will verify the remaining requirements yet unverified in ground testing and in the December launches.

4.5 Safety

Safety is the highest priority in the VDS design. The VDS was designed from a system level to reduce the hazard of an asymmetrical failure of the drag blades which might result in the vehicle pitching over and endangering the safety of those attending the launch. This hazard has been mitigated at a system level by designing the VDS with one central actuator and one central gear rather than one actuator for each drag surface. Because the actuator and linking method are the largest points of potential failure, the VDS design has mitigated the risk of an asymmetrical failure.

VDS Actuation Risk Assessment						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity**	**Probability**	**Risk Level**	**Mitigation**
Structural damage to the airframe during actuation during flight and during pre-flight test.	1. Improper installation, that result in tolerance issues 2. Securing hardware properly 3. Drag blades over rotating/over retracting	1a. Tearing into the airframe resulting in sever zippering. 1b. Prevent drag blades from opening during flight, overshooting the altitude and breaking the waiver. 2. Damage to equipment and possible loss to the	3	4	Moderate	All hardware being checked and proper clearances must be verified by a sub-team lead and a captain.
VDS actuates on rail	1. Electrical and/or programing failure.	1a. Vehicle escapes path of rail and resulting in a unstable flight. 1b. Potential injury to personal or spectators if the rocket were to go on a rogue flight path.	2	5	Low	Consistent testing and validation of the system functions to ensure a premature deployment does not occur.

VDS failing to retract during recovery	1. Drag blades over extending breaking the motor gearbox.	Damage to vehicle sections as they hit each other on descent. Potential to injury to personal or spectators. Shock cord and shroud lines tangling on drag blades causing a free fall of the vehicle.	5	2	Moderate	The team will implement limit switches on both extrema of movement to prevent the overextending or over retracting of the air blades.
VDS fails to actuate	1. Gear binding. 2. Electrical failure.	1. Mission failure. The vehicle is delivered to an apogee above 1 mile. 2. Vehicle apogee exceeds waiver.	2	4	Moderate	The motor and mass will be chosen to not exceed the waiver in the event of VDS failure.
DC motor induces noise on sensors	DC motor oscillates quickly.	Mission failure. Noise induced causes false and potentially unneeded actuation leading to lower achieved altitude.	5	2	Moderate	A 0.1 μF capacitor will be soldered across the motor leads to reduce noise.
Drag blades are pressed into airframe slots during flight	VDS assembly isn't fastened properly to the launch vehicle.	Airframe ruptures during ascent and the structural integrity of the launch vehicle is endangered.	2	5	Moderate	Proper installation of the VDS will be ensured before every flight.

Table 25: Variable drag system risk assessment.

5 Technical Design: Launch Vehicle

The launch will be constructed primarily from carbon fiber, fiberglass, aluminum, and plywood. In order to maximize the braking power of the VDS and achieve an apogee of 5280 feet, the launch vehicle will be optimized to minimize mass. The launch vehicle can be divided into seven distinct sections, which is outline below in Figure 25; nose cone section, payload recovery bay, deployment bay, payload bay, booster recovery bay, VDS bay, and propulsion bay.

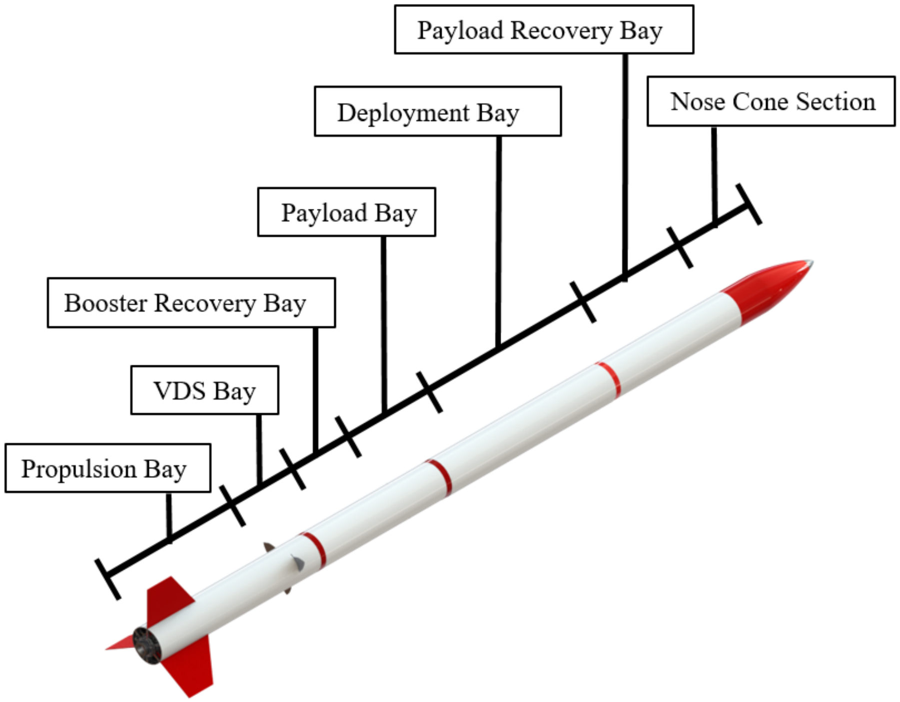


Figure 25: Full Scale Launch Vehicle.

5.1 Mission Success Criteria

1. The launch vehicle shall carry a payload up to an apogee altitude of 5280 [ft] ±33ft with zero anomalies.
2. All recovery events shall occur at the programmed altitudes.
3. The launch vehicle shall have a stable ascent.
4. The launch vehicle shall be completely reusable once it is recovered.

5.2 Statement of Work Verifications

Requirement Number	Requirement	Method of Verification
1.1	The vehicle shall deliver the science or engineering payload to an apogee altitude of 5,280 feet above ground level (AGL).	Analysis The launch vehicle will be efficiently documented and all material and component masses will be recorded throughout the design and manufacturing. Accurate OpenRocket simulations and hand calculations will be maintained to ensure a correct motor selection. The VDS will be optimized and tested to minimize deviation of apogee altitude from 5,280 feet.
1.2	The vehicle shall carry one commercially available, barometric altimeter for recording the official altitude used in determining the altitude award winner.	Inspection The launch vehicle will descend under a single recovery system, using a drogue and main parachute. A Perfectflite StratoLogger CF altimeter will be used to record the apogee altitude for the competition. For complete redundancy, a secondary backup altimeter shall be included as well.
1.3	All recovery electronics shall be powered by commercially available batteries.	Inspection The primary and redundant altimeters will be powered by 12 volt batteries.
1.4	The launch vehicle shall 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 parachutes will be designed to ensure every section of the launch vehicle lands with a kinetic energy below the maximum kinetic energy laid out in the Statement of Work. Through appropriate material selection and manufacturing techniques, the rocket will be able to land at the maximum allowable kinetic energy without incurring any damage. Landing within these constraints will leave the launch vehicle in a reusable state.
1.5	The launch vehicle shall have a maximum of four (4) independent sections.	Inspection The launch vehicle will be comprised of three independent sections: the nose cone, the payload, and the rest of the launch vehicle, which includes the recovery bay, the VDS bay, and the propulsion bay.
1.6	The launch vehicle shall be limited to a single stage.	Analysis Having a limited altitude of 5280 feet eliminates any need for staging of our launch vehicle. Motor

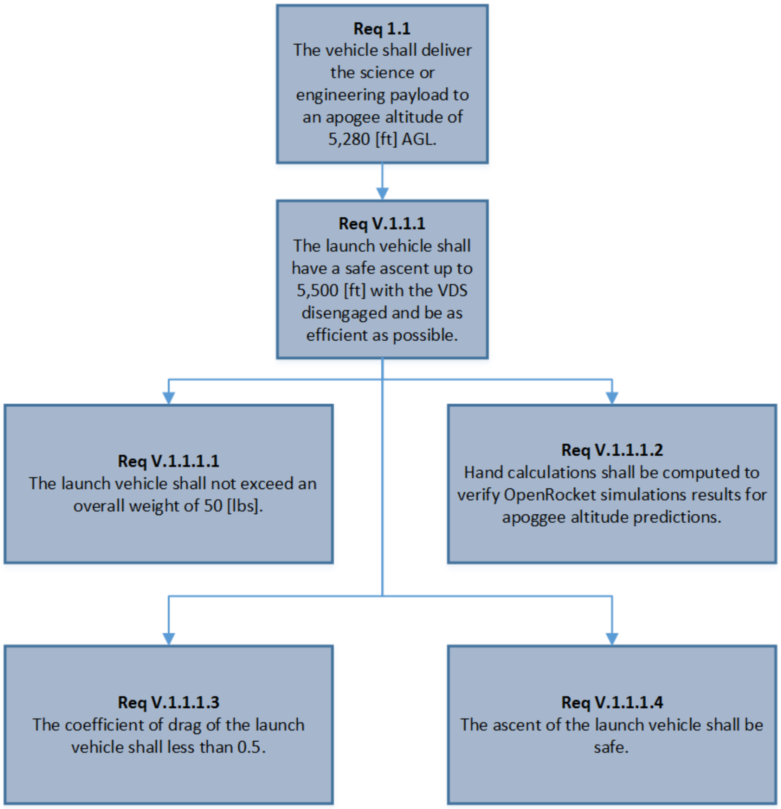
		selections have been made to accomplish all necessary altitude requirements on a single stage launch vehicle.
1.7	The launch vehicle shall be capable of being prepared for flight at the launch site within 4 hours, from the time the Federal Aviation Administration flight waiver opens.	Demonstration A comprehensive launch procedure checklist will be constructed by the team to allow for accurate and expedited vehicle assembly while preparing for flight.
1.8	The launch vehicle shall be capable of remaining in launch-ready configuration at the pad for a minimum of 1 hour without losing the functionality of any critical on-board component.	Demonstration The power supplies for the payload electronics, altimeters, and flight event devices have been chosen to eliminate the chances of power failure for an extended period of time.
1.9	The launch vehicle shall be capable of being launched by a standard 12 volt direct current firing system.	Inspection The launch vehicle will utilize proven launch igniters purchased from Wildman Rocketry. The igniters are designed to ignite the vehicle's motor by use of a standard 12 volt direct current firing system
1.10	The launch vehicle shall require no external circuitry or special ground support equipment to initiate launch (other than what is provided by Range Services).	Demonstration The launch vehicle will not require external circuitry or special ground support equipment to initiate launch.
1.11	The launch vehicle shall use a commercially available solid motor propulsion system using ammonium perchlorate composite propellant (APCP) which is approved and certified by the National Association of Rocketry (NAR), Tripoli Rocketry Association (TRA), and/or the Canadian Association of Rocketry (CAR).	Inspection The team will use an AeroTech L2200-G motor for its full scale launch vehicle.

1.12	Pressure vessels on the vehicle shall be approved by the RSO.	Inspection The current design of the launch vehicle does not require the use of any pressure vessels. If the design changes to include such a system, NASA and the RSO will be notified, and the criteria mentioned in the Statement of Work will be met.
1.13	The total impulse provided by a Middle and/or High School launch vehicle shall not exceed 5,120 Newton-seconds (L-class).	The total impulse of the AeroTech L2200-G is 5,104 Newton-seconds.
1.14	The launch vehicle shall have a minimum static stability margin of 2.0 at the point of rail exit.	Analysis OpenRocket simulations will be created to verify that the static stability margin at the point of rail exit is above 2.0.
1.15	The launch vehicle shall accelerate to a minimum velocity of 52 fps at rail exit.	Analysis OpenRocket simulations will be created to verify that the velocity at rail exit is higher than 52 fps.
1.16	All teams shall successfully launch and recover a subscale model of their rocket prior to CDR. The subscale model should resemble and perform as similarly as possible to the full-scale model, however, the full-scale shall not be used as the subscale model.	Demonstration A 1:2 scaled model of the full scale launch vehicle has been designed. The subscale launch vehicle will be used to test stability and integration of various systems seen in the full scale launch vehicle.
1.17	All teams shall successfully launch and recover their full-scale rocket prior to FRR in its final flight con-figuration. The rocket flown at FRR must be the same rocket to be flown on launch day.	Demonstration The team plans to conduct several full scale test flights throughout the season to test the rigidity and effectiveness of the VDS and payload design.
1.18	Any structural protuberance on the rocket shall be located aft of the burnout center of gravity	Demonstration The only structural protuberance on the launch vehicle are the drag blades of the VDS. The launch vehicle was designed to place the VDS as far aft as possible. As a result, all structural protuberances are located aft of the burnout center of gravity.

1.19	The launch vehicle shall not utilize forward canards. The launch vehicle shall not utilize forward firing motors. The launch vehicle shall not utilize motors that expel titanium sponges (Sparky, Skidmark, MetalStorm, etc.) The launch vehicle shall not utilize hybrid motors. The launch vehicle shall not utilize a cluster of motors. The launch vehicle shall not utilize friction fitting for motors. The launch vehicle shall not exceed Mach 1 at any point during flight. Vehicle ballast shall not exceed 10% of the total weight of the rocket.	Inspection The launch vehicle will comply with all of the vehicle prohibitions listed in requirement 1.19 in the statement of work.

5.3 Launch Vehicle Requirements

5.3.1 Ascent Requirements



Requirement Number	Requirement	Method of Verification
1.1	The vehicle shall deliver the science or engineering payload to an apogee altitude of 5,280 [ft] AGL.	<u>Test</u> Flight altimeters will record the apogee of the vehicle during both test launches and the competition launch.
V.1.1.1	The launch vehicle shall have a safe ascent up to 5,500 [ft] with the VDS disengaged and be as efficient as possible.	<u>Test</u> The launch vehicle integrated with the VDS will be tested through several test launches.
V.1.1.1.1	The launch vehicle shall not exceed an overall weight of 50 [lbs].	<u>Inspection</u> The launch vehicle will be efficiently documented and all material and component weights will be recorded throughout the design and manufacturing. The weight of the fully assembled launch vehicle will be measured before every test launch.

V.1.1.1.2	Hand calculations shall be computed to verify OpenRocket simulations results for apogee altitude predictions.	Analysis This requirement will verified
V.1.1.1.3	The coefficient of drag of the launch vehicle shall be less than 0.5.	Demonstration CFD simulations will simulate flight conditions and compute the coefficient of drag of the entire launch vehicle. A wind tunnel will be used to simulate a flight and verify CFD simulations.
V.1.1.1.4	The ascent of the launch vehicle shall be safe.	Test Test launches will be carried out to ensure the launch vehicle performs as intended. The launch vehicle will be designed and constructed in accordance will all NAR safety regulations.

5.3.1.1 *Derivation of Requirement V.1.1.1*

To achieve an apogee altitude of 5,280 feet, the launch vehicle will aim for an apogee altitude of 5,500 feet and utilize the VDS to achieve an apogee altitude of 5280 feet. An altitude was 5,500 feet was chosen to minimize the risk of earning a score of zero in case of failure of VDS to function properly. As stated in the Statement of Work in Requirement 1.2.6.3, an apogee altitude greater than 5,600 feet will warrant a score of zero for the altitude portion of the competition. By choosing an altitude of 5,500 feet, a 100 feet buffer is provided in the worst case scenario if the VDS were to fail.

5.3.1.2 *Derivation of Requirement V.1.1.1.1*

In order the launch vehicle to achieve an altitude of 5,500 feet, a maximum weight has been defined for the launch vehicle. By creating OpenRocket simulations and utilizing MATLAB Simulink simulations, it was determined that a weight of 50 pounds with an overall coefficient of drag of 0.5 is the largest weight that the launch vehicle can in order to achieve an apogee altitude of 5500 feet while still employing an L-class motor.

5.3.1.3 *Derivation of Requirement V.1.1.1.2*

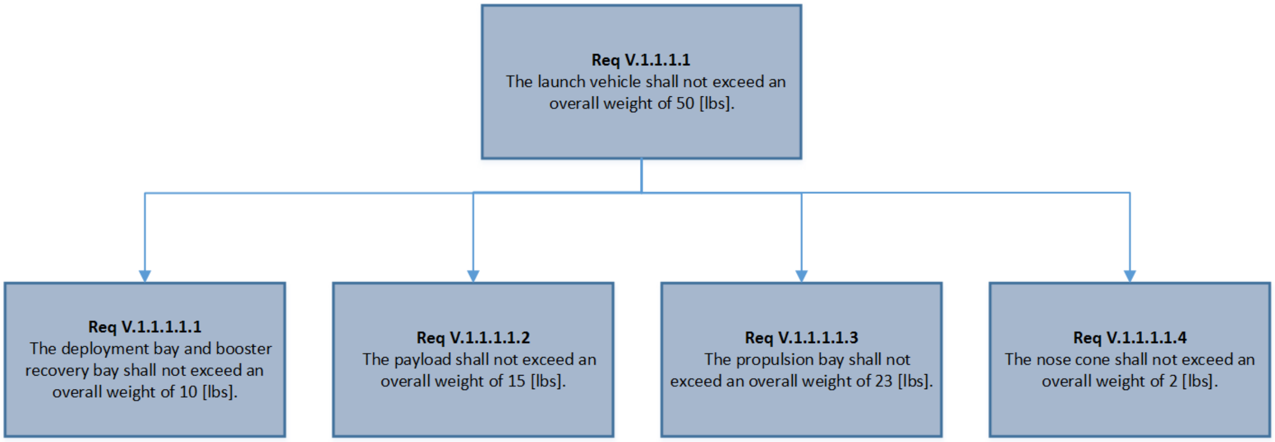
While OpenRocket simulations are able to provide apogee altitudes and stability for vehicle configurations, hand calculations will be performed to ensure that the OpenRocket simulation was correctly implemented.

5.3.1.4 *Derivation of Requirement V.1.1.1.3*

Several prototype full scale test launches were completed during the summer to better determine the flight characteristics of the launch vehicle in a specific configuration. It was found that the coefficient of drag of the prototype vehicle was 0.5. By decreasing the coefficient of drag of the rocket, the launch vehicle is more efficient and thus requires a smaller motor in order to achieve a target apogee altitude. The full scale launch vehicle will be optimized to minimized the coefficient of drag.

5.3.1.5 *Derivation of Requirement V.1.1.1.4*

Safety is the highest priority during the design and construction of the launch vehicle. In order to for the mission to be considered a success, the launch vehicle is required to safely ascend to 5,280 feet to deliver the payload.



Requirement Number	Requirement	Method of Verification
V.1.1.1.1.1	The deployment bay and booster recovery bay shall not exceed an overall weight of 10 [lbs].	Inspection Every component will be predicted based on material densities and weighed before every flight.
V.1.1.1.1.2	The payload shall not exceed an overall weight of 15 [lbs].	Inspection Every component will be predicted based on material densities and weighed before every flight.
V.1.1.1.1.3	The propulsion bay shall not exceed an overall weight of 23 [lbs].	Inspection Every component will be predicted based on material densities and weighed before every flight.
V.1.1.1.1.4	The nose cone shall not exceed an overall weight of 2 [lbs].	Inspection Every component will be predicted based on material densities and weighed before every flight.

5.3.1.6 *Derivation of Requirement V.1.1.1.1.1.1*

By analyzing weights of past recovery equipment, the team decided to allocate a maximum mass of 10 pounds to all recovery equipment and bays within the launch vehicle.

5.3.1.7 *Derivation of Requirement V.1.1.1.1.1.2*

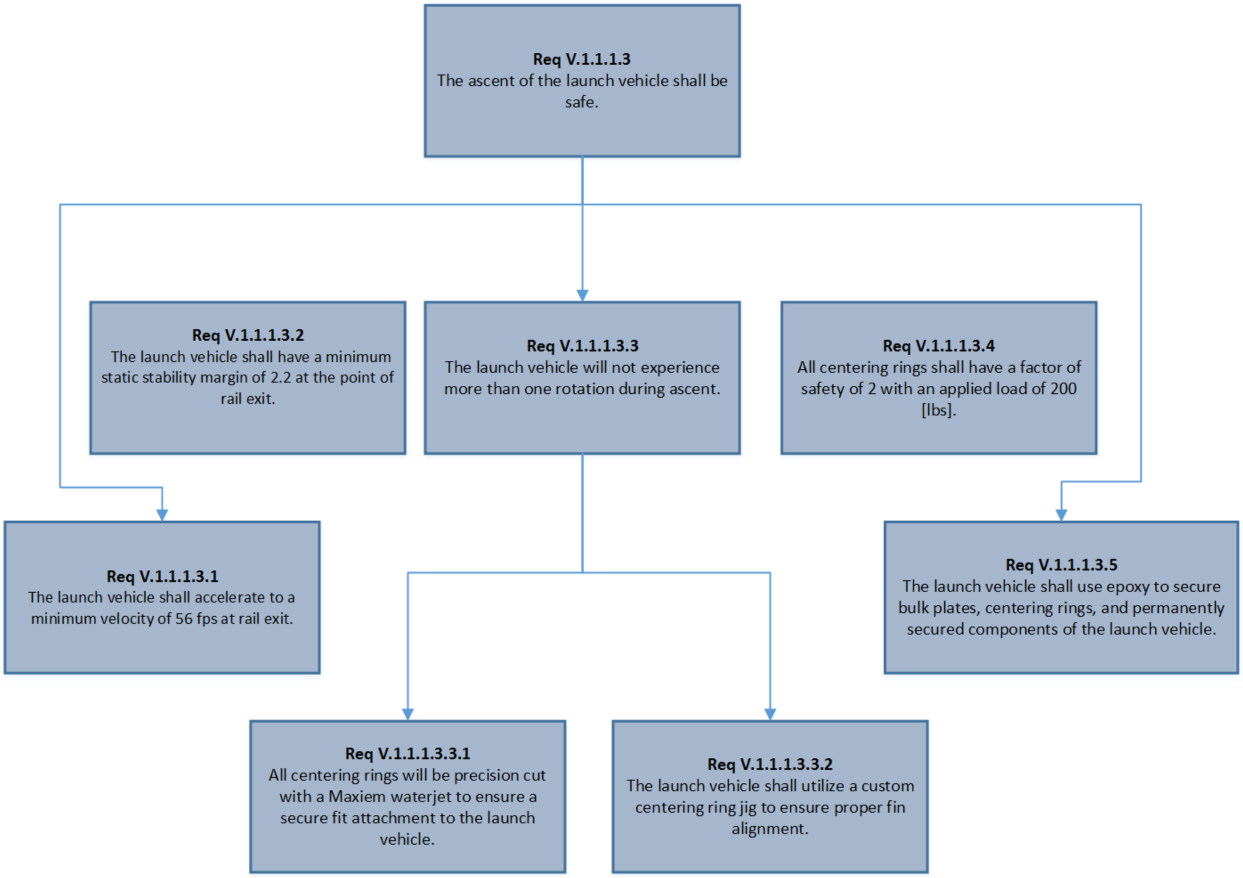
A prediction of the launch vehicle with all necessary compartments, such as the propulsion bay, recovery bays, and nose cone, was determined based on material densities and past component weights. Necessary compartments are defined as any section of launch vehicle that contributes to a safe launch. The difference between the calculated maximum overall weight of the launch vehicle and the predicted weight of the all of the necessary compartments was allocated to the payload, which was 15 pounds.

5.3.1.8 *Derivation of Requirement V.1.1.1.1.1.3*

The propulsion bay consists of the motor and the systems that secure the motor and fins to the launch vehicle. CAD programs were used to determine the weight of all structural components. The components weights were simulated based on a worst-case-scenario off a thrust from the AeroTech L2200, which is the largest motor in the L-class of motors. It was determined that the weight of the propulsion bay assembly with the largest L-class motor would be 20 pounds. Therefore, the weight of the propulsion bay for the launch vehicle will not exceed 20 pounds.

5.3.1.9 *Derivation of Requirement V.1.1.1.1.1.4*

The nose cone weight was determined by using the density of the team's custom filament wound carbon fiber airframe and CAD programs. The maximum allocated weight of the nose cone was given a buffer of 2 pounds to account for any design changes.



Requirement Number	Requirement	Method of Verification
V.1.1.1.3.1	The launch vehicle shall accelerate to a minimum velocity of 56 fps at rail exit.	<u>Analysis</u>

		OpenRocket simulations will be created to calculate the exit rail velocity and hand calculations will be used to verify the OpenRocket simulations.
V.1.1.1.3.2	The launch vehicle shall have a minimum static stability margin of 2.2 at the point of rail exit.	Analysis OpenRocket simulations will be created to calculate the static stability margin and hand calculations will be used to verify the OpenRocket simulations.
V.1.1.1.3.3	The launch vehicle will not experience more than one rotation during ascent.	Test An on-board camera recording system will determine how rotations the launch vehicle experiences during test launches. All centering rings will be precision cut with a Maxiem waterjet to ensure a secure fit attachment to the launch vehicle and the launch vehicle will utilize a custom jig to ensure proper centering ring alignment.
V.1.1.1.3.4	All centering rings shall have a factor of safety of two with an applied load of 200 pounds.	Analysis FEA simulations will be performed to ensure that each centering ring has a factor of safety two with an applied load of 200 pounds.

5.3.1.10 *Derivation of Requirement V.1.1.1.3.1*

To ensure a safe and stable flight, the has decided to set a minimum rail exit velocity of 56 feet per second. The minimum rail exit velocity outlined in the Statement of Work is 52 feet per second, however, a buffer of 4 feet per second was provided to account for any irregularities in simulations or hand calculations of the flight characteristics of the launch vehicle.

5.3.1.11 *Derivation of Requirement V.1.1.1.3.2*

The minimum static stability margin outlined in the Statement of Work is 2.0, however, the launch vehicle will be required to adhere to a minimum static stability margin of 2.2. To account for design changes and simulation errors, the team will aim to achieve a static stability margin of 2.2.

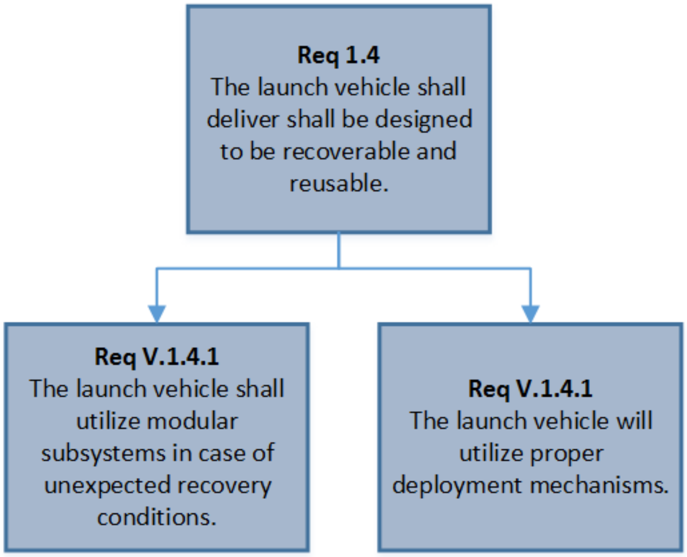
5.3.1.12 *Derivation of Requirement V.1.1.1.3.3*

By analyzing past flight test launch data, it was determined that the launch vehicle passively averages one rotations on ascent. The launch vehicle will be constructed with proper techniques to minimize rotations per ascent of the launch vehicle to maximize efficiency of the launch vehicle by reducing transfer of linear kinetic energy to rotational kinetic energy of the launch vehicle.

5.3.1.13 *Derivation of Requirement V.1.1.1.3.4*

Due to the weight reduction slots that have been incorporated into the design of the centering of the propulsion bay, proper hand calculations and FEA simulations will be utilized to ensure each has a factor of two with half of the maximum thrust of current motor. In the case of epoxy failure of a centering ring failure, the other two centering rings will still be able to transmit the load of the motor to the rest of the launch vehicle.

5.3.2 Descent Requirements



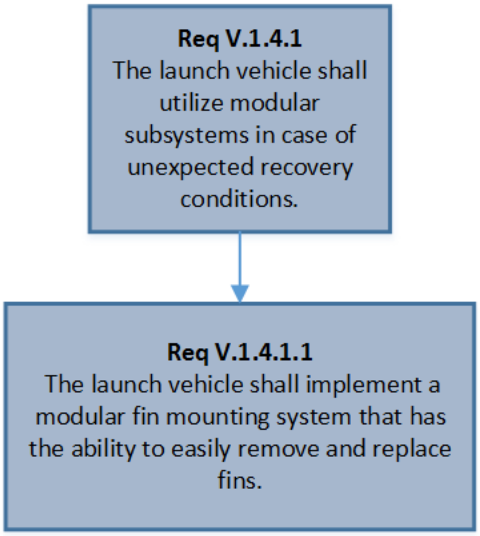
Requirement Number	Requirement	Method of Verification
V.1.4	The launch vehicle shall be designed to be recoverable and reusable.	Test Several test launches will be conducted to verify the reusability and recoverability.
V.1.4.1	The launch vehicle shall utilize modular subsystems in case of unexpected recovery conditions.	Demonstrate Systems will be implemented to easily repair component of the vehicle.
V.1.4.2	The launch vehicle will utilize proper separation mechanisms.	Test Multiple test launches will be conducted to verify that all separation mechanisms operate as intended.

5.3.2.1 *Derivation of Requirement V.1.4.1*

In testing, unexpected recovery scenarios may harm vehicle subsystems. This requirement was derived in order for the launch vehicle to have the capability to be used in extensive testing.

5.3.2.2 *Derivation of Requirement V.1.4.2*

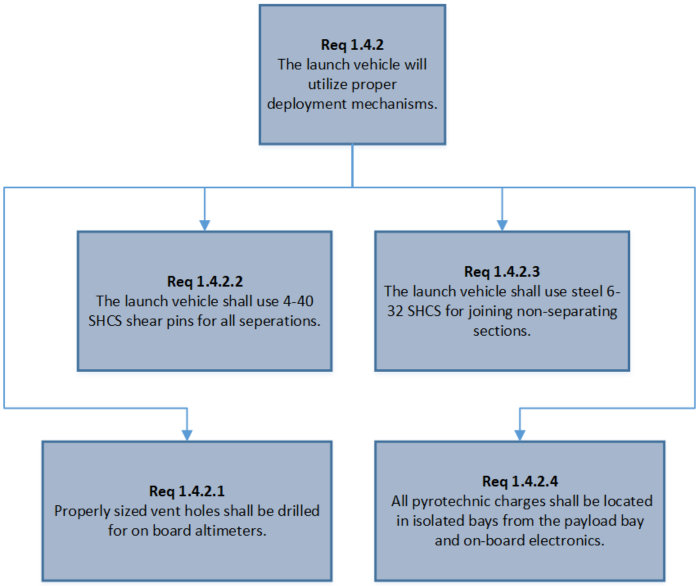
In order to complete the full flight mission, the launch vehicle shall be able to separate the interior subsystems. This requirement was derived in order to ensure that the launch vehicle is equipped with proper separation mechanisms.



Requirement Number	Requirement	Method of Verification
V.1.4.1.1	The launch vehicle shall implement a modular fin mounting system that has the ability to easily remove and replace fins.	<u>Inspection</u> A removable fin system was designed to easily install and remove fins from the launch vehicle.

5.3.2.3 *Derivation of Requirement V.1.4.1.1*

Having the ability to swap out fins greatly improves the recoverability and reusability of the launch vehicle. If the impact force of the ground on the fins is higher than expected and causes failure in the fins, the launch vehicle will still be able to be reusable if a modular fin mounting system is utilized. Due to this advantage, the team will utilize a fin mounting system that can easily swap out fins.



Requirement Number	Requirement	Method of Verification
V.1.4.2.1	Properly sized vent holes shall be drilled in the launch vehicle for on board altimeters.	<u>Inspection</u> The proper vent hole size for each bay which contains an altimeter will be calculated and implemented within the launch vehicle.
V.1.4.2.2	The launch vehicle shall use nylon 4-40 SHCS shear pins for all separations.	<u>Inspection</u> 4-40 SHCS shear pins will be installed into all separating sections of the launch vehicle before every launch.
V.1.4.2.3	The launch vehicle shall use steel 6-32 SHCS for joining non-separating sections.	<u>Inspection</u> Steel 6-32 SHCS shear pins will be installed into all separating sections of the launch vehicle before every launch.
V.1.4.2.4	All pyrotechnic charges shall be located in isolated bays from the payload bay and on-board electronics.	<u>Demonstration</u> The launch vehicle will be designed so that all pyrotechnic charges isolated from any on-board electronics.

5.3.2.4 *Derivation of Requirement V.1.4.2.1*

If vent holes are incorrectly sized, the altimeters could misinterpret the pressure reading and set off the ejection charge at the wrong time. Therefore, it is imperative that the vent holes are properly sized for the respective bay are they are located in.

5.3.2.5 *Derivation of Requirement V.1.4.2.2*

4-40 SHCS shear pins were chosen for the separation due to ease of installation.

5.3.2.6 *Derivation of Requirement V.1.4.2.3*

Steel 6-32 SHCS were chosen for all non-separating sections due to the different thread size and diameter when compared to the 4-40 SHCS shear pins. This prevents incorrect installation of stell 6-32 SHCS into separating sections.

5.3.2.7 *Derivation of Requirement V.1.4.2.4*

To prevent altimeter and payload failure, all pyrotechnic charges will be separated from the payload and on-board electronics.

5.4 System Level Trade Studies

5.4.1 Nose Cone Profile Trade Study

To maximize efficiency of the launch vehicle, the nose cone profile was optimized to reduce the overall coefficient of drag of the launch vehicle. The three nose cones considered for the full scale full scale launch vehicle include the LD Haack nose cone, the elliptical nose cone, and the conical nose cone, all of which are 12 inches in length. The two main factors that were considered during selection of the nose cone profile include efficiency and overall mass.

In order to determine the efficiency of each nose cone profile, a CFD simulation was run to determine the coefficient of drag of the entire launch vehicle with each nose cone profile. Each CFD simulation utilized an air flow speed of 700 feet per second, which is an approximation of the burnout velocity of the launch vehicle due to potential motor changes. The pressure results from a CFD simulation at 700 ft/s of the entire launch vehicle with each nose cone profile is shown below in Figure 26, Figure 27,and Figure 28.

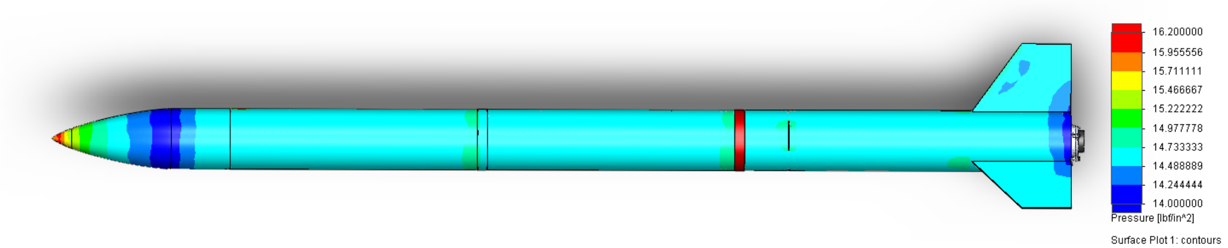


Figure 26: LD Haack nose cone CFD surface pressure plot.

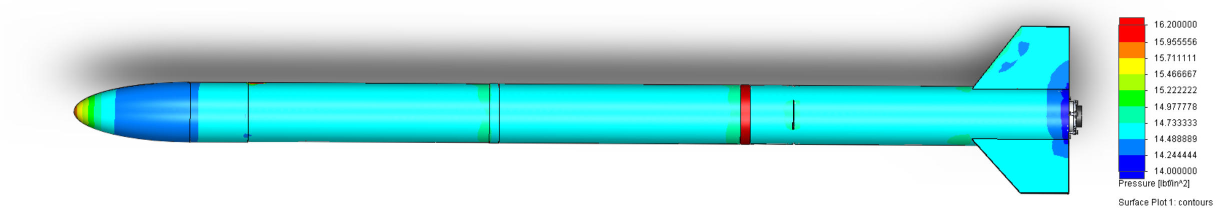


Figure 27: Elliptical nose cone CFD surface pressure plot.

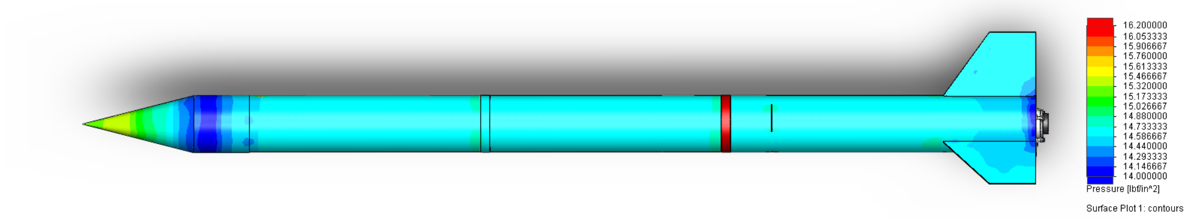


Figure 28: Conical nose cone CFD surface pressure plot.

The coefficient of drag of the launch vehicle determined from the CFD simulations for each nose cone profile is shown below in Table 26.

Nose Cone Profile	Coefficient of Drag at 700 ft/s
12 inch LD Haack	0.41852
12 inch Elliptical	0.4397
12 inch Conical	0.4459

Table 26: Simulated coefficient of drag for each nose cone profile.

As seen above, the 12 inch LD Haack nose cone has the lowest coefficient of drag when compared to the elliptical and conical nose cones. The weight of each nose cone profile was also compared. The weight of each nose cone profile was determined with a density of .43 oz/in^3, which is approximately the density of custom filament wound carbon fiber airframe. The weight of each nose cone profile with a wall thickness of 0.05 inches is shown below in Table 27

Nose Cone Profile	Weight (oz)
12 inch LD Haack	7.47
12 inch Elliptical	8.99
12 inch Conical	5.94

Table 27: Nose cone profile weights.

A trade study comparing the LD Haack nose cone, elliptical nose cone, and conical nose cone is shown below in Table 28.

Nose Cone							
Options:		12 in. LD Haack		12 in. Elliptical		6 in. Conical	
Mandatory Requirements							
Overall length does not exceed 12 inches.		Yes		Yes		Yes	
Coefficient of Drag less than 0.5.		Yes		Yes		Yes	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Drag Coefficient (0-10)	40.00%	9	3.6	8	3.2	7	2.8
Manufacturability (0-10)	20.00%	6	1.2	4	1.2	5	1
Price (0-10)	10.00%	5	0.5	6	0.5	5	0.5
Mass (0-10)	30.00%	3	0.9	4	0.6	4	1.2
Total Score		**6.2**		**5.8**		**5.5**	

Table 28: Nose cone Kepner-Tregoe table.

5.4.2 Fin Mounting System Trade Study

The fin mounting systems considered for implementation for the launch vehicle consisted of through-the-wall fin mounting, exterior bracket fin mounting, and the removable fin system. A Kenper-Tregoe trade study comparing all three systems can be seen below in Table 29.

Fin Mounting System							
Options:		Through-the-wall fin mounting		Removable Fin System		Exterior bracket mounted fins	
Mandatory Requirements							
Able to swap out fins after a flight		No		Yes		Yes	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Fin Connection Rigidity (0-10)	40.00%	10	4	9	3.6	10	4
Manufacturability (0-10)	20.00%	9	1.8	7	1.4	5	1
Price (0-10)	10.00%	7	0.7	4	0.4	1	0.1
Weight (0-10)	30.00%	5	1.5	8	2.4	2	0.6
Total Score		**8**		**7.8**		**5.7**	

Table 29: Kepner-Tregoe trade study comparing various fin mounting systems.

The removable system was chosen for the fin mounting system to be utilized in the launch vehicle due to it modularity and low weight when compared to other fin mounting systems.

5.5 Design

5.5.1 Airframe

In the past, the airframe and motor mount were constructed using 5 layers of 6k carbon fiber roving. The tubing was wound, from the innermost to outermost layer, at 45°, 35°, 45°, 35°, and 65° using a 4-axis X-Winder, as seen below in Figure 29.

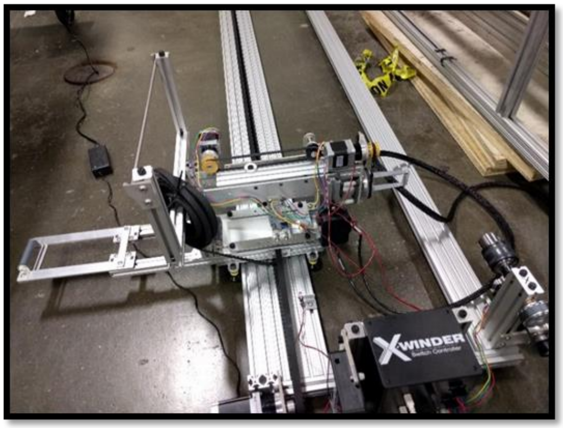


Figure 29: 4-axis X-Winder

With the objective of improving manufacturing efficiency and minimizing weight, further research and tests will be conducted on resins, filament winding angles, strength predictions and strength testing.

The current research of the effects of different winding angles on the strength and deformation of filament wound composite tubes has shown a direct relationship between decreased filament winding angle relative to the mandrel axis and resistance to axial tension and compression, as well as resistance to longitudinal bending. However, decreasing winding angle causes a decrease in winding filament consolidation on the mandrel during each pass of the X-Winder and an increased likelihood of filament slippage and bunching. Research has also shown a direct relationship between increased filament winding angle relative to the mandrel axis and resistance to circumferential tension and compression. A quasi-isotropic layup with 50% of fibers running in the direction of the load, 40% running 45˚ to the load and 10% running transverse to the load has been determined to have ideal properties.

A series of trial windings and tests will be performed to establish the lowest winding angles that will optimize both mechanical performance and manufacturing efficiency. Test tubes will be wound at the lowest possible angles gradually increasing the steepness of each tubes filament winding angle until a range of angles are established that are both consistently producible and low enough to take advantage of optimal mechanical properties of fiber reinforced plastics loaded in the direction of their fibers. Once an acceptable range of angles has been established, the tubes will be tested for fiber volume ratio and porosity to establish the true lowest ideal winding angle. The completed tubes will be cut transverse to the axis and the resulting rings will be cut into strips. The strips will be tested using the burnout method, based on ASTM's Standard Test Method for Ignition Loss of Cured Reinforced Resins (D2584-02), which compares the mass of the composite with the mass of the fibers left after combustion. The strips will also be optically inspected via photomicrographs of the polished ends of the samples to check for porosity and confirm the findings of the burnout method. The angle with the fewest defects and highest volumetric ratio below 65% will be selected and used for the 50% of fibers that would ideally run parallel to the axis of the mandrel.

Aramid, E-Glass and Carbon were considered for application in the airframe. A trade study of the fibers is shown in Table 30.

Fiber Type							
Fiber:		Carbon		E-Glass		Aramid	
Mandatory Requirements							
Able to withstand loads experienced during flight		Yes		Yes		Yes	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Strength (0-10)	20.00%	9	1.8	5	1	2	0.4
Fatigue Resistance (0-10)	10.00%	3	0.3	6	0.6	9	0.9
Electrical Insulation (0-10)	10.00%	3	0.3	9	0.9	9	0.9
Environmental Resistance (0-10)	10.00%	9	0.9	9	0.9	6	0.6
Adhesive Properties (0-10)	20.00%	10	2	10	2	5	1
Strength to Weight (0-10)	20.00%	10	2	5	1	8	1.6
Price (0-10)	10.00%	3	0.3	9	0.9	6	0.6
Total Score		**7.6**		**7.3**		**6**	

Table 30: Fiber trade study table.

Carbon is highly resistant to compressive, tensile and lateral deformation, as well as moderately resistant to fatigue. However, carbon is the most expensive of the options and has poorer impact and electrical insulating properties, causing issues with radio attenuation. Carbon's greatest advantage is the wide range of filament properties that are available, making it ideal for custom applications with specific material property requirements. Carbon fiber will be used for its high tensile and compressive strength, resistance to harsh environmental factors, strong adhesive properties and low weight.

The team will use MTS equipment to determine the tensile and compressive strength of the new carbon fiber airframe design wound on the X-Winder. Using this information as a baseline, the ideal filament characteristics and number of layers will be calculated using Mark E. Tuttle's AMTAS CLT program.

Polyester, vinyl ester, epoxy and epoxy vinyl ester resin (EVER) were considered for application in the airframe. A trade study of the characteristics of the resins is shown in Table 31.

Resin Type									
Resin:		Epoxy		EVER		Polyester		Vinyl Ester	
Mandatory Requirements									
Able to withstand loads experienced during flight		Yes		Yes		Yes		Yes	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Strength (0-10)	30.00%	9	2.7	9	2.7	5	1.5	7	2.1
Fatigue Resistance (0-10)	10.00%	8	0.8	8	0.8	6	0.6	4	0.4
Adhesive Properties (0-10)	20.00%	9	1.8	9	1.8	3	0.6	6	1.2
Environmental Resistance (0-10)	20.00%	7	1.4	9	1.8	3	0.6	5	1
Shrinkage (0-10)	10.00%	8	0.8	8	0.8	2	0.2	4	0.4
Price (0-10)	10.00%	4	0.4	3	0.3	9	0.9	6	0.6
Total Score		**7.9**		**8.2**		**4.4**		**5.7**	

Table 31: Resin trade study table.

EVERs combine many of the advantages of epoxy, vinyl ester and polyester resins without many of the disadvantages. EVERs have similar mechanical properties, similar heat resistance, better performance in harsh environments and better chemical resistance than epoxy resins. EVERs, like polyester resins, are also easier to work with due to their viscosity, shelf life and cure times, but are more forgiving to changes in environmental temperatures during curing. EVER will be used because of its tensile strength, resistance to fatigue, high adhesiveness, resistance to environmental factors and low volumetric shrinkage

The airframe will be constructed from a composite of carbon fiber and EVER for their mechanical and adhesive properties, light weight and resistance to harsh environmental factors. The composite will be wound on the X-Winder using a quasi-isotropic layup with 40% of fibers running 45˚ to the axis, 10% of fibers running perpendicular to the axis and 50% of fibers running as close to parallel to the axis as is optimal. The most optimal angle for the 50% of fibers running close to parallel to the axis will be determined using the ASTM D2584-02 burnout method and optical examination using photomicrographs.

5.5.2 Nose Cone Design

The nose cone will be secured to the launch vehicle via three 4-40 SHCS nylon shear pins. The nose cone will attach to the forward end of the deployment bay. The team decided to choose a nose cone profile that provided an ideal coefficient of drag at transonic speeds. The nose cone shape for the launch vehicle will be a 2:1 fineness ratio LD Haack series nose cone, which can be seen below in Figure 30.

Figure 30: Nose cone rendering.

The nose cone will be constructed from 6k carbon fiber filament using the X-Winder. In order to facilitate the filament winding manufacturing process, a nose cone mandrel was designed, which can be seen below in Figure 31.

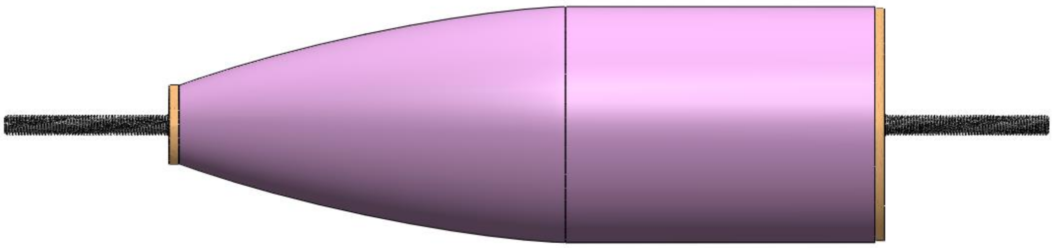


Figure 31: Nose Cone mandrel assembly.

The mandrel will be manufacturing using a ShopBot CNC Router from 3" extruded polystyrene. Two slices of extruded polystyrene will be made and glued together to form the shape of the nose cone. A central wooden plank that runs through a slot in the middle of the mandrel will rotate the entire mandrel assembly. Two 1/2" - 20 carriage bolts will be using to secure to the mandrel assembly to the X-Winder.

The tip of the nose cone will be manufactured from PLA plastic using a MakerBot Replicator 3D printer. It will be secured to the body of the nose cone via a 1/4” -20 all thread that is mounted to a wooden bulkhead secured to the interior of the nose cone by epoxy

A CFD simulation was performed to determine the force that the nose cone tip would experience at burnout velocity.

5.5.3 Avionics

Custom altimeter sleds have been designed to house the altimeters. The stratologger sleds will be 3D printed from PLA plastic using a MakerBot Replicator 3D printer. A rendering of the altimeter sled is shown below in Figure 32.

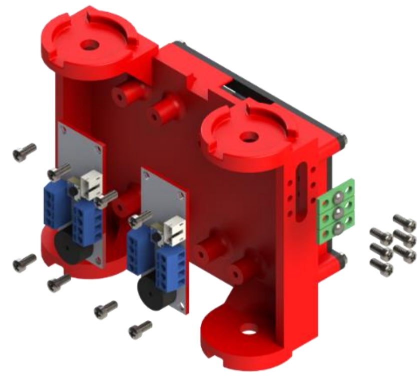


Figure 32: Altimeter sled assembly exploded view.

Each Perfectflite StratoLogger CF will be mounted using four 4-40 screws onto four extruded standoffs. Each altimeter will be powered by an individual Duracell 9V battery. Duracell batteries have been selected due to their reliability. Since the leads are internally soldered, the chance of battery failure from vibrations during flight is less likely than with a battery that does not have internally soldered leads. The batteries will be mounted on the opposite side of the altimeter sled, as shown in Figure 33.

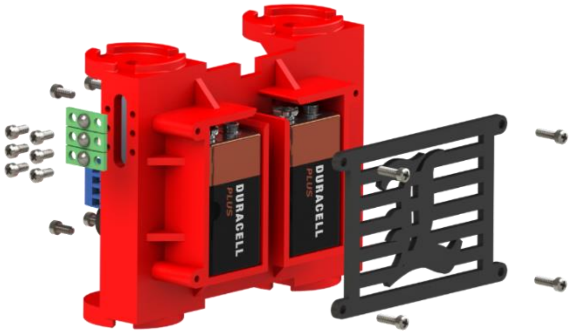


Figure 33: Battery view of altimeter sled.

5.5.4 Propulsion Bay

The two primary goals achieved with the propulsion bay are to serve as the attachment point for the removable fin system and house the motor and motor casing to propel the launch vehicle. The propulsion bay airframe will be constructed from 6.0 inch diameter filament wound carbon fiber tubing. In order to ensure that the fin slots are cut at the specified location, a jig has been created to mark where the slots would be placed using a Universal Laser Systems laser cutter. The jig is seen below in Figure 34.

Figure 34: Fin slot alignment jig.

Once drawn, the fin slots will be cut using a rotary Dremel tool with an abrasive cut off tool attachment. The thickness of the stencil, 0.125 inch, used in the jig is identical to that of the fins used in the launch vehicle, ensuring a near perfect fit with the fins.

Motor Tube

The motor mount tube will be constructed from 3.0 inch diameter filament wound carbon fiber tubing. The motor tube will be cut to a length of 16 inches, which is the length of the motor. This ensures proper axial alignment of the motor within the launch vehicle.

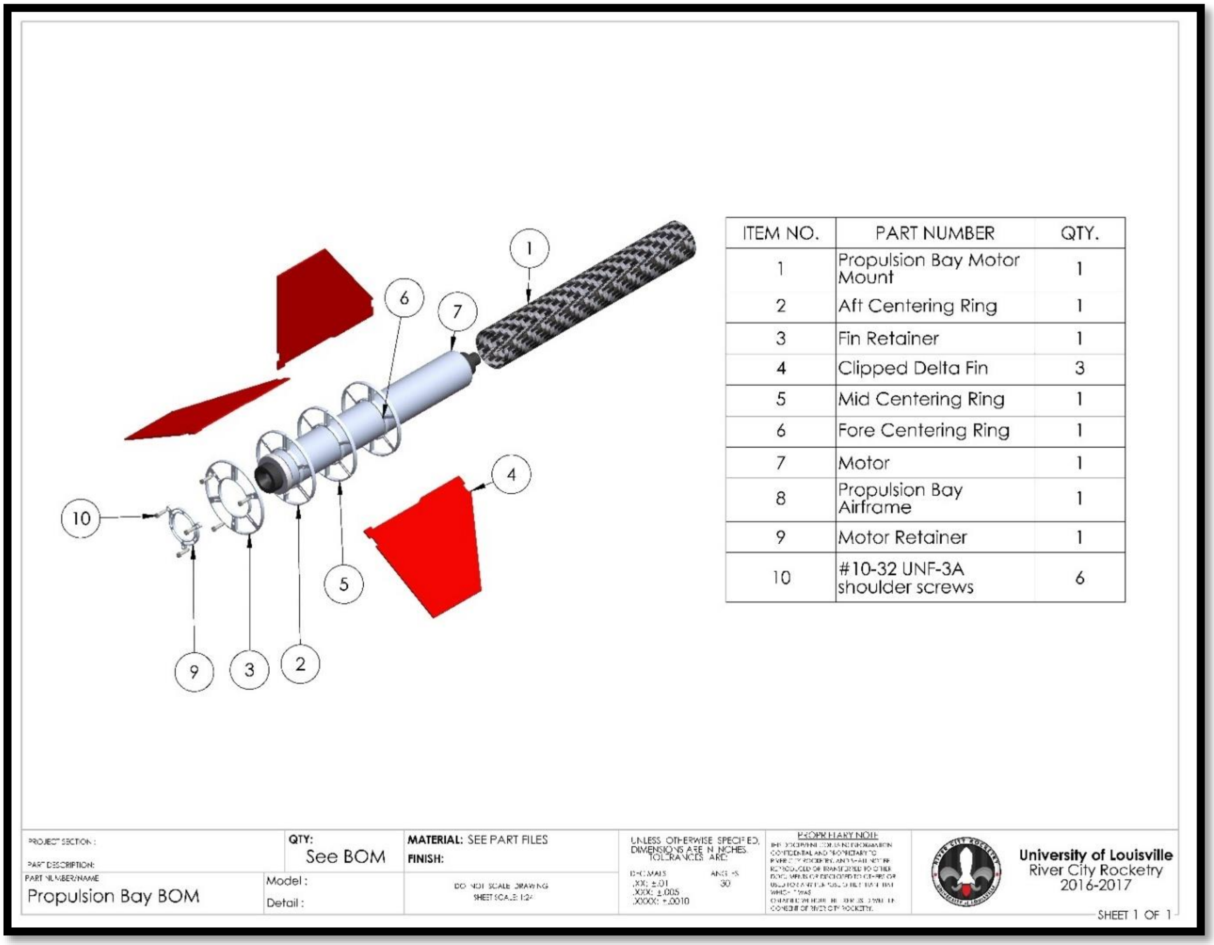
Removable Fin System

In order to reduce weight, and remove epoxy joints, a precision fin mounting system has been designed for the launch vehicle. This system eliminates the possibility of damaging fins or epoxy joints during transportation of the launch vehicle or during the landing of the launch vehicle. Additional fins will be readily available at launch, allowing for any damaged fin to immediately be replaced. Along with having the ability to replace damaged fins before a launch, the removable fin system also allows different fin designs to be utilized during test launches to account for mass changes throughout the year.

Figure 35: Propulsion bay assembly.

Figure 35 shows an assembled rendering of the removable fin system as it appears in the propulsion bay. The assembly consists of three centering rings, a rear fin retainer, and a motor casing retainer. The centering rings are the only components epoxied to the motor mount tube and airframe. Proper alignment of the centering rings is critical to the success of the removable fin system. To ensure proper alignment, the fins will be placed in the centering rings during the curing process of the epoxy.

With the motor installed in the casing and motor tube, the motor retainer mounts to the fin retainer via three #10-32 UNF-3A shoulder screws 1 inch in length. All fasteners in the system are made from 18-8 stainless steel. An exploded propulsion bay assembly and BOM is shown below in Figure 36.



ITEM NO.	PART NUMBER	QTY.
1	Propulsion Bay Motor Mount	1
2	Aft Centering Ring	1
3	Fin Retainer	1
4	Clipped Delta Fin	3
5	Mid Centering Ring	1
6	Fore Centering Ring	1
7	Motor	1
8	Propulsion Bay Airframe	1
9	Motor Retainer	1
10	#10-32 UNF-3A shoulder screws	6

Figure 36: Propulsion Bay BOM.

The fins are held in place in the propulsion bay by placing the fin tab into their proper slots in each centering ring. The fins are inserted into the centering rings and. Each fin is secured into the propulsion to the launch vehicle by four connection points. Each forward fin tab is pushed forward into the slot located in the fore centering. The slot in the mid centering ring and aft centering ring provides proper alignment for each fin with the motor mount. The fin is locked into place by sliding the aft fin tab into the alocated fin slot in the fin retainer. The fin retainer is mounted to the aft centering ring via three #10-32 UNF-3A socket head cap screws 1 inch in length. A schematic showing the fin tabs and connection points of the fins into the centering rings is show below in Figure 37.

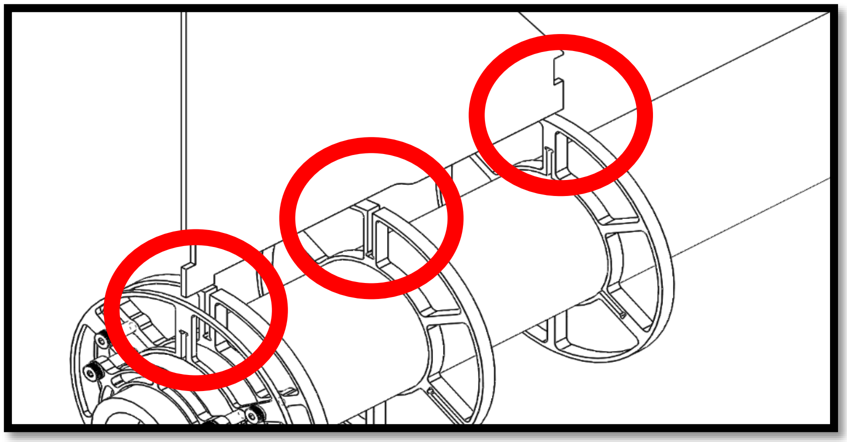


Figure 37: Removable fin system connection points.

Centering Ring Design

The centering rings will be custom manufactured from a Maxiem Water Jet from 6061 – T6 aluminum. All of the centering rings have specifically sized slots radially separated 120° to insert the three fins into the propulsion bay. A detailed drawing of the fore centering ring can be seen below in Figure 38.

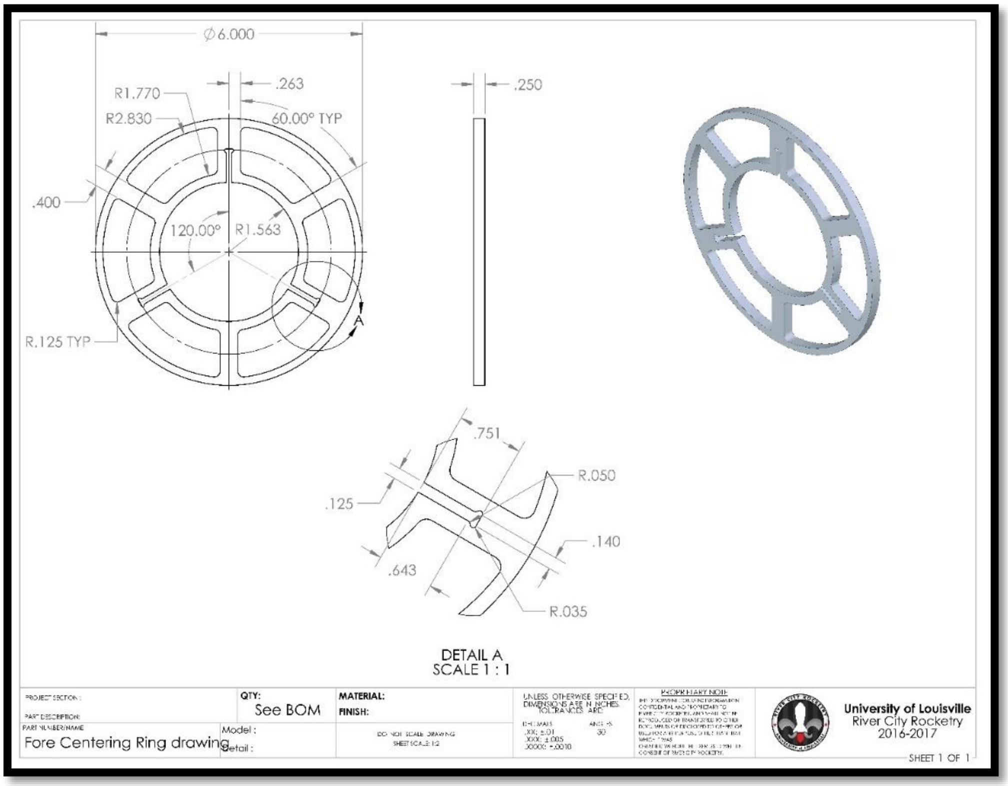


Figure 38: Detailed drawing of fore centering ring

Each centering ring has a set of three equally spaced weight reduction slots. While reducing weight, the weight reduction slots will also effect the strength of the section. To combat this issue, Finite Element Analyses (FEA) were performed for each centering ring with the following parameters conditions shown blow in Table 32.

Component	**Simulated Load (N)**	**% of Maximum Motor Thrust**
Fore centering ring	1550.9	50
Mid Centering Ring	1550.9	50
Aft Centering Ring	1550.9	50

Table 32: FEA simulation parameters.

The stress and displacement results can be seen below in Figure 39 and Figure 40.

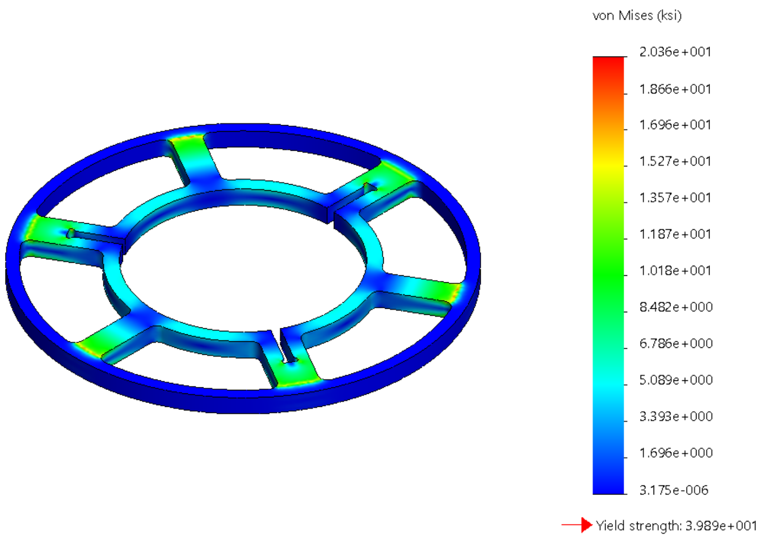


Figure 39: Finite element analysis stress plot.

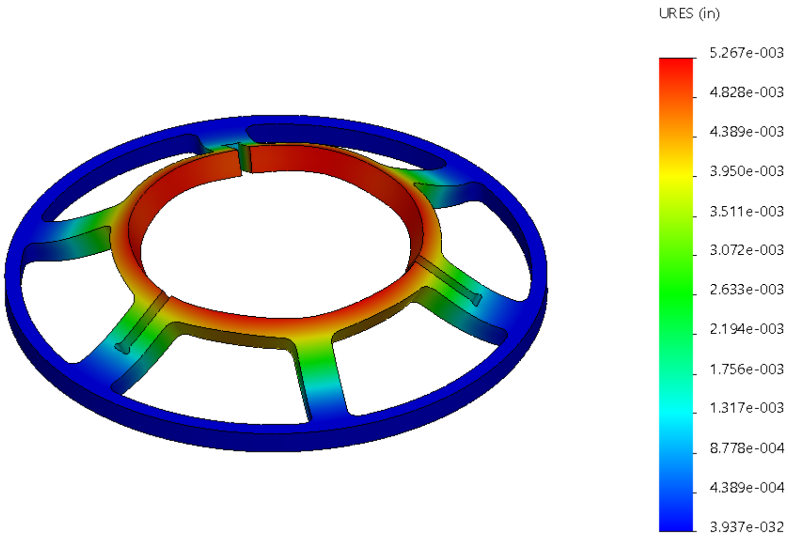


Figure 40: Finite element analysis displacement plot.

The minimum factor of safety throughout each centering ring with the simulated load and maximum displacement is shown below in Table 33.

Component	Minimum Factor of Safety	Maximum Displacement (in)
Fore centering ring	2.0	0.0053
Mid Centering Ring	2.0	0.0058
Aft Centering Ring	2.0	0.0054

Table 33: FEA centering ring results.

Motor Retention System

To properly secure the motor casing to the propulsion bay, a custom motor retainer has been designed. Since drogue recovery will not be facilitated by motor ejection, the motor retainer will be subjected to the following loads:

1. Supporting the weight of the launch vehicle while waiting for launch.
2. Support the weight of the motor casing with motor installed.
3. Withstand impact force of parachute deployment.

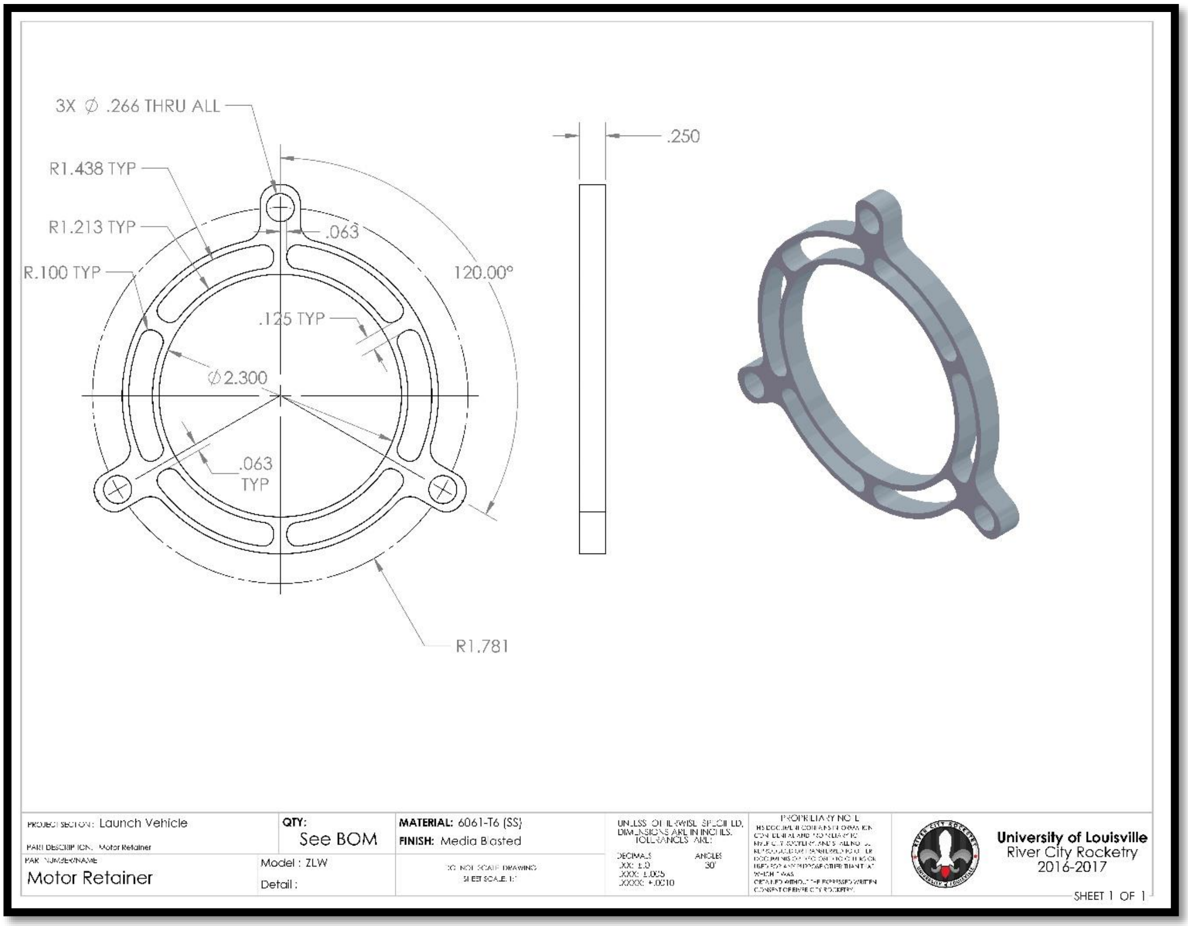


Figure 41: Detailed drawing of motor retainer.

Prior to motor installation, the launch vehicle fins and fin retainer must be installed. With the casing installed, the motor retainer will be attached to fin retainer via three #10-32 UNF-3A shoulder screws 1 inch in length. The motor retainer will be machined from 6061- T6 aluminum, using an OMAX Abrasive Waterjet.

Fin Design

In order to reduce drag and better compensate for the VDS, vehicle will utilize three fins. The fins will be constructed from carbon fiber. A material thickness of 1/8" was chosen for the fins as the launch vehicle will travel below supersonic speeds. The fins will be cut using an OMAX Abrasive Waterjet. This manufacturing method has been proven to be faster and more precise than traditional manufacturing methods. Figure 42 shows a detailed drawing of the launch vehicle fins.

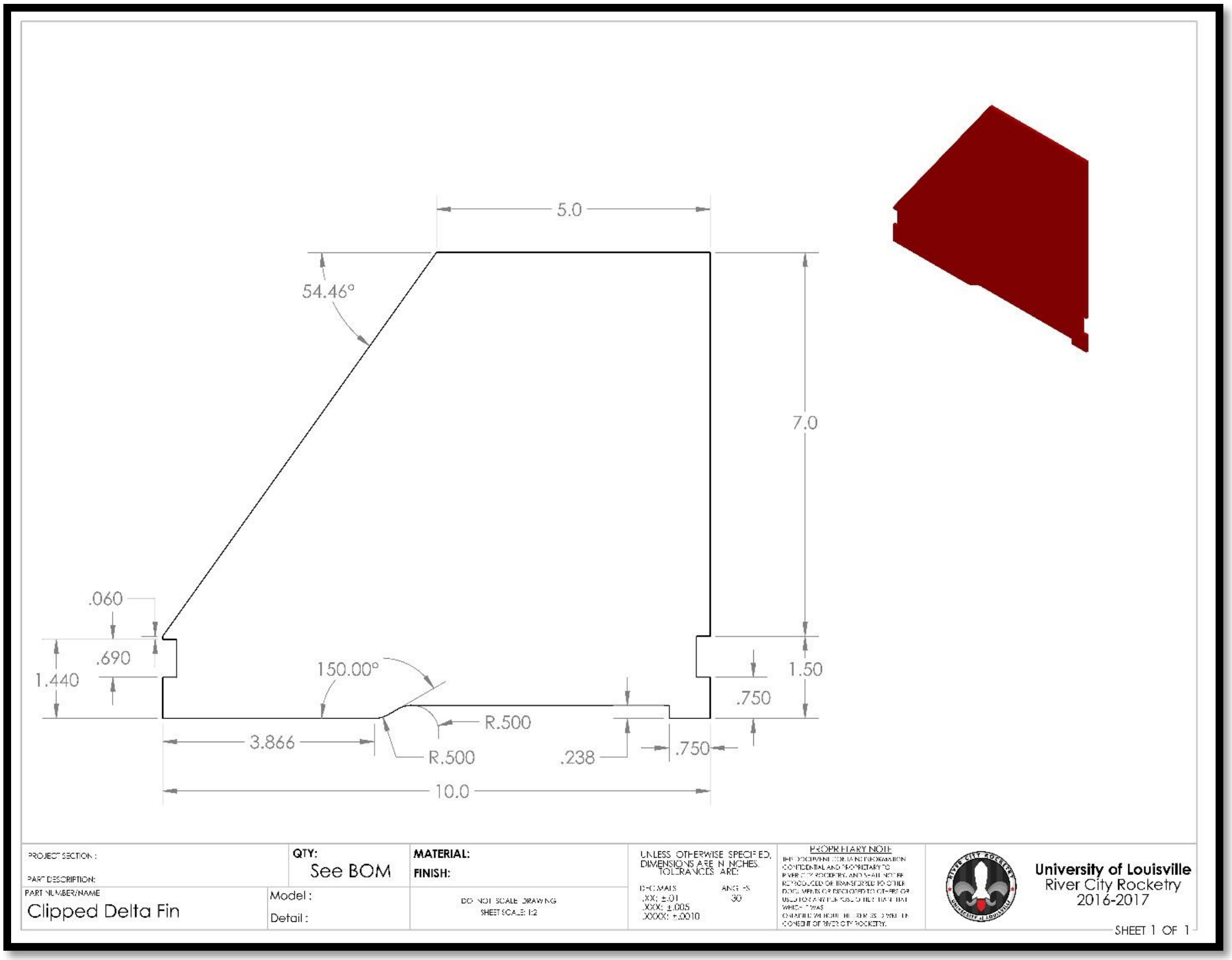


Figure 42: Detailed drawing of launch vehicle fin.

5.5.5 Subscale Vehicle

In order to test the design and aerodynamic characteristics of the launch vehicle, a one half scale model was designed. To facilitate a standard dual deployment recovery configuration, the payload and deployment bays featured in the full scale model was replaced with an altimeter bay and drogue parachute bay and the VDS bay was replaced with a main parachute bay. Additionally, recovery bay sizes were adjusted to allow adequate room for all recovery equipment. The subscale vehicle will utilize an AeroTech I285-R motor. The final subscale launch configuration is shown in Figure 43.

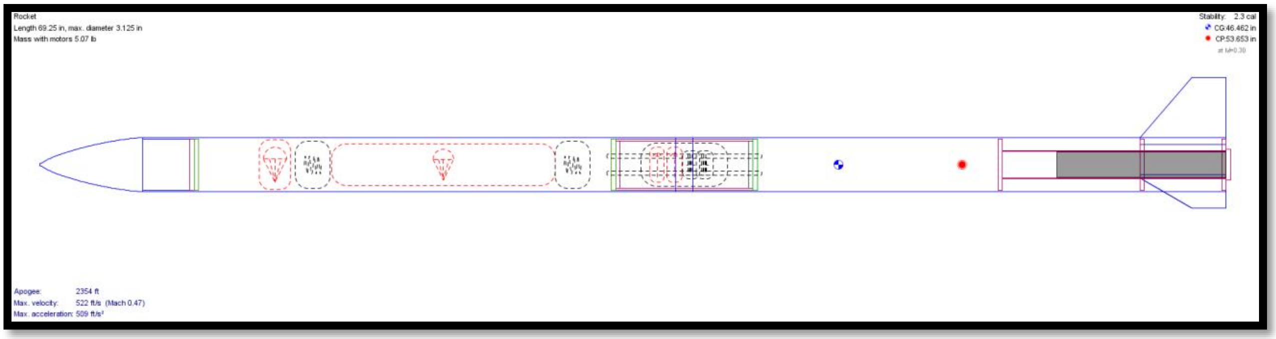


Figure 43: Subscale OpenRocket configuration.

It is imperative for the launch vehicle's flight verification that the subscale vehicle's flight see's similar aerodynamic stresses. The team designed the subscale to have a higher peak velocity and a similar maximum acceleration, as seen below in Table 34.

Property	Full Scale	Sub Scale
Diameter (in)	6	3
Length (in)	138	69
Burnout Weight (lbs)	45.9	5.98
Static stability margin (at rail exit)	2.2	2.3
Maximum velocity (ft/s)	721	522
Maximum acceleration (ft/s^2)	469	509
Exit Rail Velocity (ft/s)	97.6	108

Table 34: Comparison of properties between the full scale and subscale launch vehicles.

5.6 Mission Performance Predictions

5.6.1 Applicable Formulations

Three core values must be calculated to assess the stability and success of the rocket: peak altitude, center of gravity, and center of pressure. The peak altitude is found through a precise sequence of equations. The average mass, m_a, is first calculated using

$$m_a = m_r + m_e - \frac{m_p}{2} \tag{1}$$

where m_r is the rocket mass, m_e is the motor mass, and m_p is the propellant mass. The aerodynamic drag coefficient (kg/m) is then computed by

$$k = \frac{1}{2}\rho C_D A \tag{2}$$

where ρ is the air density (1.22 kg/m^3), C_D is the drag coefficient, and A is the rocket cross-sectional area (m^2). Equations 1 and 2 are utilized to calculate the burnout velocity coefficient (m/s) using

$$q_1 = \sqrt{\frac{T - m_a g}{k}} \tag{3}$$

where T is the motor thrust, and g is the gravitational constant (9.81 m/s^2). Equations 1, 2, and 3 are then used to compute the burnout velocity decay coefficient (1/s) using

$$x_1 = \frac{2kq_1}{m_a} \tag{4}$$

Equations 3 and 4 are used to calculate the burnout velocity (m/s) using

$$v_1 = q_1 \frac{1 - e^{-x_1 t}}{1 + e^{-x_1 t}} \tag{5}$$

where t is motor burnout time (s). The altitude at burnout can then be computed by

$$y_1 = \frac{-m_a}{2k} \ln\left(\frac{T - m_a g - k v_1^2}{T - m_a g}\right) \tag{6}$$

Once the burnout altitude is calculated, the coasting distance must be determined beginning with the calculation of the coasting mass using

$$m_c = m_r + m_e - m_p \tag{7}$$

The coasting mass replaces the average mass in equations 3 and 4; this results in equations 8 and 9 for the coasting velocity coefficient and coasting velocity decay coefficient, respectively:

$$q_c = \sqrt{\frac{T - m_c g}{k}} \tag{8}$$

$$x_c = \frac{2kq_c}{m_c} \tag{9}$$

Equations 8 and 9 can then be utilized to determine the coasting velocity (m/s) using

$$v_c = q_c \frac{1 - e^{-x_c t}}{1 + e^{-x_c t}} \tag{10}$$

The coasting distance can then be computed using

$$y_c = \frac{m_c}{2k} \ln\left(\frac{m_c g + k v^2}{T - m_c g}\right) \tag{11}$$

The peak altitude is then determined using

$$PA = y_1 + y_c \tag{12}$$

The center of gravity location is calculated using

$$cg = \frac{d_n w_n + d_r w_r + d_b w_b + d_e w_e + d_f w_f}{W} \qquad (13)$$

where W is the total weight, d is the distance between the denoted rocket section center of gravity (nose, rocket, body, engine, and fins, respectively) and the aft end. The center of pressure measured from the nose tip is calculated using

$$X = \frac{(C_N)_N X_N + (C_N)_F X_F}{(C_N)_N + (C_N)_F} \qquad (14)$$

where C_{NN} is the nose cone center of pressure coefficient (2 for conical nose cones), X_N is the computed by

$$X_N = \frac{2}{3} L_N \qquad (15)$$

where L_N is the nose cone length. C_{NF} in equation 14 is the fin center of pressure coefficient calculated using

$$(C_N)_F = \left[1 + \frac{R}{S+R}\right]\left[\frac{4N\left(\frac{S}{d}\right)^2}{1 + \sqrt{1 + \left(\frac{2L_f}{C_R + C_T}\right)^2}}\right] \qquad (16)$$

where R is the radius of the body at the aft end, S is the fin semispan, N is the number of fins, L_F is the length of the fin mid-chord line, C_R is the fin root chord length, and C_T is the fin tip chord length. X_F in equation 14 is calculated using

$$X_F = X_B + \frac{X_R(C_R + 2C_T)}{3(C_R + C_T)} + \frac{1}{6}\left[(C_R + C_T) - \frac{(C_R C_T)}{(C_R + C_T)}\right] \qquad (17)$$

where X_B is the distance from the nose tip to the fin root chord leading edge. X_R is the distance between the fin root leading edge and the fin tip leading edge measured parallel to body. Equations 14 through 17 are also known as the Barrowman Equations (The Theoretical Prediction of the Center of Pressure, 1966). Note that Equation 14 is a simplified form because the rocket has no transition in diameter in the body; thus, the transitional terms have been omitted. These equations are used to verify the OpenRocket simulation conducted of the full scale launch vehicle.

5.6.2 Flight Simulations

Using the OpenRocket software, mass measurements from previous years and component material densities were using to calculate the overall mass of the launch vehicle. While not every component mass has been accounted for in the OpenRocket simulation, such as epoxy and fasteners, the launch vehicle will still be able to achieve its target altitude through its use of the VDS. Table 35 lists the various weights of each section of the launch vehicle.

Section of launch vehicle	Length of section (in)	Weight (lbs)
Nose Cone Section	12	1
Payload Recovery Bay	25	4.44
Deployment Bay	34	2.33
Payload Section	12	7.86
Booster Recovery Bay	26	6.27

VDS Bay	12	3
Propulsion Bay with Motor	33	21
Total	**138**	**45.9**

Table 35: Launch vehicle overall dimensions.

An OpenRocket model of the full scale launch vehicle was created to verify Equations 1 through 17 as well as determine the overall flight characteristics. The specifications of the Open Rocket Simulation of the launch vehicle are shown in Table 36.

Center of Gravity (in from nose)	86.615
Center of Pressure (in from nose)	109
Rail exit velocity with 12 foot rail (ft/s)	97.6
Max. acceleration (ft/s2)	469
Predicted apogee altitude (ft)	5561
Thrust to weight ratio	14.65

Table 36: Launch vehicle flight specifications.

The launch vehicle will include three clipped delta fins. The clipped delta fin shape was chosen due to its efficiency and durability. Three fins were chosen rather than four to accommodate the VDS system to allow even airflow over all three fins. The drag blades of the VDS are offset by 60 degrees relative to the fins. One concern the team faced during the integration of the launch vehicle with the VDS was disruption of airflow around the fins during flight. At 700 ft/s, which is an approximation for the burnout velocity of the launch vehicle, a CFD and validates that the turbulent air flow from the drag blades do not interfere with the air flow over the fins, as shown in Figure 44 below.

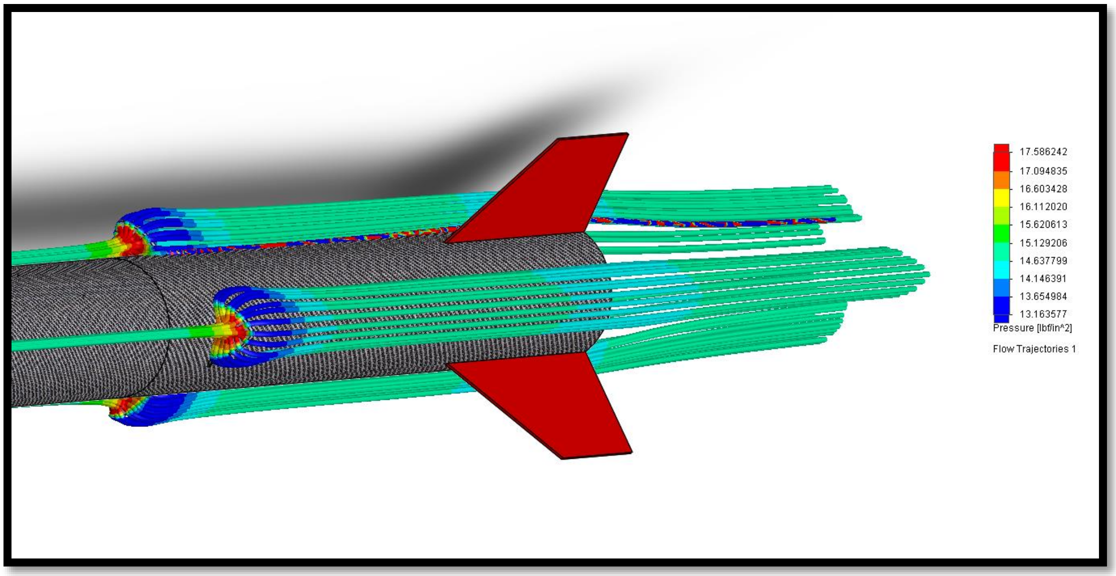


Figure 44: VDS air flow CFD simulation results.

5.6.3 Motor Selection

Several OpenRocket simulations were conducted with different motor configurations in order to choose the motor that produced an appropriate apogee altitude. Simulations were conducted for motors ranging from 4600 Newton-seconds to 5120 Newton-seconds, which is the maximum total impulse for a motor set by the Statement of Work. Table 37 below shows the combination of every available motor and the corresponding loaded launch vehicle weight that results in an apogee altitude of 5,500 feet.

Motor	Overall Launch Vehicle Weight (lbs)
Aerotech L1420	39.9
AeroTech L952	44.8
AeroTch 2500	40.4
AeroTech L1365	42.0
Cesaroni L3150	42.3
Cesaroni L1410	42.6
Cesaroni L610	41.9
Cesaroni L2375	43.6
Cesaroni L1395	43.3
Cesaroni L1115	44.8
Cesaroni L1685	46.7
AeroTech L1500	45.2
AeroTech L2200	46.5

Table 37: Motor and weight combination required to achieve apogee altitude of 5,500 feet.

The motor choice has been made with consideration of motor availability and the maximum allowable weight to reach approximately 5,500 feet. Due to the mass of the payload and desired apogee altitude of the VDS, the full scale launch vehicle will utilize the Aerotech L2200 Mojave Green motor. This motor was chosen due to its desired total impulse, brand reliability, and availability. With this motor, the launch vehicle will reach an estimated apogee altitude of 5561 feet. This apogee altitude was chosen to utilize the

VDS, which will deploy the drag blades to decrease the apogee altitude to 5280 feet. The thrust vs time curve and the specifications of the Aerotech L2200 Mojave Green motor can be seen blow in Figure 45 and Table 38, respectively.

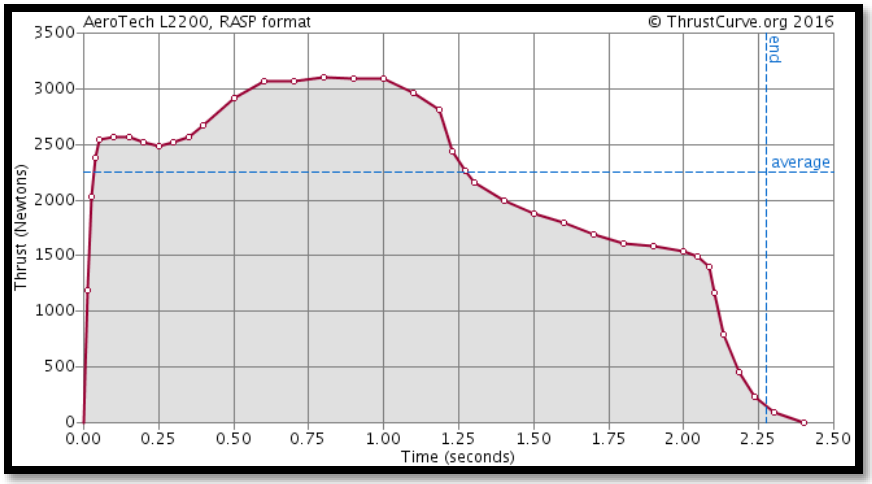


Figure 45: Aerotech 2200G Thrust Curve.

Diameter	75.0 mm
Total Weight	167.59 oz
Propellant Weight	88.75 oz
Average Thrust	2200.0 N
Maximum Thrust	3101.8 N
Total Impulse	5104.1 N-sec
Burn Time	2.3 sec

Table 38: Aerotech L2200 Mojave Green Specifications.

5.6.4 Flight Simulations

As eluded to in the VDS Section, the launch vehicle will aim for an apogee altitude higher than 5280 ft in order to account for variances in launch condition. The launch vehicle will be designed to reach an apogee altitude of approximately 5,500 ft to provide a 100 ft buffer for the penalty associated with an apogee altitude of 5,600 ft in case the VDS does not deploy. Technical Design: Variable

An OpenRocket model, which is shown below in Figure 46, was created to verify Equations 1 through 17, which calculate the location of the center of gravity, location of the center of pressure, and apogee altitude of the launch vehicle.

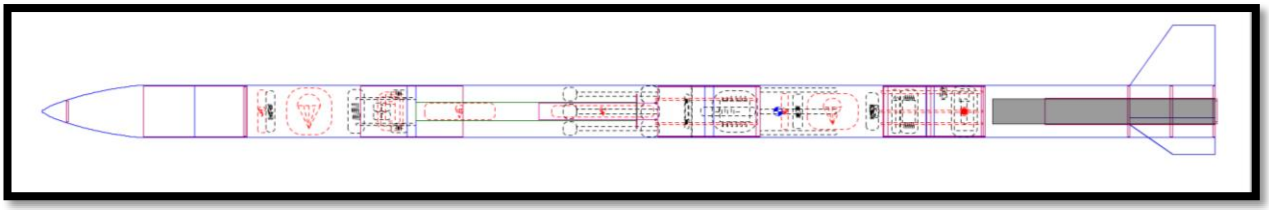


Figure 46: OpenRocket full scale launch vehicle configuration.

The following plots shown in Figure 47 through Figure 50 display various simulations results without any VDS involvement, indicating proper motor selection and vehicle stability.

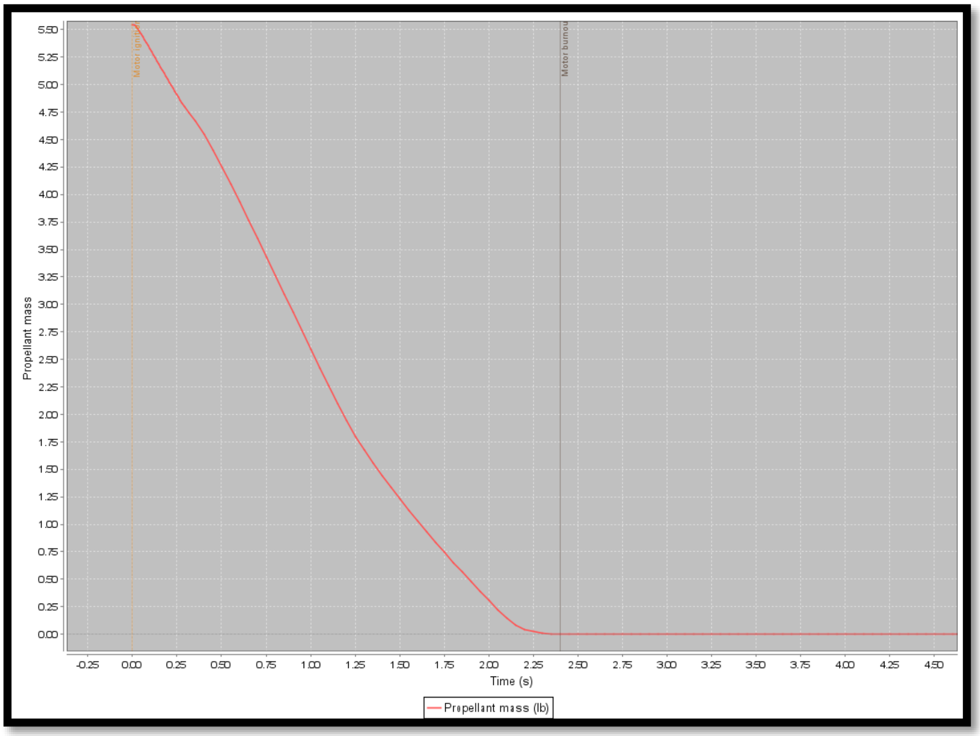


Figure 47: A plot of propellant mass versus time

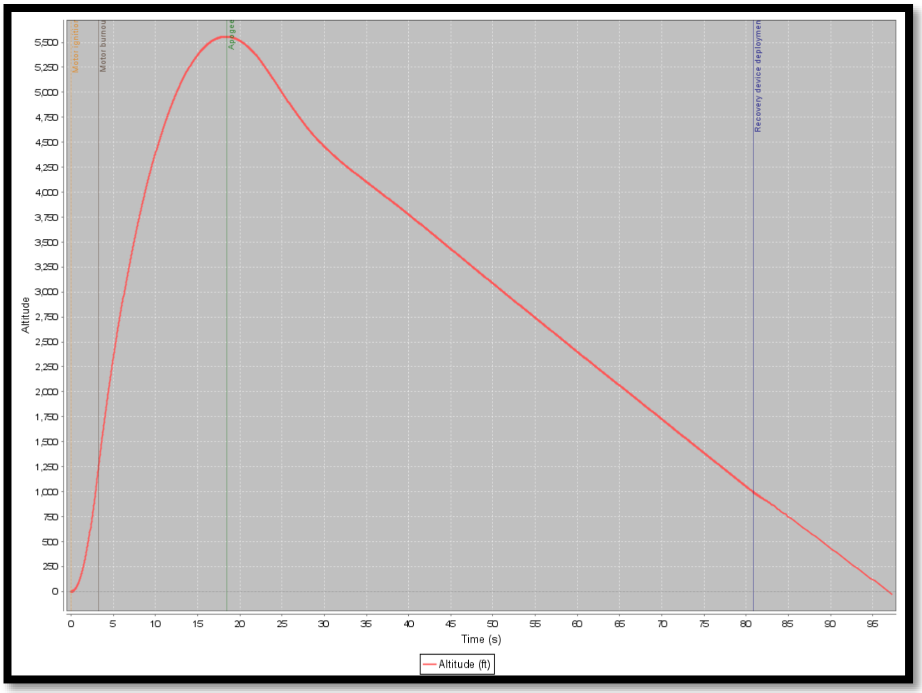


Figure 48: Altitude vs time without VDS.

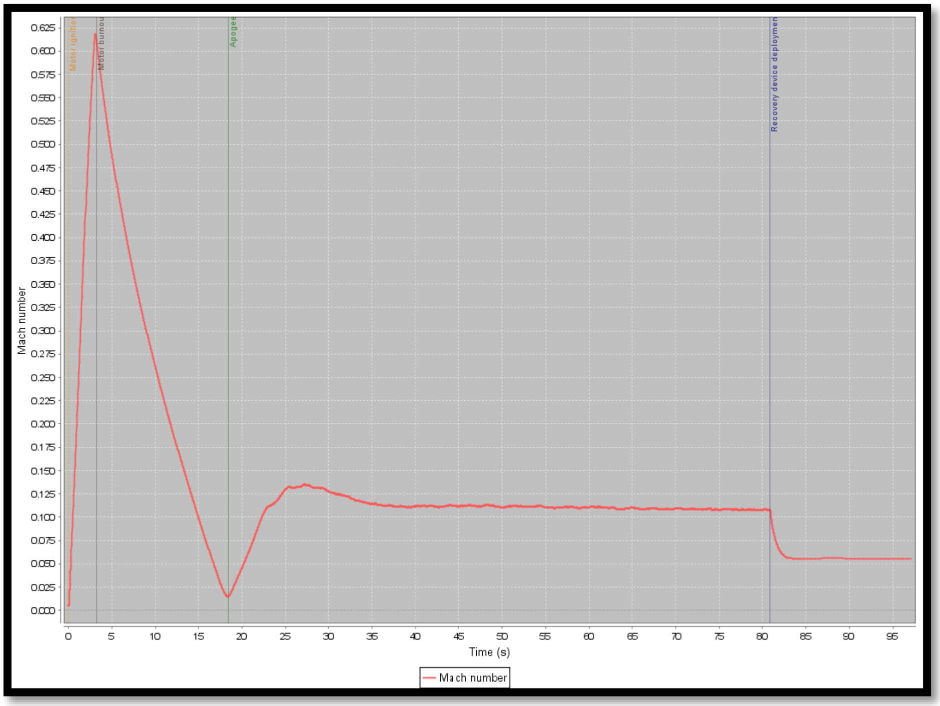


Figure 49: A plot of Mach number versus time.

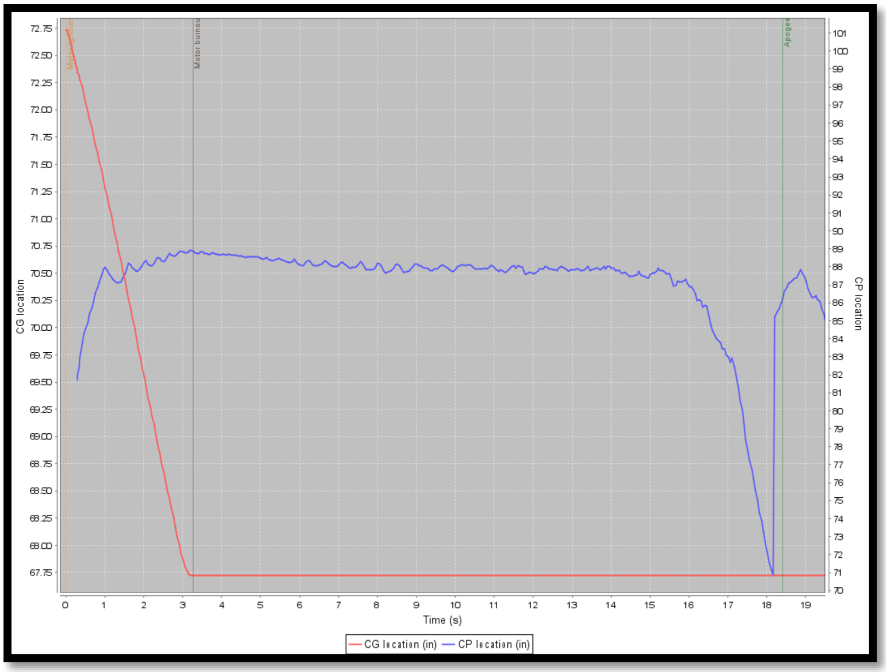


Figure 50: A plot of CG and CP locations versus time.

5.6.4.1 *VDS Flight Performance Predictions*

Because OpenRocket is only capable of simulating the flights of standard rockets, the team has developed its own simulation to provide mission performance predictions for the vehicle with the VDS system. This simulation incorporates the kinematics of vehicle ascent, the responses of the control scheme, and the mechanics of the VDS's actuators. This simulation has been used to predict flight performance, tune controls parameters, and derive performance requirements. The Mathworks Simulink blocks for the simulation are shown below in Figure 51.

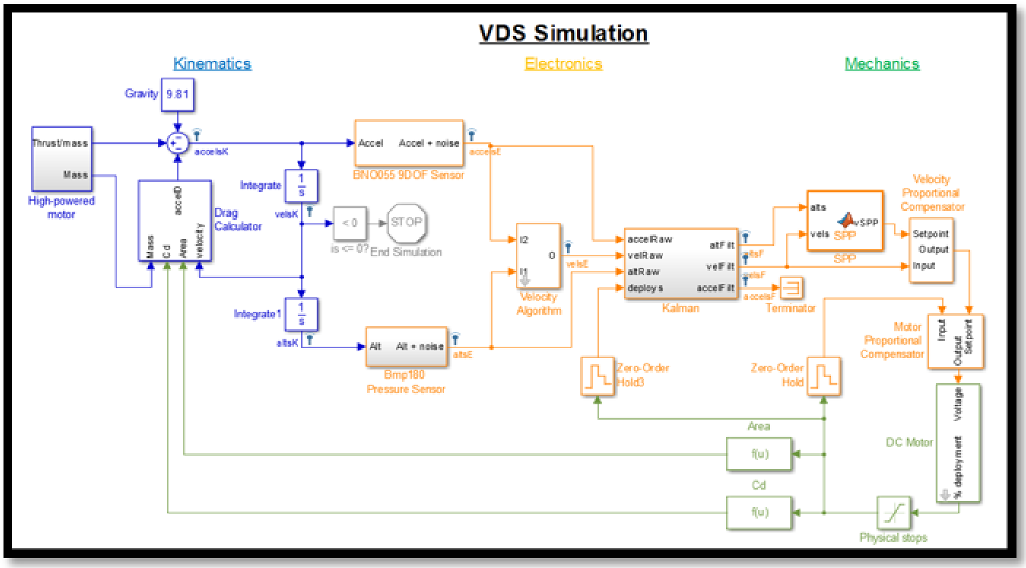


Figure 51: VDS simulation Simulink blocks.

This simulation includes factors such as state noise, data sample frequency, DC motor response time, drag coefficient, and motor selections. It has been used to evaluate trade-offs between different system level design choices such as in system level design trade study and in sensor schemes. It has also been used to derive certain performance requirements such as in the derivation of the VDS sensor fidelity requirements.

An example output of this simulation is shown below in Figure 52 and Figure 53.

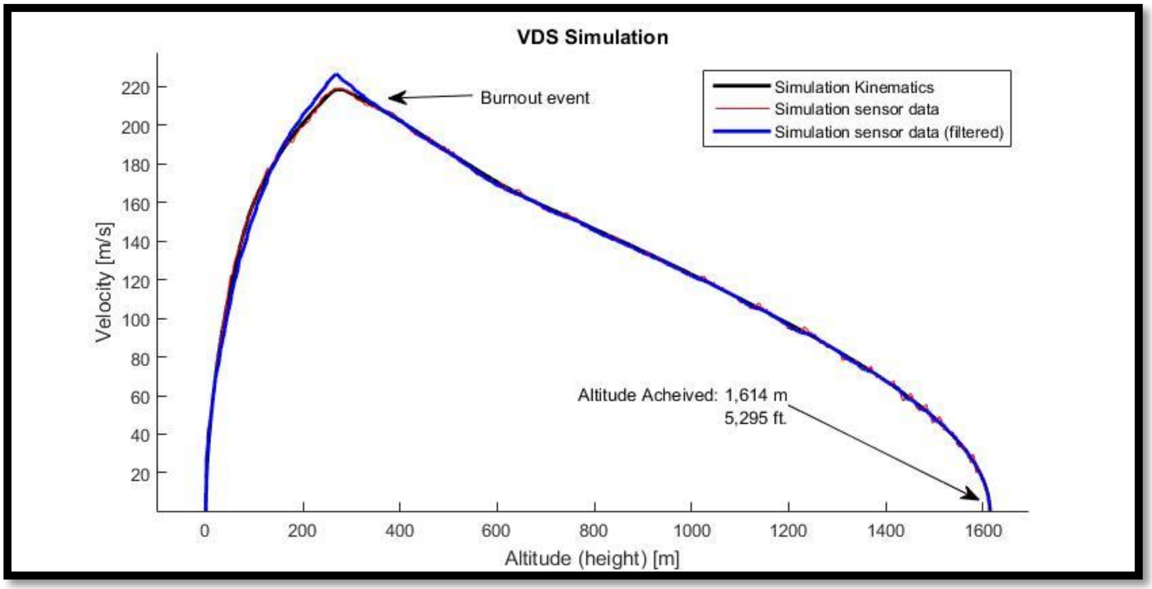


Figure 52: VDS simulation flight.

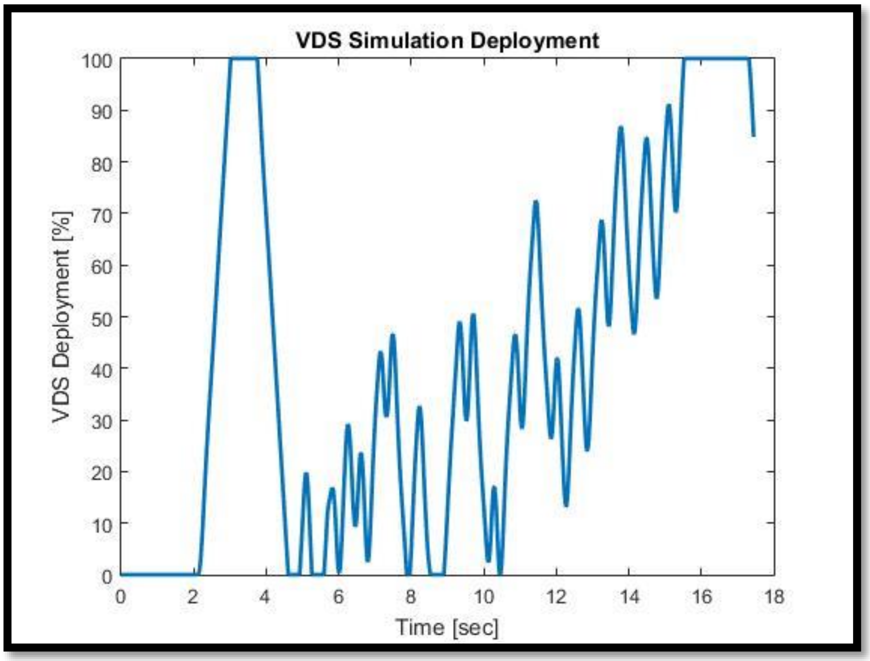


Figure 53: VDS Simulation deployment.

5.7 Project Plan

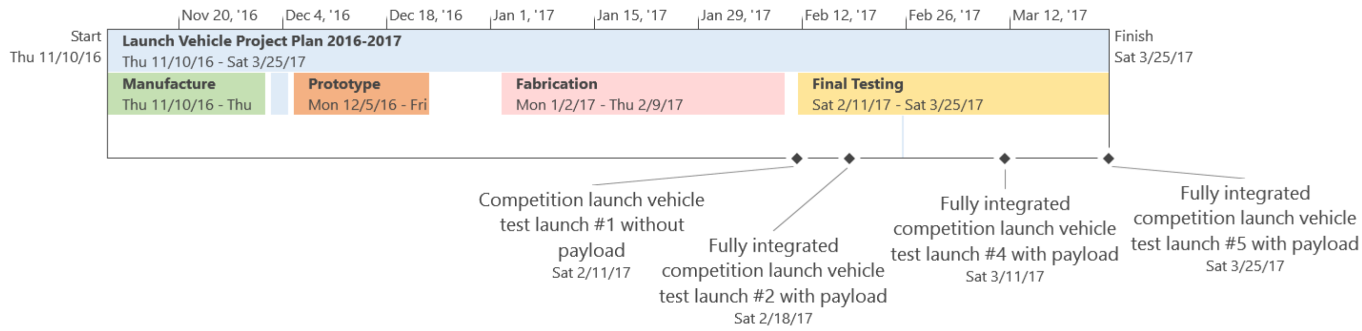


Figure 54: Launch vehicle overview schedule.

5.8 Safety

Stability and Propulsion Risk Assessment						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity**	**Probability**	**Risk**	**Mitigation**
Motor fails to ignite.	1. Faulty motor. 2. Delayed ignition. 3. Faulty e-match. 4. Disconnected e-match.	1,3,4. Rocket will not launch. 2. Rocket fires at an unexpected time.	3	4	Low	Follow NAR safety code and wait a minimum of 60 seconds before approaching the rocket to ensure that the motor is not simply delayed in launching. If there is no activity after 60 seconds, have the safety officer check the ignition system for a lost connection or a bad igniter. If this does not fix the failure mode, be prepared to remove the ignition system from the rocket motor, retrieve the motor from the launch pad and replace the motor with a spare. Igniters have been securely installed throughout the season, having a 100% success rate.

Motor catastrophic failure occurs.	1. Faulty motor. 2. Packing the motor incorrectly. 3. Igniter not inserted properly.	Rocket and interior components significantly damaged. Possible injury to personnel or spectators.	1	5	Moderate	Confirm that all personnel are at a distance allowed by the Minimum Distance Table as established by NAR in order to ensure that no one is hurt by flying debris. Extinguish any fires that may have been started when it is safe to approach. Collect all debris to eliminate any hazards created due to explosion. The motors the team have selected are from a reliable supplier. The team has had a 100% success rate.
Rocket doesn't reach high enough velocity before leaving the launch pad.	1. Rocket is too heavy. 2. Average thrust of motor is too low. 3. High friction coefficient between rocket and launch tower. 4. Rail buttons shear during liftoff.	1,2. Unstable launch.	1	5	Moderate	Too low of a velocity will result in an unstable launch. Simulations are run to verify the motor selection provides the necessary exit velocity. Should the failure mode still occur, the issue should be further examined to determine if the cause was due to a faulty motor or in the booster needs to be redesigned.
Fins shear during flight.	Improper material selection and size for the fins and centering rings.	Unstable rocket, causing the flight path to become unpredictable.	1	5	Moderate	Confirm all personnel are alert and at a distance allowed by the Minimum Distance Table as established by NAR. Examine external epoxy beads for cracks prior to launch.
Airframe buckles during flight.	Airframe encounters stresses higher than the material can support.	Rocket will become unstable and unsafe during flight.	1	5	Moderate	Through prediction models, appropriate material selection, and a secure factor of safety, this failure mode can be nearly eliminated.
Internal bulkheads fail during flight.	Forces encountered are greater than the bulkheads can support.	1. Internal components supported by the bulkheads will no longer be secure. 2. Parachutes attached to bulkheads will be left ineffective.	1	5	Moderate	The bulkheads will be designed to withstand the force from the motor firing with an acceptable factor of safety. 1. Electrical components could be damaged and will not operate as intended during flight. 2. A catastrophic failure is likely. A portion of the rocket or the fairing would become ballistic.

Centering rings fail.	Epoxy is not properly applied to centering rings.	Motor is propelled through the inside the launch vehicle.	1	3	High	This probability will be mitigated through verification of the subscale construction techniques followed by a successful flight.
Motor retainer fails.	Joint did not have proper preload or thread engagements.	Motor casing and spend motor falls out of launch vehicle when the main parachute opens.	1	5	Moderate	The motor retainer will be properly installed and designed to withstand all induced loads during flight.

Table 39: Stability and Propulsion Risk Assessment

Vehicle Assembly Risk Assessment						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity**	**Probability**	**Risk**	**Mitigation**
Rocket drop (INERT)	Mishandling of the rocket during transportation.	Minimal damage and scratches to components of the rocket.	4	5	Low	The rocket has been designed to be durable in order to survive loads encountered during flight and upon landing. Careful handling should be practiced while transporting the rocket.
Rocket drop (LIVE)	Mishandling of the rocket during transportation.	1. Minimal damage and scratches to components of the rocket if no charges go off. 2. Charges prematurely go off, resulting in a serious safety threat to personnel in the area and significant damage to the rocket.	1	5	Moderate	The rocket has been designed to be durable in order to survive loads encountered during flight and upon landing. Careful handling should be practiced while transporting the rocket.
Black powder charges go off prematurely	1. Altimeters send a false reading. 2. Open flame sets off charge.	1,2. Charges prematurely go off, resulting in a serious safety threat to personnel in the area and significant damage to the rocket.	1	5	Moderate	All electronics will be kept in their OFF state for as long as possible during preparation. Open flames and other heat sources will be prohibited in the area.

Seized nut or bolt due to galling or cross threading	Repetitive uninstalling and reinstalling of parts made of materials prone to galling.	Component becomes unusable, potentially ruining expensive, custom machined parts. Amount of rework depends on the location and component that seized.	2	4	Moderate	Through proper choice in materials, appropriate pre-load, and proper installation, the risk of galling can be eliminated.
Pinched shock cord lines or shroud lines	Poor packing of the parachute and its shroud lines. Not following packing procedure check list.	Line over occurs on deployment bag, causing no deployment of main parachute. Shock cord gets tangled causing damage to vehicle and its components.	1	5	Moderate	Training on packing the parachute along with a detailed check list to follow during launch preparation. Keeping two personal's eyes on the packing of the recovery scheme.

Table 40: Vehicle assembly risk assessment table.

6 Technical Design: Recovery

6.1 Self-Derived System-Level Requirements

In addition to the requirements provided in the statement of work (**SOW**) outlined in Table 55 at the end of this section, requirements specific to the payload challenge have been derived to ensure the success and safety of the multirotor. These requirements are named with the convention of R.1.x.x to differentiate them from the SOW requirements. The highest level of the R.1 requirements are featured below.

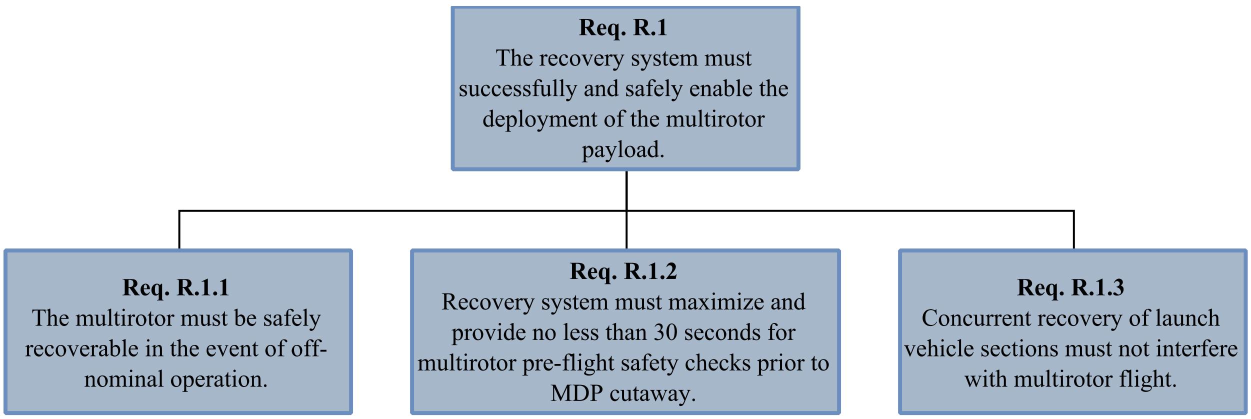


Figure 55: System level requirements

The self-derived requirements and their proposed methods of verification are covered in Table 41 below.

Requirement Number	Requirement	Verification
R.1	The recovery system must successfully and safely enable the deployment of the multirotor.	Test Fully integrated full-scale test flights will be conducted to confirm that no collisions or off-nominal cases are caused by the recovery system.
R.1.1	The multirotor must be safely recoverable in the event of off-nominal operation.	Analysis All recovery phases of multirotor will be designed and calculated to be under kinetic energy requirement, or capable of aborting to such a mode.
R.1.2	Deployment bay recovery system must maximize and provide no less than 30 seconds for multirotor pre-flight safety checks prior to MDP cutaway.	Analysis Recovery system parameters will be optimized and calculated to verify that the elapsed multirotor descent time when under MDP is ≥ 30s.
R.1.3	Concurrent recovery of launch vehicle sections must not interfere with multirotor flight.	Test Fully integrated full-scale test flights will be conducted to confirm that predictive recovery data given to multirotor enables collision avoidance.

Table 41: R.1 requirement verifications

6.2 Design Overview

To accommodate the deployment of the multirotor, the launch vehicle will have to be staged into 2 untethered sections at apogee. At apogee, the launch vehicle will separate at its midsection, and the booster section will recover separately from the upper deployment bay. This midsection separation is necessary to enable the multirotor deployment and to reduce the kinetic energy during the high velocity drogue descent of the payload section. Both sections feature a dual deployment from a single recovery bay through the utilization of Advanced Retention Release Devices (**AARD**s). Upon reaching the target multirotor deployment altitude, the multirotor will be jettisoned via black powder charges, and will descend under a multirotor deployment parachute (**MDP**). Once the multirotor preflight safety checks are complete to ensure nominal operation, the MDP cutaway maneuver will be executed and the multirotor will begin autonomous flight.

6.2.1 Staging Procedure

In order for the vehicle and multirotor to be recovered safely and in a reusable state, 5 sequentially staged deployment events will have to occur, with a potential 6th* auxiliary deployment in the multirotor abort configuration. The sequence is detailed below.

Event	Altitude	Phase	Description
1	5,280 ft.	Booster Drogue Event	Launch vehicle separates at the midsection of the vehicle via black powder charge and shear pin configuration. Booster section begins drogue descent.
2	5,280 ft. +2 sec. delay	Deployment Bay Drogue Event	Upper section of launch vehicle containing deployment bay separates at nose cone, beginning its drogue descent.
3	~1700 ft.	Deployment Bay Main Event	ARRD disengages deployment bay drogue shock cord, and causes main deployment.
4	~1350 ft.	Multirotor Deployment	Multirotor is ejected from deployment bay via black powder charges, bringing the deployment bay under the kinetic energy requirement, and initiating deployment of the MDP
5	~600 ft.	Booster Main Deployment	ARRD disengages booster drogue shock cord, and causes main deployment.
6*	N/A	Multirotor Abort	If multirotor kinetic energy exceeds 75 ft-lb during autonomous flight, multirotor reserve parachute (**MRP**) will deploy.

Table 42: Recovery staging procedure.

The sequence of recovery events described in Table 42 are procedurally illustrated below in Figure 56.

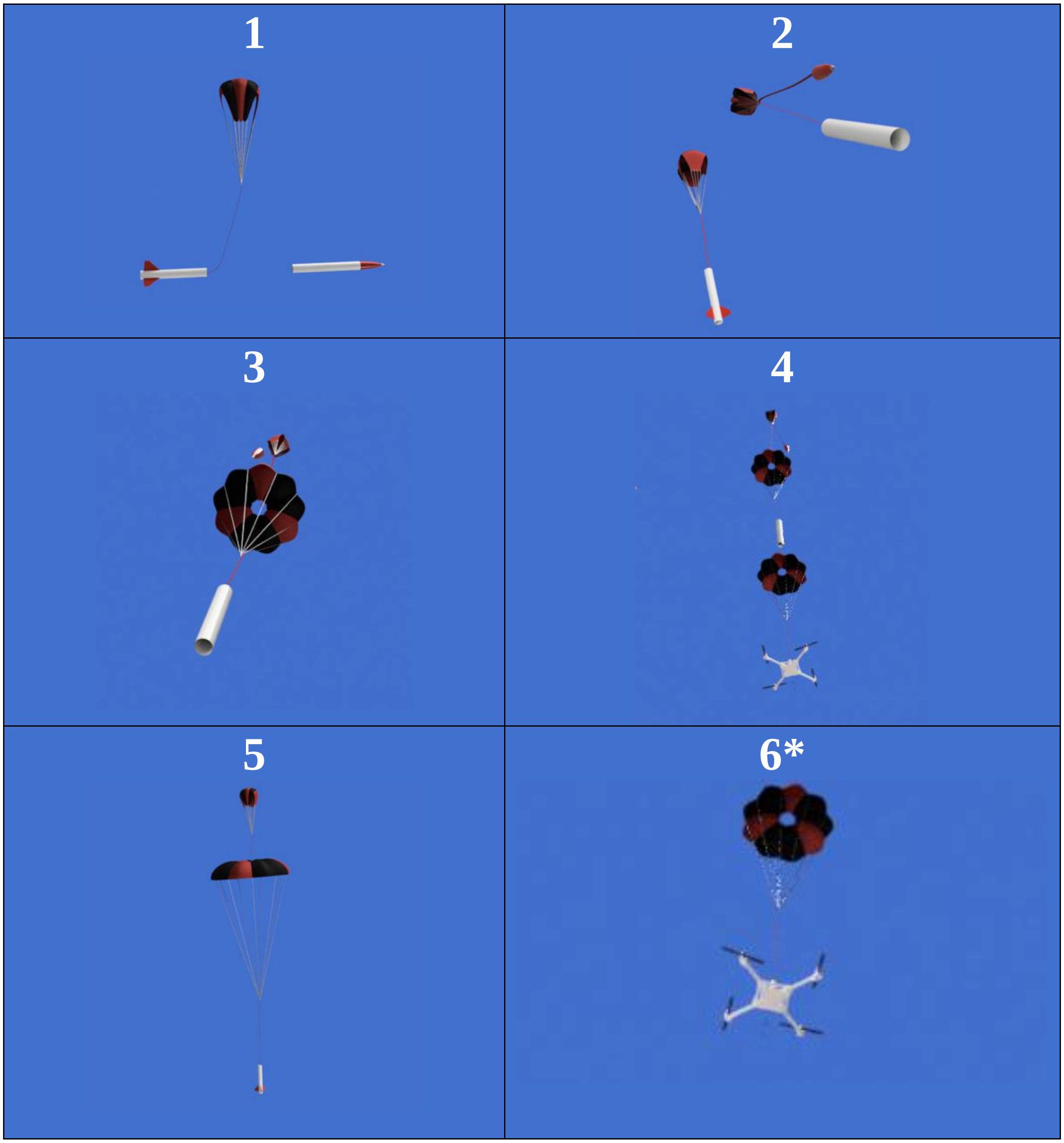


Figure 56: Illustration of sequential recovery events

6.2.2 ARRD Deployment

Due to the complex sequence of recovery events needed to ensure the safety of multirotor and the need to minimize recovery weight for precise target apogee acquisition, it was decided to perform a single-bay dual deployment for both the booster section and payload section. In order to enable this, the recovery systems for both sections will be stowed with the drogue and main parachute in the same recovery bay. In this configuration, both systems will use ARRDs to anchor the shock cord of the drogue to the bulk plate underneath the main deployment bag. The ARRD in Figure 57 below is a robust assembly available from RATTworks that serves as a load bearing connection point until a black powder charge forces the eyelet out of the assembly, freeing the drogue shock cord. This ensures that the drogue does not act as the main pilot chute until the desired deployment altitude.

Figure 57: ARRD disassembled to show black powder chamber and releasable eyelet

As seen below in Figure 58, the ARRD assembly provides a temporary connection point directly to the vehicle's bulkplate for the drogue parachute. This connection maintains slack in the shock cord that runs from the eyelet on the ARRD to the top of the main deployment bag, passively tethering the drogue to the main deployment bag.

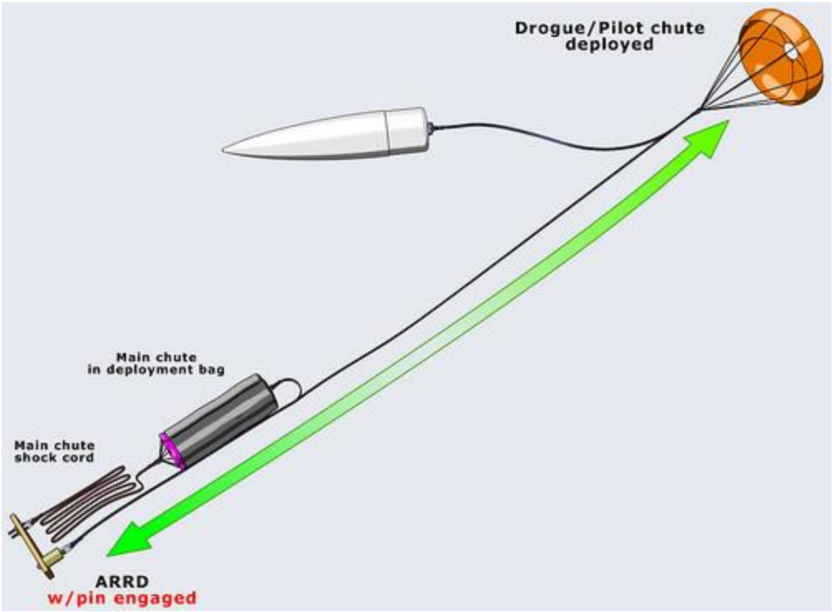


Figure 58: AARD retention configuration of drogue and main during drogue descent.

During main event, the drogue shock cord is freed from the ARRD, activating the tether to the deployment bag and allowing the drogue to now act as a pilot chute for the main parachute. The pilot pulls the deployment bag from the bay, and at line stretch pulls the bag off of the main parachute and frees the nosecone to descend under the retired drogue as shown below in Figure 59

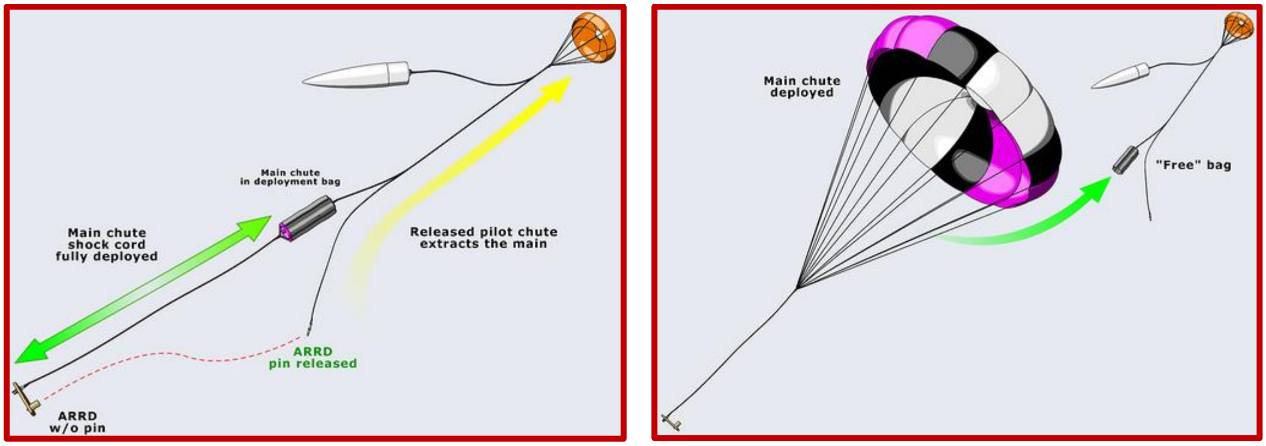


Figure 59: ARRD deployment of main parachute

Each system will be triggered by a redundant set of PerfectFlite StratologgerCF's. The PerfectFlite StratologgerCF's altimeter records its altitude at a rate of 20Hz with a 0.1% accuracy. In previous testing, the altimeter was found to be accurate to ±1 foot. The StratoLogger can be configured to provide a constant serial (UART) stream (9600 baud rate ASCII characters) of the device's current altitude over ground. Each StratoLogger will be powered by an individual Duracell 9V battery. Duracell batteries have been selected due to their reliability and the feature that their leads are internally soldered.

6.3 Parachute Selection

For the recovery of the launch vehicle, a Kepner-Tregoe trade study was necessary for each parachute to determine the ideal parachute for each specific application. The performance characteristics of the parachutes that were considered are featured below in Table 43.

Parachute Type	C_d	C_x	Angle of Oscillation
Annular	0.85–0.95	1.4	<±6°
Cruciform	0.6–0.85	1.1–1.2	≤±3°
Vortex Ring	1.5–1.9	1.1–1.2	≤±2°
Toroidal	1.2-1.3	1.8	≤±6°

Table 43: Performance characteristics comparison

6.3.1 Drogue Parachutes

Cruciform parachutes have been chosen for both drogues primarily due to their low oscillation angles and high speed performance. Due to the intensive design and manufacturing labor required to ensure the safety of the larger main descent parachutes, it was important to choose drogue parachutes that are both easy to manufacture and design in conjunction with the need for optimal performance characteristics. Cruciform parachutes are also the easiest to pack and deploy out of all the considered options by a wide margin. This is a massive benefit in recovering such a heavy launch vehicle, as this significantly reduces the risk of a drogue failure, which in turn decreases the risk of main deployment failure since an inflated drogue in the proposed ARRD configuration very nearly guarantees successful deployment of the main parachutes. The wants for each drogue parachute were weighted appropriately with these factors in mind and evaluated in a trade study, shown below in Table 44.

Deployment Bay Drogue & Booster Drogue										
Options:		Cruciform		Annular		Vortex Ring		Toroidal		
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Drag Coefficient/Efficiency (0-10)	25.00%	4	1	5.7	1.425	10	2.5	8.3	2.075
Stability (angle of oscillation) (0-10)	20.00%	7	1.4	4	0.8	10	0.4	4	0.8
Ease of Design (0-10)	15.00%	10	1.5	5	0.75	2	0.6	3	0.45
Ease of Manufacturing (0-10)	10.00%	9	0.9	5	0.5	4	0.4	5	0.5
Deployment Simplicity (0-10)	20.00%	8	1.6	9	1.8	2	0.4	9	1.8
Testability	10.00%	6	0.6	8	0.8	2	0.2	7	0.7
Total Score		**7**		**6.075**		**4.5**		**6.325**	

Table 44: Drogue parachutes trade study

For the drag coefficient and stability criteria, a baseline was established by rating the vortex ring "10" in both categories, as it has the highest drag coefficient and lowest angle of oscillation. Values were then derived for the remaining parachutes by calculating their characteristics as a percentage of the optimal values offered by the vortex ring.

6.3.2 Main Parachutes

For the main parachute of each the payload and booster, a toroidal parachute design has been chosen in lieu of the vortex ring featured in proposal due to its high drag coefficient (C_d) and relative simplicity. Vortex ring parachutes offer a substantially high C_d (C_d = 1.5 – 1.8), but are much more complex compared to toroidal parachutes. Toroidal parachutes offer a C_d of 1.2 –1.3 which approaches the theoretical maximum for non-rotating parachutes. The simplicity of toroidal parachutes when contrasted with vortex rings is an extremely desirable feature due to the significantly lower risk of failure during deployment, and compatibility with the ARRD configuration – the autorotating characteristic featured in vortex rings makes the dual purpose drogue/pilot parachute configuration extremely complex, which is only further compounded by the complex packing process, effectively maximizing the number of potential failure modes and minimizing the safety of the system. The trade study for the main parachutes of the launch vehicle is featured below in Table 45.

Main Parachutes									
Options:		Cruciform		Annular		Vortex Ring		Toroidal	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Drag Coefficient/Efficiency (0-10)	30.00%	4	1.2	5.7	1.71	10	3	8.3	2.49
Stability (angle of oscillation) (0-10)	10.00%	7	0.7	4	0.4	10	1	4	0.4
Ease of Design (0-10)	10.00%	10	1	5	0.5	2	0.2	3	0.3
Ease of Manufacturing (0-10)	10.00%	9	0.9	5	0.5	4	0.4	5	0.5
Deployment Simplicity (0-10)	30.00%	8	2.4	9	2.7	2	0.6	9	2.7
Testability	10.00%	6	0.6	8	0.8	2	0.2	7	0.7
Total Score		**6.8**		**6.61**		**5.4**		**7.09**	

Table 45: Main parachute trade study

The sizing of the main and drogue parachutes for the payload were calculated to satisfy the system requirements R.1.2 and R.1.3 shown below in Figure 60.

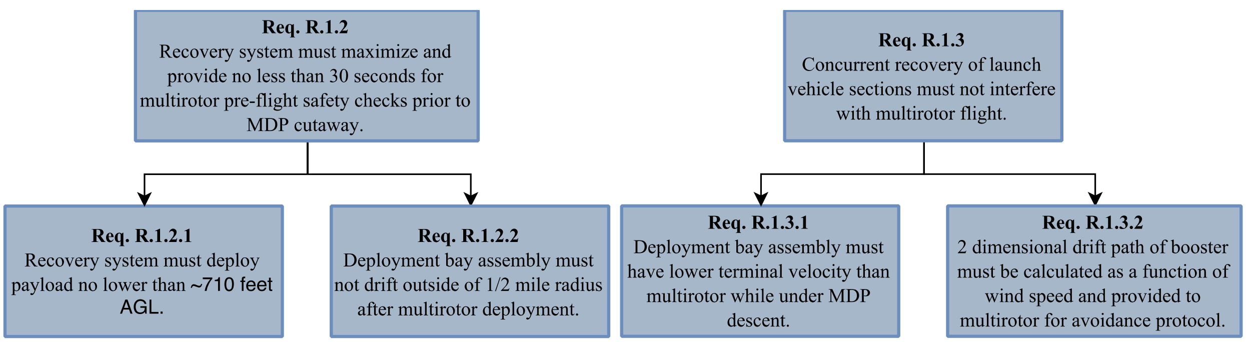


Figure 60: R.1.2 and R.1.3 requirements

The proposed verification methods for the lower level requirements in Figure 60 are outlined in Table 46 below.

Requirement Number	Requirement	Verification
R.1.2.1	Recovery system must deploy payload no lower than ~710 feet AGL.	Test Fully integrated, full scale test flights will verify that multirotor deployment above ~710 feet AGL provide sufficient duration for multirotor preflight safety checks.
R.1.2.2	Deployment bay assembly must not drift outside of 1/2 mile radius after multirotor deployment.	Analysis Drift values for 0, 5, 10, 15, and 20 mph wind speeds as a function of multirotor deployment altitude will be calculated to find maximum possible deployment altitude. Full scale tests will fully verify analysis.
R.1.3.1	Deployment bay assembly must have lower terminal velocity than multirotor while under MDP descent.	Test Drop tests will be used to verify that design calculations and predicted terminal velocities are correct.
R.1.3.2	2 dimensional drift path of booster must be calculated as a function of wind speed and provided to multirotor for avoidance protocol.	Test Full scale flights will verify accuracy of booster section drift model.

Table 46: R.1.2 and R.1.3 requirement verifications

6.3.3 Multirotor Payload

The multirotor payload must also be considered to be part of the launch vehicle, and must satisfy the requirements outlined in the statement of work for a safe recovery. To accomplish this, the MDP must keep the multirotor under 75 ft-lb of kinetic energy in the off-nominal case that the multirotor fails the preflight safety checks, and cannot execute the MDP cutaway maneuver for autonomous flight. This configuration is shown below in Figure 61.

Figure 61: Multirotor descending under MDP immediately after deployment.

To ensure that the multirotor is safe after cutaway, it will feature a small recovery bay that contains the multirotor reserve parachute (MRP) which will be identical to the MDP.

Due to the extremely small 2" diameter of the MRP and MDP bays, an efficient parachute was crucial to the viability of the multirotor. The trade study is featured below in Table 47.

MDP & MRP									
Options:		Cruciform		Annular		Vortex Ring		Toroidal	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Drag Coefficient/Efficiency (0-10)	50.00%	4	2	5.7	2.85	10	5	8.3	4.15
Stability (angle of oscillation) (0-10)	10.00%	7	0.7	4	0.4	10	0.4	4	0.4
Ease of Design (0-10)	10.00%	10	1	5	0.5	2	0.6	3	0.3
Ease of Manufacturing (0-10)	10.00%	9	0.9	5	0.5	4	0.4	5	0.5
Deployment Simplicity (0-10)	15.00%	8	1.2	9	1.35	2	0.3	9	1.35
Testability	5.00%	6	0.3	8	0.4	2	0.1	7	0.35
Total Score		**6.1**		**6**		**6.8**		**7.05**	

Table 47: Multirotor parachute trade study

The sizing of the identical MDP and MRP were calculated to satisfy the lower level requirements of R.1.1 shown in Figure 62 below.

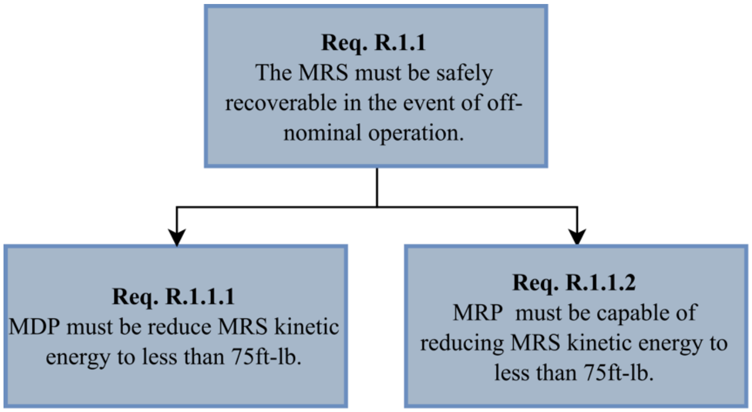


Figure 62: R.1.1 Requirements

The proposed verification methods for the lower level requirements in Figure 62 are outlined in Table 48 below.

Requirement Number	Requirement	Verification
R.1.1.1	MDP must be reduce multirotor kinetic energy to less than 75ft-lb.	<u>Test</u> Drop tests will be conducted to verify that descent velocity yields multirotor kinetic energy < 75ft-lb.
R.1.1.2	MRP must be capable of reducing multirotor kinetic energy to less than 75ft-lb.	<u>Test</u> Drop tests will be conducted to verify that descent velocity yields multirotor kinetic energy < 75ft-lb.

Table 48: R.1.1 Requirement Verifications

6.4 Design

The cruciform drogue for the deployment bay was then sized with the constraint that the retired drogue would become the nosecone main once separated, and would need to be properly sized to ensure that the nosecone's kinetic energy satisfies SOW 2.3.

The nominal diameter was calculated using

$$D_o = \sqrt{\frac{4m_v m_s g}{\pi E C_D \rho}} \quad (10)$$

Where D_o is the nominal diameter of the parachute, m_v is the total mass of the vehicle (or all tethered sections being recovered), m_s is the mass of the most massive section, g is gravitational acceleration, E is kinetic energy, C_D is the drag coefficient of the parachute, and ρ is the air density at sea level.

In the recovery case of this specific rocket, the rocket is being recovered in 4 individual sections, ($m_v = m_s$) so this equation for each section reduces to

$$D_o = \sqrt{\frac{4m^2 g}{\pi E C_D \rho}} \tag{11}$$

This drogue size ensures that deployment bay drogue descent speed is maximized (provided that main opening forces are reasonable and the safety of the system can be guaranteed), and thus minimize drift during drogue state and maximize the deployment altitude for the multirotor.

The steady state velocity of the deployment bay under main parachute was calculated using

$$v_e = \sqrt{\frac{2mg}{C_D S_O \rho}} \tag{12}$$

Where v_e is terminal velocity under parachute.

In order to solve for parachute sizes such that the empty deployment bay descends slower than the multirotor and avoids collision, we can form an inequality from equation (3)

$$\sqrt{\frac{2m_1 g}{C_{D_1} S_{O_1} \rho}} < \sqrt{\frac{2m_2 g}{C_{D_2} S_{O_2} \rho}} \tag{13}$$

Where the subscripts 1 and 2 denote the empty deployment bay and deployed multirotor, respectively.

(4) can be rearranged to find the surface area required for the empty deployment bay parachute as follows:

$$\frac{m_1 S_{O_2}}{m_2} < S_{O_1} \tag{14}$$

Where the C_D terms have dropped out since identical parachute types will be used for each.

Lastly, the equation

$$D_o = \sqrt{\frac{4S_o}{\pi}} \tag{15}$$

Can be used to relate D_o and S_o.

The parameters for each parachute were calculated using these equations and are detailed below in Table 49 and Table 50 below.

Drogue Descent Phase					
Section of Rocket	**Mass (lb_m)**	**S_o (ft^2)**	**D_o (ft)**	**V_e (ft/s)**	**Fx**
Booster	20.5	3.2	1.9	93.6	1.8
Deployment Bay (loaded)	16.1	1.5	1.3	129.0	7.0

Table 49: Predicted metrics for drogue recovery phase

Main Descent Phase					
Section of Rocket	**Mass (lb_m)**	**S_o (ft^2)**	**D_o (ft)**	**V_e (ft/s)**	**E (ft·lb)**
Nose Cone	2.0	1.5	1.4	42.5	69.9
Booster	20.5	65.9	9.0	14.6	69.9
Deployment Bay (loaded)	16.1	6.8	2.9	40.2	418.4
Deployment Bay (unloaded)	4.9			22.3	39.4
Multirotor	7.9	9.9	3.5	23.4	69.9

Table 50: Predicted metrics for main recovery phase

The opening force for parachute deployment is described by the equation

$$F_x = \frac{(C_D S)_O \rho v^2 C_x X_1}{2} \tag{16}$$

Where $C_D S_o$ is the drag area of the fully open parachute, v is the velocity at parachute deployment, C_x is the opening force coefficient (dimensionless), and X_1 is the force reduction factor (dimensionless). The opening force reduction factor is as low as 0.02 for finite mass cases, and is 1 for infinite mass cases where the parachute is acting as if there is zero deceleration. The infinite mass case was used for the drogue parachute deployments as a worst case scenario. $X_1 = 0.032$ is a reasonable force reduction factor for more severe personnel parachute mass loading scenarios. This force reduction factor was used to represent a worst case scenario in the main deployment calculations.

For booster drogue deployment, v was calculated using

$$v = \sqrt{2g(l_s + l_c)} \tag{17}$$

Where l_s and l_c are the preliminary estimates for suspension line length and shock cord line length, respectively. This equation yields the velocity gained from the booster freefalling along the distance of its shock cord and suspension lines until line stretch, where the canopy begins inflation.

For the deployment bay section, v was calculated using

$$v = \sqrt{(gt)^2 + 2g(l_s + l_c)} \tag{18}$$

Where $t=2$ for the 2 second delay of the deployment bay at apogee. This neglects any horizontal motion induced from weathercocking during ascent, but is still a sufficient approximation.

The opening force calculations are shown below in Table 51.

Opening Forces						
Event	**C_D**	**S_O (ft^2)**	**Velocity At Opening (ft/s)**	**C_x**	**X_1**	**F_x (lb_f)**
Booster Drogue	0.6	3.2	25.8	1.2	1.0	1.9
Deployment Bay Drogue		1.5	72.9			7.0
Deployment Bay Main	1.2	6.8	129.0	1.8	0.032	9.6
Deployment Bay Unloading			40.2			0.9
Multirotor Main		10.0	2.0			2.0
Booster Main		66.0	93.6			48.7
Reserve Deployment		10.0	24.3			0.5

Table 51: Opening forces at each recovery event

Drift values from the launch point can be easily calculated for each section via simple vector addition, provided the simplifying assumptions are made that each section moves at nearly the speed of the influencing crosswind, and that the vehicle follows a perfectly vertical ascent path. These values are detailed in Table 52 below. Gray boxes represent items tethered or otherwise coupled in drogue phase. Termination of the gray box represents main event.

Predicted Drift Values						
Crosswind Velocity (mph)	**Section**	**Descent Duration (s)**		**Drift (ft)**		**Total Drift (ft)**
		Drogue	**Main**	**Drogue**	**Main**	
0	Booster	49.4	40.6	0		0
	Nose Cone	30.1	31.3			
	Deployment Bay	30.1	59.6			
	Multirotor (off-nominal descent)	30.1	56.8			
5	Booster			362	298	659
	Nose Cone			221	229	450
	Deployment Bay				437	658
	Multirotor (off-nominal descent)				416	637
10	Booster			724	595	1319
	Nose Cone			441	459	900
	Deployment Bay				875	1316
	Multirotor (off-nominal descent)				833	1274
15	Booster			1086	893	1978
	Nose Cone			662	688	1350
	Deployment Bay				1312	1974
	Multirotor (off-nominal descent)				1249	1911
20	Booster			1448	1190	2638
	Nose Cone			883	918	1801
	Deployment Bay				1749	2632
	Multirotor (off-nominal descent)				1665	2548

Table 52: predicted drift values.

6.5 Testing

Though the parachutes that have been selected for the completion of the mission are not particularly prone to catastrophic failure modes, discrepancies in the calculated design parameters from the actual final product characteristics have the potential to cause catastrophic failure modes in the multirotor and recovery of the launch vehicle.

For this reason, ground testing and subscale testing will need to be performed to confirm that there are no critical discrepancies in the performance characteristics of the fabricated parachutes.

Preliminary testing plans are outlined in Table 53 below.

Ground Testing			
Test Item	**Parameter**	**Reason**	**Method**
Toroidal Parachute Design	**C_d**	Vital in terminal velocity predictions essential to multirotor safety and deployment bay collision avoidance.	Parachute will be drop tested with incremental center suspension line lengths to attain C_d consistent with expected C_d.
	Angle of Oscillation	Unexpected oscillation severity could cause unstable deployment of multirotor and cause off-nominal operation.	Parachute will be drop tested to verify that angle of oscillation is permissible for subsequent subscale test flights.
	F_x	Drogue phase terminal velocity for deployment bay is significant – accurate prediction of main opening force is critical for safe recovery.	A subscale parachute will be drop tested to find actual value of X_1.
Cruciform Parachute Design	**C_d**	Critical in predicting terminal velocity of deployment bay and resultant opening force. Highly influential in accurately estimating time available for multirotor preflight safety checks and booster avoidance protocol.	Subscale parachutes with varying W/L ratios will be drop tested to attain C_d consistent with expected C_d.
	Angle of Oscillation	Severe oscillation under drogue phase is unlikely to damage multirotor, but must be verified to be controllable within acceptable range.	Parachute will be drop tested to verify that angle of oscillation is permissible for subsequent subscale test flights. W/L ratio can be altered accordingly to control potential excessive oscillation.
Parachute Deployment	**Drogue**	Drogue must properly deploy to guarantee main deployment.	Ground deployment tests will be conducted to ensure that drogue is properly jettisoned without interfering with main recovery gear.
	Main	Main must be able to be pulled from recovery bay by drogue.	Drop tests will be conducted to simulate drag force of drogue to ensure it is sufficient to extract main deployment bag.
	ARRD	ARRD must be tested to be operational as expected, as well as sufficiently reliable.	ARRD will be repeatedly tested to ensure reliably proper release.

Table 53: Preliminary testing items

6.6 Subscale

In order to verify both the R.1 requirements and the SOW requirements provided for recovery, subscale tests will have to be performed to further verify the tests in Table 53 and to test the ARRD deployment concept. The subscale will feature a double-staging single recovery bay designed to exactly replicate the configuration proposed for the full scale vehicle. The parachute parameters outlined in Table 54 were calculated using the same equations discussed in 6.4 on page 110.

Subscale						
	Drogue (2400ft)		Main (400ft)			
Mass: 5.98 lbm	D_o (ft)	V_e (ft/s)	D_o (ft)	V_e (ft/s)	F_x (lb force)	E (ft lb)
	1.5	68.0	2.6	27.1	2.17	70

Table 54: Subscale Recovery System Parameters

6.7 SOW Verifications

The SOW requirements addressed in this design are featured below.

Requirement		Verification
2.1	The launch vehicle shall stage the deployment of its recovery devices, where a drogue parachute is deployed at apogee and a main parachute is deployed at a much lower altitude.	The vehicle will descend in two untethered sections under drogue parachutes to 1700 and 600 ft AGL, respectively where they will then stage and descend under main recovery.
2.2	Each team must perform a successful ground ejection test for both the drogue and main parachutes. This must be done prior to the initial subscale and full scale launches.	Ground ejection tests will be conducted for every potential deployment event, including deployment of the MDP, and MRP
2.3	At landing, each independent sections of the launch vehicle shall have a maximum kinetic energy of 75 ft-lbf.	Each section under main will have no more than 75 ft-lb of kinetic energy, including off-nonimal cases in which the multirotor is unsafe to deploy, or nominal flight encounters a failure mode and deploys MRP.
2.4	The recovery system electrical circuits shall be completely independent of any payload electrical circuits.	The main vehicle recovery circuits will feature independent circuitry, and the multirotor abort electronics will be independent from all other GNC systems aboard the multirotor.
2.5	The recovery system shall contain redundant, commercially available altimeters. The term "altimeters" includes both simple altimeters and more sophisticated flight computers.	Each main vehicle recovery event will be controlled by redundant pairs of StratologgerCF altimiters. The multirotor abort system will feature more sophisticated, redundant, kinetic

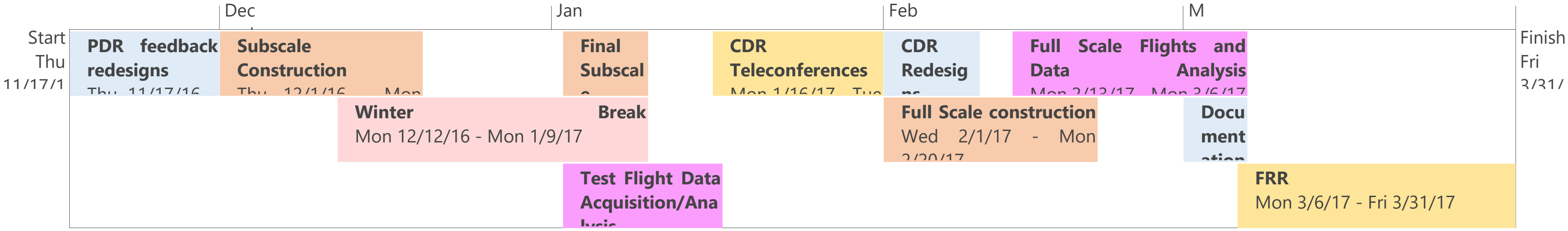
		energy-dependent deployment computers.
2.6	Motor ejection is not a permissible form of primary or secondary deployment.	Black powder charges will be used for each deployment.
2.7	Each altimeter shall be armed by a dedicated arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad.	Each arming switch will be accessible via standard vent holes with screwdriver, or magnetic clasp door if deemed possible prior to CDR.
2.8	Each altimeter shall have a dedicated power supply.	Each StratologgerCF will be powered with 12V Duracell battaries.
2.9	Each arming switch shall be capable of being locked in the ON position for launch.	Each arming switch will be accessible via standard vent holes with screwdriver.
2.10	Removable shear pins shall be used for both the main parachute compartment and the drogue parachute compartment.	Both main vehicle recovery bays will feature removable shear pins, as will the bay for the MDP and reserve parachute.
2.11	An electronic tracking device shall be installed in the launch vehicle and shall transmit the position of the tethered vehicle or any independent section to a ground receiver.	Each independent section, including the multirotor, will carry a GPS dog tracker.
2.12	The recovery system electronics shall not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	The main recovery electronics will be shielded from any potential EMI in dedicated, isolated avionics bays.

6.8 Safety

Recovery Risk Assessment						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity**	**Probability**	**Risk Level**	**Mitigation**
Rocket does not separate..	1. Not enough pressurization to break shear pins. 2. Coupling has too tight of fit.	Rocket follows unsafe ballistic path.	1	5	Low	1. The separation section of the rocket will be designed to ensure that the black powder charge provides sufficient pressurization, allowing the rocket to separate. 2. The coupling between the sections will be sanded down to have a loose fit. All personnel at the launch field will be notified immediately in the event of ballistic trajectory.
Altimeter or e-match failure	Parachutes will not deploy.	Rocket follows unsafe ballistic path.	1	5	Low	Multiple altimeters and e-matches are included into systems for redundancy to eliminate this failure mode.
Parachute does not open	1. Parachute stuck in deployment bag. 2.Parachute tangled.	Rocket follows unsafe ballistic path.	1	4	Moderate	Deployment bags will be specially made for the parachutes. This will allow for an organized packing that can reduce chance of deployment failure modes.
Rocket descends too quickly	Parachute is improperly sized.	Damaged launch vehicle	2	5	Low	The parachutes have each been carefully selected and designed to safely recover its particular section of the rocket..

Rocket descends too slowly	Parachute is improperly sized.	The rocket will drift farther than intended, potentially facing damaging environmental obstacles.	3	3	Low	The parachutes have each been carefully selected and designed to safely recover its particular section of the rocket. Should this be too large, the parachute will have to be resized.
Parachute has a tear or ripped seam	Potential partial or total failure modes	Damaged launch vehicle	2	5	Low	Careful inspection prior to packing should be eliminate this failure mode..
Parachute or suspension lines become burnt	Potential partial or total failure modes	Damaged launch vehicle	2	5	Low	Through careful packing and the appropriate use of Nomax material, this failure mode is unlikely.
Recovery system separates from the rocket	1. Bulkhead becomes dislodged. 2. Parachute disconnects from the U-bolt.	1,2. Parachute completely separates from the component, causing the rocket to become ballistic.	1	5	Low	The cables and bulkhead connecting the recovery system to each segment of the rocket are designed to withstand expected loads with an acceptable factor of safety
Landing of "rest of vehicle" with deployed telescoping deployment rod	1. Vehicle components get damaged on impact	1a. Joining bulk plate is sheared off. 1b. If drifting over the crowd occurs, injury to personal and spectators.	2	4	Low	Proper sizing of parachutes reduce the kinetic energy of the telescoping deployment rod.

6.9 Recovery Timeline Overview



7 Technical Design: Payload

7.1 Selection, Design and Rationale of payload.

Target Detection and upright landing has been selected for this year's experimental payload. Figure 63 displays the preliminary design for this projects competition payload.

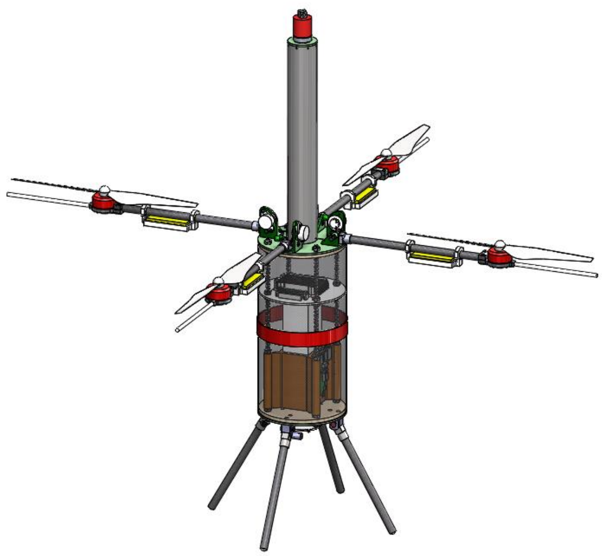


Figure 63: Rendering of the Experimental Payload Assembly

Table 55 displays the statement of work provided for the Target Detection and Upright Landing payload. In order for the mission to be considered a success, all of the objectives listed in the statement of work must be achieved.

NASA Student Launch Handbook Requirement No.	Requirement
3.2.1	An onboard camera system capable of identifying and differentiating between three randomly placed targets.
3.2.2	After identifying and differentiating between the three targets, the launch vehicle section housing the camera/cameras shall land upright.
3.2.3	Data from the camera system shall be analyzed in real time by a custom designed on-board software package that shall identify and differentiate between the three targets.

Table 55: Statement of Work Requirements.

7.2 System Level Trade Study

To accomplish the requirements listed in the statement of work, the payload was broken down into three individual subsystems and analyzed at a system level through multiple trade studies. Figure 64 displays the three general subsystems the payload will be divided into in order to accomplish the requirements specified in the statement of work. Table 56 displays the three main payload subsystems and their system descriptions.

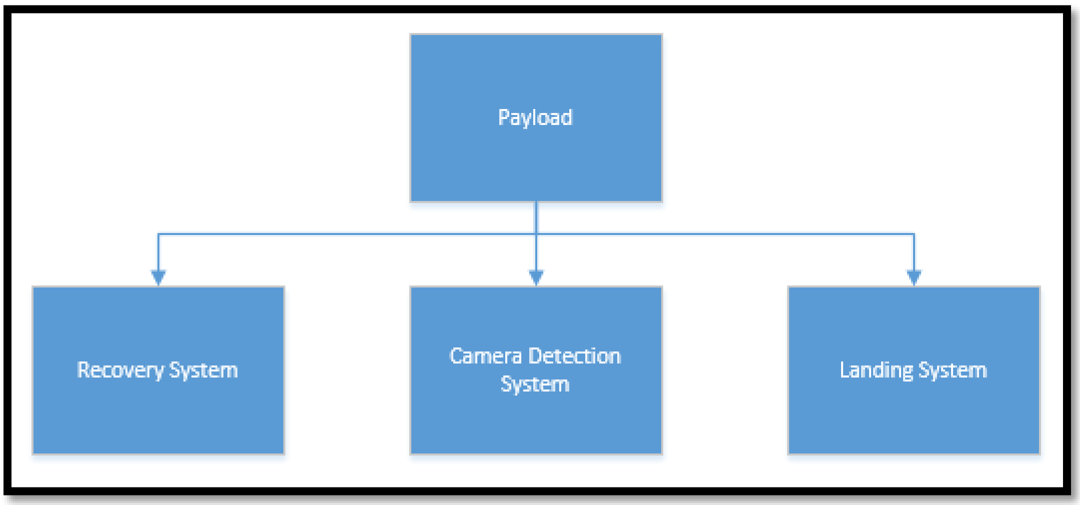


Figure 64: Payload General Subsystem break-down.

Payload Primary Subsystem	General Subsystem Requirement
Upright Landing System	The Upright Landing System shall be responsible for the controlled and stable landing of the payload.
Recovery System	The Recovery System shall be responsible for guiding the payload safely to the ground and controlling the rate of descent.
Target Detection System	The Target Detection System shall be responsible for the recognition of three randomly placed targets during after the initial launch vehicle ascent and before the payload landing

Table 56: General Subsystem descriptions.

7.2.1 Bounding Conditions

7.2.1.1 *Tipping Analysis*

In order for the payload to meet all upright landing requirements from the SOW, it is imperative that the selected landing leg system provides stability upon landing for all possible tipping scenarios. The following tipping analysis was conducted for general worst case landing scenarios with 20 mph winds. Worst case scenarios were determined from the SOW and safety regulations imposed by the RSO and NASA. Table 57 lists out the variables used, and Figure 65 below illustrates variables used for analyzing the system.

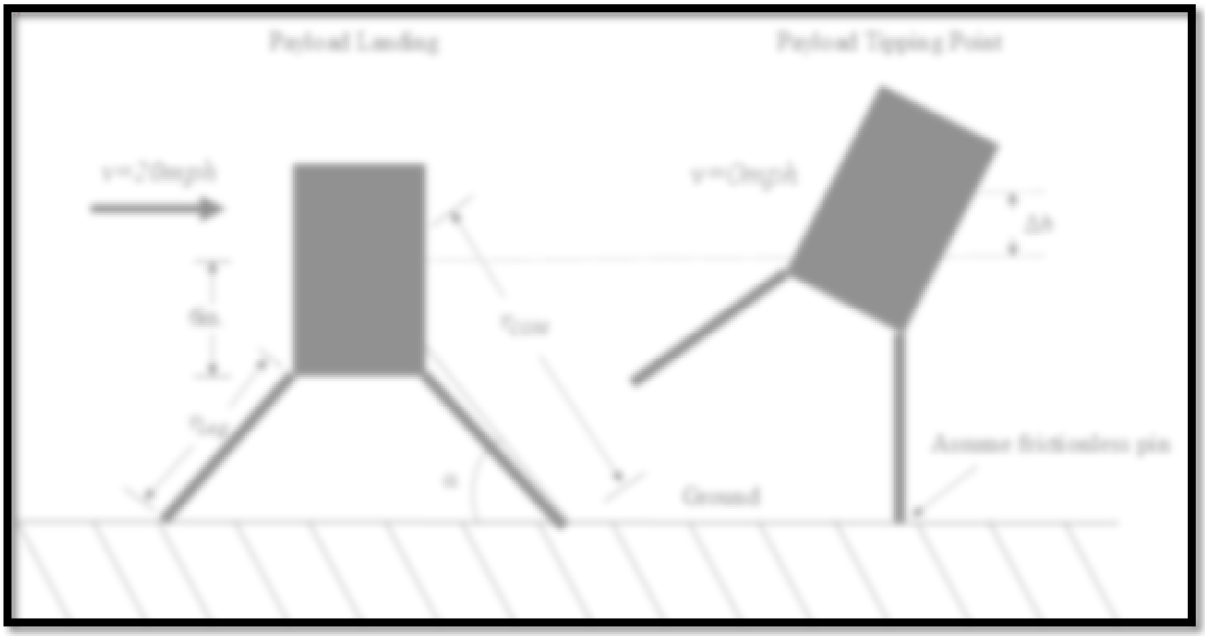


Figure 65: Variables used for tipping analysis.

Variable	Definition	Range/ Calculation [Unit]
r_{COM}	Distance from payload center of mass to pivot point	Calculated [in.]
v	Lateral velocity of payload	20 [mph]
Δh	Center of mass change in height	Calculated [in.]
α	Angle between center of mass and ground	$0 < \alpha < 90$[degrees]
g	Acceleration due to gravity	9.81 $[\frac{ft}{s^2}]$

Table 57: Variables used in the tipping analysis.

Assumptions made during tipping analysis are outlined in the Table 58 below.

Assumption	**Justification**
The payload is traveling with wind at 20 mph.	20 mph is the maximum wind speed allowed to launch in. This is a conservative estimate of the maximum lateral velocity a payload would experience during landing.
The payload landing legs contact the ground without any incident angle.	This assumption was made in order to simplify the problem.

Falling velocity of payload upon impact can be neglected.	Falling velocity of the payload does not directly contribute to the tipping of the payload
The payload rotates about one leg.	This assumption was made in order to simplify analysis and easily find bounding conditions of tipping.
The landing leg to ground pivot point was be treated as frictionless pinned joint.	This assumption was made in order to simplify analysis and easily find bounding conditions of tipping.
The payload rotates up to the tipping point where the center of mass of the payload is located directly over the point of contact of a leg.	This assumption defines the energy state of the system.
The payload's center of mass is 6 in above top of leg.	The payload will be housed inside a 12 in. coupler. With internal components evenly distributed throughout the payload we can assume the center of mass to half-way up the payload body.
The payload's center centered axially about the 6 in diameter tube.	This assumption was made assuming the design of the payload will be axisymmetric.

Table 58: Tipping analysis variables assumptions.

The system was analyzed with the conservation of energy. Using the assumed 20 mph lateral velocity of the payload, its kinetic energy be calculated just before it hits the ground. Next, the payload was evaluated just before its tipping point. Here, it is assumed the system has reached its maximum pivot height and only contains potential energy. The potential energy is then rewritten in terms of the center of mass radius, and the center of mass angle. Equation 1 below shows this relationship.

$$r_{COM} = \frac{(v \sin \theta)^2}{2g(1 - \sin \theta)} \tag{2}$$

Using this relationship, a solution set of center of mass radius, and their respective angles from the ground are generated which resist tipping in worst case scenarios. The results can be seen in Figure 66.

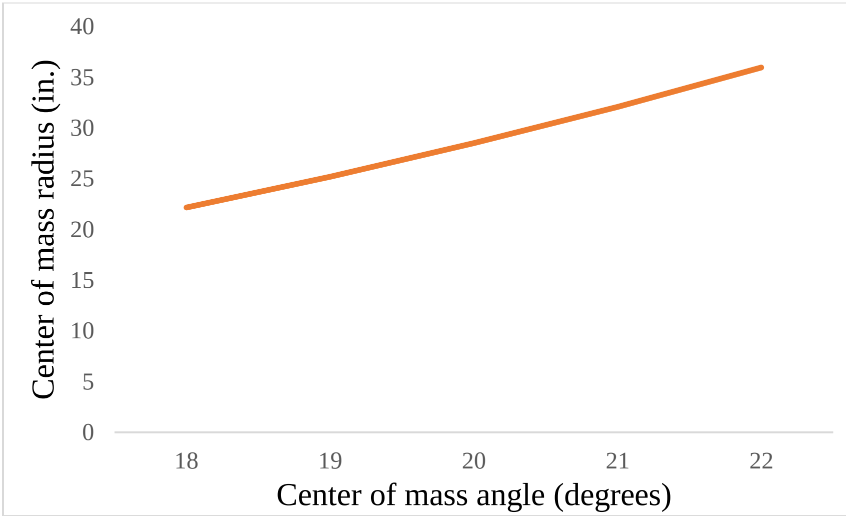


Figure 66: Center of mass radius vs. center of mass angle.

From this solution set, leg lengths and angles can be found which will not tip. Figure 67 below shows a schematic of dimensions used to derive leg lengths and leg angles which resist tipping.

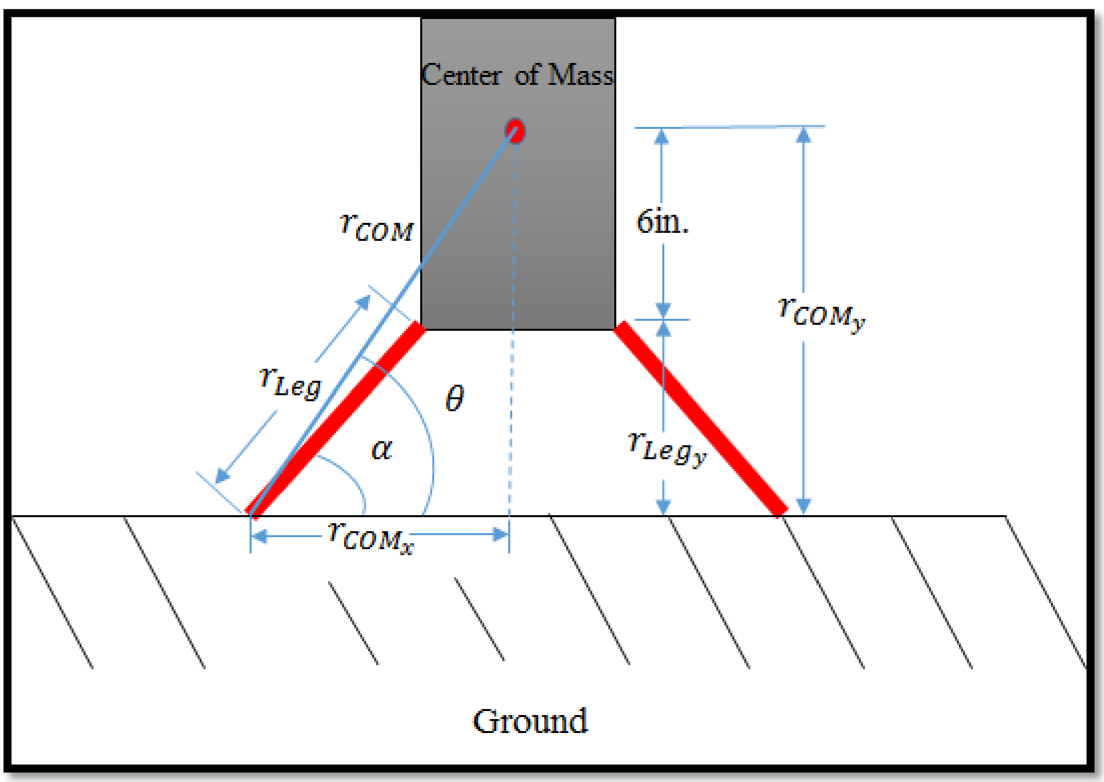


Figure 67: Leg length and angles defined to resist tipping.

Equation (1), (2), and (3) were used to determine possible leg geometries which resist tipping at 20 mph.

$$r_{Leg_y} = r_{COM} sin\alpha - 6 \tag{3}$$

$$r_{Leg_x} = r_{COM} cos\alpha - 3cos\alpha \tag{4}$$

$$\theta = tan^{-1}\left(\frac{r_{COM} sin\alpha - 6}{r_{COM} cos\alpha}\right) \tag{5}$$

Variable	**Description**	**[Unit]**
r_{Leg}	Leg length	[in.]
r_{Leg_y}	Y-component of leg length	[in.]
r_{Leg_x}	X-component of length	[in.]
α	Angle between leg and ground	[degrees]
θ	Angle between center of mass and ground	[degrees]

Table 59: Possible leg geometries for worst case scenario.

A final range of possible leg geometries are displayed in Figure 68.

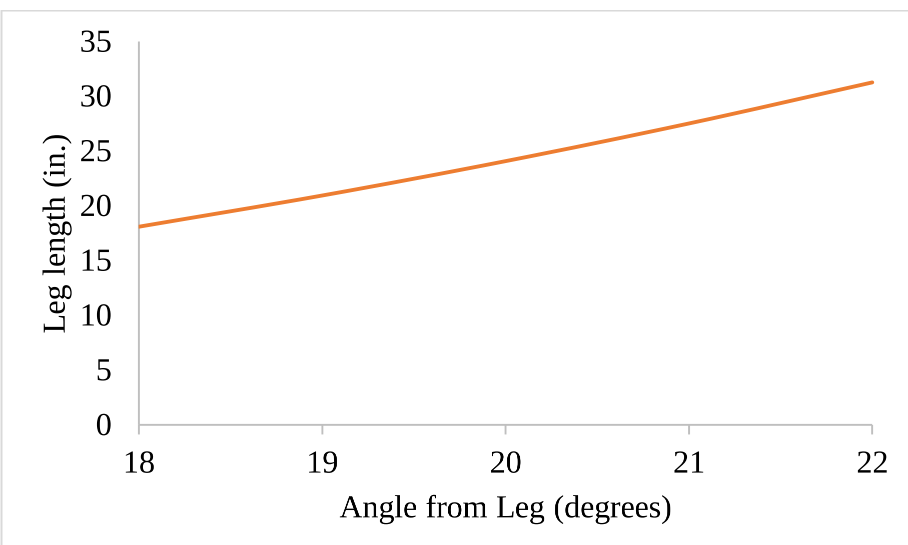


Figure 68: Leg length vs. leg angle.

From Figure 68 we see the resulting leg lengths versus angle which will not tip. Note that angles range from 35 degrees to 50 degrees. Geometries which resulted in leg lengths longer than 36in.

were excluded due to the impracticality of integrating a leg that length. Leg geometries which did not allow for 6in. of clearance were also discarded.

7.2.1.2 *Landing Load Analysis*

Another key element to the upright landing system of the payload is the structural stability of the landing system being utilized. Landing leg loads were calculated in order to create design constraints and requirements for potential landing leg systems. Figure 69 displays the mechanical landing leg system which was analyzed.

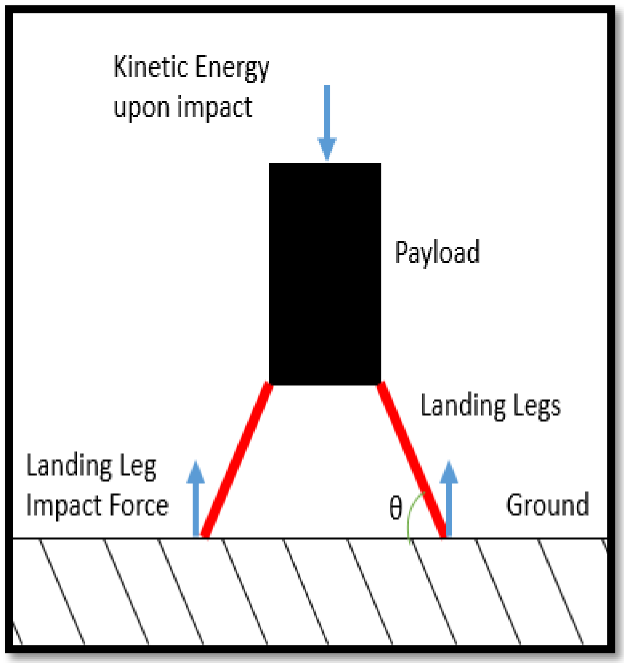


Figure 69: Impact analysis diagram.

Table 60 displays the variables used to determine landing leg loads for a discrete landing leg system.

Variable	**Description**	**Range/ Calculated [Unit]**
L	Axial Leg Length	6<L<36 [inches]
θ	Angle of point of contact between landing leg and ground. The range of this parameter was constrained by the tipping analysis derived in section 1.2.1.1	18<θ<22 [degrees]
δ_{Zrigid}	Payload axial deflection for rigid legs	0.1< δ_Z <1 [inches]

$\delta_{Zgaspiston}$	Payload axial deflection for gas piston legs.	$1< \delta_l <12$ inches
KE	Payload impact kinetic energy	25<KE<75 [ft*lbf]
F_{axial}	Axial compressive leg force	Calculated [lbf]
F_Z	Payload axial impact force	Calculated [lbf]

Table 60: Variables used on the landing leg load analysis.

Figure 70 below is the free body diagram of the forces and variables used in the leg load analysis.

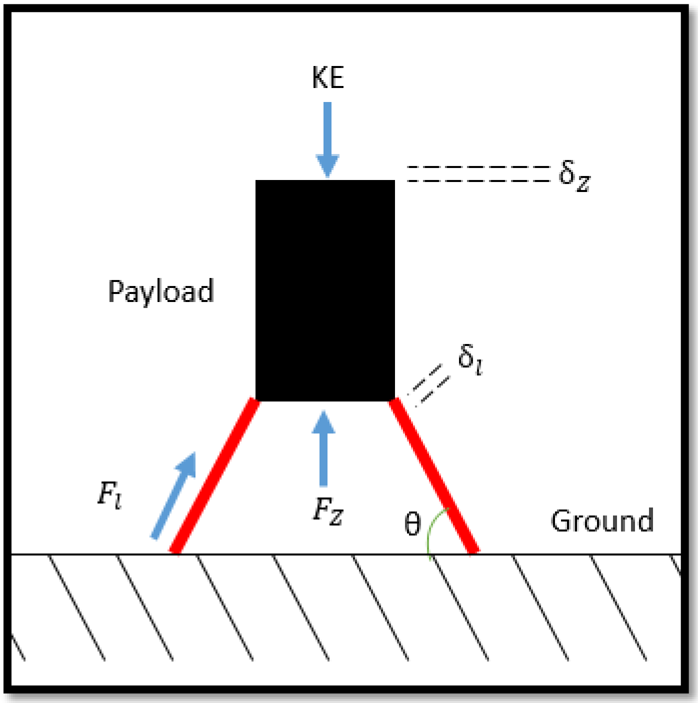


Figure 70: Free body diagram of landing leg analysis

Table 61 displays the assumptions made in to simplify the analysis model and the justification for each assumption.

Assumption	Justification
The ground is assumed to be infinitely stiff	This assumption was made due to the nature of uncertainty in understanding the mechanics of the ground the payload will be landing on.
Analysis was conducted assuming discrete legs were utilized such as individual rigid legs or gas piston legs	This assumption was made in order to calculate individual axial leg loads.

acting through single points of contact with ground.	
Analysis was conducted assuming the system consisted of four landing legs.	Many different discrete leg systems could be used. A four leg system was analyzed due to the balance between system weight and landing stability
Analysis assumed that impact occurred with no lateral velocity.	This assumption was made to allow equal load sharing between all the legs upon impact.
Analysis assumed Landing legs to act as "Pinned two force" axial members	While unrealistic with potential real world designs, assuming the legs to act as individual two force members allows pure axial loads to be calculated with transverse shear loads being neglected.

Table 61: Assumptions and justifications of in the landing load analysis.

Payload axial impact force was calculated using the following equation2.

$$F_z = \frac{2KE}{\delta z} \tag{6}$$

For both types of legs the separate sets of leg deflections were plugged into δz to determine landing leg loads for both systems of landing legs. Axial landing leg load was calculated using the following equation3.

$$F_l = \frac{F_z}{4 \sin \theta} \tag{7}$$

Figure 71 displays the three dimensional solution set of axial leg load plotted against payload axial deflection vs. leg angle for shock absorbing gas spring piston legs. Figure 72 plots the same parameters from Figure 71 with rigid landing leg deflection values.

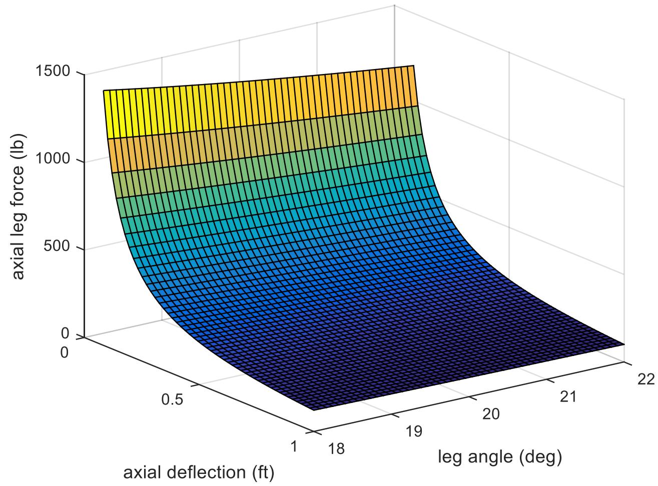


Figure 71: Axial landing loads plotted against leg angle vs. axial deflection for gas spring landing legs.

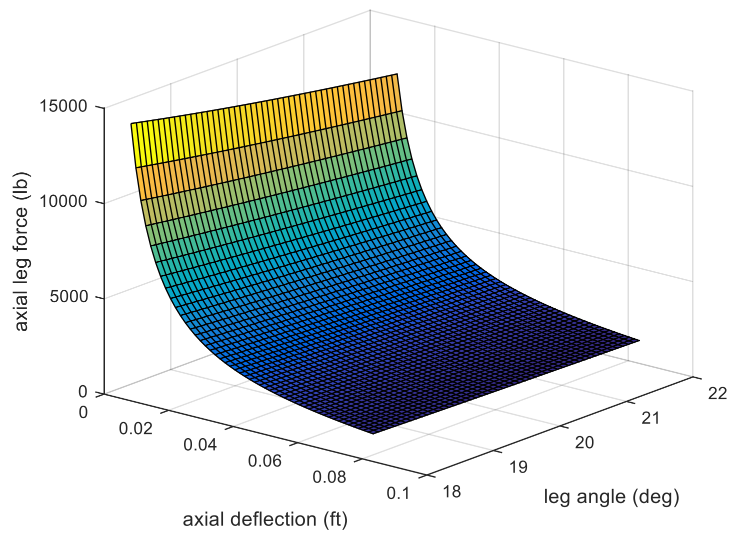


Figure 72: Axial landing loads plotted against leg angle vs. axial deflection for rigid landing legs.

Table 62 displays the minimum leg load solution along with back calculated leg dimensions.

Landing Leg System	Minimum Individual Axial Leg Load (lbf)	Leg Angle (deg)	Leg Length (in)
Gas Spring Leg System	100.1	22	31.25
Rigid Leg System	1201	22	

Table 62: Calculated minimum leg load based on tipping constrains.

7.2.1.3 *Bounding Condition Results*

From the analysis provided in section 1.2.1.1 and 1.2.1.2, the optimized landing leg solutions in worst case launch condition outputs an answer which would not be feasible. Landing legs at these angles would induce very high transverse shear loads and could potentially bottom out the payload, cause significant damage, and fail the mission. Table 63 displays the derived requirements from the analysis in section 1.2.1.1 and 1.2.1.2.

Requirement Number	Requirement	Method of Verification
BC.1	Payload must have active control over its vertical velocity	Demonstration Demonstration of the payloads ability to control vertical velocity.
BC.2	Payload must have active control over its lateral velocity	Demonstration Demonstration of the payloads ability to control lateral velocity

Table 63: Implemented bounding conditions.

Derivations of requirement BC.1
This requirement was derived based on the analysis conducted above in sections 1.2.1.1 and 1.2.1.2.

Derivation of Requirement BC.2
This requirement was derived based on the analysis conducted above in section 1.2.1.1 and 1.2.1.2.

7.2.2 Landing Leg Trade Study
Three general solutions were considered in the design of the upright landing system. The three general solutions to the upright landing system consisted of deployable rigid landing legs, deployable gas spring legs, and a pneumatically inflated landing apparatus.

7.2.2.1 *Deployable Rigid Legs*
Deployable rigid landing legs were studied as an option for the payload landing leg system. Rigid landing legs would serve as a reliable system due to their simple design and minimum number of components. Rigid legs take up minimal space while stowed in a launch configuration. This allows for easy integration into the launch vehicle. Manufacturing a fixed leg landing system utilizes common manufacturing methods, while other considered systems would not. A problem with a fixed leg landing system is their inability to absorb impact. No damping system means the payload body and components may be subject to high impact loads during landing.

7.2.2.2 *Deployable Gas Spring Legs*
Another consideration for the payloads landing leg system was deployable gas springs. Gas springs are metal pistons which use compressed gas inside the piston as a damping system. Unlike a metal spring, which bounces back to equilibrium, when a gas spring is compressed it slowly expands to its equilibrium position. A main benefit of this system is its damping ability. Compared to other

systems, gas springs were rated the highest for impact absorption. They would give the payload a higher margin of kinetic energy which it could safely land without damage. Issues with gas springs include their size, weight, and affordability. Compared to rigid legs, gas springs are inefficient with space and weight. Aerospace grade aluminum gas springs were reviewed, yet these still yielded a higher weight than what was allotted for the landing leg system.

7.2.2.3 *Inflatable Legs*

An inflatable base was proposed as a landing system for the payload. This landing system is extremely space and weight efficient. An inflatable base can handle high impact loads. The stiffness of the inflatable leg system raises design and manufacturing issues. Inflatable systems are good for protecting a payload but they are not ideal for landing a payload in a specific orientation. Complexity of design, and lack of prior experience with similar systems pose issues for this landing system.

All of the potential landing leg systems are analyzed in the trade study in Table 64.

Landing Leg System							
Options:		Deployable Rigid Legs		Deployable Gas Spring Legs		Inflatable Legs	
Mandatory Requirements							
Withstand Landing Impact Loads		Yes		Yes		Yes	
Landing Stability against Tipping		Yes		Yes		Yes	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Mitigating Risk	20%	9	1.8	6	1.2	2	0.4
System simplicity	15%	7	1.05	3	0.45	2	0.3
Weight	15%	8	1.2	3	0.45	5	0.75
Testability	15%	7	1.05	7	1.05	6	0.9
Impact Absorption	10%	2	0.2	9	0.9	6	0.6
Manufacturability	10%	8	0.8	3	0.3	5	0.5
Size	10%	5	0.5	3	0.3	8	0.8
Affordability	5%	7	0.35	2	0.1	7	0.35
Total Score			**6.95**	**4.75**		**4.6**	

Table 64: Landing Leg System trade study table.

Based upon the results of the study table, deployable rigid legs have been chosen as the landing leg system. This option is the obvious best choice due to the excessive weight of the gas piston legs and the unfamiliar design territory of the inflatable legs. In addition, optimum flight performance requires a lightweight system. This design choice also aids in maintaining a smaller launch vehicle and payload weight.

7.2.3 Recovery System Trade Study

Seven recovery systems were considered in recovery subsystem trade study. The seven general solutions to the upright landing system consisted of deployable parachute, deployable maneuverable parachute, and deployable parachute with pneumatic thruster guidance, pneumatic propulsion with a redundant parachute, a deployable parachute with downward thrusting multirotor guidance system, and a multirotor system with a redundant parachute.

7.2.3.1 *Deployable Parachute*

A simple deployable parachute was considered as a recovery system. After main parachute deployment, the payload would be separated and be recovered through this deployable parachute. While this single parachute controls the vertical descent velocity, it has no control over its lateral velocity. Therefore, this recovery system is susceptible to winds inducing lateral velocity on the payload. A robust landing leg system would be utilized to accomplish the vertical landing under worst case drift scenarios. Also, with no lateral velocity control, this recovery system has no way to guarantee that the cameras will be able to see the targets during descent.

Additionally, a parachute's vertical velocity control is passive and maintains a constant descent velocity for the entire recovery. Having a constant velocity poses issues because a desirable slow landing velocity means a parachute that is more susceptible to wind drift. With these two criteria's in mind, tradeoffs in flight characteristics pose design issues with parachute sizing and LLS design.

7.2.3.2 *Deployable Maneuverable Parachute*

Maneuverable gliding parachutes such as parawings and air ram chutes were considered as a potential recovery system. These parachutes not only have control over vertical descent velocity but also have directional control. This was favorable in order to navigate the payload towards the targets upon payload deployment.

Research on these parachutes discovered that they are used both commercially and recreationally to land sensitive payloads and sport skydivers. These uses of maneuverable parachutes are for much larger payloads than the proposed ten-pound payload. A large design problem for this recovery system would be to adapt this parachute style to the small scale application that they have not been previously used for.

The landing scenario for this recovery system poses design issues as well. Since, lateral velocity is induced with the directional control of these parachutes, a robust landing leg system needs to be in place to negate tipping upon landing

In addition, a robust GNC system would need to be developed for this recovery system. Developing GNC for a system such as this is an immense task to be completed in the time frame of this competition. Testing the flight characteristics and GNC of this system poses a significant problem to this recovery solution.

Actuators would need to be added to this system to induce the directional degrees of freedom of these parachutes. These actuators would add significant weight to the system and complicate deployment procedures.

7.2.3.3 *Deployable parachute with pneumatic thruster guidance*

This recovery system combines vertical velocity control of a parachute with directional control from pneumatic thrusters. The pneumatic thrusters would be the guidance system that ensures that the onboard cameras will see the targets during descent. For this recovery system, the vertical velocity constant from passive control of the parachute. The thrusters would be mounted at the CG of the recovery system to ensure that the pneumatic thruster actuation produces only lateral movements and not rotational movement.

This recovery system is very susceptible to wind drift. The amount of directional thrust from the pneumatic thrusters is not enough to effectively negate the lateral parachute drag. As in the single parachute recovery system, a robust LLS needs to be included to negate tipping upon landing due to the wind drift susceptibility. Additionally, the same constant descent velocity issue that arises in the deployable parachute recovery system arises in this recovery system as well.

7.2.3.4 *Pneumatic propulsion with a redundant parachute*

This recovery system was proposed to have a system with active control over its lateral and vertical movements through pneumatic propulsion.

Pneumatic propulsion is achieved in the same way that propulsion is achieved in the deployable parachute with pneumatic thruster guidance. In this recovery system, there would be more thrusting nozzles and they would be orientated at an angle that induces both vertical and lateral thrust components. These thrust components give the payload its active vertical and lateral control.

The payload would be separated after main parachute deployment. From this position, the pneumatic thrusters would propulsive guide the payload to a position that would have the onboard cameras detect the targets. After target detection, the pneumatic thrusters would propulsive land the payload safely on the ground.

A backup parachute would be included in this recovery system and would be deployed in the event of a flight or deployment anomaly to safely recover the payload under the kinetic energy requirement.

7.2.3.5 *Deployable parachute with downward thrusting multirotor guidance system*

This recovery system was proposed to mitigate flight risk with a parachute while having active control over lateral velocity with a multirotor. The multirotor would thrust towards the ground in order to maintain tension in the parachute shock cord. While thrusting downwards, the multirotor would be able to actively pitch its orientation to produce the lateral thrust component needed to negate the lateral drag from the parachute.

In the event of a deployment or flight anomaly with the multirotor, the parachute would land the payload safely under the kinetic energy requirement with the attached parachute.

Worst case scenario analysis was done on this recovery system. The proposed parachute design was a cruciform parachute with a coefficient of drag of 0.6. This design was selected based upon simplicity of design and manufacturability coupled with its acceptable coefficient of drag. A parachute surface area was chosen utilizing the drag equation.

The drag equation was analyzed in a steady state descent condition with a projected payload mass of ten pounds. The calculated surface area that would land the payload under the kinetic energy requirement was 29 square feet.

The next scenario analyzed was the flight system in the maximum wind speeds of 20 mph that would be seen on launch day.

The propulsion for the multirotor was chosen to be the DJI E800 with 4.6 pounds of thrust per motor. This propulsion system was selected based upon acceptable thrust and propeller size.

With the parachute surface area and propulsion selected, the worst case wind speed scenario was analyzed. It was found that the descent velocity exceeded the kinetic energy requirement when the multirotor actuated to fight the lateral component of parachute drag.

7.2.3.6 *Multirotor system with a redundant parachute*

This recovery system was proposed to have absolute control over the payload during flight. Multirotor systems fly all over the world every day and complete successful vertical landings with remarkable ease. In addition, multirotor have extremely desirable flight characteristics of a stable flight platform for viewing and ability to fly in significant wind conditions.

The main design issues that come with this recovery system are the storage of the multirotor during flight and the deployment of the multirotor from the launch vehicle. If the multirotor is safely deployed during from the launch vehicle, the rest of the mission requirements are easily achieved through the multirotor flight characteristics.

After deployment, the multirotor would fly to the targets for the onboard cameras to differentiate the targets. The multirotor would then vertically land the payload on the ground.

A backup parachute would be included in this recovery system for the same safety reason that is included in pneumatic propulsion with a redundant parachute recovery system.

With all of these recovery systems in mind, a recovery system trade study table was conducted and is shown below in

Payload Recovery System						
Options:	Single Parachute	Parachute with Pneumatic Thruster	Pneumatic Thruster with Backup Parachute	Parachute with Multi-rotor guidance	Multi-rotor with backup parachute	Maneuverable Parachutes
Mandatory Requirements						

Maintain Kinetic Energy requirement		Yes		Yes		Yes		No		Yes		Yes	
Lateral Position Control		No		Yes		Yes		Yes		Yes		Yes	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Mitigating Risk	20.00 %	9	1.8	6	1.2	2	0.4	6	1.2	6	1.2	7	1.4
System simplicity	15.00 %	8	1.2	2	0.3	2	0.3	0	0	6	0.9	4	0.6
Testability	15.00 %	3	0.45	5	0.75	5	0.75	4	0.6	9	1.35	3	0.45
Landing control	15.00 %	0	0	5	0.75	7	1.05	5	0.75	9	1.35	3	0.45
Available Documentation	10.00 %	8	0.8	1	0.1	3	0.3	1	0.1	9	0.9	4	0.4
Manufacturability	10.00 %	8	0.8	2	0.2	2	0.2	7	0.7	7	0.7	5	0.5
Flight Mobility	10.00 %	0	0	4	0.4	4	0.4	4	0.4	10	1	5	0.5
Affordability	5.00%	9	0.45	2	0.1	1	0.05	5	0.25	5	0.25	9	0.45
Total Score		**0**		**3.8**		**3.45**		**0**		**7.65**		**4.75**	

Table 65: Payload recovery system trade study.

From this trade study table, a multirotor with a redundant recovery system was chosen for the recovery system.

7.2.4 Target Detection System Trade Study

7.2.4.1 *TDS Design*

Two target detection system solutions were created in response to mechanical and technical constraints proposed at different design stages. The two possible target detection systems consist of an ascent based detection system and a descent based detection system. These solutions were based on when the system would be required to detect targets. A trade study was conducted to compare each system.

7.2.4.2 *Descent System*

This system was proposed and seen as the best course of action initially. Attached to the bottom of the landing vehicle, a single camera would be mounted and relay pertinent information to flight and targeting systems.

Stability would be provided by the vehicle, and there was little concern with having the entire target zone visible as the zone could be scoured until all targets were identified.

Downsides to this system was the difficulty and complexity required of other systems to make this a reliable solution.

7.2.4.3 *Ascent System*

Originally this system was adopted based on changing requirements and the relative simplicity of this system compared to the descent based one.

This system was based on obtaining the targets while the vehicle was in the ascent phase and thus made several additional requirements of the TDS. It would require that cameras be mounted at both vertical and horizontal offsets to ensure that the entire target zone would be covered. A much smaller time frame of usable footage would be generated as well; since there are both upper and lower bounds introduced through this solution.

It was determined that 7 cameras would be required for this system to have total radial coverage, and would need to be at a minimum altitude before the entire target zone is visible. This minimum altitude would be dependent on the mounting angle of each camera relative to the fuselage, assuming a 90° launch angle.

Table 66 shows a trade study documenting the pros and cons of each system.

Target Detection System					
Options:		Ascent		Descent	
t					
Capable of Target Detection		Yes		Yes	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**
Benefits					
Control (0-10)	25.00%	0	0	6	1.5
Target Acquisition Window (0-10)	40.00%	2	0.8	10	4
Costs					
Additional Hardware (0-10)	25.00%	9	2.25	3	0.75
Cost ($)	5.00%	354.83	17.74	50.69	2.53
Weight (g)	5.00%	338.52	16.926	48.36	2.418
Calculations	**Total**				
Benefit	6.3	0.32		0.87	
Cost	42.62	0.05		0.02	
Total Score		**6.01**		**49.61**	

Table 66: TDS Trade study.

The final design iteration and the results of the trade study show that a descent based system would be the most appropriate and would provide the team with the best chance possible to detect each target.

Working constraints were the primary concern, and meeting those of the ascent system could not be determined with any substantial degree of confidence without excessive testing. Constraints provided ask that the minimum working altitude be 825 ft., with a total of 6 seconds time given for target detection.

7.2.4.4 *Computer Vision Library and Implementation*

OpenCV was selected to be the image processing library for its extensive documentation as well as its flexibility. OpenCV is a popular and powerful open source library, and obviously capable of meeting the needs of this challenge.

The target selection process in and of itself shall consist of several tests performed by the vision program. A binary image will be generated, colors identified, and rough estimates will be made to best ensure that each target is correct.

7.3 Mission Requirements

From the trade studies above, the following subsystems were derived from the primary subsystems to guarantee the system success. Figure 73 displays the derived subsystems and their relationship to the original primary subsystems. Table 67 displays the selected subsystems along with their descriptions.

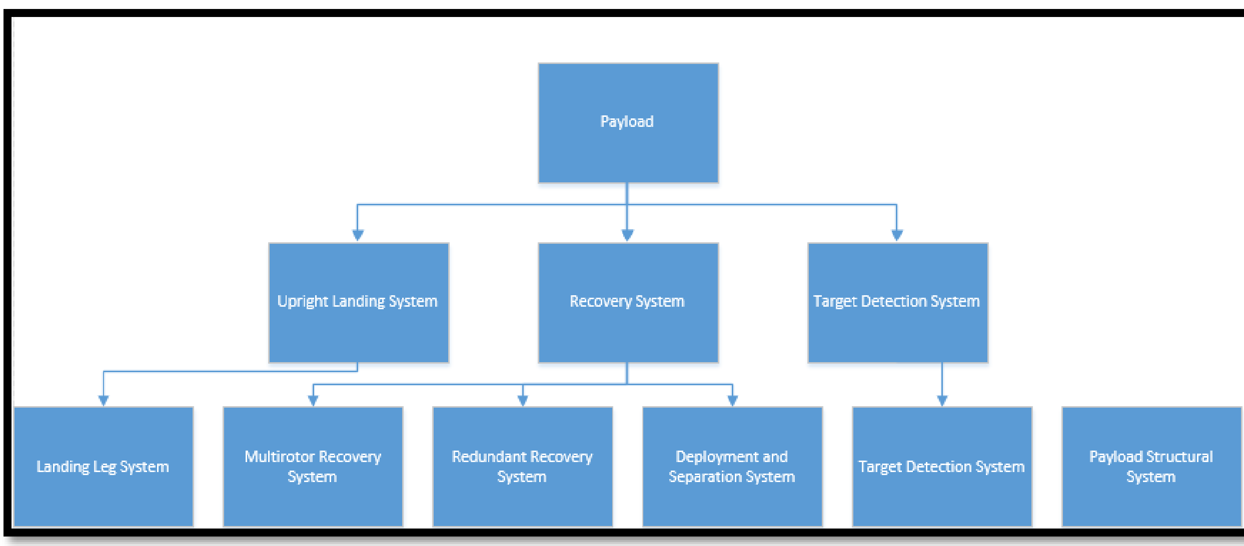


Figure 73: Derived payload subsystems.

Derived Payload Subsystem	Payload Subsystem Descriptions
Multirotor Recovery System (MRS)	The MRS shall be responsible for guiding the payload safely to the ground and controlling the rate of descent using a multirotor based system. The MRS

	shall also be responsible for moving the target detection system into a line of sight of the targets.
Redundant Recovery System (RRS)	The RRS shall be responsible for acting as a redundant recovery method in the event of a flight anomaly within the MRS through the deployment of an onboard backup parachute.
Deployment and System (DCS)	The DCS shall be responsible for the safe and stable deployment of the payload out of the Launch Vehicle. The DCS also encompasses the initial separable deployment parachute the payload falls under during its initialization phase.
Target Detection System (TDS)	The Target Detection System shall be responsible for the recognition of three randomly placed targets during after the initial launch vehicle ascent and before the payload landing.
Landing Leg System (LLS)	The LLS shall be responsible for absorbing impact loads and providing upright stability upon landing.
Payload Structures System (PSS)	The PSS shall be responsible for carrying all flight loads induced on the payload and will contain all subsystems of the payload.

Table 67: Derived payload subsystems and their descriptions.

7.3.1 Subsystem Requirements, Verifications, and Derivations.

7.3.1.1 *Multirotor Recovery System (MRS)*

Requirement Number	Requirement	Method of Verification
MRS.1	The payload shall be recovered and landed upright via the autonomous Multirotor Recovery System.	Test The MRS's functionality will be demonstrated through a minimum of two successful full scale test flights.
MRS.2	The MRS shall have the capability to safely deploy from the launch vehicle.	Test Ground testing of the deployment mechanisms will be accomplished to verify repeatability of the system.
MRS.3	The MRS shall have the ability to navigate and land in winds up to 20mph.	Test Test flights will be conducted to verify predicted flight performance of the payload in worst case wind scenarios.

MRS.4	The MRS shall not inhibit the Redundant Recovery System.	Test Ground testing of RRS deployment will be done with the MRS to verify deployment clearances.
MRS.5	The MRS must conduct preflight checks to ensure all systems are ready for flight, including the flight after DCS separation.	Test Flight systems tests will be conducted prior to every flight.
MRS.6	The MRS shall immediately navigate away from the deployment parachute after cutaway to prevent a potential collision.	Analysis Analysis will be done for the locations that the MRS must navigate to in order to avoid collisions.
MRS.7	The MRS flight computer shall meet all related requirements set forth by the TDS.	Inspection These requirements will be considered during the choice of flight computer.

Table 68: MRS requirements.

Derivation of Requirement MRS.1
In order to verify that the MRS will be able to complete its mission, full scale tests must be completed to ensure that the system is able to safely and repeatedly perform its system requirements. This requirement was derived in order to This requirement was chosen due to analysis conducted in section 1.2.1.1 and 1.2.1.2. These led to the conclusion that leg geometries which could resist tipping in worst case wind scenarios, would not be able to withstand landing loads, and vice versa, In order to land without tipping in all scenarios an active recovery system was selected.

Derivation of Requirement MRS.2
The payload must be stowed within the launch vehicle during flight. This requirement was imposed to drive a design which will provide safe and reliable deployment of payload propulsion system.

Derivation of Requirement MRS.3
BC.1 was derived in order to mitigate worst case lateral velocity landing conditions of the payload which analysis showed in section 1.2.1.1 would show tipping. This requirement was derived in order to satisfy BC.1.

Derivation of Requirement MRS.4
The RRS's primary driving design attribute will be centered on robustness. In order for the payload to have a safe landing under all conditions, the RRS must be able to deploy at any point in flight. This requirement was derived to guarantee the safe and controlled landing of the payload for all risks imposed by the MRS.

Derivation of Requirement MRS.5

Before the MRS takes flight, the system must conduct preflight checks to verify system wide functionality to guarantee of the system and those in the vicinity of the payload. This requirement was derived to guarantee the safe flight of the MRS.

Derivation of Requirement MRS.6
When the MRS performs its cutaway from the deployment parachute, there is a risk of colliding with the parachute. The MRS must know where it can and cannot fly to avoid collision with the deployment parachute.

Derivation of Requirement MRS.7
Since the Target Detection System (TDS) will be reliant on the MRS flight computer hardware, the MRS must support all related requirements that the TDS defines.

7.3.1.2 *Redundant Recovery System (RRS)*

Requirement Number	Requirement	Method of Verification
RRS.1	The payload shall contain a Redundant Recovery System in the event of a deployment or flight anomaly within the MRS.	Test Flight Test of the RRS will be performed in order to verify system functionality. A pre-flight checklist will be developed which will be completed prior to all test and competition flights.

Table 69: RRS requirements.

Derivation of requirement RRS.1
In order for the payload to have the level of safety necessary, a safe recovery must be able to complete in all flight scenarios.

7.3.1.3 *Target Detection System (TDS)*

Requirement Number	Requirement	Method of Verification
TDS.1	TDS shall have ability to detect and differentiate between all three randomly placed targets.	Test TDS shall undergo significant ground testing to ensure targets are identified precisely and accurately.
TDS.2	TDS shall provide sufficient camera angle for target detection.	Demonstration Flight recordings shall be created and analyzed to determine camera angle configurations.
TDS.3	TDS shall remain stable during all flight functions.	Test

		Flight data and recording will be analyzed to obtain accurate information on video feed quality.
TDS.4	TDS shall be able to determine information related to target location and pass that back to the MRS.	<u>Test</u> Integration testing will be completed and subscale tests will be conducted to determine system readiness and coordination.
TDS.5	TDS shall record all video/images it receives during flight.	<u>Test</u> Ground testing will be conducted to ensure all video is recorded accurately.
TDS.6	TDS shall provide sufficient evidence that targets have been identified.	<u>Test</u> Ground testing shall be performed and reviewed to collect sufficient, accurate data.
TDS.7	TDS shall be able to differentiate shapes and colors at a specified altitude.	<u>Test</u> Ground & subscale testing will be conducted to determine effectiveness of detection system. Results will be analyzed and adjustments made depending on results.

Table 70: TDS requirements.

Derivation of requirement TDS.1
In order for the MRS to successfully complete the task of hovering above ground targets, the GPS coordinates of each target must be known. In order to provide these GPS coordinates, the TDS must be able to identify and differentiate between the targets.

Derivation of requirement TDS.2
The ability of the TDS to successfully detect targets depends on the vehicle being stable at all times. Turbulent flight complicates the detection process and may lead to detection failures.

Derivation of requirement TDS.3
The ability of the TDS to successfully detect targets depends on the vehicle being stable at all times. Turbulent flight complicates the detection process and may lead to detection failures.

Derivation of requirement TDS.4
The ability of the MRS to hover above targets is dependent on it receiving GPS coordinates for the targets from the TDS.

Derivation of requirement TDS.5
Flight footage is essential for debugging/optimizing the target detection algorithm; availability of footage allows for quick simulations to test performance.

Derivation of requirement TDS.6

Post-flight information is required to verify that the TDS correctly identified targets. This evidence could simply consist of GPS coordinates for each target.

Derivation of requirement TDS.7
Ground targets will be specific shapes/colors; thus, it is essential that the TDS be able to differentiate between these qualities in order to successfully identify targets.

7.3.1.4 <u>*Landing Leg System (LLS)*</u>

Requirement Number	7.3.1.5 <u>*Requirement*</u>	**Method of Verification**
LLS.1	LLS shall be stowable into the recovery bay	<u>Inspection</u> Fitment of the leg assemblies will be inspected through CAD models and will be verified during assembly.
LLS.2	The LLS shall not interfere with the main vehicle recovery system	<u>Inspection</u> Inspection of the recovery bay will ensure necessary space allocation for the LLS and launch vehicle recovery system.
LLS.3	The LLS shall lock into the landing configuration after deployment from the main vehicle recovery bay	<u>Test</u> Black powder testing will be accomplished to verify proper deployment and separation of the LLS from the recovery bay.
LLS.4	The LLS shall provide enough rigidity and stability to support the entire payload system upon landing	<u>Demonstration</u> LLS stability will be demonstrated through flight tested.

Table 71: LLS requirements.

Derivation of requirement LLS.1
In order for the LLS to be able to function in all parts of the launch, it must be stowable into the recovery bay. This stowing ability requires the LLS to function with the other launch systems.

Derivation of requirement LLS.2
Due to the LLS being able to stow into the recovery bay, the LLS must not interfere with the main vehicle recovery system. There is a large safety concern that the landing legs may negatively affect the deployment of the main recovery system. This requirement ensures the absolute necessity of a safe recovery of the booster section.

Derivation of requirement LLS.3
The payload will not be able to perform its vertical landing if the landing legs are not supported in their correct position. This requirement was derived to ensure that the landing legs are locked into place during landing.

Derivation of requirement LLS.4
During landing, the payload must be sufficiently supported from its landing legs to maintain its vertical landing position. This requirement was derived to give the payload its proper support needed for a vertical landing.

7.3.1.6 *Payload Structures System (PSS)*

Requirement Number	Requirement	Method of Verification
PSS.1	The PSS shall carry the flight and landing loads experienced during launch and payload recovery. (Section of airframe)	Analysis of worst case thrust and weight scenarios will be applied to the PSS to ensure that it can carry all the flight loads.
PSS.2	The PSS shall organize and house all payload subsystem components to be easily accessible.	An inspection of the assembled payload will be done to ensure that all of the subsystem components fit
PSS.3	Every subsystem in the system shall be easily accessible for servicing.	A demonstration of servicing each subsystem will be conducted
PSS.4	PSS shall allow for the arming of all recovery electronics prior to flight (fulfills SOW 2.7)	A demonstration of all recovery electronics being armed from outside the airframe will be conducted.
PSS.5	System shall absorb opening shock force from RRS deployment at max kinetic energy	Materials analysis on PSS bulkplates will verify the opening force absorption.

Table 72: PSS requirements.

Derivation of requirement PSS.1
In order for the payload to complete its deployment, flight, and landing operations, its internal structure must provide sufficient support in all situations. This requirement was derived to prevent the payload subsystems being rendered unable to perform their mission due to lack of support.

Derivation of requirement PSS.2
All subsystems must be housed and supported for them to perform their respective requirements. This requirement ensures that the PSS provides the rigid support and protection of these subsystems.

Derivation of requirement PSS.3
With all of the subsystems housed within the PSS and the large amount of testing that will be done on the payload, every subsystem must be easily serviceable. This will serve to facilitate repairs and changes that may need to be done to the subsystems.

Derivation of requirement PSS.4

This requirement was derived to fulfill SOW requirement 2.7, which requires that all recovery electronics must be able to be armed from outside of the airframe.

7.3.1.7 *Deployment and Separation System (DCS)*

Requirement Number	Requirement	Method of Verification
DCS.1	System shall safely deploy payload from deployment bay.	Deployment testing of the payload will be conducted following RCR safety guidelines
DCS.2	System shall cutaway from payload following MRS initialization	Both flight and static ground testing of payload cutaway will be conducted.
DCS.3	System shall separate from payload under a command sent from a ground station directed by a RSO	Demonstration of RSO directed deployment will be shown during full scale launch.
DCS.4	System shall not inhibit MRS deployment of arms and legs prior to separation.	Appropriate modeling and testing of deployment will verify that the DCS and MRS do not interfere with one another.
DCS.5	The deployment parachute connection must be strong enough to support the deployment parachute, but still be severable to release the RRS.	Load testing simulating the parachute opening force and deployment testing of the RRS will be conducted.

Table 73: DCS requirements.

Derivation of Requirement DCS.1
In order for the payload to complete its flight, it must safely be deployed from the deployment bay. This deployment maneuver must be tested numerous times to ensure that it will compete this requirement every time.

Derivation of Requirement DCS.2
The payload must be able to release itself from the deployment parachute to begin its flight sequence.

Derivation of Requirement DCS.3
The control of this cutaway operation must be controlled from the ground station for safety reasons.

Derivation of Requirement DCS.4
In order for the payload to achieve its flight, the DCS must not inhibit any functions of the MRS. This includes the deployment of the MRS propulsion arms. These arms must be able to actuate to their flight positions during DCS actuation.

Derivation of Requirement DCS.5
For the deployment sequence, the payload initially hangs underneath the deployment parachute. This parachute must be rigidly connected to the payload through the DCS. In addition, this connection must be severable in order to deploy the RRS.

7.4 Multirotor Recovery System (MRS)

As described in Table 67, the MRS is the primary system responsible for maneuvering the payload to an appropriate point of view for the TDS and for safely recovering the payload to the ground. The MRS design consists of a deployable multirotor system. The MRS encompasses all of the mechanical and electrical systems which will provide the payload with autonomous flight capabilities.

7.4.1 MRS Design

The physical design of the MRS is shown in its flight position below in Figure 74.

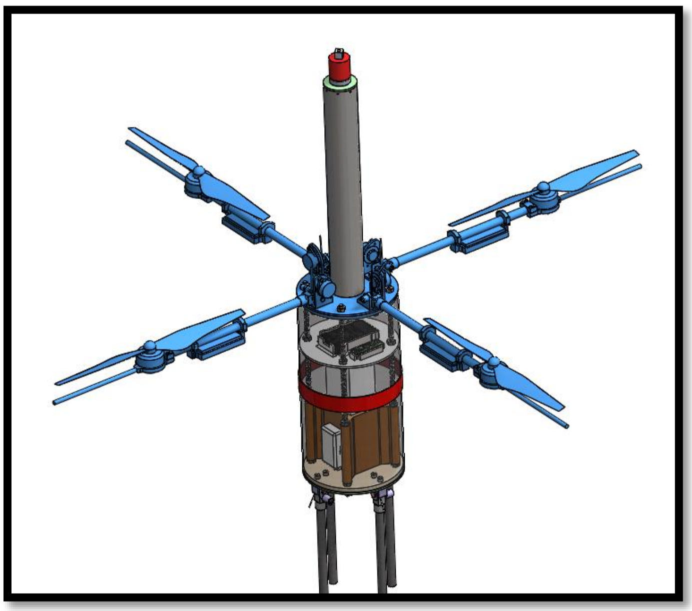


Figure 74: MRS highlighted in blue in its flight configuration.

Figure 75 is a block diagram of the RRS electronics.

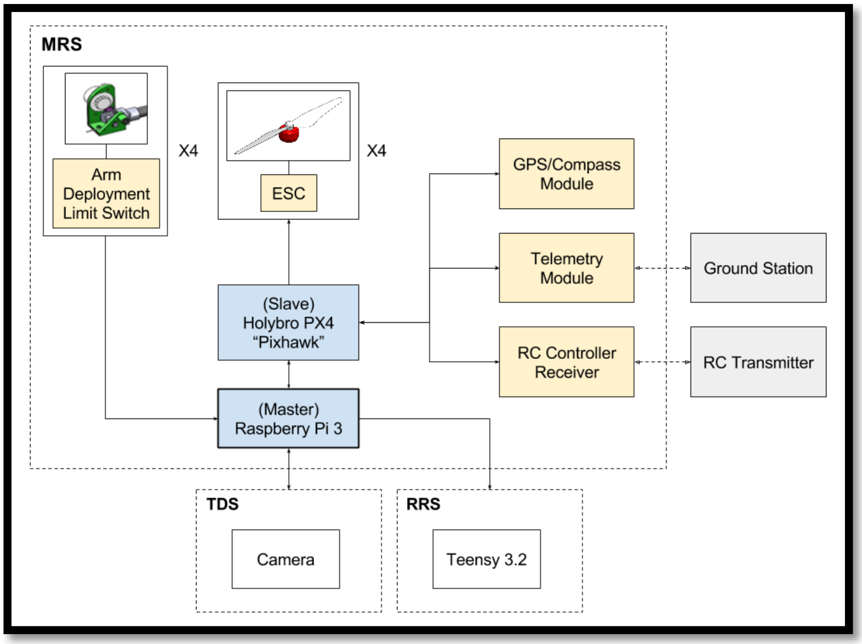


Figure 75: MRS System Block Diagram.

7.4.2 Guidance, Navigation, and Control (GNC)

The GNC is the autonomous navigation system that controls the MRS. This is the brain of the system and is based on a flight computer that communicates commands to a flight controller. The system utilizes values acquired from sensors such as the altimeter, the GPS module, the electronic speed controllers (ESC), and limit switches to make decisions about the appropriate course of action given any situation.

The GNC logic is first described in words and visually represented in logic flowchart shown in Figure 77.

7.4.2.1 *Pre-Launch*

Upon startup the GNC will take and store "home" values for the key navigation components necessary for autonomous navigation back to the launch zone, namely GPS coordinates and altitude.

7.4.2.2 *Launch*

The system sleeps during the launch of the rocket until the deployment of the MRS arms trigger limit switches attached to the multirotor. This signifies that the arms have been successfully deployed and locked into place.

7.4.2.3 *Flight Initialization*

As soon as the multirotor successfully verifies that the arms are in place, the system begins its flight initialization checks in accordance with requirement MRS.5. These include determination of the vector of travel from the GPS as well as spinning the motors up to verify that they are functioning properly. This is done utilizing the ESC's current sensing capability to determine if the motors, while in operation, are within a range of currents defined to be normal for freely spinning, unhindered propellers. Once the GNC is satisfied with the initialization values, it will signal the ground control unit to request approval from the RSO to initialize autonomous flight.

7.4.2.4 *Autonomous Flight*

Upon receipt of RSO approval, the propulsion system will start the MRS motors and the deployment parachute will be released. The system will immediately navigate to a safe zone outside the path of the falling deployment parachute. The system will then autonomously navigate to the observation position.

7.4.2.5 *Target Acquisition*

Once in the observation position, the MRS will hover and hold position to allow the TDS to view the launch zone. The MRS will then wait for one of three responses: targets acquired with the GPS locations of the targets, targets not acquired with a distance and direction of suspected targets, or targets not acquired with distance and direction unknown. From this point, if distance and direction are not known, the MRS will begin an outward box method scan in intervals of 50ft until targets are found, range limits are reached or targets are found as shown in Figure 76 below.

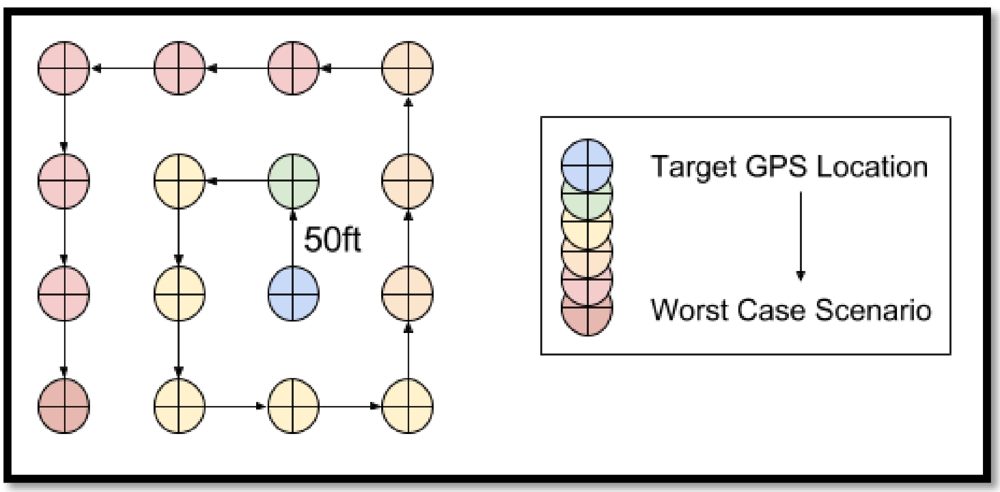


Figure 76: Outward Box Scan Method

If the TDS returns a suspected distance and direction of the targets, the MRS will navigate to the designated location and allow the TDS to scan again.

7.4.2.6 *Autonomous Landing*

Once the targets are found, the MRS will navigate to the GPS location of each of the targets and hover over each one before moving to the next. The MRS will then land the payload on the final target.

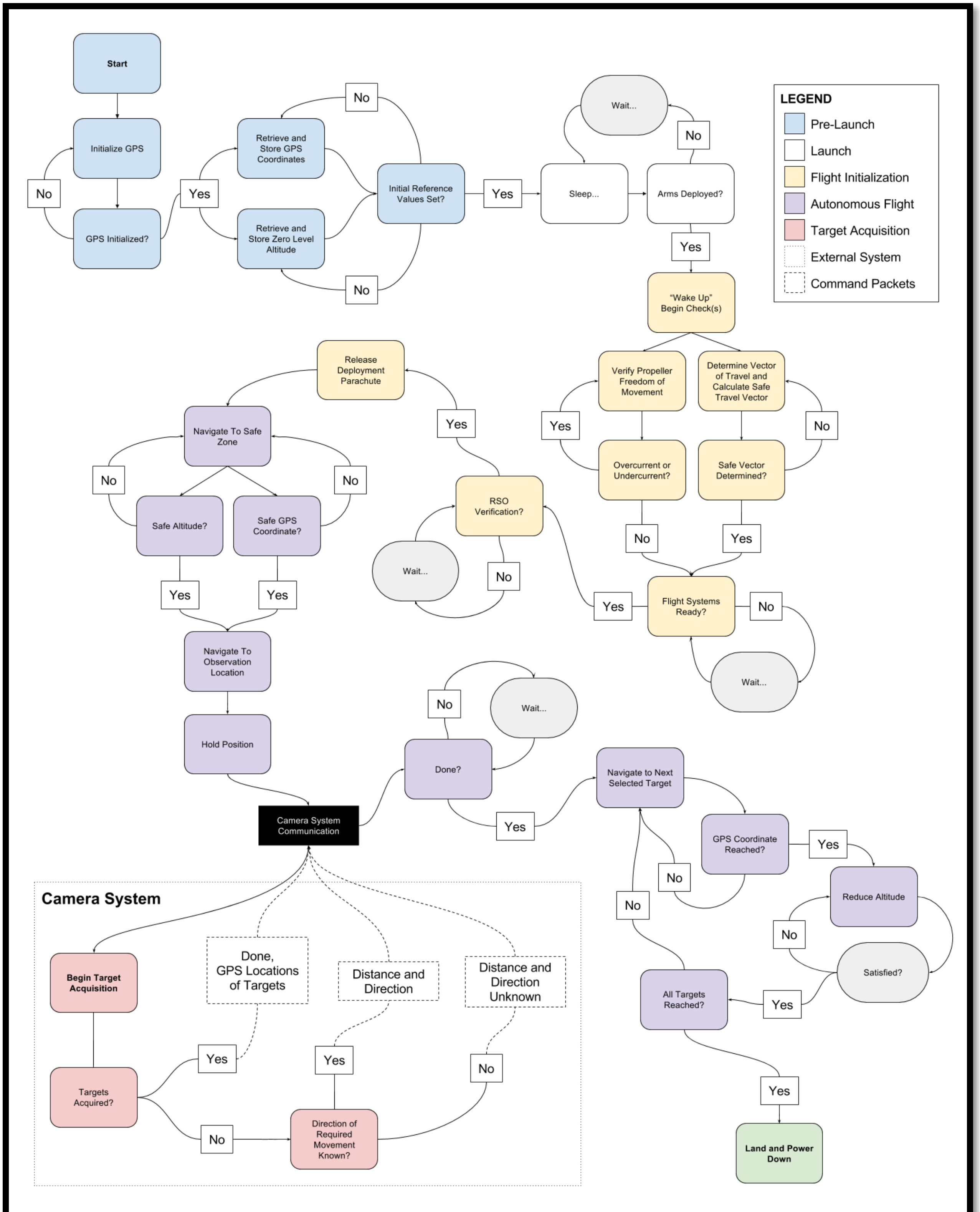


Figure 77: MRS GNC logic flowchart.

7.4.3 Motor Configuration

After selecting the flight control system, the team then had to select the appropriate motor configuration to meet the defined requirements. The motor configuration refers to the layout of the motors and arms of the MRS. The configurations investigated are outlined below in section 7.4.3.1 with accompanying figures.

7.4.3.1 *Options*

The three options compared for the motor configuration of the MRS include a quadcopter in an "X" layout (Figure 78), an octocopter in an "X" layout (Figure 79), and a hexacopter (Figure 80). Table 74 is a trade study detailing some of the advantages and disadvantages of each system relative to the criteria we used to choose the MRS motor configuration.

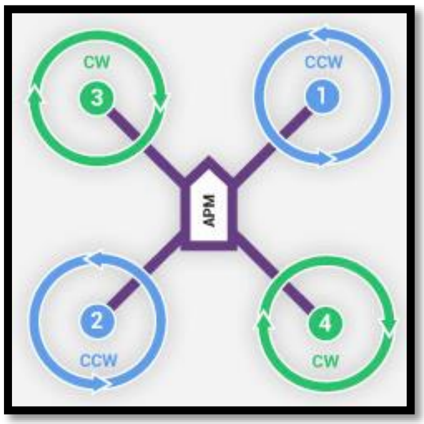


Figure 78: Quadcopter (X4) Configuration (4 Motors, 4 Arms)

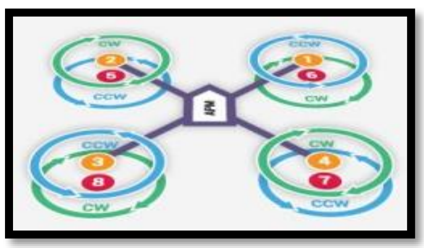


Figure 79: Coaxial Octocopter (X8) Configuration (8 Motors, 4 Arms)

Figure 80: Hexacopter Configuration (6 Motors, 6 Arms)

Motor Configuration							
Options:		Quadcopter (X4)		Coaxial Octocopter (X8)		Hexacopter	
Mandatory Requirements							
Supported by Flight Controller		Yes		No		Yes	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Benefits							
Stability (0-10)	25.00%	6	1.50	7	1.75	8	2.00
Motor Failure Redundancy (0-5)	10.00%	0	0.00	4	0.40	4	0.40
Ease of Design (0-10)	15.00%	8	1.20	4	0.60	5	0.75
Costs							
Required Size of Deployment Bay (0-10)	10.00%	5	0.50	9	0.90	7	0.70
Weight (est.)	35.00%	4	1.40	8	2.80	6	2.10
Monetary Cost (0-10)	5.00%	4	0.20	8	0.40	6	0.30
Calculations	**Total**						
Benefit	8.60	0.31		0.32		0.37	
Cost	9.30	0.23		0.44		0.33	
Total Score		**1.39**		**0.00**		**1.10**	

Table 74: Motor configuration trade study.

As can be seen above in Table 74, weight is a very important part of the choice of motor configuration. In order for the MRS to meet the requirements set upon it, a general rule of a 2:1 thrust-to-weight ratio is desired. This allows the multirotor to hover at 50% throttle under ideal conditions leading to responsive movement capabilities and increased power efficiency.

The coaxial octocopter adds great difficulty to the design of the deployment bay. When working within the confines of a 6 in. diameter deployment bay, the width of two motors stacked on each other would be nearly impossible to fit. The difficulty in physical design alongside the difficulty in finding a flight controller to support the configuration allowed this design consideration to be eliminated relatively quickly.

The hexacopter design was considered much more thoroughly. The design was alluring because of the amount of stability the configuration provides along with motor failure redundancy. This design also provides a greater thrust to weight ratio than the competing designs. The issue that ultimately led to the elimination of this configuration was the increased design complexity necessary to fit the folding arms within the physical confinements of the deployment bay.

The choice to use the quadcopter configuration over the others is a decision driven largely by simplicity of design. The quadcopter configuration can be designed to fit within the deployment bay relatively easily which leads to a much higher appeal over the other options. The quadcopter also reduces the amount of failures possible during deployment of the MRS in accordance with requirement MRS.2.

7.4.4 Propulsion System

Requirement Number	Requirement	Method of Verification
MRS.3.1	The propulsion system of the MRS shall provide a thrust-weight ratio of at least 2:1.	Analysis Analysis will be performed while choosing the propulsion system to be used.

Table 75: Propulsion system requirements.

Derivation of Requirement MRS.3.1

This requirement was derived to meet the common multirotor industry standard of maintaining a proper thrust to overall weight ratio which provides the multirotor with adequate agility.

This system encompasses the components that provide propulsion to the MRS. It includes brushless motors, rotors, and electronic speed controllers (ESC) which handle driving the motors. The choice of a suitable propulsion system is critical to the success of the mission as it must be capable of not only safely carrying the weight of the entire payload but also maneuvering with it for an undetermined period of time.

The three options being considered to meet our system requirements are the DJI E800 kit, the DJI E1200 kit, and a motor, rotor, and ESC combination consisting of the T-Motor M4008, a 14x4.8 rotor, and the DJI 640S ESC which is referred to as "Option #3" in the table below.

Table 76 is a trade study on our options for the propulsion system.

Propulsion System (Motor, Rotor, ESC)							
Options:		DJI E800 Kit (3510 Motor, 13x4.5 Rotor, 620S ESC)		DJI E1200 Kit (4216 Motor, 13x4.5 Rotor, 640S ESC)		Option #3 (T-Motor M4008, 14x4.8 Rotor, DJI 640S ESC)	
Mandatory Requirements							
Thrust > 15 lb.		Yes		Yes		Yes	
Max Rotor Diameter ≤ 15 in.		Yes		No		Yes	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Benefits							
Thrust (kg, 4 motors, 22.2V)	40.00%	8.4	3.36	15.6	6.24	6.4	2.56
Estimated Thrust-Weight Ratio	20.00%	2.3	0.46	3.7	0.75	1.8	0.35
Costs							

Weight (g)	10.00%	672	67.20	644	64.40	684	68.40
Blade Diameter (in)	15.00%	13	1.95	17	2.55	13	1.95
Monetary Cost ($)	5.00%	$319.00	15.95	$356.00	17.80	$407.00	20.35
Calculations	**Total**						
Benefit	13.72	0.34		0.62		0.26	
Cost	260.55	0.48		0.48		0.51	
Total Score		**0.71**		**0.00**		**0.50**	

Table 76: Propulsion system trade study.

Based on this trade study, the team has chosen the DJI E800 kit for the MRS propulsion system.

The DJI E1200 system has promising specifications but can be eliminated immediately by the large rotor size. Option #3 has a low overall thrust and estimated thrust-weight ratio and thus is outperformed by the DJI E800 system.

7.4.4.1 *Chosen Propulsion System Performance*

During the employment of the propulsion system trade study, an analysis was done on each motor to predict propulsion characteristics including the total thrust and thrust-weight ratio based on the DJI E800 specifications provided by DJI and the current estimated weight of the MRS.

Table 77: DJI E800 shows this analysis for the propulsion system chosen in section by the Table 76 trade study.

DJI E800 Propulsion Analysis	
Configuration	
Motor Configuration	Quadcopter (X4)
Characteristics	
Weights	
Total Weight (g)	3646.06
Total Weight Per Rotor (g)	911.52
Thrust Characteristics	
Total Thrust (g)	8,400.00
Thrust-Weight Ratio	2.304
Output	
Max Allowed Extra Weight (g)	553.94

Table 77: DJI E800 propulsion analysis.

As can be seen in table above, the thrust-weight ratio of the MRS is expected to be 2.304. This ratio is well above the requirement set by MRS.3.1.

7.4.5 Multirotor Arm Design

In order to accomplish mission requirement MRS.2, a deployable multirotor arm system was designed. Figure 81 and Figure 82 display a detailed view of the deployed Multirotor Arm Assemblies along with an individual detailed view of the assembly respectively Table 78. displays the requirements that were imposed on the Arm Pivot Assembly.

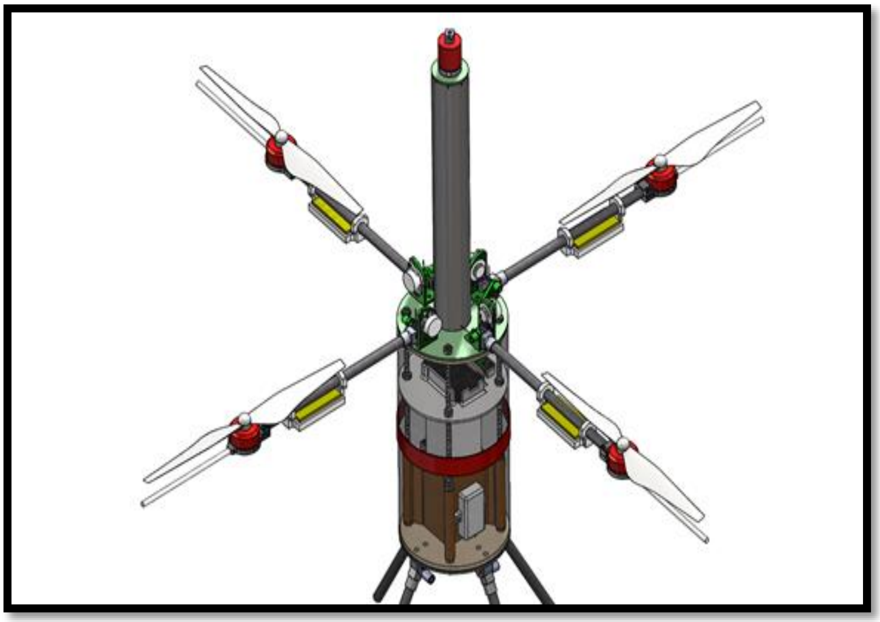


Figure 81: Upper Bulkplate Assembly with Deployed Propulsion Arm Assemblies

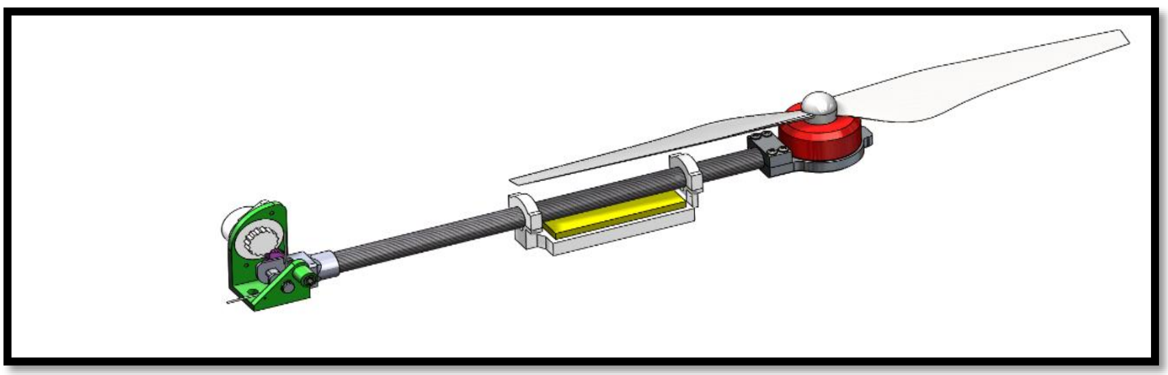


Figure 82: Deployed Multirotor Arm Assembly

Requirement Number	Requirement	Method of Verification
MRS.2.1	The Arm Pivot Assembly must provide stable actuation of the propulsion assembly upon payload deployment	The Pivot Arm Assembly will be prototyped and tested to verify deployment times and locking mechanism

MRS.2.2	The MRS Propulsion Arms shall deploy and lock into their flight positions upon separation from the deployment bay.	Testing of arm and leg deployment will be conducted to ensure repeatability of deployment.
MRS.2.3	Electrical feedback of successful arm deployment must be relayed to the ground station.	Testing the functionality and validity of limit switch feedback will be conducted.
MRS.2.4	The Arm Pivot Assembly must be capable of handling the max thrust and torque loads from the motor assembly	Analysis will be provided where the arm pivot assembly maintains a minimum FOS of 2 on yield for all worst case scenarios.

Table 78: Arm Pivot Assembly requirements.

Derivation of requirement MRS.2.1
Once the Arm Pivot Assembly is deployed from the airframe, the arms will be susceptible to slamming.

Derivation of Requirement MRS 2.2

Failure of the arms to lock into flight position will disable the MRS's flight capability. Recovery under redundant parachute reduces probability of successful target detection.

Derivation of Requirements MRS 2.3

Feedback from arms is required to ensure payload is ready for flight. Failure to recognize an arm hasn't successfully deployed could result in severe flight failure.

Derivation of Requirements 2.4

The arm pivot's inability to hold max thrust could result in mechanical failure during flight.

The arm design consists of the two major subsystems displayed in Table 79. Figure 83 displays the two major subsystems of the multirotor arm system.

Sub Assembly No.	Sub Assembly Name	Description
1	Arm Pivot Assembly	The Arm Pivot Assembly actuates the stowed arm into the deployed position during the deployment of the payload. The arm Pivot assembly transfers the propulsive loads of the motor into the payload upper bulkplate during flight.
2	Propulsion Assembly	The Propulsion Assembly consists of the motor and the arm tube which mounts into the arm pivot assembly.

Table 79: Propulsion Arm Subassemblies and descriptions.

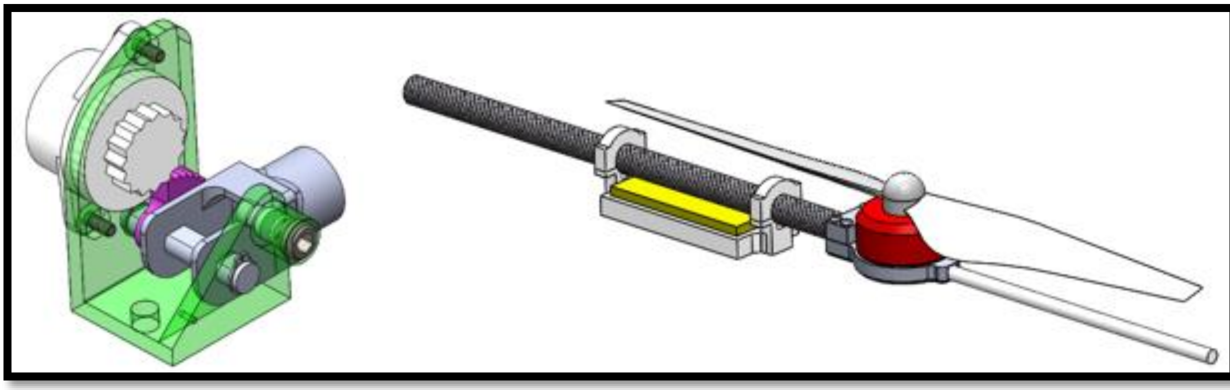


Figure 83: Subassemblies 1 and 2 (left to right) of the Propulsion Arm Assembly respectively.

7.4.5.1 *Separation Process*

The actuation of the multirotor arms into the deployed position moves through three distinct phases during the separation of the payload from the deployment bay. Figure 84,Figure 85,Figure 86, and Figure 87 illustrates the three distinct phases of the multirotor arm actuation process. The three phases are described in Table 80.

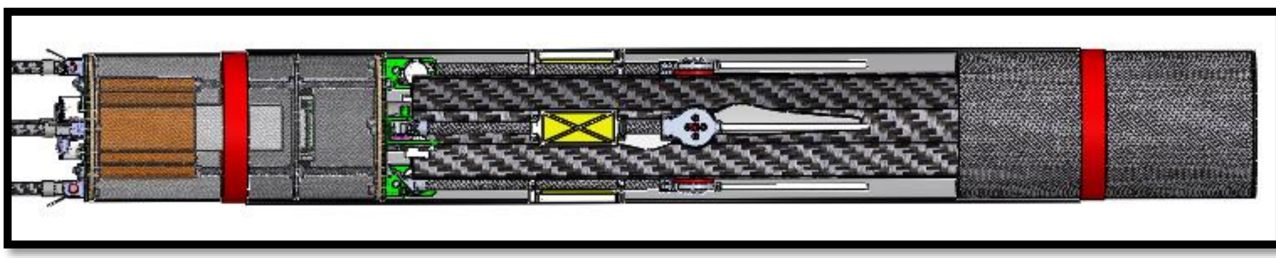


Figure 84: Payload stowed within deployment bay during separation event.

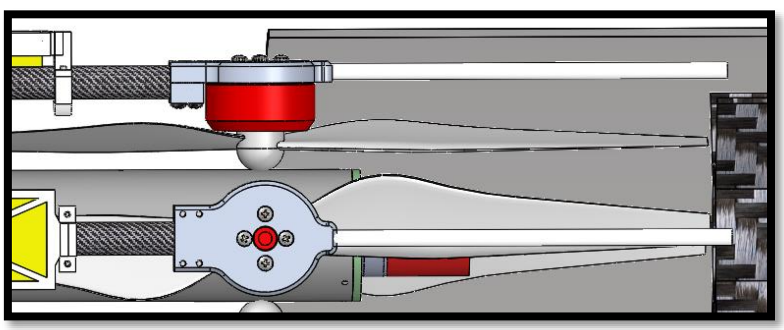


Figure 85: Phase 1 of the separation process detailing the multirotor arms clearing the deployment bay airframe.

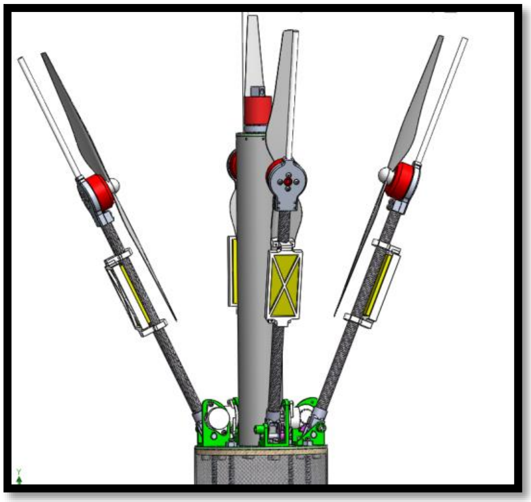


Figure 86: Phase 2 of the separation process detailing the actuation of the multirotor arms into the stowed position.

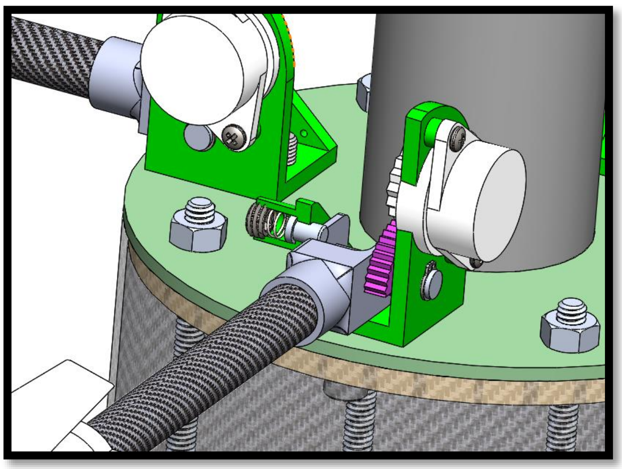


Figure 87: Phase 3 of the separation process detailing the actuation of the lock pin into the arm pivot with a section view of the arm clevis.

Separation Phase	Process Overview
Phase 1	Phase 1 begins when the vehicle deploys the payload from the deployment bay via black powder.
Phase 2	Phase 2 consists of the actuation of the multirotor arms into the deployed configuration.
Phase 3	Phase 3 consists of the lock pin mechanism actuating into the Arm Pivot.

Table 80: Separation Process overview descriptions.

7.4.6 Future Design

Future MRS Designs will contain limit switches to provide feedback to the flight controller confirming deployment. This feature will satisfy mission requirement mission requirement MRS

7.4.7 Flight Control System

The first task in designing the MRS to function according to the logic is to choose an appropriate flight control system to handle the proposed requirements. The flight control system consists of a main flight controller, a GPS/compass module, and a telemetry module for communication of diagnostic information to a ground station.

The mandatory criteria in decision making for this system includes the availability of an external command interface to allow the programing of autonomous decision making to command the movement and actions of the multirotor, and GPS to aid in the autonomous movement.

The trade study shows all of the criteria that went into the final decision and compares three options: the DJI A3, the Holybro PX4 "Pixhawk", and the TopXGun T1.

Flight Control System (MCU, GPS, Telemetry)							
Options:		DJI A3		Holybro PX4 "Pixhawk"		TopXGun T1	
Mandatory Requirements							
External Command Interface		Yes		Yes		No	
GPS		Yes		Yes		Yes	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Benefits							
Available Documentation (0-10)	40.00%	9	3.60	8	3.20	2	0.80
Extra I/O (#)	7.50%	4	0.30	8	0.60	12	0.90
Peripheral Availability (0-5)	10.00%	2	0.20	4	0.40	1	0.10
Ease of Use (0-10)	7.50%	7	0.53	6	0.45	3	0.23
Costs							
Weight (g)	30.00%	186	55.80	96	28.80	75	22.50
Monetary Cost ($)	5.00%	$899.00	44.95	$204.00	10.20	$309.00	15.45
Calculations	**Total**						
Benefit	11.30	0.41		0.41		0.18	
Cost	177.70	0.57		0.22		0.21	
Total Score		**0.72**		**1.87**		**0.00**	

Table 81: Flight control system trade study table.

Based on this trade study, the team is opting to move forward with the Holybro PX4 flight control system by a wide margin.

The DJI A3 system is an overall solid system, but is ultimately not feasible due to its extremely high monetary cost. The TopXGun T1 system does not meet all mandatory requirements and therefore can be immediately thrown out.

7.4.8 Flight Computer

7.4.8.1 *Options*

The three options being considered to handle the role of the flight computer for the MRS are the Raspberry Pi 3 Model B, the Banana Pi M3, and the BeagleBone Black.

Table 82 below contains a trade study outlining the requirements imposed on this component and the final choice.

Flight Computer							
Options:		Raspberry Pi 3 Model B		Banana Pi M3		BeagleBone Black	
Mandatory Requirements							
Compatible w/ PX4 Flight Controller		Yes		Yes		Yes	
Multithreading Capabilities		Yes		Yes		Yes	
Native Camera Interface		Yes		Yes		Yes	
Image Processing Capabilities		Yes		Yes		Yes	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Benefits							
Available Documentation (0-10)	50.00%	9	4.50	7	3.50	6	3.00
Prior Knowledge of System (0-10)	30.00%	9	2.70	4	1.20	2	0.60
I/O Availability	5.00%	40	2.00	40	2.00	92	4.60
Costs							
Weight (g)	15.00%	45	6.75	45	6.75	39.68	5.95
Monetary Cost ($)	5.00%	$47.50	2.38	$73.00	3.65	$55.00	2.75
Calculations	**Total**						
Benefit	15.50	0.64		0.42		0.32	
Cost	28.23	0.07		0.08		0.07	

Total Score	**8.72**	**5.00**	**4.57**

Table 82: Flight computer trade study.

There are many factors that need to be considered when choosing a flight computer. For this specific application, compatibility with the flight controller is a must, as well as supporting necessary image processing libraries for the TDS.

The availability of documentation is an extremely important factor in this selection because of the systems that it will be integrated into. The communal use of the computer system could cause issues which are resolved much faster if there is lots of documentation available. The dual use of this particular computer for the MRS and the TDS makes prior knowledge of the computer that much more important.

Because the team has much more experience using the Raspberry Pi and the availability of documentation is so much better than the next best option, the Raspberry Pi is the clear choice for this application. Using this computer allows the use of a Raspberry Pi Camera system, as it offers native support, a wealth of documentation, and sufficient configuration options.

7.4.9 Battery

7.4.9.1 *Options*

The battery selection of the MRS is an important consideration because of the fact that it is the heaviest electrical component in the system. The battery is also the key component in the estimation of flight time of the MRS. Although there are many options that could be considered in the design of the MRS, the three below are among the best based on the factors outlined in the trade study below. Based on the propulsion system trade study in Table 83, it was decided that a 6S Li-Po battery would be necessary.

The options that are being considered include the Tattu 8000mAh 15C, the Tattu 7000mAh 10C and the Turnigy Nano-tech 6000mAh 25C 6S Li-Po batteries. Table 83 is a trade study outlining the criteria being considered for the battery.

Battery							
Options:		Tattu (8000 mAh)		Tattu (7000 mAh)		Turnigy Nano-tech (6000 mAh)	
Mandatory Requirements							
Peak Current Output ≥ 90 A		Yes		Yes		Yes	
Capacity ≥ 6 Ah		Yes		Yes		Yes	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Benefits							
C Rating	5.00%	15	0.75	10	0.50	25	1.25
Battery Capacity (Ah)	50.00%	8	4.00	7	3.50	6	3.00
Costs							

Weight (g)	30.00%	942	282.60	914	274.20	1013	303.90
Volume (cm)	10.00%	446.5	44.65	401.1	40.11	450.1	45.01
Monetary Cost ($)	5.00%	$99.39	4.97	$71.99	3.60	$80.61	4.03
Calculations	**Total**						
Benefit	13.00	0.42		0.35		0.38	
Cost	1,003.07	2.66		2.54		2.82	
Total Score		**0.16**		**0.14**		**0.13**	

Table 83: Battery trade study.

As can be seen in the trade study above, the most important factor in our design consideration is the capacity of the Li-Po. This is a major factor when trying to get the most flight time possible. However, the drawback of increasing the capacity is that, as a general rule, the size (both weight and volume) of the battery is increased.

Based on the weight and battery capacity alone, the Turnigy battery can be quickly ruled out as an option. The battery is both bigger in size and lower in capacity than either of the Tattu batteries. Although a good trade study shows variety in all aspects, two Tattu batteries are being assessed because of the fact that, compared to most of the other 6S Li-Po batteries that can be found, Tattu batteries are consistently smaller and have higher capacities. In the decision between the 8 Ah and 7 Ah batteries, the higher capacity battery was marginally heavier, however not enough to outweigh the benefit of the higher capacity.

7.5 Target Detection System (TDS)

7.5.1 Microprocessor

As briefly discussed in the flight computer subsection 7.4.7, the microprocessor system requires a microprocessor capable of performing both the roles of flight computer and target detector. The Raspberry Pi 3 Model B is the MPU that best meets these requirements as evidenced by the flight computer trade study (Table 82). Both the TDS software and the flight computer software will run concurrently on the flight computer. Traditional means of inter-process communication will be used to relay data (GPS coordinates, etc.) between the two processes.

7.5.2 Camera Configuration

Due to the selection of the Raspberry Pi 3, the TDS will utilize one Raspberry Pi Camera; the camera will be mounted at the bottom of the landing craft and attached to a gyroscope in order to ensure that the camera is perpendicular to the ground at all times. The Raspberry Pi is supported natively by the Pi Camera peripheral. There are two versions of this peripheral. The first version (v1) is a 5MP camera that features an OmniVision OV5647 sensor with a field of view of 54x41°. The second version (v2) is an 8 MP model that employs a Sony IMX219 sensor; this model offers a slightly higher field of view of 62.2x48.8°.

The camera was selected based on the trade study below.

Camera System					
Options:		PiCam v1 (5MP)		PiCam v2 (8MP)	
Mandatory Requirements					
Pi Compatible		Yes		Yes	
Variable Output		Yes		Yes	
Infinity Focus		Yes		No	
Wants	**Weights**	**Value**	**Score**	**Value**	**Score**
Benefits					
Output Options (0-10)	90.00%	6	5.4	8	7.2
Costs					
Cost ($)	5.00%	$22.95	1.15	$24.66	1.23
Weight (g)	5.00%	3	0.15	3	0.15
Calculations	**Total**				
Benefit	12.6	0.43		0.57	
Cost	2.68	0.48		0.52	
Total Score		0.89		0.00	

Table 84: Camera system trade study.

Each sensor is capable of outputting a variety of different resolutions and frame rates, though v2 offers slightly better options and configurations. The v2 camera, however, does not support infinite focus, which is imperative for this application; without infinite focus, the level of control and precision required by aerial vehicles cannot be guaranteed, meaning that v2 can effectively been discarded. Infinite focus is a feature that allows a camera's focal point to be set a non-determinate distance away from the sensor; this implies that the distance an object is from the camera module has no bearing on the sharpness of the image.

Thus, it was for this reason that the older, less powerful camera module (v1) was chosen for the TDS.

Extensive testing will be done at all stages of development; test footage will be generated to maximize chances of success. Critical components of the system will be tested individually by conducting subscale and subsystem tests. Variables such as input quality and frame rate will be analyzed to determine what camera configurations are optimal.

7.6 Payload Structure System (PSS)

7.6.1 PSS Design

The payload structures system is responsible for housing and integrating all of the payload subsystems. The PSS design is separated into the interior and exterior components. The PSS interior consists of structural components and the exterior consists of structural mounts for the other payload subsystems. The PSS is shown below in Figure 88: PSS components and structures with coupler section removed (left) and coupler section in place (right). with the overall dimensions are listed in Table 85.

Height (in)	Diameter (in)	Mass (lbs)
12.25	6.00	4.12

Table 85: Overall PSS dimensions.

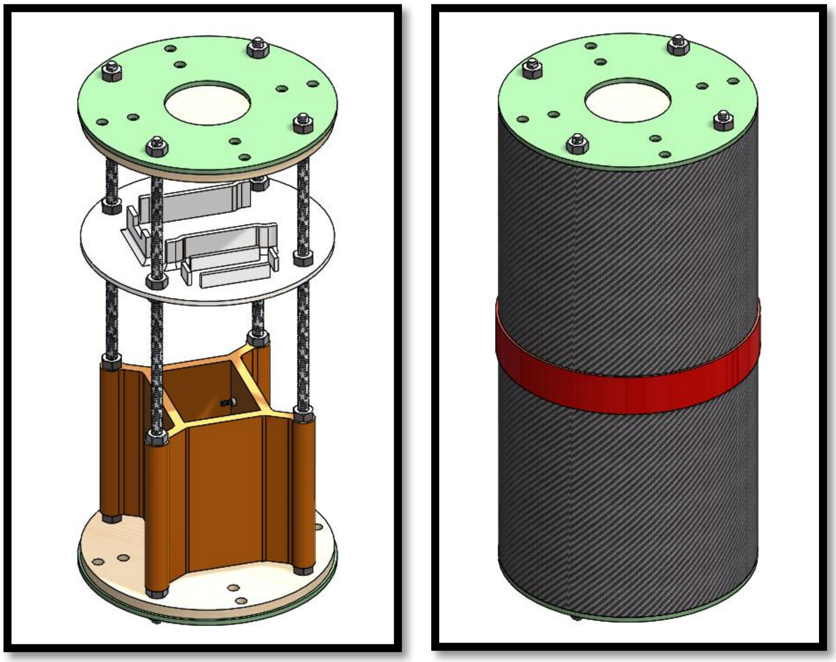


Figure 88: PSS components and structures with coupler section removed (left) and coupler section in place (right).

7.6.1.1 *PSS Interior*

Table 86 displays the sub-requirements of the PSS explained in this section.

Requirement Number	Requirement	Method of Verification
PSS.2.1	Provide structural support for the entire payload.	Demonstration Multiple test flights will be ran to demonstrate structural components ability to handle loads during flight and launch. .
PSS.2.2	Provide housing for all flight electronics	Testing Testing will be conducted on all electronic sleds to guarantee they properly fit all electrical components and can withstand loads experienced during flight.

Table 86: PSS interior structure.

Derivation of Requirements PSS 2.1

In order for the payload to hold all internal components required for flight. It must maintain structural support for all internal components through varying flight scenarios.

Derivation of Requirements PSS 2.2

This requirement was derived to ensure all mechanical and electrical components are protected during launch and flight.

7.6.1.2 Structural Support

There are three main components to the interior structural support of the PSS; the top bulkplate assembly, the bottom bulkplate assembly, and ¼ 20 aluminum all thread. The top and bottom bulkplate assemblies consist wooden and fiberglass bulkplates. The wooden inner bulkplates align the structure in the coupler by fitting into the coupler section. The fiberglass bulkplates fit on the top and bottom ends of the coupler section. Four aluminum threaded rods run though to both ends of the fiberglass bulkplates. These rods are bolted on each end and on both sides of the bulkplate to keep the structure in a rigid position. Aluminum all thread was chosen for its light weight, stiffness, and fastening ability.

7.6.1.3 Electronics Housings

Two 3D printed electronics housings are mounted in the PSS interior. These housings are secured in place on the aluminum all thread rods. The first housing, the flight controlled bulkplate, mounts the Pixhawk flight controller and teensy. The flight controller bulkplate is shown below with electronics mounted in Figure 89.

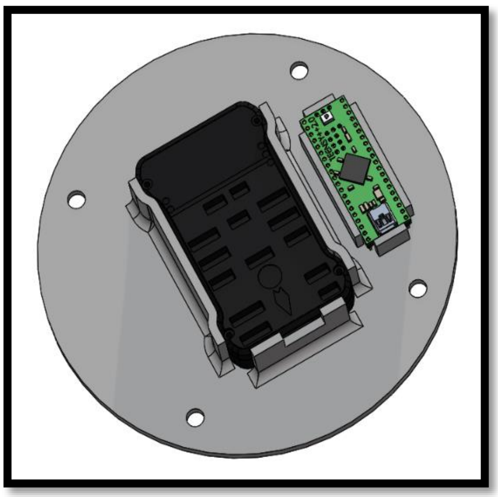


Figure 89. Bulkplate mounting the Pixhawk flight controller and the teensy.

These components are currently held in place through press fits on their 3D printed walls. This bulkplate has the ability to be placed in different vertical positions on the all thread. This was designed with in order to have the ability to mount the flight controller at the exact center of gravity of the payload.

The second 3D printed housing mounts the flight battery, Raspberry Pi, and GPS/compass module. The exploded view of this housing and electronic components are shown below.

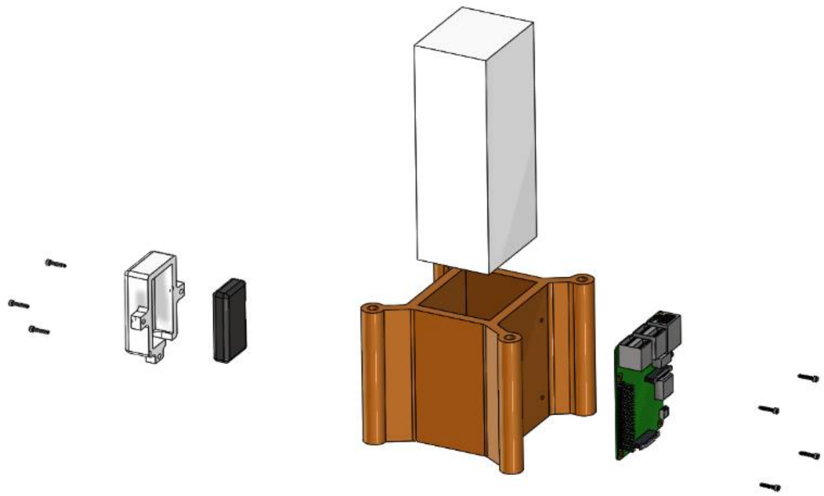


Figure 90. Exploded view of the 3D printed housing for the flight battery, Raspberry Pi, and GPS/Compass module.

The white box is a mockup of the flight battery, the Raspberry Pi is on the right and the GPS/compass module is on the left. 4-40 fasteners are used to secure these electrical components down; except for the flight battery which will be fastened by a velcro strap.

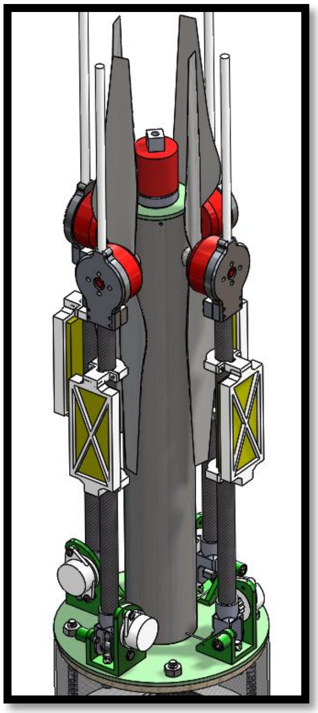
7.6.1.4 *PSS Exterior*

The exterior airframe of the PSS is a coupler section of the main launch vehicle airframe. This carbon fiber coupler section protects and supports the interior payload components. A witness ring is in the center of this coupler section. Shear pins will connect the recovery bay airframe and deployment bay airframe to either side of this witness ring. The dimensions of the coupler section are laid out in the (Insert reference to Drawing).

Five systems are supported on the exterior of the PSS. The LLS, TDS, MRS, RRS, and DCS are all mounted or have components on the PSS exterior. These subsystems are discussed below on what bulkplate they are mounted on.

7.6.1.5 *Top Bulkplate*

The MRS, RRS, and components of the DCS are mounted on the top bulkplate of the PSS. The MRS arm clevises are rigidly mounted to the top bulkplate with ¼ -28 bolts. The higher thread count is needed to get the most amount of thread surface area to grip to the bulkplate.



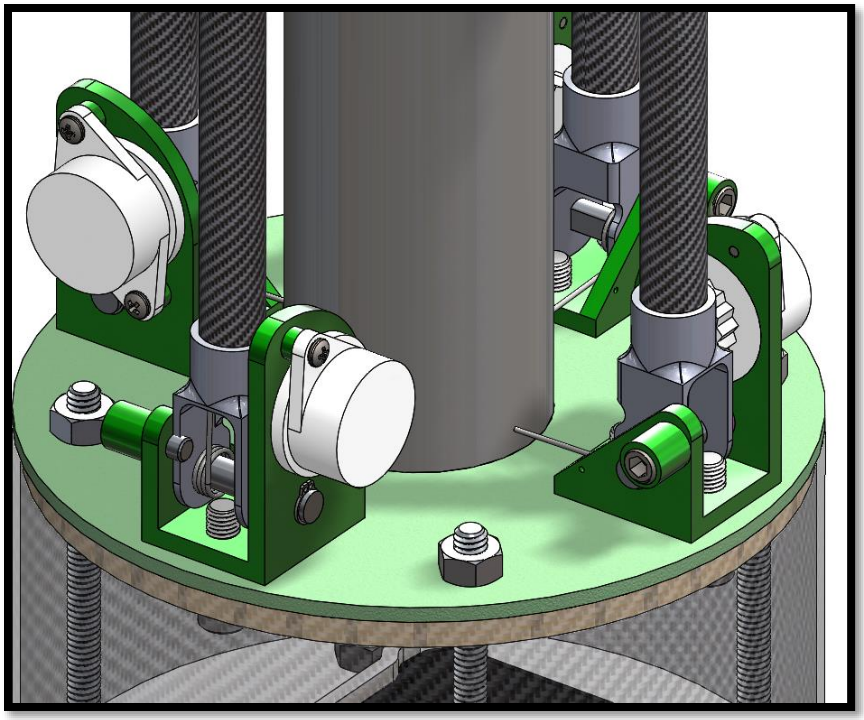


Figure 91. The top PSS bulkplate shown with the MRS in the stowed position, the RRS tube, and the DCS components at the top (right). A close up of the top bulkplate with the systems mounted on it (left).

The RRS deployment tube must be rigidly mounted to the top bulkplate assembly to mitigate cocking risks during the DCS deployment. This rigid mount is accomplished through mounting the RRS tube to the wooden bulkplate with epoxy through a cut in the top fiberglass bulkplate.

7.6.1.6 *Bottom Bulkplate*

The LLS and TDS are mounted on the bottom bulkplate assembly. Figure 92 below shows these systems mounted on the bottom bulkplate assembly with the legs in their deployed position.

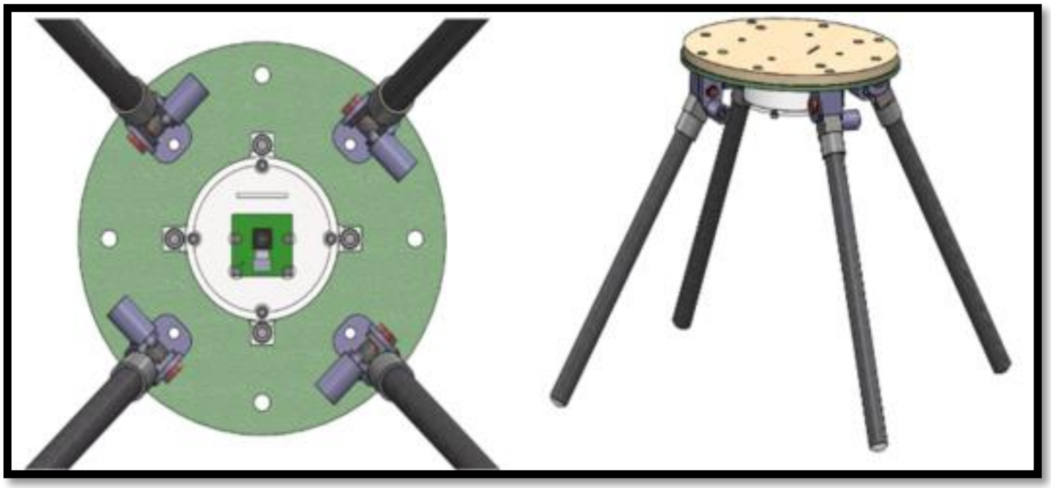


Figure 92: Bottom bulkplate with TDS and deployed legs(left) and bottom bulkplate in the landing orientation with legs deployed(right).

Each leg clevis will be fastened with two 12-24 bolts for a rigid connection to the bulk plate during all stages of actuation. The TDS mount has a surrounding wall and acrylic cover to prevent the camera from being blurred during deployment.

7.7 Redundant Recovery System (RRS)

In order to be considered a success, the RSS must meet the mission requirements outlined in Table 87 below.

Requirement Number	Requirement	Method of Verification
RRS.1.1	The RRS shall deploy a parachute during the event of a payload flight anomaly within the MRS.	Test Ground and flight testing will be accomplished to verify RRS deployment logic and functionality.
RRS.1.2	The RRS must have control system redundancy.	Inspection A preflight inspection will be performed to ensure redundant control systems are functioning nominally.

RRS.1.3	The RRS shall deploy if the vertical kinetic energy of the payload exceeds 75ft-lb.	Test Payload flight and drop tests will be performed to verify the systems conditional logic.
RRS.1.4	In the event of a flight anomaly, the RRS shall cut power from the MRS.	Test Flight anomalies will be simulated through ground testing to validate system functionality.

Table 87: Sub-requirements of the RRS.

7.7.1 RRS Design

As briefly described in Table 67, the RRS is an onboard electronic system that performs constant monitoring of the MRS to detect potential flight anomalies. In the event that an anomaly is encountered, the RRS will step in by shutting down necessary electronics of the MRS and deploy a parachute to return the MRS safely to the ground.

7.7.2 RRS Electronics

Since the electronic requirements of the RRS are very similar to those in the VDS, the RRS will employ the use of some of the same electronic components. It will consist of a Teensy microcontroller, a BMP180 barometric altimeter, and several MOSFETs that together make up the electronic redundancies of the payload.

The RRS electronics are outlined below in Figure 93.

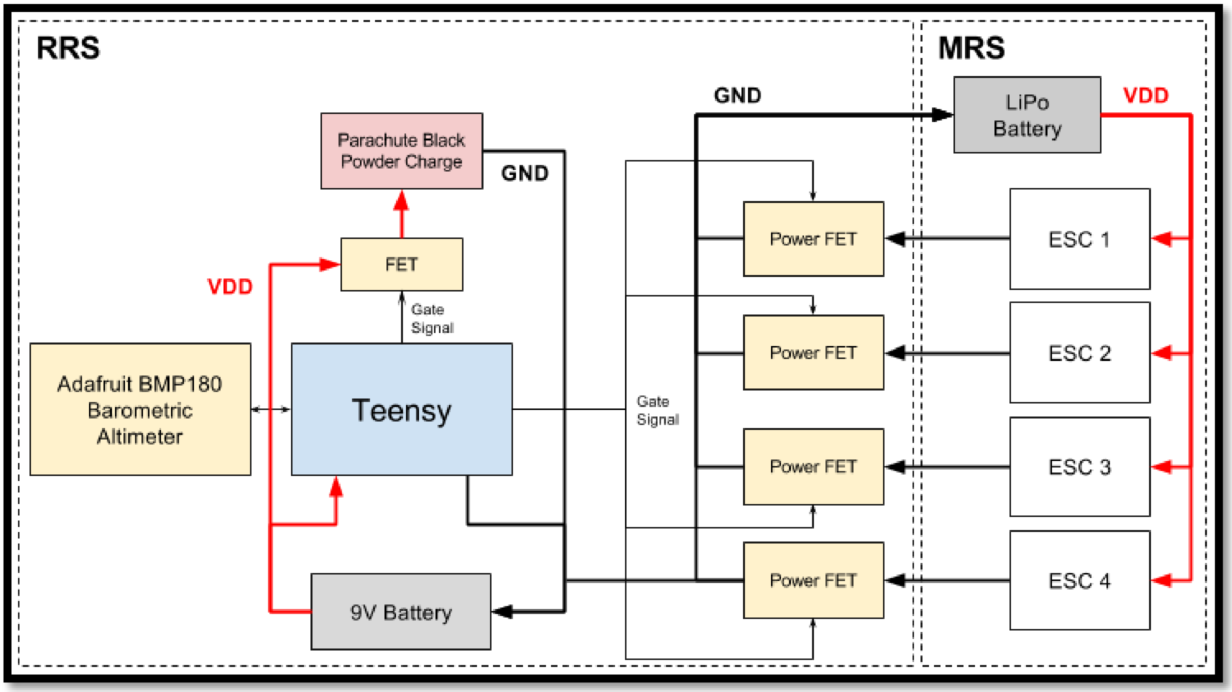


Figure 93: RRS Electronic Block Diagram

In order to improve on safety in the event of an issue during flight, the entire electronic sub-system of the RRS will be duplicated and both will run in parallel onboard the rocket.

7.7.3 RRS Logic

This section outlines the logic that the RRS will follow. The logic is generally described in words and represented by the logic flowchart shown in Figure 94.

7.7.3.1 *Pre-Flight*

The RRS controller will initialize as soon as the rocket is put onto the launch pad and readied for launch. It will then begin monitoring the altitude.

7.7.3.2 *Unsuccessful Deployment Monitoring*

Once the RRS reaches an altitude over 200ft, the system will begin to monitor the MRS for flight initialization status. If the system reaches an altitude of lower than 200ft and the MRS flight initialization status is still false, the RRS will release the deployment parachute, cut the electrical power to the MRS and deploy the RRS parachute.

7.7.3.3 *MRS Malfunction Monitoring*

Upon successful initialization of the MRS, the logic of the RRS will no longer look at the raw altitude, but rather derive the falling velocity from the change in altitude to determine the kinetic energy. The system will ensure that the kinetic energy is less than the 75ft-lb in accordance with requirement RRS.1.3. If the kinetic energy is outside of this requirement, the system will cut the electrical power to the MRS and deploy the RRS parachute.

7.7.3.4 *MRS Distress Signal Monitoring*

The MRS will have the ability to send a distress signal to the RRS in the event that any of the systems within the MRS detect flight conditions that would lead to an unsuccessful landing. For example: tumbling or massive uncorrectable drift as a result of a broken propeller or some other off nominal case. This signal would cause the RRS to cut power to the MRS and deploy the RRS parachute.

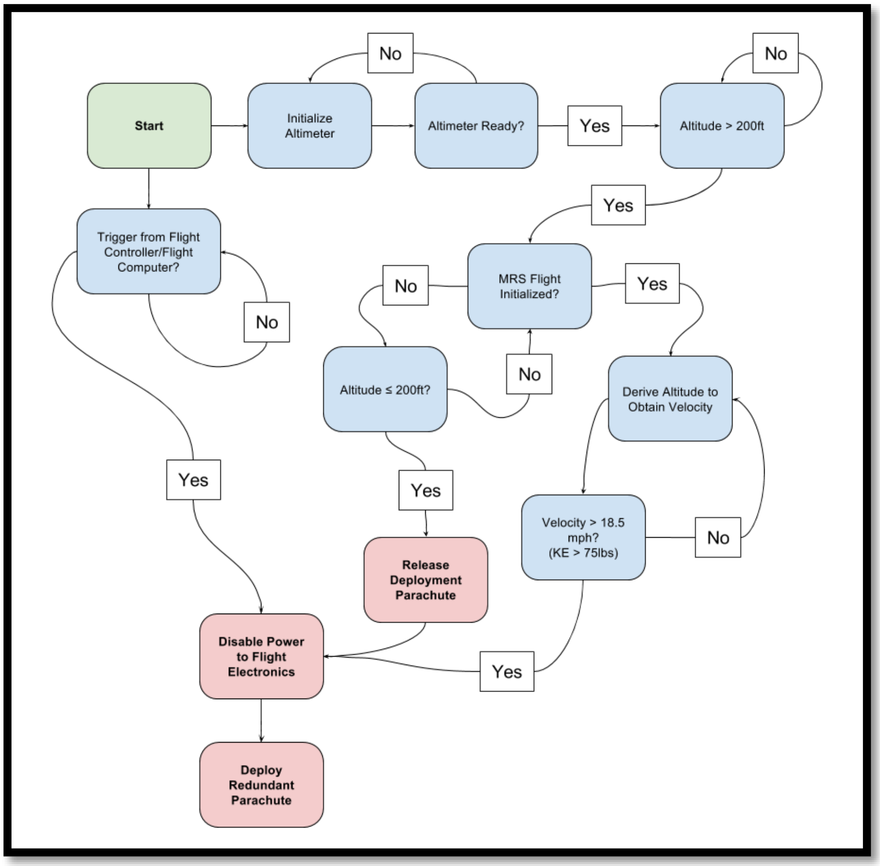


Figure 94: RRS logic flowchart.

7.7.4 RRS Parachute

The redundant recovery parachute is packed inside 14 inch RRS tube. This tube is mounted onto the top bulkplate assembly of the PSS. The DCS AARD bulkplate seals the top of the RRS tube through a shear pin connection. The RRS tube is shown in Figure 95 mounted to the PSS with the DCS components on top.

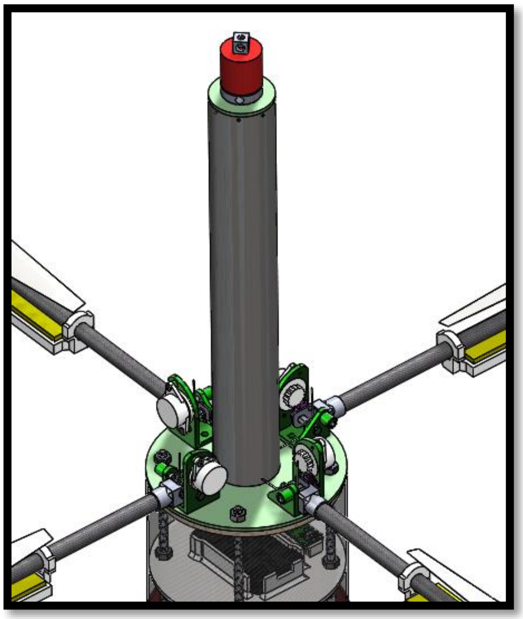


Figure 95: RRS tube shown in payload assembly.

7.7.4.1 *Parachute Deployment*

The RRS parachute is deployed through black powder deployment. This black powder deployment shears the ARRD bulkplate and propels the RRS parachute out to deploy.

7.8 Landing Leg System (LLS)

7.8.1 LLS Design

In order to be considered a success, the LLS must meet the sub-requirements outlined in Table 88 below.

Requirement Number	Requirement	Method of Verification
LLS.2.1	LLS must not contain any protrusions or edges that might cause interference with recovery systems	Inspection During manufacturing inspection of all component surfaces, edges, and joints will guarantee no protrusions or sharp edges.
LLS.2.2	Limit switches must confirm to flight controller that legs have successfully deployed	Test Testing will ensure that limit switch location allows for consistent differentiation of the deployed configuration vs. the stowed configuration.

LLS.1.1	LLS must remain functional after black powder separation	Test Further testing will verify that limit switches and mechanical components can function properly after a black powder separation
LLS.1.2	LLS housing and legs must fit into sheathes along inner airframe of recovery bay	Inspection & Testing Inspection of leg fitment into sheathes, and testing of leg deployment will verify that leg properly fit into recovery bay sheathes.
LLS.3.1	LLS deployment torsion spring must provide enough torque to rotate leg into deployment configuration.	Testing of leg deployment will guarantee that torsion springs can consistently deploy legs.

Table 88: LLS sub-requirements.

Derivation of Requirements LLS.2.1
Snagging or hanging the LLS on the recovery system could cause catastrophic failure of MRS or launch vehicle. Meeting this requirement mitigates the risk of this failure.

Derivation of Requirements LLS.2.2
Failure of the arms to deploy would keep the payload from being recovered under the MRS. This requirement was implemented to guarantee that in the event the arms do not deploy, the payload will not deploy.

Derivation of Requirements LLS.1.1
This requirement was derived because the LLS is located near black powder charge for separation. The LLS must be able to function after charge has ignited.

Derivation of Requirements LLS.1.2
Orientation of the payload during launch dictates that the legs be located below the payload inside the recovery bay. The LLS must fit into the sheathes of the recovery bay to not inhibit the main booster recovery system.

Derivation of Requirements LLS.3.1
The LLS must be able to deploy from its stowed position to its landing position. This requirement was derived to ensure that the torsion springs provide enough torque to actuate the legs to the required position.

7.8.1.1 *Landing Leg System*

The payload LLS is comprised of four carbon fiber legs attached to the bottom of the payload. In flight configuration the LLS is stowed in the outer portion of the launch vehicles recovery bay. Separation of the recovery bay airframe will allow springs to actuate the legs into a landing configuration. The LLS contains two main components, the spring housing, and leg pivot. Figure 96: LLS overall viewFigure 96 below shows the LLS by itself and attached to the payload.

Figure 96: LLS overall view

Table 89 below displays the overall dimensions of one leg in the LLS.

Mass (lb)	Length (in.)	Width (in)
0.081	8.83	1.00

Table 89:Overall dimension of single leg.

7.8.2 <u>LLS Components</u>

7.8.2.1 *<u>Spring Housing</u>*

The spring housing is shown below in Figure 97.

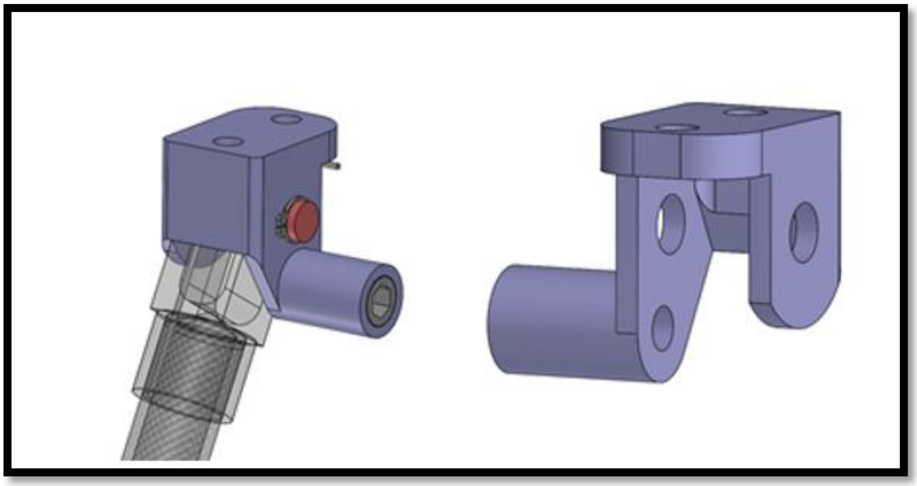


Figure 97: Spring housing attached to leg (left) spring housing alone (right).

The spring housing is designed to meet the following mission requirements outlined in Table 90.

Requirement Number	Requirement	Method of Verification
LLS 4.1	LLS must maintain a FOS of 2 for deployment and landing loading scenarios	FEA simulations will be conducted to ensure proper FOS on all components.
LLS 3.1	Pin must be able to consistently lock leg pivot into landing configuration.	Ground testing will be conducted to simulate deployment and prove pin can consistently snap into place.
LLS 3.4.	Limit switches must confirm to flight controller legs are fully actuated before cutaway.	Ground testing will ensure that limit switches are located so they consistently detect leg deployment.

Table 90: Mission requirements for spring housing.

Derivation of Requirements LLS.4.1
This requirement was derived to ensure all mechanical systems can confidently function in all scenarios.

Derivation of Requirements LLS.3.1
This requirement was implanted to ensure repeatability of LLS actuation success.

Derivation of Requirements LLS.3.4
Limit switches provide necessary feedback on LLS actuation. This ensures payload will not cut away without proper support for landing.

Table 91 below displays general dimensions of the spring housing sub-system.

Mass (lbs)	Length (in.)	Width (in)
0.081	8.83	1.00

Table 91: Overall dimensions of spring housing sub-system.

The main function of the spring housing is to attach each leg to the bottom of the payload, house a deployment torsion spring, and locking pin. In order to meet mission requirement LLS 4.1, the spring housing will be made out of billet 6061-T6 aluminum using a CNC mill. Using aluminum ensures adequate strength and stability upon deployment, descent, and landing.

The spring housing has a 0.25in. thru hole that holds the leg pin. This pin connects the legs to the payload and allows the leg to swivel from a stowed configuration to landing configuration, this feature satisfies requirement LLS.1.2. A breakdown of all mentioned components are shown in Figure 98.

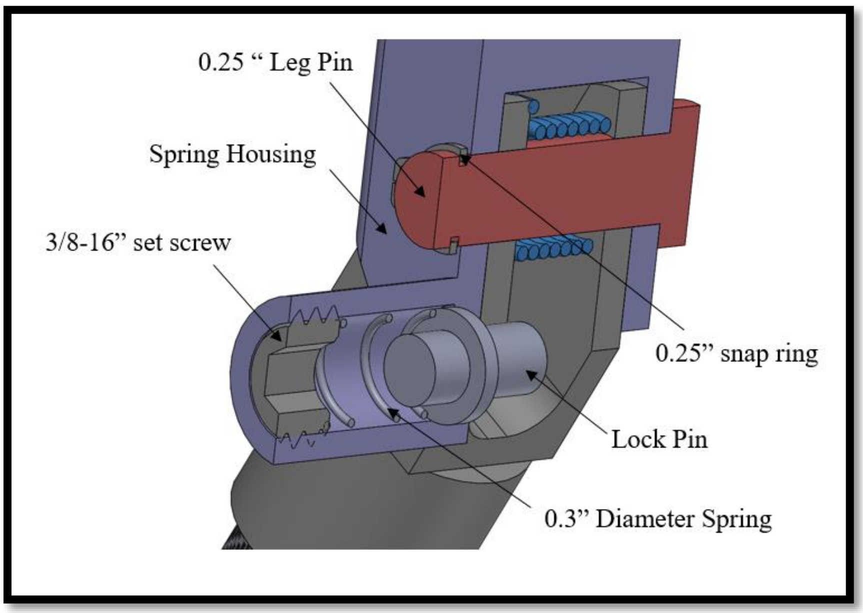


Figure 98: Breakdown of LLS components

The leg pin will be lathed out of aluminum. A flange on one end, and a 0.25in. snap ring on the other end will keep the pin in the housing. A groove will be machined into an end of the pin to ensure the snap ring maintains its position. A snap ring was selected for its lightweight and simplicity. Future LLS designs will contain limit switches to provide feedback to the flight controller confirming deployment.

Locking Mechanism

The extrusion on the outside of the spring housing contains the locking mechanism. The locking mechanism is shown here in a sectional view.

7.8.2.2 *Landing Leg Sub-Assembly*

Attached to the spring housing is the landing leg sub-system of the LLS. Figure 99 shows the landing leg sub system and its components.

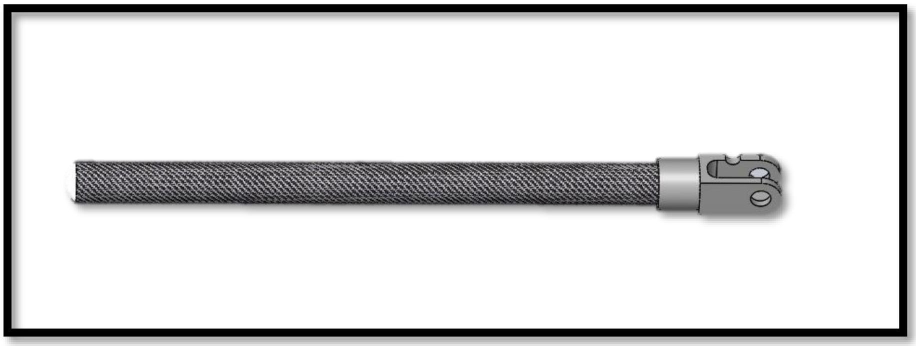


Figure 99: Landing leg sub-system

The landing leg sub-system meets the following mission requirements.

Requirement Number	Requirement	Method of Verification
LLS.1.2	The landing leg must maintain small diameter to fit into recovery bay sheathes.	Inspection The fitment of the attached recovery bay and payload will be inspected to ensure landing legs have adequate room inside sheathes.
LLS.3.2	The landing leg must be lightweight to maximize the deployment springs effectiveness.	Testing Ground testing will be conducted to guarantee the deployment spring can effectively deploy the legs.
LLS.4.1	Landing leg must keep the bottom of the payload at least 6in above the ground.	Inspection Inspection of the final payload
LLS.3.3	Landing legs and locking mechanisms must be constructed to resist wear throughout multiple deployments.	Test Numerous ground tests will verify that continuous usage will not affect performance of locking pin.

Table 92: Landing leg sub-system mission requirements.

Derivation of Requirements LLS.1.2

This requirement was derived to enable the LLS to easily stow into the recovery bay. This maximizes the amount of room for the booster recovery system.

Derivation of Requirements LLS.3.2

This requirement was derived to make it easier for the torsion springs to actuate the legs to landing position.

Derivation of Requirements LLS.4.1

This requirement was derived to fulfill analysis conducted in section 1.2.1.1.

Derivation of Requirements LLS.3.5

This requirement was derived to ensure the LLS withstands a rigorous testing schedule.

7.8.2.3 Design

The main function of the landing leg sub-system is to serve as the legs for the payload. The landing leg sub-system comprises of three components, this includes the leg pivot, carbon fiber leg, and 3D printed foot. An exploded view of the landing leg sub-assembly components can be seen below.

The leg pivot connects the carbon fiber leg to the spring housing. In order to meet requirement LLS 3.3, the leg pivots will be milled out of 6061-T6 aluminum to ensure repetitive use will not affect LLS performance. This grade was selected for its machinability and strength. A 0.5in. hole is milled into the bottom of the leg pivot. This will fit over the carbon fiber leg and epoxy the two components together. The top end of the leg pivot contains a 0.25in. thru-hole. This hole fits over the leg pin and allows the landing leg sub-assembly to rotate during deployment.

The landing leg is a 7.5in. x 0.5in diameter carbon fiber tube. In order to meet mission requirement LLS 3.2, carbon fiber was selected as the leg material. A 3D printed insert will be epoxied into the end of the leg act as a foot for the landing leg sub-assembly. This prevents dirt or any unwanted objects from entering inside the leg. Figure below shows the 3d printed foot insert.

7.9 DCS

7.9.1 DCS Trade Study

Five different DCS designs were considered. These consist of shock cord, Sky Crane pneumatic cushion, shock cord and pneumatic cushion, rotatable airframe flaps, and parachute deployment.

Shock Cord Deployment

This deployment was proposed for its simplicity. After section separation at apogee, the deployment bay would be separated by black powder charges. The MRS would be jettisoned with this black powder charge and begin to fall away from the remaining sections. At this point, the MRS is attached to a shock cord that is rigidly attached to a bulkplate in the deployment bay. The MRS's fall is stopped by the shock cord, then initializes its flight systems and an ARRD performs the cutaway when it is cleared to fly. One problem with this method is it induces a large shock force on the deployment bay bulk plate and payload.

Sky Crane Pneumatic Cushion

This system was proposed to mitigate the shock force from the MRS deployment black powder charge. Black powder deployment with the sky crane presents similar issues as seen with the shock cord method. As the MRS falls, it is rigidly connected to the sky crane. The sky crane is an aluminum rod that deploys the MRS. As the sky crane follows the MRS down its falling path, it encounters the pneumatic cushion. A model of this pneumatic cushion is shown below in Figure 100.

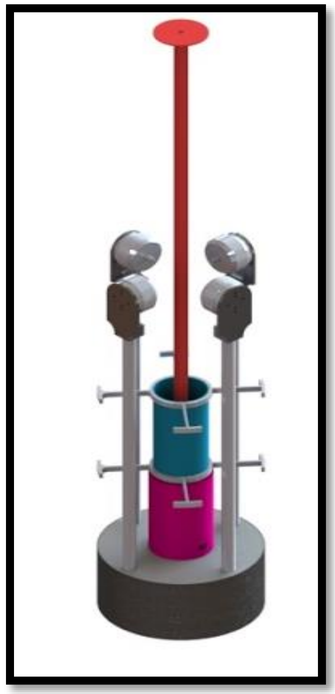


Figure 100: Sky crane with pneumatic cushion.

The pneumatic cushion is achieved through a small cross sectional area difference between the red bulkplate and the blue pneumatic cushion tube. This area difference controls the fluid flow of the ambient air out of the carbon fiber tube, enabling the built up pressure inside to cushion the fall. This cushion significantly decreases the shock force that is seen on the deployment bay bulk plate during deployment.

The pneumatic cushion arm needs to be long enough to give the motor arms clearance to deploy. This criteria resulted in a length of a 20 inch arm. With this long crane arm, the volume and weight of the deployment bay increases significantly. Once the sky crane comes to a rest, the MRS initializes and cuts away through linear solenoid pins.

Shock Cord and Pneumatic Cushion Deployment

This deployment system was proposed to mitigate the shock force from the MRS black powder deployment and to reduce the size of the deployment system. This deployment is similar to the Sky Crane pneumatic cushion, but with the length of the sky crane arm significantly reduced. A shock cord replaces most of the sky crane arm except for the pneumatic cushion. As the MRS falls during deployment, the shock cord extends and eventually actuates the pneumatic cushion when it has reached its full length. After the pneumatic cushion actuates, the MRS initializes and cuts away from the shock cord with a tender descender.

Rotatable Airframe Flap Deployment

This deployment method was proposed to negate the falling of the payload out of the deployment bay. Cutouts on the deployment bay airframe and an interior coupler enable the MRS arms to deploy out of the airframes. The coupler section is able to rotate to open and close the cutouts on the deployment airframe. The complexity of this design and risk of having cutouts on the airframe pose design issues with this deployment scheme.

Parachute Deployment

This deployment system was proposed for its simplicity. Parachute deployment is a standard method of deployment in rocketry. The simplest way to have the MRS initialize its flight systems before separation is to have it initialize under its own parachute. The MRS deploys under its own parachute from the deployment bay after the nosecone section separates from the booster section and deploys its parachute. The MRS now initializes under its own parachute. After MRS initialization, it cuts away from this parachute with an ARRD and begins its flight sequence.

Deployment and Separation System											
Options:		Just a Shock Cord		Sky Crane pneumatic cushion		Shock cord and pneumatic cushion		Rotatable Flaps		parachute	
Mandatory Requirements											
Safely allow arms to deploy		Yes		Yes		Yes		Yes		Yes	
Separate the payload		No		Yes		Yes		Yes		Yes	
Allow for multirotor initialization		Yes		Yes		Yes		Yes		Yes	
Categories	**Weights**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**	**Value**	**Score**
Mitigating Risk	20.00%	3	0.6	2	0.4	4	0.8	3	0.6	7	1.4
System simplicity	15.00%	7	1.05	2	0.3	5	0.75	6	0.9	5	0.75
Testability	15.00%	3	0.45	7	1.05	7	1.05	7	1.05	4	0.6
Weight	15.00%	7	1.05	3	0.45	5	0.75	5	0.75	9	1.35
Available Documentation	10.00%	4	0.4	0	0	0	0	0	0	9	0.9
Manufacturability	10.00%	8	0.8	3	0.3	6	0.6	5	0.5	7	0.7
Length	10.00%	8	0.8	3	0.3	6	0.6	6	0.6	10	1
Affordability	5.00%	9	0.45	7	0.35	8	0.4	5	0.25	5	0.25
Total Score		**5.6**		**3.15**		**4.95**		**4.65**		**6.95**	

Deployment is an extremely catastrophic environment, the DCS needs to be as simple and robust as possible. Rotatable flaps and pneumatic cushion deployment are intricate designs with a fair amount of complexity to them. Through the tried and true way of deployment by parachutes coupled with the results of the trade study, a parachute deployment will be used for the DCS.

7.9.2 DCS Design

7.9.2.1 *DCS Layout*

The DCS layout consists of the deployment bay structures and payload structures.

The deployment bay structures consist of the deployment tube, two centering bulkplates, and the MRS arm guides. The deployment tube is a two inch inner diameter carbon fiber tube that runs from the top of the deployment bay to the top of the payload in its stowed position. This tube is rigidly connected to the top bulkplate of the deployment bay and loosely connected around the RRS tube. This tube serves as the black powder deployment cavity. The MRS arm guides ensure that the MRS propellers do not become cocked or hit each other during any point of the deployment. The deployment bay is shown in Figure 101.

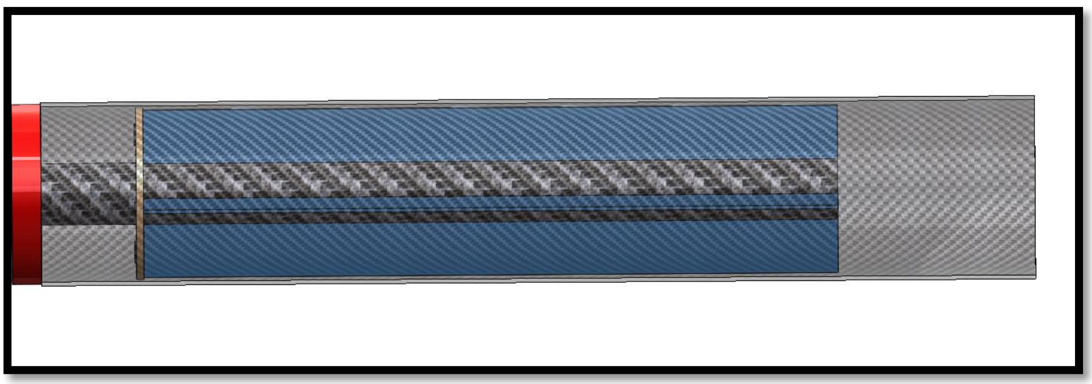


Figure 101: Deployment bay with MRS arm guides highlighted in blue and the deployment tube.

Paylaod Structures

The DCS structures on the paylaod body are the ARRD and ARRD bulkplate. These two components are located at the top of the RRS tube. The ARRD is rigidly mounted to the ARRD bulkplate through a 5/16-18 bolt. The ARRD bulkplate is mounted to the RRS tube with three shear pins that thread through an interior aluminum collar. Each nylon shear pin can hold a minimum of 50 pounds each. With a deployment parachute oupning force of one pound, three shear pins will provide the support needed to withstad the deoployment parachute opening force. These three shear pins will be sheared away in the evernt of the RRS deployment. These components are shown below Figure 102.

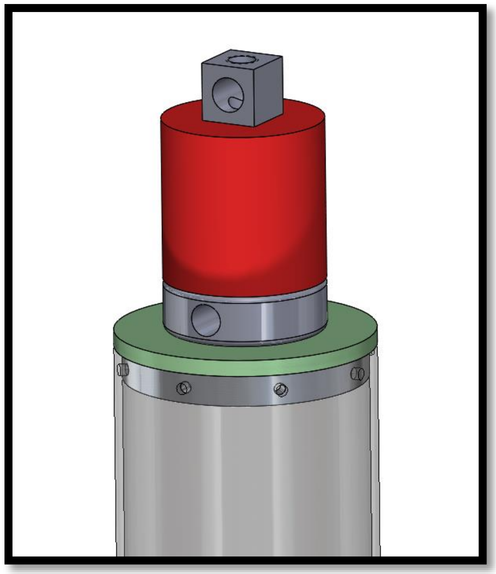


Figure 102: ARRD, ARRD bulkplate, and the aluminum shear pin collar mounted onto the RRS tube.

The final PSS structure is the deployment parachute. This parachute is stowed in the cavity betweeen the RRS tube and the deployment bay tube. The deployment parachute deploys after the payload is jettisoned from the deployment bay. The top shackle of the ARRD connects the payload to the deployment parachute. The design of this parachute is explained in

7.9.3 DSS Operation Scheme

The release of the payload from the launch vehicle to its flight is a two-step process. The release steps are black powder jettison and Advanced Retention and Release Device (ARRD) cutaway.

7.9.3.1 *Black Powder Jettison*

The Black powder jettison of payload occurs during deployment bay main recovery phase. The black powder deployment jettisons the payload from its stowed flight position (Figure 103) to its initialization position (Figure 104).

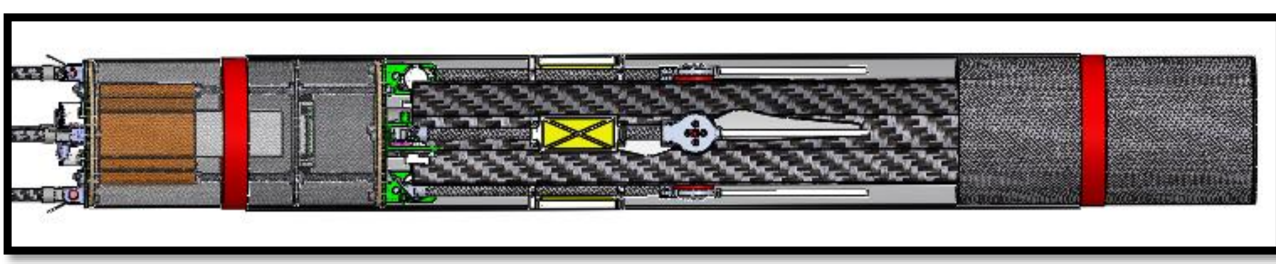


Figure 103: Payload in its stowed position in the deployment bay.

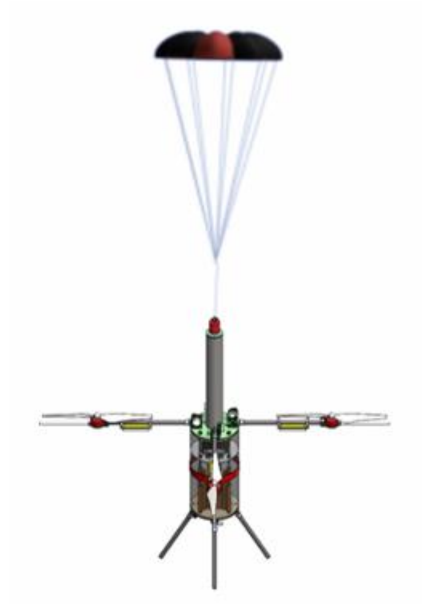


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Figure 104: Payload in the initialization position connected to the deployment parachute.

The black powder charge is ignited in the cavity between the deployment bay tube and the RRS tube. The black powder charge severs the shear pins that hold the deployment bay airframe and the payload coupler together. The black powder ignition also sends the payload out of the deployment bay. The deployment bay tube acts as a linear bearing, guiding the payload down and out of the deployment bay. Another function of the deployment tube is to encapsulate all of the black powder and dog barf in its cavity. The black powder discharge effects must be encapsulated in this tube to protect the MRS electronics.

7.9.3.2 *ARRD Cutaway*

After the black powder jettison, the payload falls in the deployment phase underneath the deployment parachute. After the payload has initializes its flight systems and the RSO gives the command that the payload can be released, the ARRD will cutaway the payload from the deployment parachute and begin its flight. The ARRD is tensile tested up to two thousand pounds, which makes it an acceptable mechanism for this cutaway.

7.10 Testing

All subsystems of the payload will be rigorously tested to validate mission requirements. A comprehensive schedule of verification of design tests is shown below.

- Black powder deployment test
- Test landing stability from full scale launches

7.10.1 MRS MASTER TEST PLAN

7.10.1.1 *Introduction*

This is the master test document for all MRS tests. The following are the majors test campaigns which will be accomplished through this document: Deployment, initialization, and autonomous flight.

7.10.1.2 *Subsystems Testing*

The following sections describe the components and processes in the MRS that must be tested.

7.10.1.3 *Items to be Tested*

- Deployment
    - Arm deployment
    - Leg deployment
    - Limit switch verification of arm and leg deployment
    - System functionality with DSS
- Initialization
    - Initialization time
    - Actuation of propulsion system after deployment
    - Testing current draw on propulsion systems.
- Autonomous flight
    - RC Controlled Flight Test
    - Autonomous Flight Test
    - Full Scale Competition Flight Test
    - Propulsion system thrust
    - Flight computer functionality
    - Navigation to GPS points
    - GPS initialization and accuracy
    - Vertical landing

7.10.1.4 *Safety Notes*

Safety glasses must be worn when operating around the MRS propulsion system. Pinching points on the arm and leg deployment clevises must be treated with caution. It is recommended that gloves are word during testing of the arm and leg deployment. Proper flight procedures will be followed with respect to the FAA's guidelines on multirotors.

7.10.2 RRS Master Test Document

7.10.2.1 *Introduction*

This is the master test document for the Redundant recovery system. The RSS's responsibility is landing the payload safely under in the event of a flight anomaly.

7.10.2.2 *Subsystems Testing*

The following sections describe the components and processes in each the RRS that must be tested. Deployment below refers to the deployment of the redundant parachute.

7.10.2.3 *Items to be Tested*

- Deployment during flight
- Deployment logic verification
  - Stability logic
  - Max kinetic energy logic

7.10.2.4 *Safety Notes*

Deployment of the redundant parachute is achieved through black powder charges. Appropriate safety precautions surrounding black powder charges must be followed. For the flight deployment, launch safety must be followed with an experimental recovery system.

7.10.3 DCS Master Test Plan

7.10.3.1 *Introduction*

This is the master test document for all DCS system tests. The DCS is responsible for deploying the payload from the main airframe and performing the cutaway as the payload takes flight.

7.10.3.2 *Subsystems Testing*

The following sections describe the tests that will be performed for the LLS. Deployment below refers to the payload leaving the deployment bay. Separation refers to the DCS releasing the payload for flight.

7.10.3.3 *Items to be Tested*

- Ground black powder deployment of payload
- Ground separation of payload
- Deployment parachute shock force resistance
- Flight black powder deployment of payload
- Flight separation of payload

7.10.3.4 *Safety Notes*

Appropriate safety precautions surrounding black powder charges must be followed. Flight safety procedures must be followed in all flight tests.

7.10.4 PSS Master Test Document

7.10.4.1 *Introduction*

This is the master test document for the payload structures systems. These tests are derived from the PSS requirements. These hierarchical tests are the tests that all future PSS tests will stem from.

7.10.4.2 *Subsystems Testing*

The following sections describe the tests that will be performed on the PSS.

7.10.4.3 *Items to be Tested*

- Bulkplate resistance to parachute opening force
- Flight testing of electronics housing
- Max thrust payload structural testing
- Mitigating black powder clouding the camera

7.10.4.4 *Safety Notes*

Appropriate safety precautions surrounding black powder charges must be followed. Safety glasses must be worn when operating around the MRS propulsion system for flight tests.

7.10.5 TDS MASTER TEST DOCUMENT

7.10.5.1 *Introduction*

This is the master test document for all TDS Tests. Tests are to be formulated and conducted to test elements and implementation of the TDS. Each test will serve as a simulation of working conditions to create accurate as possible results and operating conditions so that members of the TDS sub team can draw conclusions and make assumptions based on the results.

7.10.5.2 *Subsystem Testing*

The following sections describe information regarding what is being tested and requirements that are to be met to classify the test as passed or failed.

7.10.5.3 *Items to be Tested*

- Flight Deployment

  Camera system will be tested and test footage will be recorded to use for later ground tests. A full flight test will feature many variables, but will provide an excellent test and simulation to base continued development and evaluations on. Behavior related to descent vehicle deployment and navigation will be specifically evaluated.

- Subscale Drone Deployment

  A drone test will provide accurate, real world data to be used for analysis and target detection rates.

- Ground Testing

  Initial testing that will provide a framework for development and implementation. Ground testing is used an umbrella term in the sense that numerous tests will be completed to test individual aspects of the TDS such as maximum range and detection window.

- Angular Velocity and Acceleration

  Angular velocity and acceleration testing will be conducted to determine worst case and operational standards for the descent system. Footage generated from these tests will be utilized to farther development and establish guidelines for the TDS.

7.10.5.4 *Safety Notes*

Potential for TDS to cause Raspberry Pi to malfunction; Raspberry Pi malfunctions may lead to malfunctions with flight computer. TDS may provide incorrect headings to flight computer. TDS could potentially identify incorrect targets

7.11 Payload Timeline Overview

Below is the payload timeline overview with the major timeline project categories and events of the project.

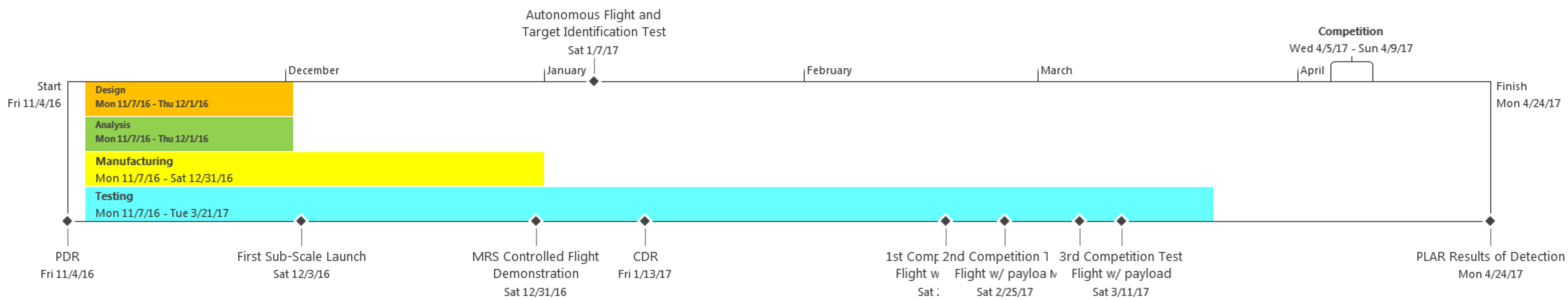


Figure 105: Payload Timeline Overview.

7.11.1 <u>Project Outline</u>

The four major project timelines outlined in this timeline are the Design, Analysis, Manufacturing and Testing categories.

Design and Analysis

The Design and Analysis phases of the experimental payload are planned to last from November 7 through December 1, 2016. During this time, detailed designs of the major components of the payload subsystems will be accomplished along with appropriate analysis to substantiate the success of the finalized designs.

Manufacturing

The Manufacturing phase of the experimental payload is planned to last from November 7 to December 31, 2016. During this time, the payload propulsion, landing, and electrical components will be prototyped. After prototypes have been tested and final designs have been approved, the final payload will be manufactured.

Testing

The Testing phase of the experimental payload is planned to last from November 7 to March 31, 2016. During this time, all mechanical electronics components will be tested through nominal and off nominal scenarios. Due to the complexity of the overall system, reliability proven through testing is vital to the mission success of the payload. The major testing milestones are the manually controlled flight demonstration, the autonomous flight and target identification test, and the fully integrated full scale test flight demonstrating mission duration. The manually controlled flight demonstration and autonomous flight test with target identification test are scheduled to be accomplished December 31 and January 7 respectively prior to CDR as a proof of design.

7.12 Safety

All primary payload subsystem were analyzed through the team Safety Risk Assessment Matrix. Below are summaries of potential risks of each subsystem.

Payload Deployment and Cutaway System Risk Assessment
The hazards outlined in this section will discuss the risks associated with the deployment of the payload from the vehicle. The payload deployment interfaces with multiple systems, making it prone to hazards. This can be found in Table 93.

Payload Multirotor Recovery System Risk Assessment
The hazards outlined in this section discuss the risks associated with flight of the payload through the MRS. The MRS will navigate the payload through multiple aerial maneuvers making prone to environmental hazards along with hazards related to multirotor flight. This can be found in Table 94.

Payload Redundant Recovery System Risk Assessment
The hazards outlined in this risk assessment is associated with the RRS that monitors the state of the payload during the initial deployment from the vehicle and during the autonomous flight. This assessment strictly deals with the electrical components that monitor a pre-determined set of criteria that will deploy a backup parachute if any of the criteria are met. Please refer to the recovery risk assessment for the deployment of the backup parachute. This can be found in Table 95.

Payload Landing Leg System Risk Assessment
The hazards outlined in this risk assessment is associated with the LLS which absorbs impact loads and provides stability during the upright landing of the payload. The LLS stows into the recovery bay and deploys during the main separation event of the launch vehicle making this system susceptible to recovery interference hazards.

Payload Deployment and Cutaway System risk assessment matrix.						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity**	**Probability**	**Risk Level**	**Mitigation**

Payload fails to separate from launch vehicle	1.Poor tolerance of concentricity between payload body and deployment tube during manufacturing 2. Stratologgers fail to ignite black powder charge. 3.Shear pins fail to shear	In event payload bay doesn't separate from deployment bay, both exceed kinetic energy requirement	1	4	Moderate	Custom jigs will be designed to guarantee the concentricity between the payload coupler and the deployment tube. The fitment of the payload into the deployment bay will be inspected and the deployment will be simulated in order to mitigate risk.
Black powder separation charge damaging flight hardware	Heat from charge could burn through wire insulation shorting out or severing circuit connections.	Motor failure, or failure to detect arm deployment	1	4	Moderate	A deployment tube will be implemented to contain and isolate the black powder separation charge from reaching flight electronics. Black powder testing will also be conducted to ensure separation charges do not effect flight hardware.
Recovery system tangles with arms	1. Arms don't deploy properly/fast enough resulting in tangling with recovery components.	Payload falls without deployed parachute exceeding kinetic energy requirement. Resulting in possible injury to personal and spectators.	1	5	Moderate	A deployment tube will be added to separate recovery components from payload arms. Multiple tests will be conducted to ensure arms consistently deploy.

Table 93: Payload Deployment and Cutaway System risk assessment matrix.

Payload Multirotor Recovery Risk Assessment						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity**	**Probability**	**Risk Level**	**Mitigation**

MRS loses control/ power	1. Disconnected electrical connection, 2. Poor weather conditions	1.Loss of thrust will cause payload to lose control. Could result in injury to spectators. 2. Weather anomalies cause payload to lose control	1	4	Moderate	1.At least two full scale test flights will demonstrate the capability of the MRS's electrical systems to maintain control. 2.RSO will check weather conditions just before cut away and give the "go-ahead."
Payload Arms do not deploy during separation	1.Torsion spring doesn't provide sufficient torque to rotate the arms down 2.Lock pin does not seat into the arm. 3. Torque damper produces too much resisting torque	1.Partially deployed payload arms can interfere with recovery systems. 2.Payload could fall resulting in spectator injury.	2	3	Moderate	Limit switches will be integrated into arm pivot to determine if arm has deployed. These switches communicate with flight computer to ensure MRS will not cut away unless arms are fully deployed
Spinning rotors causing injury	Accidentally hitting the start button during handling of the MRS.	Hands or body parts in path of spinning rotor could result in severe injury.	3	2	Moderate	1.MRS will give initialization beeps before motors spin up to warn nearby personal 2.Handling procedures will be developed for the MRS to mitigate risk.

Component falls off MRS during flight	Flight vibrations cause fasteners to loosen	MRS components could fall and injure spectators	2	4	Moderate	1.Lock-tite will be applied to every threaded fastener to prevent loosening and mitigate risk. 2.Multiple flight tests followed by an inspection of threaded connections will guarantee that fasteners on MRS can withstand flight vibrations.

Table 94: Payload Multirotor Recovery System risk assessment matrix.

Redundant Recovery Risk Assessment						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity**	**Probability**	**Risk Level**	**Mitigation**
RRS electronics fail to signal RRS deployment.	Flight electronics interfere with RRS electronics	Kinetic energy requirement is exceeded	1	4	Moderate	Flight electronics and RRS electronics will have separate sleds and power supplies.
Black powder/ E-match failure	Severed wire connection during launch	Payload could fall resulting in spectator injury.	1	3	Low	End to end check will be accomplished before every flight.
Propulsion system interferes with RRS deployment	Layout of payload systems results in RRS and propulsion system interference.	RRS fails to deploy in the event of a flight anomaly.	1	4	Moderate	RRS deployment tube designed to offset deployment location of parachute away from rotors.

Table 95: Redundant Recovery System risk assessment matrix.

Landing Leg System Risk Mitigation						
Payload Leg breaks on impact	Descending too quickly Motor/ rotor failure occurs on descent of payload.	Damage to payload and fragments projecting outward. Injury to personal or spectators from fragmented pieces.	2	3	Moderate	Through testing the team can validate the the landing procedure and produce a lower risk level.
Payload tipping over after landing occurs	1. Weather related due to wind. 2. Propellers do not shut off properly and result in potential fragmentation when tipping over occurs.	1. Payload remains on its side failing the upright landing challenge. 2. Fragmentation of propellers occur.	3	3	Low	1.RSO will determine if weather is safe for cut away. 2.Continuous testing of the landing procedure to ensure a successful upright landing.
Payload lands on launch stand power supplies/other launch vehicles	1. Avoidance controls misinterpret a launch stand from a safe landing area. 2. Power shut off due to an electronic/coding failure	1. Explosion from on board batteries/launch stand batteries. 2. Damage to other launch vehicles and components. 3. Severe injury to personal or spectators.	1	4	Moderate	Through a multitude of ground avoidance tests, the payload will learn recognition faster. Implementation of redundant recovery can also lower the risk level.

8 Educational Engagement

Throughout the course of the past four years, the University of Louisville River City Rocketry Team has managed to reach out to over 5,000 students and adults in the local community. The team's outreach gives back to the state of Kentucky by teaching the youth about engineering, math, technology, logical thinking, and of course rocketry. River City Rocketry continues to maintain relationship built with organizations in the community as well as developing new relationships through paths like our request an event page on our website. The focus is never on how many people can be reached, but the quality of education that can be brought to each and every individual.

Figure 106: Denny and Ben building paper rockets at Boyce College.

8.1 Classroom Curriculum

The University of Louisville River City Rocketry Team has developed a variety of programs that are to be incorporated in this year's outreach program. Included is a list of the different activities in which the team has participated in the past and will continue to do this year.

Raytheon MathMovesU

Over the past couple years River City Rocketry has participated in MathMovesU, which has been sponsored by Raytheon Missile Systems and the University of Louisville J.B. Speed School. This event takes place at the Engineering School's campus where the team utilized Duthie Hall for and the intramural fields. The entire day Saturday breaks down into the following steps outlined below:

Step 1: Introduction from RCR and Raytheon

After students arrive and check in a brief presentation of River City Rocketry of what we do and how our rockets compare to the Estes rockets, as shown below in Figure 107. Following the team's presentation, Raytheon Missile systems will give a brief presentation on what they do on a daily basis and how STEM based classes are beneficial to building rockets in the future.

Figure 107: Kevin showing the comparison between Estes rockets and RCR's rockets.

Step 2: Construction and lesson of Estes kits:

Students get to learn the basics of how rockets operate in flight as they construct their Estes kit parallel with the lesson. This enables the students to learn what each component is as well as why that component is used. During the construction process we have volunteers located at specific building stations while a singular team member walks the class through the construction process as shown below in Figure 108.

Figure 108: River City Rocketry showing the construction process of the Estes kit.

Step 3: Safety briefing and launching of Estes kits:
After construction is finished a safety briefing is performed for the students as well as the adults to ensure that all personal, volunteers, students, and parents are safe throughout the event. While rockets are being launched everyone that is within the bounds of the intermural field is required to wear safety glasses and stay behind a pre-defined white line.

As each rocket is set up on the launch stand, one student per stand at a time, the student will take control of the launcher and wait for the Range Safety Officer (RSO) to allow the safety keys to be inserted. A countdown from 5 starts and at 0 the students can igniter there rockets into the sky as shown below in Figure 109.

Figure 109: Launching of Estes kits during MathMovesU at J.B. Speed School.

The steps that are used in the MathMovesU event by River City Rocketry are used at all major rocketry building outreach events.

STEM night

This is the first year River City Rocketry will be participating in STEM night at Farmer Elementary. The team will have a table on display of our past rockets as well as have an interactive section that will allow students to play and build bristle bots as shown below in Figure 110.

Figure 110: BristOnele bot kit for interactive outreach events.

The team will use bristle bots for a one night outreach event where hundreds of students are coming to and from the teams table. These are very useful for when the team has an average of 30 seconds per student to interact with them and allows for two different options of interactions. The options of interaction is outlined below:

Option 1

The students will be able to take pre-made bristle bots and race them against fellow classmates to test their reaction time and understanding of how the bots operate. This method allows for students to think in the mind set of "reverse engineering" by optimizing the balancing of the bristle bot. This option is also less time consuming and allow for students to experience multiple booths.

Option 2:

The students are allowed to build a bristle bot from scratch and learn how a small circuit interface integrates with the mechanical vibrational motor. This relationship between electrical and mechanical components shows the students how two different systems integrate in the real world of engineering.

Lego Mindstorm Programming

Every year, local students work in teams on building and programming Lego Mindstorm robots to complete specific tasks as defined by the FIRST Lego League competition. The team continually plays a role in educating students on these teams in the fundamentals of robot design and programming. The team regularly meets with the students to mentor them throughout the process. The students write programs, perform testing, and continue to tweak the programs until the robot performs the desired task.

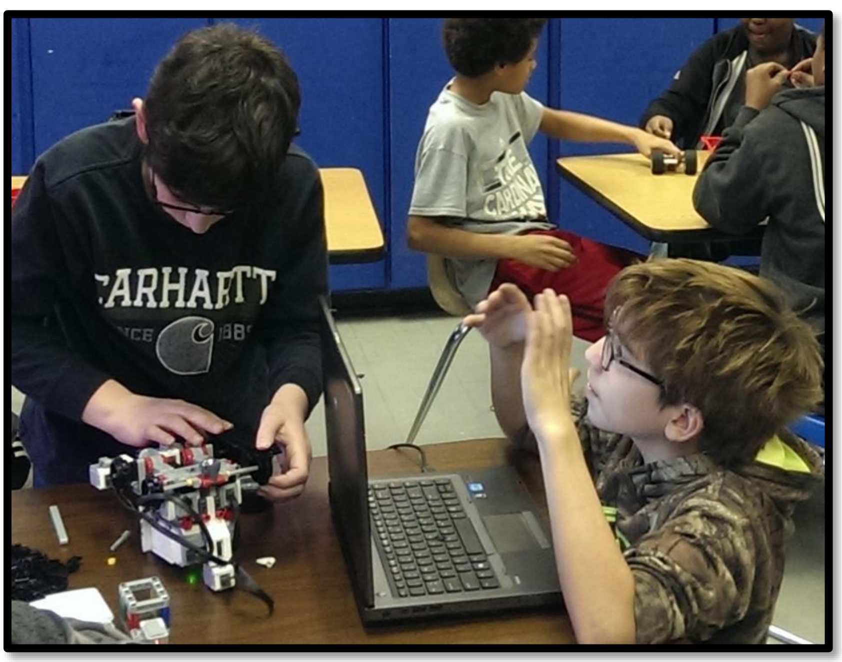


Figure 111: Students discuss designs and modifications to their program.

8.2 Outreach Opportunities

8.2.1 Engineering Exposition (E-Expo)

Since 2006, the J.B. Speed School of Engineering Student Council has hosted the largest student-run event on the University of Louisville's campus called Engineering Exposition. The event is geared towards celebrating strides in engineering as well as getting the local youth interested in the field. During the event, the professional engineering societies on UofL's campus set up educational activities and scientific demonstrations for the elementary and middle school students to participate in.

The University of Louisville River City Rocketry Team will host its sixth annual water bottle rocket competition for middle school students. Teams from local middle schools can participate in teams of up to three students to design and build their own water bottle rockets out of two liter bottles and other allowable materials. Workshops will be held with schools interested to teach the students about the components of a rocket and aerodynamics in preparation for the competition. The students will get to show off their rockets at the E-Expo event throughout the day and will conclude the day with the competition. Teams will compete for awards in highest altitude, best constructed rocket, and landing closest to the launch pad. This event has been a huge success in the past and many schools have voice interest in continuing their involvement so we are looking for our best turn out yet this year.

Figure 112: Denny building rockets with students at E-Expo 2016.

In addition to the water rocket competition, the team will host a paper rocket station for people of all ages. This has been the most popular station at the exposition in the past and are looking to continue to build up that reputation.

8.2.2 Boy Scouts and Cub Scouts:

In the past, the University of Louisville River City Rocketry Team has worked with local Boy Scout and Cub Scout troops to assist the earning of the Space Exploration merit badge. The team has assisted in developing a program that meets the requirements to earn the merit badge. The scouts get to learn about the history of space, current space endeavors, and build and launch an Estes rocket. The team has plans to continue to work with these groups throughout the year.

While cub scouts are not eligible to earn their merit badge, we still enjoy getting to teach them about rocketry. We have had the pleasure of working with scout troops in educating the kids about the fundamentals of rocketry, while also giving them the opportunity to build and launch their own paper rockets. We plan to continue to build our relationships with these troops this year.

8.2.3 Big Brothers Big Sisters Partnership:

Big Brothers Big Sisters is active in the Louisville community and is constantly striving to bring opportunities to underprivileged kids. The team recently put on a program at The Big Carnival for kids that had not yet been paired with a mentor through the program. This is the second year in a row that the team has participated in this event. Both years, this event has been a huge success in brining STEM to under privileged kids.

Figure 113: Zak assisting in the construction of a paper rocket at The Big Carnival.

"Kevin and UL Rocket Team,

On behalf of The Big Leadership Team of Big Brothers Big Sisters of Kentuckiana, we want to express our gratitude for your support of The Big Carnival. Last year the team was definitely the favorite and this year you all did not disappoint! All of the children enjoyed designing and launching their rockets! Your support of The Big Carnival means so much to us but even more to the waitlist children who attended with their families.

Thank you from The Big Leadership Team & Big Brothers Big Sisters!"

8.2.4 Louisville Mini-Maker Faire

Annually, Louisville hosts a Mini-Maker Faire. The team always participates by taking the previous year's project out to show off to anyone attending the event. A mixture of people attend this event ranging from small children to adults with experience in the field. This gives the team am opportunity to talk to the community about our project and what it does. This is an informal setting which is perfect for interacting with visitors and answering their questions about the project, what the team does, and about rocketry in general.

8.2.5 Kentucky Science Center

During the 2015-2016 season, the team first came in contact with Andrew Spence, manager of public programs and events, which assisted in several events in the Louisville area. For this season the team will participate in the Youth Science Summit, Advanced Manufacturing, and Engineers week at Kentucky Science Center. The team will be able to reach out to hundreds of young rocketeers and teach them about rocketry, engineering, and skills needed to succeed as an engineer.

8.2.6 FIRST Lego League Competition

The team initially become involved with the FIRST Lego League Competition during the 2014-2015 season. This was such a successful event that River City Rocketry has been invited back last season and is looking forward to participate for a third year in a row. The FIRST Lego League competition is an all-day event and the team performs several activities throughout the day. Throughout the majority of the day, the team has a display set up so that when students are in between events, the team can talk to them about the previous year's project. This is a good way to show the students how programming can be applied into something beyond their Lego Mindstorm robots.

During the competition period, team members assist in the judging process. The team helps to judge a portion of the competition called core values. In this, students are tested in a variety of ways to see how well they work together as a team and how dedicated they are to their project. Students are given a variety of tasks to complete as a team and are then questioned on their methodology and teamwork. This is important to show the students the importance of being able to work together as a team and qualities of a successful team.

At the end of the day, while all of the teams are waiting for the final results of the competition, River City Rocketry representatives give a presentation to all of the students, parents, and educators present. Here the team is able to talk about what River City Rocketry does as a team and relate that to the students' projects. This is an opportunity to share how the team designs, manufactures, and test just the same as the competitors. It is important that the students realize that the skills learned by participating FIRST Lego League competition can be applied to the real world and that it aligns with STEM career paths.

Figure 114: Emily and Kevin presenting at FIRST Lego League Regional Competition.

8.2.7 Louisville Astronomical Society

The team has been invited to be the guest speaker at a Louisville Astronomical Society (LAS) meeting. This event is for both those that are members of LAS as well as the public. This is an opportunity for the team so share what was accomplished during the 2015-2016 season as well as what the team is looking to do during the 2016-2017 season. The setting will allow for technical conversations about the project.

8.2.8 Executive Board of Advisors

The team was invited by the Dean of the University of Louisville J.B. Speed School of Engineering to present to his board of advisors. The advisors included CEO's and management from various companies from the region. This presentation consisted of a technical review of the previous year's design, what the team is about, the tasks that the team are required to complete, and the successes of the season. This provided the team excellent exposure to a variety of companies in the region

9 Project Plan

9.1 Timeline

River City Rocketry has developed an overview schedule that outlines the basic schedule for the entire team. As indicated below in Figure 115 the team has outlined the overall NASA Student Launch Schedule as milestones to illustrate where the team should be in reference to NASA's deadlines. Everything below NASA's milestones are the team's schedule which is split up into the sub-systems of the team. Each sub-system is broken up into general topics and monitored on progress with the light blue line which indicates the current progress that is required for November 4th, 2016.

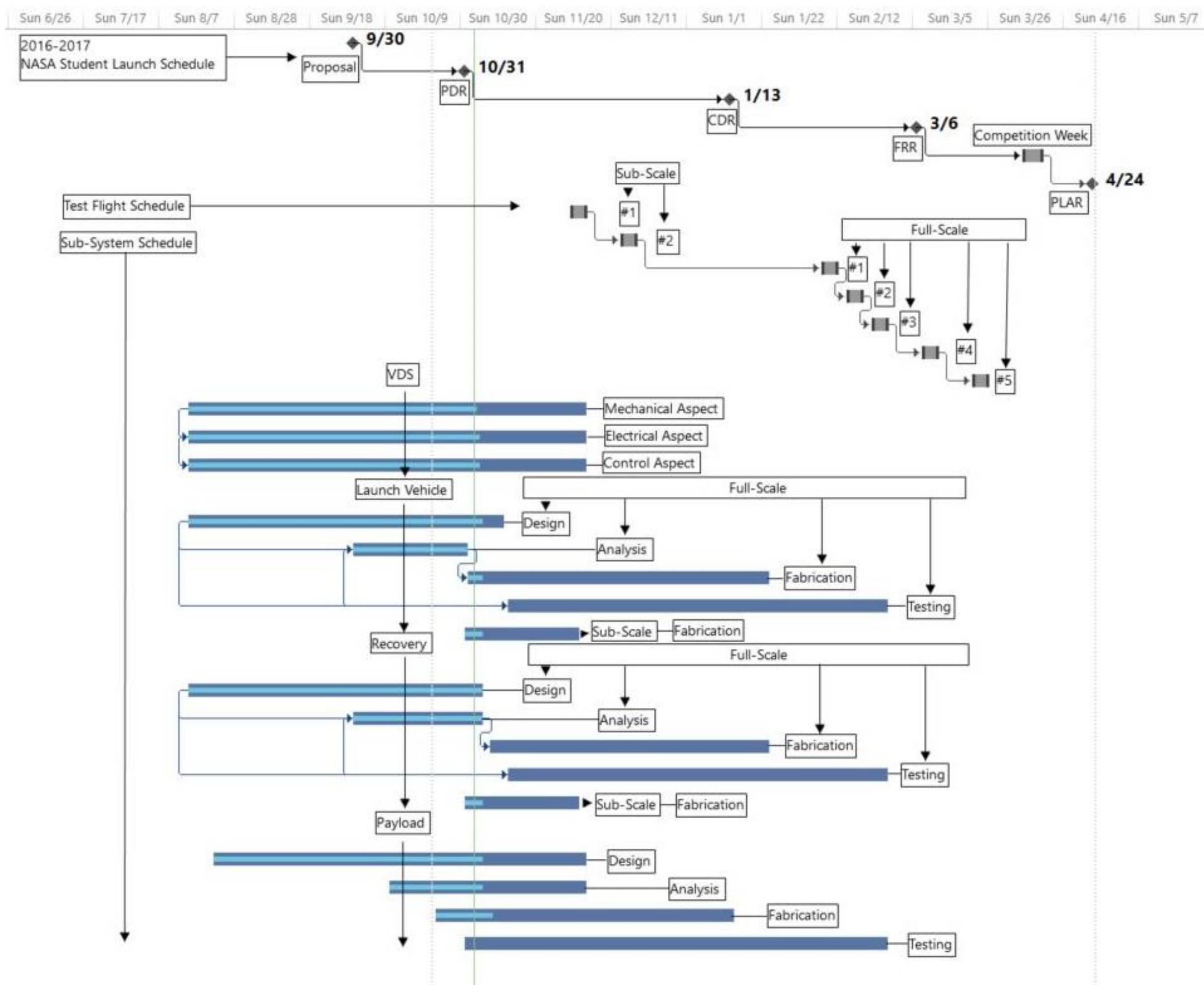


Figure 115: 2016-2017 River City Rocketry overview schedule.

9.2 Comprehensive Budget

Variable Drag System Budget			
Description	**Quantity**	**Per Unit Cost**	**Total Cost**
Raspberry pi	2	$35.00	$70.00
1/4" Thick 6061 T-6 Aluminum Drag Flaps	3	$7.23	$21.69
1/4" Thick 12" x 48" Delrin	1	$85.22	$85.22
1/8" Dowel Pins 3/4" Length (pkg of 25)	2	$10.63	$21.26
M3-16 mm Socket Head Cap Screws (pkg of 50)	1	$10.20	$10.20
AndyMark DC Motor	3	$28.00	$84.00
KRD-19852 Teensy 3.6	2	$29.95	$59.90
Adafruit 9-dof absolute orientation IMU Fusion Breakout BNo055	2	$39.95	$79.90
Raspberry Pi 3 - Model B - ARMv8 with 1G RAM	2	$39.95	$79.90
SD/microSD 8Gb	2	$9.95	$19.90
Short Feather Male Headers - 12-pin and 16-pin Male Header Set	20	$0.50	$10.00
gps	1	$39.95	$39.95
banana to alligator clip cables	1	$3.95	$3.95
banana to IC hook cables	1	$4.95	$4.95
HP Pavilion 21.5-inch LED HDMI VGA Monitor (used) (black)	2	$88.49	$176.98
Neiko® 01924A Self-Adjusting 3-in-1 Automatic Wire Stripper, Cutter and Crimping Tool	1	$14.99	$14.99
PanaVise 381 Vacuum Base PanaVise	1	$66.99	$66.99
Omron SS-5GL Limit Switch	2	$1.80	$3.60
Hall effect encoder cable	1	$5.00	$5.00
DC/DC converter breakout	1	$29.95	$29.95
		Overall Cost	**$888.33**

Full Scale Vehicle Budget			
Description	**Quantity**	**Per Unit Cost**	**Total Cost**
6K Carbon Ribbon Toe, 4.65lbs	2	$279.00	$558.00
Fiberglass Toe, 15lbs	1	$245.00	$245.00
1/8" Thick 24" x 36" Fiberglass	4	$35.78	$143.12
6" Plywood Bulkplate - 1/2" Thick (Coupler)	5	$5.90	$29.50
6" Plywood Bulkplate - 1/2" Thick (Airframe)	5	$5.90	$29.50
6" 6061 T-6 Aluminum Centering Rings -1/4" Thick	4	$5.17	$20.68
Aerotech L1420R-P	6	$249.99	$1,499.94
75mm 5120 motor casing	1	$550.00	$550.00
1/4"-20 x 4' Threaded Rod (Aluminum)	3	$4.46	$13.38
1/4"-20 Hex Nuts (Aluminum) (pkg of 100)	1	$6.74	$6.74
4-40 Black Nylon Shear Pins (pkg of 100)	1	$5.42	$5.42
3/8"-16 for 2.5" OD Black-Oxide (18-8 SS) (pkg of 25)	5	$1.55	$7.75
1/4" Flat Washer (Alumium) (pkg of 100)	1	$6.64	$6.64
3/8" Flat Washer Black-Oxide (18-8 SS) (pkg of 100)	1	$8.49	$8.49
6" x 12" Carbon Fiber Coupler	4	$110.00	$440.00
Featherweight Screw Switches	4	$5.00	$20.00
Professional Paint Job for Competition	1	$250.00	$250.00
		Overall Cost	**$3,834.16**

Subscale Vehicle Budget			
Description	**Quantity**	**Per Unit Cost**	**Total Cost**
Fiberglass Tow, 15lbs	1	$245.00	$245.00
54mm Motor Mount Tube	1	$15.50	$15.50
1/8" Thick 24" x 36" Fiberglass	3	$35.78	$107.34
3" Plywood Bulkplate - 3/16" Thick (Coupler)	5	$1.64	$8.20
3" Plywood Bulkplate - 3/16" Thick (Airframe	5	$1.66	$8.30
2" Plywood Centering Rings - 3/16" Thick	4	$1.62	$6.48
1/4"-20 x 4' Threaded Rod (Aluminum)	2	$4.46	$8.92
1/4"-20 Hex Nuts Black-Oxide (pkg of 50)	2	$4.53	$9.06
1/4"-20 for 1.5" ID Black -Oxide U-Bolt (Stee	5	$1.14	$5.70
4-40 Black Nylon Shear Pins (pkg of 100)	1	$5.42	$5.42
1/4"-20 Flat Washer (Aluminum) (pkg of 100)	1	$6.64	$6.64
PerfectFlight Stratologger	4	$54.95	$219.80
Electric Matches	15	$1.25	$18.75
4FA Powder (1lb)	1	$29.94	$29.94
9V Duracell Batteries (x4)	3	$12.73	$38.19
		Overall Cost	**$733.24**

Recovery Budget			
Description	**Quantity**	**Per Unit Cost**	**Total Cost**
PerfectFlite Stratologgers	4	$54.95	$219.80
1" x 25' TUNSC Nylon Shock Cord	4	$19.95	$79.80
18" X 18" FCP Nomac	2	$10.95	$21.90
1/4"-20 Eyebolts	2	$9.71	$19.42
1/4"-20 U-Bolt	1	$0.75	$0.75
5/16"-18 U-Bolt	1	$1.04	$1.04
Flame Resistant Fabric 54"	3	$10.99	$32.97
64" x 1yd Ripstop Fabric	75	$9.00	$675.00
Type II Nylon Shroud Line (100 Yards)	2	$31.50	$63.00
1/4" Quick Links	3	$3.10	$9.30
9/32" Quick links	5	$3.10	$15.50
Electric Matches	50	$1.25	$62.50
11/16" Vials (pkg of 36)	1	$14.47	$14.47
4FA Black Powder (1lb)	1	$24.20	$24.20
9V Duracell Batteries (x4)	3	$12.73	$38.19
Garmin Astro GPS Unit	2	$189.99	$379.98
1/4"-20 Hex Nuts (pkg of 50)	1	$11.46	$11.46
1/4"-20 Washers (pkg of 100)	1	$8.25	$8.25
3" Plywood Bulkplate - 1/4" thick (Airframe)	2	$1.99	$3.98
1/8" Thick 24" x 36" Fiberglass	1	$42.49	$42.49
Nylon Thread	1	$20.99	$20.99
		Overall Cost	**$1,744.99**

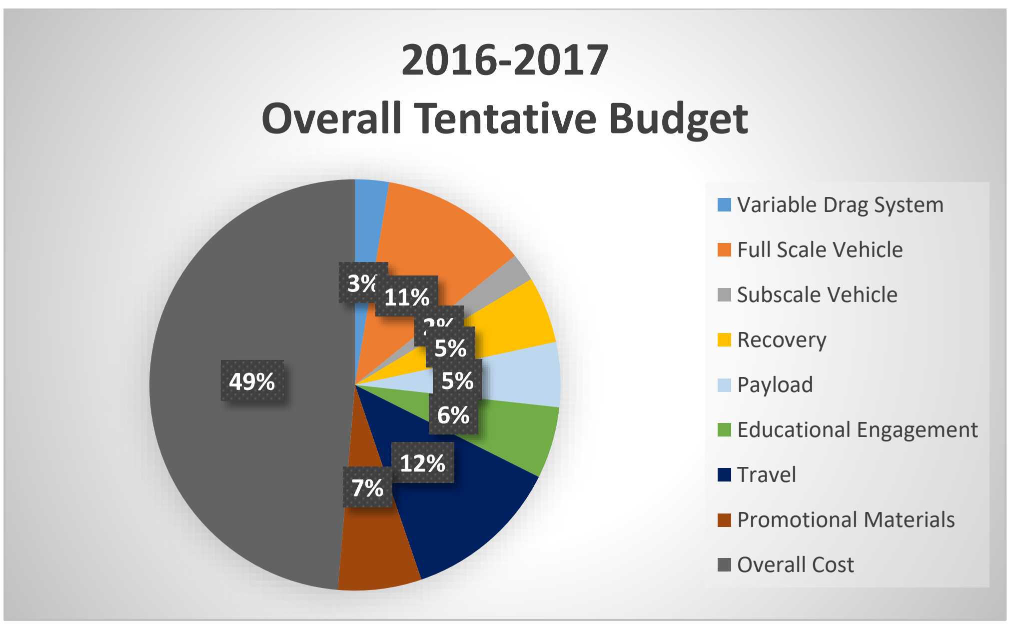
Payload Budget			
Description	**Quantity**	**Per Unit Cost**	**Total Cost**
DJI E800 Propulsion System	1	$469.00	$469.00
ESC's	0	$10.00	$0.00
Tattu 8000mAh 22.2V Lipo Battery Pack	1	$99.39	$99.39
Rasberry pi	2	$35.00	$70.00
Rasberry pi cam	1	$20.00	$20.00
6061-T6 Aluminum 1 -1/2" x 2' x 2'	1	$650.00	$650.00
Carbon Fiber Woven Sheet	0	$58.00	$0.00
Flight computer	1	$204.00	$204.00
GPS sensor module	0	$100.00	$0.00
fastening hardware	1	$50.00	$50.00
Torsion spring	8	$2.00	$16.00
Helical compression spring	4	$2.00	$8.00
Carbon Fiber Tube .375x.503x60inches	2	$54.99	$109.98
		Overall Cost	**$1,696.37**

Educational Engagement Budget			
Description	**Quantity**	**Per Unit Cos**	**Total Cost**
Orbit 1" 24V Electronic Valve	3	$12.97	$38.91
7/8" Tire Valve (pkg of 2)	2	$2.09	$4.18
1 NPT Pipe Size Threading Bushing (Brass)	3	$7.70	$23.10
2-1/2" Tube ID x 1/2 Male Pipe Size Barbed Fitting (Br	3	$4.66	$13.98
2-1/2" Male x 1 NPT Female Bushing (PVC)	3	$2.80	$8.40
7/32" to 5/8" Hose Clamp (pkg of 10)	1	$5.87	$5.87
1/4" Wide x 14 Yards Teflon Tape	1	$5.19	$5.19
2 Pipe Size x 4' Length (PVC)	1	$36.94	$36.94
2 Pipe Size Cap (PVC)	3	$0.94	$2.82
Plastic Pipe Cement	1	$12.94	$12.94
3/4 Male Adapter to Female Slip (PVC)	6	$0.30	$1.80
3/4 Pipe End male x 1/2 Female Bushing (PVC) 3	3	$0.36	$1.08
3/4 Pipe Size x 5' Length (PVC)	1	$3.25	$3.25
1/2 Pipe Size x 4' Length (PVC)	1	$9.08	$9.08
2 Pipe End Male x 3/4 Female Slip Bushing (PVC)	3	$1.57	$4.71
6mm, SPDT-NO Push Button Switch	3	$6.18	$18.54
15" Length Red Nylon Cable Tie (pkg of 25)	1	$6.12	$6.12
9V Battery (pkg of 12)	1	$14.36	$14.36
9V Battery Snap, I-Style	6	$0.68	$4.08
24 GA 25' Stranded Wire (Black)	1	$3.18	$3.18
24 GA 25' Stranded Wire (Red)	1	$3.18	$3.18
Starhwak Model Rocket Kit (pkg of 25)	2	$149.67	$299.34
Estes Tandem Model Rocket Launch set	2	$26.18	$52.36
1/2A3-4T Engine Bulk Pack (pkg of 24)	0	$57.79	$0.00
Scotch Tape (pkg of 3)	40	$4.74	$189.60
BristleBot Kit	50	$19.99	$999.50
Estes B6-4 Engines Bulk Pack	1	$56.93	$56.93
Estes AB-3 Engines Bulk Pack	1	$57.59	$57.59
		Overall Cost	**$1,877.03**

Travel Expenses Budget			
Description	**Quantity**	**Per Unit Cost**	**Total Cost**
Hotel (Competition in Huntsville, AL) [unit is per week, quantity per room]	6	$450.00	$2,700.00
Hotel (Testing in Manchester, Tennessee, Music City Missiles Club) [unit is for max 2 days	3	$186.00	$558.00
Gas per gallon (Competition in Huntsville, AL)	200	$2.39	$478.00
Gas per gallon (For all out of town testing)	160	$2.39	$382.40
		Overall Cost	**$4,118.40**

Promotional Materials Budget			
Description	**Quantity**	**Per Unit Cost**	**Total Cost**
Shirts	20	$20.00	$400.00
Polos	40	$40.00	$1,600.00
Stickers	750	$0.25	$187.50
		Overall Cost	**$2,187.50**

Overall Tentative Budget	
Budget	**Total Cost**
Variable Drag System	$888.33
Full Scale Vehicle	$3,834.16
Subscale Vehicle	$733.24
Recovery	$1,744.99
Payload	$1,696.37
Educational Engagement	$1,877.03
Travel	$4,118.40
Promotional Materials	$2,187.50
Overall Cost	$16,191.69



9.3 Funding

The team utilizes the innovation and success of River City Rocketry to propose funding to multiple commercial companies and grants throughout the year. Each year the team puts effort to reach a remainder balance of $10,000 for next year's team, the breakdown of how the team will sustain its budget is outlined below in Table 96, the team's history of results in the competition, summary of accomplishments performed in the past season, and a detailed budget outlining the expenses of the past season. The sponsorship packet can be found on our website "http://www.rivercityrocketry.org"www.rivercityrocketry.org and is consistently updated from year to year.

Sustainable Budget					
Inflow					
Donor	**Description of Donation**	**Date Submitted**	**Date Received**	**Amount Requested**	**Accepted**
J.B. Speed School	The University of Louisville J.B. Speed School donates based off presentation of materials and amount requested/needed by the organization.	Thursday, September 22, 2016	Friday, October 28, 2016	$5,000.00	Y
Raytheon Missle Systems	Assistance in outreach event MathMovesU.	Thursday, October 13, 2016	Thursday, October 27, 2016	$1,000.00	Y
SpaceX	Grant for university teams not only NASA Student Launch but a multitude of competitions. They have no specific ceiling on the amount to request.	Tuesday, November 1, 2016	TBD	$10,000.00	TBD
2015-2016 RCR Remaining Balance	Remaining balance of the teams expenditures from the 2015-2016 NASA Student Launch Competition	N/A	N/A	$23,799.00	Y
			Overall Income	**$29,799.00**	
Outlfow					
			Expected Team Expenses	**$16,191.69**	
			End of the Season Expected Total	**$13,607.31**	

Table 96: 2016-2017 River City Rocketry sustainable budget.

By striving to reach and go beyond our $10,000 goal, the team is able to perform research on potential payloads that would otherwise not be in the competition if we were to start development the day the Statement of Work was released. Not only can we perform research over the summer, but the team can make equipment improvements as well that will facilitate manufacturing, design, and overall cost in the long run.

The community has supported River City Rocketry in the past and besides grants or commercial sponsors the following individuals have reached out to the team and continue to do so year after year.

Community Outreach: River City Rocketry has enabled a donate button on www.rivercityrocketry.org to allow anyone contribute to funding this year's team. This is a way for people to make small personal donations in any amount that they feel is necessary.

U of L Today with Mark Hebert: River City Rocketry performed a radio interview with U of L today with Mark Hebert where the discussion of past year's success as well as this year's season tasks took place. The team received an increase in followers not only on our Facebook page but on all sources of social media.

Wave 3 – MathMovesU: The event MathMovesU, which is discussed in further detail in Educational Outreach, brought in Wave 3 News where River City Rocketry got local television coverage over the duration of the event. This promoted the team's educational outreach as well as showed how much community support the team is receiving during this year's season.

WHAS 11 – Mini Maker Faire: River City Rocketry participated in the 2015 and 2016 Louisville Mini Maker Faire. WHAS 11 covered this event, which showcased the team on local television where the team demonstrated last year's Autonomous Ground Support Equipment. The team further grew its support and received constant emails to either join or arrange an outreach event.

Discovery Channel – Daily Planet: On launch day of the 2014-2015 season, River City Rocketry was followed around by Discovery Channel Daily Planet to catch every angle that goes into launch day. The team received international coverage both over the internet as well as broadcasted nationally in Canada.

Louisville Cardinal: The Louisville Cardinal is the independent student newspaper at the University of Louisville. The newspaper is widely read and respected by the students at the university. In years past, River City Rocketry took the opportunity to sit down for interviews with the Louisville Cardinal. This has allowed students from all over the university to see what the team is doing and the progress they have made.

Registered Student Organization: In the Spring of 2012, River City Rocketry became a Registered Student Organization (RSO) at the University of Louisville. Since receiving RSO status, the team has been able to reach out to the Student Senate as well as several of the university's Student Councils to gain support and increase the knowledge of rocketry at UofL. The team has received very positive feedback and was elected "Best New RSO" in its first year as an RSO.

Speed School Student Council: Since the birth of River City Rocketry, Speed School Student Council (SSSC) has supported the team. By maintaining a good relationship with SSSC and attending team building retreats, River City Rocketry is able to receive funding from J.B. Speed School of Engineering.

9.4 Community Support

Throughout the past five years of the team's involvement in NASA Student Launch Projects, the team has developed a strong network within the University of Louisville, local industry, and the local community. Year after year, the team acknowledges that the success the team has seen would not have been possible without the support of the community.

Due to the mandatory co-op program that the University of Louisville's J.B. Speed School of Engineering has, the team has made many connections with different companies. As a result of team members spending a year of their undergraduate career working in the industry, lasting relationships have been formed between companies and the team. This is a huge contribution to the team's growing network. A compiled list of our community supporters and method of support is shown in **Table 97**.

Supporter	Method of Support
Art's Rental Services	Discounted trailer rental.
Big Brothers Big Sisters Louisville	Invite to participate in outreach opportunities.
Bro Ties	Apparel donation.
Darryl Hankes	Team mentor, high power rocketry knowledge and experience, discounted rocketry materials.
Dr. Yongsheng Lian	Team advisor for five years, oversees budget, campaigns for funds, and builds relations within university and industry.

Engineering Garage Manager (Mike Miller)	Machine shop equipment and storage and workshop space.
FirstBuild	Material donation, manufacturing support, equipment time and training.
Gregg Blincoe	Support with manufacturing processes and advice from previous team leadership experience.
Emily Robison	Assist in writing and technical criticism and advice from previous team leadership experience.
Austin Eschner	Provides technical criticism and knowledge in manufacturing challenges.
Jefferson County Public Schools	Invites team to teach students STEM in their classrooms.
Kyle Hord	Provides knowledge and expertise on recovery design and manufacturing.
Lowes	Discounted tooling and materials.
Metal Supermarkets	Discounted metal.
NASA (SL Team)	Critical review of technical package.
Nick Greco	Provides knowledge and expertise on vehicle design and team management.
Speed School Administrative Assistant (Diane Jenne)	Runs team university bank account, orders materials and components, purchases are tax free.
Speed School Communications and Marketing (Kari Donahue)	Helps the team receive exposure, promotes events, organizes press releases.
Speed School Director of Outreach (Gary Rivoli)	Establishes connections with local schools for educational events, financially sponsors outreach.
Dr. Kelly	Generous donor, on the board of trustee's advisors for the University of Louisville, and rocket enthusiast.
Alumni	Supporters of the University of Louisville.

Table 97: RCR community supports.

9.5 Project Sustainability

Since the start of River City Rocketry, the end goal**Error! Reference source not found.**season, the team is always looking for more ways to develop community and financial support to ensure the continued presence in this competition.

Local Exposure

River City Rocketry continues its exposure in a multitude of ways. The most primitive are through the following experiences that occur from year to year.

- Educational outreach events

- Community outreach events
- Local news media
- University press releases

River City Rocketry over the years has received a significant amount of exposure by appearing on WDRB local news, Discover Channel (Canada), NASA TV, the University of Louisville's webpage and in the University of Louisville magazine.

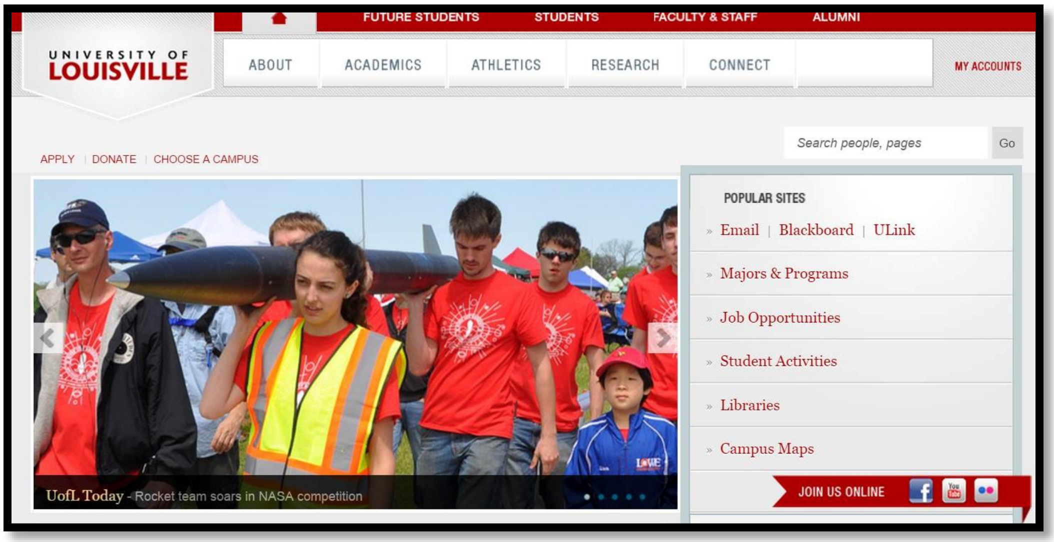


Figure 116: River City Rocketry on the front page of the University of Louisville website.

To further gain additional media exposure locally, the team will develop follow up stories on current team events to continually gain interested media. The team finds that one of the most rewarding methods of increasing exposure is through working with youth. Because of the success of last year, the team plans to cooperate with the Kentucky Science Center in coordinating outreach events for this upcoming season that will hopefully gravitate future members to River City Rocketry. Media coverage and publicity regarding previous years' achievements will likely gain the attention of newly interested participants and further the team's success in the NASA Student Launch competition.

Recruitment and Retention

A secondary form of exposure is to highlight the importance of the rocket project. While local exposure increases future team membership and initial awareness, university exposure explains the importance of the rocket team as well as the excitement that ensues. The team retains members' interest by having a series of interest meetings on top of constant improvement of the team, for example the Variable Drag System (VDS) over the summer. With ongoing projects and periodic launches that may even include their own level one certification flights as shown below in Figure 117, Figure 118, members take great interest in the team and tend to contribute multiple years to the team. To ensure the entire team maintains on the same page bi-weekly meetings will take place where each sub-team lead will present a technical presentation of the progress they have made of a period of time and where they are headed. This assists in presentation practice as well as to mitigate design flaws by having the entire team to tag up.

Figure 117: RCR member Alex Basil getting ready for his level one certification flight.

Figure 118: RCR member Justin Johnson getting ready for his first level one certification flight.

However, no matter how many young, enthusiastic members the team gains, it won’t bode well for the future of the team unless each individual is learning and engaged. The team is looking to do the following in order to help students grow in all aspects of the competition:

- Team members, whether new or old, work together to fulfill any projects or learn the basics of rocketry.
- Students all own a small portion of the project.
- Training on manufacturing techniques.
- Regular targeted training sessions on various aspects of rocketry (ex. Recovery, simulation, electronics, etc.).
- Involved in technical writing – revise with mentor to learn technical writing skills.
- Involved in presentations – improve technical and informal presentation skills.

By getting new members involved in all aspects of the project and working closely with a mentor, they will develop into the next generation of leaders for the team, which is crucial to success in the future. This has proven to be successful as all of the current leadership has been brought in and mentored closely by former and current team members.

Conclusion

River City Rocketry started the 2016-2017 season with several goals in mind. This includes the continued effort to keep setting the standard for safety in the NASA Student Launch Competition, as River City Rocketry has for the past 3 years. In terms of Community Outreach, the team has pledged to engage 2,000+ students in STEM-centered outreach events, encouraging enthusiasm for the rocketry, science, and the larger STEM fields. Some of our events are outlines below:

1. Participating in such events as MathMovesU.
2. Youth Science Summit with the Kentucky Science Center.
3. Engineering Expo with annual water rocket competition.
4. Pursuing more ambitious recruiting, developing relationships with contacts in Louisville and the University, and establishing relationships with aerospace companies.

River City Rocketry also has design goals that set them apart from other competitors. The main proponent of this is to design a Variable Drag System (VDS), which raises the bar for reliable and consistent apogee accuracy in every NSL flight. Payload design is the other key identifier that the team has set out to highlight: the goal of creating a reliable target detection system, with the capability of upright landing at a defined location, to further push for an increase in not only higher engineering standards but to also promote the research-based nature of the University of Louisville. Finally, the team's intention is to foster a healthy growth of the team, from all disciplines of study; expanding the team's cumulative knowledge of rocketry and ensuring a sustained and continuous improvement in the team's ability to achieve its goals.

The team has been working throughout the year to meet these goals through multiple resources. Safety has been a crucial part of all RCR projects. By drawing an emphasis on safety on every part of a project, the team is able to construct the most safety-conscious design, evaluating every system and procedure and subjecting them to a thorough vetting process.

Appendix I – Launch Procedures

9.6 Safety Checklist: Multirotor Recovery System

To be checked and initialed by MRS Safety representative.

MRS Representative Signatures:

1. ____________________________ *2.* ______________________________

Required Equipment:

- *Battery Voltage Monitor*
- *Li-Po Battery Charger*
- *RC transmitter*
- *Spare Propellers*
- *Spare Motors*
- *Associated fastening tools*
- *Multi-meter*
- *4-40 nylon shear pins*
- *Recovery Insulation (Dog Barf)*
- *Black Powder*
- *E-match*
- *Spare 3D printed electronic sleds*

Prior to leaving for launch site:

1. ___ Charge batteries
2. ___ Charge ground station laptop
3. ___ Charge RC transmitter
4. ___ Mechanical component checkout and damage inspection
5. ___ Verify functionality of all flight electronics
6. ___ Perform arm and leg deployment test
7. ___ Perform hover test and landing to verify multirotor functionality
8. ___ Black powder charge test ARRD
9. ___ Black powder charge test RRS.

At launch site:

1. ___ Connect Li-Po battery to MRS Electronics
2. ___ Establish connection to RC transmitter.
3. ___ Initialize home GPS coordinates.
4. ___ Perform preliminary thrust check.
5. ___ Check/tighten fasteners.

6. ___ Test camera feed on payload.
7. ___ Inspect any mechanical components for damage.

Post launch:

1. ___ Disconnect Li-Po battery.

    ⚠WARNING Ensure that Li-Po battery has been disconnected before attempting to remove propellers or handle the multirotor in any way.

2. ___ Remove propellers.
3. ___ Download flight data.

9.7 Safety Checklist: Variable Drag System (VDS) Prototype

At launch site:

1. ___ Insert micro SD card into its slot
2. ___ Connect the encoder
3. ___ Connect the 9 volt battery. Ensure that its voltage is greater than 9 volts.
4. ___ Verify that the 2 LEDs on the Arduino Pro mini and the 1 LED on the micro SD breakout are on as seen on the left. The green light on the Arduino and the red light on the SD should both be blinking rapidly. The red light on the Arduino should be solid.
5. ___ Connect the DC motor. NOTE THE WIRE COLORS. The red one should go to terminal B and the black to terminal A.
6. ___ Connect the 11.1v lipo battery. (make sure its voltage > 11.1) NOTE THE COLOR OF THE WIRES. This provides power to the DC motor.
7. ___When at pad altitude, press and hold the black button on the breadboard until the airbrakes are fully retracted. This zeros both the altimeter and the airbrake blades. It also wipes the memory card and starts a new file. Ensure that the rapid blinking briefly stops, and then starts again after the button has been released.
8. ___Insert VDS into airframe. Check that drag blades line up with their slots.
9. ___Press black button to actuate the blades, validating a proper installation.
10. ___Install coupler in recovery bay. Perform this step slowly as quick assembly will create a pressure spike causing the VDS blades to actuate.

Post launch:

1. ___ Detach the 9 volt battery.
2. ___ Take out micro SD and insert into computer with Matlab for data analysis.
3. ___Detach 11.1v lipo battery.

9.8 Safety Checklist: General Preparations

To be checked and initialed by River City Rocketry team member.

River City Rocketry Team Member Signatures:

1. ___________________________ 2. ______________________________

Prior to leaving for launch site:

Required Equipment:

- *Clear black powder capsules (x6)*
- *E-matches (x10)*
- *Drill*
- *1/8" drill bit*
- *Electrical tape*
- *Scissors*
- *Black powder*
- *Paper towels*
- *Black powder measurement kit*
- *ARRD (x3)*

Required PPE:

- *Safety glasses*

9.8.1 Black Powder Charge Preparation

1. ___ Drill a 1/8" hole in the bottom of each of the clear black powder capsules.
   ⚠**CAUTION**: Safety glasses are to be worn while drilling.
2. ___ Unwind one e-match.
3. ___ Feed wire from the e-match through the hole in the base of a capsule. Ensure the pyrotechnic end of the e-match is inside the capsule.
4. ___ Wrap electrical tape to secure the e-match in place and to ensure that black powder will not leak from the capsule.
   ⚠WARNING If the capsules are not completely sealed, black powder will leak when the capsules are filled. Leakage could potentially result in ejection charges being too small or failing altogether, causing a catastrophic failure in recovery.
5. ___ Fill capsules with appropriate amount of cc's of black powder. Fill excess space with a cellulose insulation to ensure black powder remains in contact with the pyrotechnic tip of the e-match no matter the orientation of the capsule.
6. ___ Repeat steps 2 through 4 six times.

7. ___ Store modified capsules and e-matches in explosives box.
8. ___ Remove plastic protective covers from 6 e-matches.
9. ___ Insert the two modified e-matches into the ARRD's.
10. ___ Assembly ARRD's in accordance to their product manual.
11. ___ Store loaded ARRD's in the explosives box.

⚠ DANGER E-matches are explosive. The cartages of the ARRD's and leads must be kept clear from batteries and any open flames in order to avoid accidental firing.

9.8.2 GPS Preparations

Required Equipment:

- *GPS unit(s) (x4)*
- *GPS charger*

1. ____ Check GPS unit(s) for full charge. If not fully charged, charge GPS unit(s).

Launch Day Procedures:

9.8.3 Nosecone GPS Installation

Required Equipment:

- *Nosecone GPS*
- *Nosecone GPS sled*
- *M3 screws (x2)*
- *Socket wrench set*
- *GPS tracking device*

1. ___ Check nosecone GPS for contact with tracking device.
2. ___ Securely mount GPS to GPS sled in nosecone using 2 M3 screws.
3. ___Ensure signal reception gets through nose cone before sealing nose cone bay.

9.9 Safety Checklist: Recovery

To be checked and initialed by Recovery Safety representatives.

Recovery Representative Signatures:

1. ___________________________ *2.* ______________________________

Prior to leaving for launch site:

9.9.1 Parachute Packing

Required Equipment:

- *Small fabric hair ties/Rubber bands*
- *Packing Hook*
- *Clamp*
- *Main parachute (x3)*
- *Deployment bag (x2)*

1. ___ Inspect canopies and lines for any cuts, burns, fraying, loose stitching and any other visible damage.

   Note: If any damage is identified, **immediately** inform both of the team captains and the safety officer. The rocket will be deemed safe to fly or a corrective action will be decided upon and implemented.
2. ___ Lay parachute canopy out flat.
3. ___ Ensure shroud lines are taut and evenly spaced and not tangled.
4. ___ Fold parachute per the folding procedures document in the team owncloud folder. Use clamps as necessary to ensure a tight fold.
5. ___ Place folded parachute(s) into respective deployment bag with shroud lines coming directly out of the bag.

   ⚠WARNING Ensure that the shroud lines are not wrapped around the parachute inside the deployment bag. This will result in the parachute getting stuck in the deployment bag. Verify that the parachute fits loosely in the deployment bag.
6. ___ Secure deployment flaps using shroud lines and fabric hair ties.
7. ___ Use hook to assist in securing extra length of shroud lines through loops stitched in deployment bag. Continue this pattern in the same direction around the deployment bag in order to prevent tangling.
8. ___ Attach pilot parachute to the top of the deployment bag(s) ONLY.

9.9.2 VDS and Payload Recovery Avionics Bay(s):

- *Precision flathead screwdriver*
- *Standard Phillips head screwdriver*
- *Nosecone altimeter sled*
- *StratoLogger altimeter (x4)*
- *4x40 shear pins (x24)*
- *Battery holster cover*
- *Duracell 9V battery (x4)*
- *Battery clips (x2)*
- *Multimeter*

1. ___ Verify proper shielding.

   ⚠WARNING Ensure that the entire inside of the avionics bay is properly shielded in order to protect from interference. In the incident that interference occurs, pyrotechnic devices may be actuated prematurely, causing potential harm to personnel and mission failure.
2. ___ Verify StratoLogger CF altimeters are properly programed in accordance with file in team OwnCloud folder.
3. ___ Verify 9V battery has a minimum charge of 8.7V.
4. ___ Mount StratoLoggers onto standoffs on sustainer altimeter sled using #4-40 shear pins.
5. ___ Attach batteries to battery clips and install into holster.
6. ___ Attach battery holster cover using four, #4-40 shear pin.
7. ___ Ensure screw switches are turned off and wire screw switches to switch terminal on StratoLogger.
8. ___ Wire battery to +/- terminal on StratoLogger.
9. ___ Wire main and drogue terminals on StratoLogger to terminal blocks on the nosecone
10. ___ Install altimeter sled into avionics bay.

9.9.3 Redundant Recovery System:

- *Socket wrench set*
- *Custom altimeter electronics mounting box*
- *Custom altimeter electronics*
- *4-40 shear pins (x3)*
- *Altimeter mounting box cover*
- *Single cell Li-Poly battery (x2)*
- *Recovery Insulation (Dog Barf)*
- *E-Matche(s)*

1. ___ Attach the RRS electronics to mounting sled to the PSS section bulkplate.
2. ___ Insert the PSS electronics sled onto the payload.
3. ___Install both 9V Duracell batteries into the corresponding battery mount on the electronics sled.
4. ___ Connect RRS and DCS E-matches to RRS electronics.
5. ___ Place upper bulkplate on top of payload.
6. ___ Fasten upper bulkplate to payload by installing 4x 1/4in-20 nuts onto PSS all thread rods.

Launch day procedures

9.9.4 Parachute Assembly:

Required Equipment:

- *Nomex cloth (x3)*
- *Shock cord (x5)*
- *Pilot parachute (x2)*
- *QuickLinks (x14)*

1. ___ Insert deployment bay main bag and attach ARRD tether to top of bag.
2. ___ Attach deployment bay drogue shock cord to ARRD on bulkplate via quick link.
3. ___ Install drogue.
4. ___ Insert booster main bag and attach ARRD tether to top of bag.
5. ___ Attach booster drogue shock cord to ARRD on bulkplate.
6. ___ Install drogue.

9.9.5 Nosecone Assembly

Required Equipment:

- *Precision flathead screwdriver*
- *¼"-20 nut (x2)*
- *¼"-20 washer (x2)*
- *GPS tracking device*

1. ___ Check GPS for connection with tracking device.
2. ___ Verify wiring of altimeters is correct.
3. ___ Wire a black powder charge to each terminal block.
4. ___ Install bulk plate onto threaded rods. Ensure that fiberglass plate is fully seated against the coupler tubing.
5. ___ Secure bulk plates in place using ¼-20 nuts and washers.

9.10 Safety Checklist: Overall Final Assembly Checklist

Final Assembly Representative Signatures:

1. ________________________ *2.* ___________________________

Required Equipment:

- *Allen Wrench Set – SAE*
- *Phillips Head Screwdriver (large)*
- *Flat Head Screwdriver (Large)*
- *Small Screwdriver Set (Small)*
- *Socket Wrench Set for ¼-20 Nuts*
- *Masking tape*
- *Socket Cap Screws*
- *4-40 shear pins*

1. ___ Attach propulsion bay to VDS coupler using 3x 8-32 metal bolts.
2. ___ Attach upper VDS coupler to the propulsion recovery bay using x3 8-32 shear pins.
3. ___ Attach recovery bay to the payload coupler using x4 4-40 nylon shear pins.
4. ___ Attach the payload coupler to deployment bay using x4 4-40 nylon shear pins.
5. ___ Attach the deployment bay to the nose cone using x4 4-40 nylon shear pins.
6. ___ Check that the coupling does not allow for any flexing of the rocket between any airframe and coupler tubes. Should this occur, add layers of painters tape to the coupler tubing on the payload bay until sufficient coupling is achieved.
7. ___ Tape motor igniter to the outside of the lower sustainer in a place easily seen by the field RSO.
8. ___ A final visual inspection will need to be done to ensure all systems are go.

9.11 Safety Checklist: Clear to Leave for Launch Pad:

All sections of the safety checklist preceding the "at the launch pad checklist" must be complete prior to leaving for the launch pad. A signature of completion is required for launch.

*General Pre-Launch Day Preparations:*________________________________

*Stability and Propulsion:*____

Recovery: ____

Overall Final Assembly: ____

Signatures indicating the rocket is a "Go" for launch:

Team Captain: ____

Team Co-Captain: _____________________________________

Safety Officer Signature: ________________________________

9.12 Safety Checklist: At Launch Pad Checklist

Required Equipment:

- *Pen or pencil*
- *Level 2 Certification card.*
- *GoPro camera*
- *Level*
- *Precision flathead screwdriver*

1. ___ Verify flight card has been properly filled out and permission has been granted by RSO to launch.
2. ___ Place rocket on launch pad.
3. ___ Tilt and rotate the launch pad in desired direction, or in direction ruled necessary by RSO. Use level to ensure desired launch angle. Use turnbuckles for fine adjustments.
4. ___ Ensure proper connection has been made with ground station electronics.
5. ___ Arm all electronics in the following order: payloads, cameras, and altimeters (in order as follows: AIM Xtra, StratoLoggers, and Teensey). Check for correct LED readout, beeping pattern, etc.
6. ___ Before leaving launch pad area, double check for signs that all electronics are still operating correctly.
7. ___ Arm launch pad camera and begin recording.
8. ___ Clear launch pad area and do not return until range has been reopened by the RSO.

9.13 Safety Checklist: During and After Flight (DAF):

Flight Events:

First Event: Mid-section separation - booster drogue

Observer Signature: _________________________ Time: ______

Second Event: Two second delay - deployment bay drogue

Observer Signature: ______________________ Time: ______

Third Event: 1700 feet – disengagement of deployment bay ARRD and main deployment.

Observer Signature: ______________________ Time: ______

Fourth Event: 1350 feet – Multi-rotor ejection and MDP inflation.

Observer Signature: ______________________ Time: ______

Fifth Event: 600 feet – Disengagement of propulsion bay ARRD and main deployment.

Observer Signature: ______________________ Time: ______

Potentially Sixth Event: N/A unless payload exceeds kinetic energy requirement or RSO deems unsafe.

Observer Signature: ______________________ Time: ______

Landing Events:

Launch Vehicle Assembly

Observer Signature: ______________________ Time: ______

Video Recorder Signature: _____________________________

Photographer Signature: ______________________________

Rapid Retrieval Team Member #1: _______________________

Rapid Retrieval Team Member #2: _______________________

Rapid Retrieval Team Member #3: _______________________

Required Equipment:

- *Stopwatch or phone timer.*
- *Small Phillips head screwdriver*
- *Camera*

1. Rapid Retrieval team members are to be within close vicinity to a vehicle ready to move within a few seconds notice.
2. Start stopwatch upon liftoff and call out time in 5 second intervals until T-10 seconds until first event. Continue to call out times until T-10 seconds to second event.
3. Maintain line of sight with rocket at all times. Indicate any observed anomalies out loud to alert spectators.
4. While retrieving rocket, disarm all rocket recovery systems first.
5. Prior to touching the rocket or parachute, take photo documentation of how the rocket landed.
6. Before disturbing the rocket, note any damages and anomalies with root causes. Document these for later examination.
7. Disassemble the rocket looking for any signs of wear, damage, or fatigue. Note what repairs will have to be made, if any.

After Flight Checklist: To be checked and initialed by Recovery Safety representative.

Recovery Representative Signatures:

1. ____________________________ *2.* _______________________________

1. ___ Inspect all shroud lines for any damage, or burn marks.
2. ___ Inspect all shroud attachment points for damage.
3. ___ Inspect entire canopy for any damage, or stretching.
4. ___ Inspect deployment bag for damage.

Damage found on shroud lines? Y / N

*Notes:*____

Damage found on attachment points? Y / N

*Notes:*____

Damage found on deployment bag? Y / N

*Notes:*____

Tearing or stretching found on canopy? Y/N

If yes, sketch approximate location below:

Damage Notes: ____
____
____

Repair Plan: ____
____
____

Altitude Achieved: _______________

Motor Used: ____________________

Location: ______________________

Temperature: ___________________

Pressure: ______________________

Wind Speed: ___________________

Event #1 Success: Y or N

Event #2 Success: Y or N

Captain Approval: 1. ____

2. ____

Appendix II – Safety Risk Assessments

Lab and Machine Shop Risk Assessment						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity Value**	**Probability Value**	**Risk Level**	**Mitigation**
Using power tools and hand tools such as blades, saws, drills, etc.	1. Improper training on power tools and other lab equipment. 2. Uniformed on proper tool to use.	1a. Mild to severe cuts or burns to personnel. 1b. Damage to rocket or components of the rocket. 1c. Damage to equipment	2	4	Low	1. Individuals must be trained on the tool being used. Those not trained should not attempt to learn on their own and should find a trained individual to instruct them. 1. Safety glasses must be worn at all times. 1. Sweep or vacuum up shavings to avoid cuts from debris.
Sanding or grinding materials.	1. Improper use of PPE. 2. Improper training on the use of a Dremel tool or other sanding machinery.	1a. Mild to severe rash. 1b. Irritated eyes, nose or throat with the potential to aggravate asthma. 2. Mild to severe cuts or burns from a Dremel tool and sanding wheel.	3	3	Low	1a. Long sleeves should be worn at all times when sanding or grinding materials. 1b. Proper PPE should be utilized such as safety glasses and dust masks with the appropriate filtration required.

						2. Individuals must be trained on the tool being used. Those not trained should not attempt to learn on their own and should find a trained individual to instruct them.
Working with chemical components resulting in mild to severe chemical burns on skin or eyes, lung damage due to inhalation of toxic fumes, or chemical spills	1. Chemical splash. 2. Chemical fumes.	1. Mild to severe burns on skin or eyes. 2. Lung damage or asthma aggravation due to inhalation of fumes,	2	4	Low	MSDS documents will be readily available at all times and will be thoroughly reviewed prior to working with any chemical. All chemical containers will be marked to identify appropriate precautions that need to be taken. 1. Nitrile gloves shall be used when handling hazardous materials. 1. Personnel are familiar with locations of safety features such as an eye wash station, chemical burn station and first aid kit.
						1. Safety goggles are to be worn at all times when handling chemicals. 2. When working with chemicals producing fumes, appropriate precautions should be taken such as working in a well-ventilated area, wearing vapor masks, or working under a fume hood.

Damage to equipment while soldering.	1. Soldering iron is too hot 2. Prolonged contact with heated iron 3. Soldering iron tip varies in temperature along tip	The equipment could become unusable. If parts of the payload circuit get damaged, they could become inoperable.	3	3	Low	1. The temperature on the soldering iron will be controlled and set to a level that will not damage components. 2. For temperature sensitive components sockets will be used to solder ICs to. 3. Proper de-soldering tools and wiping sponges will be available during all soldering tasks.
Dangerous fumes while soldering.	1. Use of leaded solder can produce toxic fumes. 2. Leaving soldering iron too long on plastic could cause plastic to melt producing toxic fumes.	Team members become sick due to inhalation of toxic fumes. Irritation could also occur.	3	3	Low	1. The team will use well ventilated areas while soldering. Fans will be used during soldering. 2. Team members will be informed of appropriate soldering techniques, avoiding contact of the soldering iron to plastic materials for extended periods of time.
Potential burns to team members while soldering.	Team members do not pay attention while soldering	The team member could suffer minor to severe burns.	4	3	Low	Team members will be trained how to solder and will follow all safety protocols related to soldering.
Overcurrent from power source while testing.	Failure to correctly regulate power to circuits during testing	Team members could suffer electrical shocks which could cause burns to heart arrhythmia	2	4	Low	The circuits will be analyzed before they are powered to ensure they don't pull too much power. Power supplies will also be set to the correct levels.

Use of cutting fluid.	Use cutting fluid when machining metals.	Contains carcinogens.	1	5	Low	Face shield shall be worn at all times when machining metals.
Handling Carbon Fiber and Fiberglass Tow	Use in manufacturing airframe and bulkplates	1. Splinters in skin 2. Respiratory irritation	4	3	Low	Team members are required to wear cut resistant gloves, long sleeves, and safety glasses when handling carbon fiber.
Use of white lithium grease.	Use in installing motor	1. Irritation to skin and eyes. 2. Respiratory irritation.	3	4	Low	1. Nitrile gloves and safety glasses are to be worn when applying grease. 2. When applying grease, it should be done in a well ventilated area to avoid inhaling fumes.
High voltage shock.	Improper use of welding equipment.	Death or severe injury.	1	5	Low	All team members are required to be trained on the equipment prior to use. Any time personnel is welding, there must be at least two people present.

Damage to equipment while winding airframe, X-Winder	1. Improper use of X-Winder equipment. 2. Improper training of program on X-Winder	1a. Running the carriage into the solid stops, damaging the carriage. 1b. Not tightening the chucks that connect to the mandrill; resulting in a damaged mandrill. 2. Writing incorrect program, wasting material, and damage of equipment	2	5	Low	All team members are required to be trained on the equipment prior to use. Any time someone writes or runs the X-Winder must be at least two people present.
Break bit on mill.	Spindle speed too high.	Injury to personnel and damage to equipment and/or part.	2	5	Low	All team members are required to be trained on the mill prior to use. If personnel is uncertain about the proper settings, they are to consult an experienced member prior to operation.
Metal shards.	Using equipment to machine metal parts.	Metal splinters in skin or eyes.	2	5	Low	Team members must wear long sleeves and safety glasses whenever working with metal parts.

Table 98: Lab and machine shop risk assessment.

Environmental Hazards to Rocket Risk Assessment						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity Value**	**Probability Value**	**Risk Level**	**Mitigation**
Low cloud cover.	N/A	Unable to test entire system.	1	4	Moderate	When planning test launches, the forecast should be monitored in order to launch on a day where weather does not prohibit launching or testing the entire system.
Rain	N/A	1. Unable to launch. 2. Damage electrical components and systems in the rocket.	1	4	Moderate	1. When planning test launches, the forecast should be monitored in order to launch on a day where weather does not prohibit launching or testing the entire system. 2. Have a plan to place electrical components in water tight bags. Have a location prepared to store the entire rocket to prevent water damage.
High winds	N/A	1. Have to launch at high angle, reducing altitude achieved. 2. Increased drifting. 3. Unable to launch.	1	4	Moderate	1,2,3. When planning test launches, the forecast should be monitored in order to launch on a day where weather does not prohibit launching or testing the entire system. If high winds are present but allowable for launch, the time of launch should be planned for the time of day with the lowest winds.
Trees	N/A	1. Damage to rocket or parachutes. 2. Irretrievable rocket components.	1	4	Moderate	Launching with high winds should be avoided in order to avoid drifting long distances. Drift calculations have been computed, so we can estimate how far each component of the rocket will drift with a particular wind velocity. The rocket should not be launched if trees are within the estimated drift radius.

Swampy ground	N/A	Irretrievable rocket components.	1	4	Moderat	With the potential of the salt flats being extremely soft, as well as local launch sites, the rocket should not be launched if there is swampy ground within the predicted drift radius that would prevent the team from retrieving a component of the rocket.
Ponds, creeks, and other bodies of water.	N/A	1. Loss of rocket components. 2. Damaged electronics.	1	4	Moderate	Launching with high winds should be avoided in order to avoid drifting long distances. The rocket should not be launched if a body of water is within the estimated drift radius. Should the rocket be submerged in water, it should be retrieved immediately and any electrical components salvaged. Electrical components are to be tested for complete functionality prior to reuse.
Extremely cold temperatures.	1. Batteries discharge quicker than normal. 2. Shrinking of fiberglass.	1. Completely discharged batteries will cause electrical failures and fail to set off black powder charges, inducing critical events. 2. Rocket will not separate as easily.	1	5	Moderate	1. Batteries will be checked for charge prior to launch to ensure there is enough charge to power the flight. Should the flight be delayed, batteries will should be rechecked and replaced as necessary. 2. If the temperatures are below normal launch temperature, black powder charges should be tested to ensure that the pressurization is enough to separate the rocket. If this test is successful, the rocket should be safe to launch.
Humidity	N/A	Motors or black powder charges become moist and don't ignite.	1	5	moderate	Motors and black powder should be stored in a location free from moisture to remove
UV exposure	Rocket left exposed to sun for long periods of time.	Possibly weakening materials or adhesives.	4	4	Low	Rocket should not be exposed to sun for long periods of time. If the rocket must be worked on for long periods of time, shelter should be sought.

Table 99: Environmental hazards to rocket risk assessment.

Hazards to Environment Risk Assessment						
Hazard	**Cause/ Mechanism**	**Outcome**	**Severity Value**	**Probability Value**	**Risk Level**	**Mitigation**
Harmful substances permeating into the ground or water.	Improper disposal of batteries or chemicals.	Impure soil and water can have negative effects on the environment that in turn, work their way into humans, causing illness.	4	3	Low	Batteries and other chemicals should be disposed of properly in accordance with the MSDS sheets. Should a spill occur, proper measure are to be followed in accordance with the MSDS sheets and any EHS standards.
Release of hydrogen chloride into the atmosphere.	Burning of composite motors.	Hydrogen chloride dissociates in water forming hydrochloric acid.	4	1	Moderate	While the probability of hydrochloric acid forming is high, the amount that would be produced over the course of a season is negligible. Fewer than six motors are predicted to be fired during the year, all of which are relatively small in size.
Release of reactive chemicals.	Burning of composite motors.	Reactive chemicals work to deplete ozone layer.	4	1	Moderate	While the probability of releasing reactive chemicals into the environment is high, the quantity released will result in negligible effects. Fewer than six motors are predicted to be fired during the year, all of which are relatively small in size.
Release of toxic fumes in the air.	Burning of ammonium perchlorate motors.	Biodegradation.	4	1	Moderat	Ammonium perchlorate will be burned in small quantities and infrequently. The amount of toxins released will cause minimal degradation.

Production of styrene gas.	Through the use of fiberglass in the overall design, fiberglass is manufactured by a second party.	Toxic air emissions.	4	1	Moderate	Productions methods for fiberglass produces toxic air pollutants, particularly styrene, which evaporate during the curing process. Due to the quantity of fiberglass utilized on the rocket, the amount of pollutants produced throughout manufacturing process will have a negligible effect on the environment.
Spray painting.	The rocket will be spray painted.	1. Water contamination. 2. Emissions to environment.	2	5	Low	All spray painting operations will be performed in a paint booth. This prevents any overspray from entering into the water system or air.
Soldering wires.	All wires will be soldered together to retain strength and proper connection.	1. Air contamination 2. Ground contamination	4	1	Low	The amount of vapor from the soldering process is at such a low quantities that no action will be needed.
Use of lead acid battery leakage.	Old or damaged housing to battery	1. Acid will leak onto the ground and get into the water system. 2. Chemical reaction with organic material that could potentially cause a fire.	3	4	Low	1. We are using new batteries that have been factory inspected and tested. 2. Proper lifting and storing procedures according to manufacturer's specifications will be adhered to.

Plastic waste material.	Plastic using in the production of electrical components and wiring.	1. Sharp plastic material produced when shaving down plastic components could harm animals if ingested by an animal. 2. Plastic could find its way down a drain and into the water system.	3	5	Low	1. All plastic material will be disposed of in proper waste receptacles.
Wire waste material.	Wire material used in the production of electrical components.	1. Sharp bits of wire being ingested by an animal if improperly disposed of.	3	5	Low	1. All wire material will be disposed of in proper waste receptacles.
CO2 emissions.	Travel to launch sites and competition.	Destroying the ozone layer.	4	1	Moderate	While the effects of CO2 emissions cannot be reversed, the amount produced is negligible.

Table 100: Hazards to environment risk assessment.

Appendix II – Statement of Work Requirements

Req. ID	Description	Link to reference
	Vehicle Requirements	
1.1	The vehicle shall deliver the science or engineering payload to an apogee altitude of 5,280 feet above ground level (AGL).	VDS Vehicle
1.2	The vehicle shall carry one commercially available, barometric altimeter for recording the official altitude used in determining the altitude award winner. Teams will receive the maximum number of altitude points (5,280) if the official scoring altimeter reads a value of exactly 5280 feet AGL. The team will lose one point for every foot above or below the required altitude. The altitude score will be equivalent to the percentage of altitude points remaining after and deductions.	Vehicle
1.2.1	The official scoring altimeter shall report the official competition altitude via a series of beeps to be checked after the competition flight.	Recovery & Vehicle
1.2.2	Teams may have additional altimeters to control vehicle electronics and payload experiment(s).	VDS Payload
1.2.3	At the LRR, a NASA official will mark the altimeter that will be used for the official scoring.	Vehicle
1.2.4	At the launch field, a NASA official will obtain the altitude by listening to the audible beeps reported by the official competition, marked altimeter.	Vehicle
1.2.5	At the launch field, to aid in determination of the vehicle's apogee, all audible electronics, except for the official altitude-determining altimeter shall be capable of being turned off.	Vehicle
1.2.6	The following circumstances will warrant a score of zero for the altitude portion of the competition:	VDS
1.2.6.1	The official, marked altimeter is damaged and/or does not report and altitude via a series of beeps after the team's competition flight.	Vehicle

1.2.6.2	The team does not report to the NASA official designated to record the altitude with their official, marked altimeter on the day of the launch.	Vehicle
1.2.6.3	The altimeter reports an apogee altitude over 5,600 feet AGL.	VDS
1.2.6.4	The rocket is not flown at the competition launch site.	Safety
1.3	All recovery electronics shall be powered by commercially available batteries.	Vehicle
1.4	The launch vehicle shall be designed to be recoverable and reusable. Reusable is defined as being able to launch again on the same day without repairs or modifications.	Vehicle
1.5	The launch vehicle shall 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.	Vehicle
1.6	The launch vehicle shall be limited to a single stage.	Vehicle
1.7	The launch vehicle shall be capable of being prepared for flight at the launch site within 4 hours, from the time the Federal Aviation Administration flight waiver opens.	Vehicle
1.8	The launch vehicle shall be capable of remaining in launch-ready configuration at the pad for a minimum of 1 hour without losing the functionality of any critical on-board component.	Vehicle
1.9	The launch vehicle shall be capable of being launched by a standard 12 volt direct current firing system. The firing system will be provided by the NASA-designated Range Services Provider.	Vehicle
1.10	The launch vehicle shall require no external circuitry or special ground support equipment to initiate launch (other than what is provided by Range Services).	Vehicle
1.11	The launch vehicle shall 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).	Vehicle
1.11.1	Final motor choices must be made by the Critical Design Review (CDR).	N/A
1.11.2	Any motor changes after CDR must be approved by the NASA Range Safety Officer (RSO), and will only be approved if the change is for the sole purpose of increasing the safety margin.	N/A
1.12	Pressure vessels on the vehicle shall be approved by the RSO and shall meet the following criteria:	Vehicle

1.12.1	The minimum factor of safety (Burst or Ultimate pressure versus Max Expected Operating Pressure) shall be 4:1 with supporting design documentation included in all milestone reviews.	N/A
1.12.2	The low-cycle fatigue life shall be a minimum of 4:1.	N/A
1.12.3	Each pressure vessel shall include a solenoid pressure relief valve that sees the full pressure of the tank.	N/A
1.12.4	Full pedigree of the tank shall be described, including the application for which the tank was designed, and the history of the tank, including the number of pressure cycles put on the tank, by whom, and when.	N/A
1.13	The total impulse provided by a Middle and/or High School launch vehicle shall not exceed 5,120 Newton-seconds (L-class).	Vehicle
1.14	The launch vehicle shall have a minimum static stability margin of 2.0 at the point of rail exit.	Vehicle
1.15	The launch vehicle shall accelerate to a minimum velocity of 52 fps at rail exit.	Vehicle
1.16	All teams shall successfully launch and recover a subscale model of their rocket prior to CDR.	Vehicle
1.16.1	The subscale model should resemble and perform as similarly as possible to the full-scale model, however, the full-scale shall not be used as the subscale model.	Vehicle
1.16.2	The subscale model shall carry an altimeter capable of reporting the model's apogee altitude.	Vehicle
1.17	All teams shall successfully launch and recover their full-scale rocket prior to FRR in its final flight configuration. The rocket flown at FRR must be the same rocket to be flown on launch day. The purpose of the full-scale demonstration flight is to demonstrate 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 a lower altitude, functioning tracking devices, etc.). The following criteria must be met during the full scale demonstration flight:	Vehicle
1.17.1	The vehicle and recovery system shall have functioned as designed.	Vehicle & Recovery
1.17.2	The payload does not have to be flown during the full-scale test flight. The following requirements still apply:	
1.17.2.1	If the payload is not flown, mass simulators shall be used to simulate the payload mass.	
1.17.2.1.1	The mass simulators shall be located in the same approximate location on the rocket as the missing payload mass.	
1.17.3	If the payload changes the external surfaces of the rocket (such as with camera housings or external probes) or manages the total energy of the vehicle, those systems shall be active during the full-scale demonstration flight.	

1.17.4	The full-scale motor does not have to be flown during the full-scale test flight. However, it is recommended that the full-scale motor be used to demonstrate full flight readiness and altitude verification. If the full-scale motor is not flown during the full-scale flight, it is desired that the motor simulate, as closely as possible, the predicted maximum velocity and maximum acceleration of the launch day flight.	
1.17.5	The vehicle shall be flown in its fully ballasted configuration during the full-scale test flight. Fully ballasted refers to the same amount of ballast that will be flown during the launch day flight.	
1.17.6	After successfully completing the full-scale demonstration flight, the launch vehicle or any of its components shall not be modified without the concurrence of the NASA Range Safety Officer (RSO).	
1.17.7	Full scale flights must be completed by the start of FRRs (March 6th, 2016). If the Student Launch office determines that a re-flight is necessary, than an extension to March 24th, 2016 will be granted. This extension is only valid for re-flights; not first time flights.	
1.18	Any structural protuberance on the rocket shall be located aft of the burnout center of gravity.	Vehicle
1.19	Vehicle Prohibitions	Vehicle
1.19.1	The launch vehicle shall not utilize forward canards.	
1.19.2	The launch vehicle shall not utilize forward firing motors.	
1.19.3	The launch vehicle shall not utilize motors that expel titanium sponges (Sparky, Skidmark, MetalStorm, etc.)	
1.19.4	The launch vehicle shall not utilize hybrid motors.	
1.19.5	The launch vehicle shall not utilize a cluster of motors.	
1.19.6	The launch vehicle shall not utilize friction fitting for motors.	
1.19.7	The launch vehicle shall not exceed Mach 1 at any point during flight.	
1.19.8	Vehicle ballast shall not exceed 10% of the total weight of the rocket.	
	Recovery System Requirements	
2.1	The launch vehicle shall stage the deployment of its recovery devices, where a drogue parachute is deployed at apogee and a main parachute is deployed at a much lower altitude. Tumble recovery 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 Range Safety Officer.	
2.2	Each team must perform a successful ground ejection test for both the drogue and main parachutes. This must be done prior to the initial subscale and full scale launches.	

2.3	At landing, each independent sections of the launch vehicle shall have a maximum kinetic energy of 75 ft-lbf.	
2.4	The recovery system electrical circuits shall be completely independent of any payload electrical circuits.	
2.5	The recovery system shall contain redundant, commercially available altimeters. The term “altimeters” includes both simple altimeters and more sophisticated flight computers.	
2.6	Motor ejection is not a permissible form of primary or secondary deployment.	
2.7	Each altimeter shall be armed by a dedicated arming switch that is accessible from the exterior of the rocket airframe when the rocket is in the launch configuration on the launch pad.	
2.8	Each altimeter shall have a dedicated power supply.	
2.9	Each arming switch shall be capable of being locked in the ON position for launch.	
2.10	Removable shear pins shall be used for both the main parachute compartment and the drogue parachute compartment.	
2.11	An electronic tracking device shall be installed in the launch vehicle and shall transmit the position of the tethered vehicle or any independent section to a ground receiver.	
2.11.1	Any rocket section, or payload component, which lands untethered to the launch vehicle, shall also carry an active electronic tracking device.	
2.11.2	The electronic tracking device shall be fully functional during the official flight on launch day.	
2.12	The recovery system electronics shall not be adversely affected by any other on-board electronic devices during flight (from launch until landing).	
2.12.1	The recovery system altimeters shall be physically located in a separate compartment within the vehicle from any other radio frequency transmitting device and/or magnetic wave producing device.	

2.12.2	The recovery system electronics shall be shielded from all onboard transmitting devices, to avoid inadvertent excitation of the recovery system electronics.	
2.12.3	The recovery system electronics shall 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.	
2.12.4	The recovery system electronics shall be shielded from any other onboard devices which may adversely affect the proper operation of the recovery system electronics.	
Experiment Requirements		
3.1.1	Each team shall choose one design experiment option from the following list.	
3.1.2	Additional experiments (limit of 1) are encouraged, and may be flown, but they will not contribute to scoring.	N/A
3.1.3	If the team chooses to fly additional experiments, they shall provide the appropriate documentation in all design reports so experiments may be reviewed for flight safety.	N/A
3.2	target detection and controlled landing	
3.2.1	Teams shall design an onboard camera system capable of identifying and differentiating between 3 randomly placed targets.	
3.2.1.1	Each target shall be represented by a different colored ground tarp located on the field.	
3.2.1.2	Target samples shall be provided to teams upon acceptance and prior to PDR.	
3.2.1.3	All targets shall be approximately 40'X40' in size.	
3.2.1.4	The three targets will be adjacent to each other, and that group shall be within 300 ft. of the launch pads.	
3.2.2	After identifying and differentiating between the three targets, the launch vehicle section housing the cameras shall land upright, and provide proof of a successful controlled landing.	
3.2.3	Data from the camera system shall be analyzed in real time by a custom designed on-board software package that shall identify and differentiate between the three targets.	
Safety Requirements		
4.1	Each team shall use a launch and safety checklist. The final checklists shall be included in the FRRreport and used during the Launch Readiness Review (LRR) and any launch day operations.	

4.2	Each team must identify a student safety officer who shall be responsible for all items in section 4.3.	
4.3	The role and responsibilities of each safety officer shall include, but not limited to:	
4.3.1	Monitor team activities with an emphasis on Safety during:	
4.3.1.1	Design of vehicle and launcher	
4.3.1.2	Construction of vehicle and launcher	
4.3.1.3	Assembly of vehicle and launcher	
4.3.1.4	Ground testing of vehicle and launcher	
4.3.1.5	Sub-scale launch test(s)	
4.3.1.6	Full-scale launch test(s)	
4.3.1.7	Launch day	
4.3.1.8	Recovery activities	
4.3.1.9	Educational Engagement Activities	
4.3.2	Implement procedures developed by the team for construction, assembly, launch, and recovery activities	
4.3.3	Manage and maintain current revisions of the team's hazard analyses, failure modes analyses, procedures, and MSDS/chemical inventory data	
4.3.4	Assist in the writing and development of the team's hazard analyses, failure modes analyses, and procedures.	
4.4	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 the rocketeer shall have flown and successfully recovered (using electronic, staged recovery) a minimum of 2 flights in this or a higher impulse class, prior to PDR. The mentor is designated as the individual owner of the rocket for liability purposes and must travel with the team to launch week. One travel stipend will be provided per mentor regardless of the number of teams he or she supports. The stipend will only be provided if the team passes FRR and the team and mentor attends launch week in April.	

4.5	During test flights, teams shall abide by the rules and guidance of the local rocketry club's RSO. The allowance of certain vehicle configurations and/or payloads at the NASA Student Launch Initiative does not give explicit or implicit authority for teams to fly those certain vehicle configurations and/or payloads at other club launches. Teams should communicate their intentions to the local club's President or Prefect and RSO before attending any NAR or TRA launch.	
4.6	Teams shall abide by all rules set forth by the FAA.	
	General Requirements	
5.1	Students on the team shall do 100% of the project, including design, construction, written reports, presentations, and flight preparation with the exception of assembling the motors and handling black powder or any variant of ejection charges, or preparing and installing electric matches (to be done by the team's mentor).	
5.2	The team shall provide and maintain a project plan to include, but not limited to the following items: project milestones, budget and community support, checklists, personnel assigned, educational engagement events, and risks and mitigations.	
5.3	Foreign National (FN) team members shall be identified by the Preliminary Design Review (PDR) and may or may not have access to certain activities during launch week due to security restrictions. In addition, FN's may be separated from their team during these activities.	
5.4	The team shall identify all team members attending launch week activities by the Critical Design Review (CDR). Team members shall include:	
5.4.1	Students actively engaged in the project throughout the entire year.	
5.4.2	One mentor (see requirement 4.4).	Mentor
5.4.3	No more than two adult educators.	
5.5	The team shall engage a minimum of 200 participants in educational, hands-on science, technology, engineering, and mathematics (STEM) activities, as defined in the Educational Engagement Activity Report, by FRR. An educational engagement activity report shall be completed and submitted within two weeks after completion of an event. A sample of the educational engagement activity report can be found on page 28 of the handbook.	Educational Outreach
5.6	The team shall develop and host a Web site for project documentation.	
5.7	Teams shall post, and make available for download, the required deliverables to the team Web site by the due dates specified in the project timeline.	N/A
5.8	All deliverables must be in PDF format.	N/A

5.9	In every report, teams shall provide a table of contents including major sections and their respective sub-sections.	
5.10	In every report, the team shall include the page number at the bottom of the page.	N/A
5.11	The team shall provide any computer equipment necessary to perform a video teleconference with the review board. This includes, but not limited to, a computer system, video camera, speaker telephone, and a broadband Internet connection. If possible, the team shall refrain from use of cellular phones as a means of speakerphone capability.	
5.12	All teams will be required to use the launch pads provided by Student Launch's launch service provider. No custom pads will be permitted on the launch field. Launch services will have 8 ft. 1010 rails, and 8 and 12 ft. 1515 rails available for use.	
5.13	Teams must implement the Architectural and Transportation Barriers Compliance Board Electronic and Information Technology (EIT) Accessibility Standards (36 CFR Part 1194)	
	Preliminary Design Review Requirements	
PDR.1	Summary of PDR report	Summary
PDR.1.1	Team Summary	
PDR.1.1.1	Team name and mailing address	
PDR.1.1.2	Name of mentor, NAR/TRA number and certification level	
PDR.1.2	Launch Vehicle Summary	
PDR.1.2.1	Size and mass	
PDR.1.2.2	Motor choice	
PDR.1.2.3	Recovery system	
PDR.1.2.4	Milestone review flysheet	
PDR.1.3	Payload Summary	
PDR.1.3.1	Payload title	
PDR.1.3.2	Summarize payload experiment	
PDR.2	Changes made since proposal	Changes
PDR.2.1	Highlight all changes made since the proposal and the reason for those changes	

PDR.2.1.1	Changes made to vehicle criteria	Vehicle Changes
PDR.2.1.2	Changes made to payload criteria	Payload Changes
PDR.2.1.3	Changes made to project plan	Project Changes
PDR.3	**Vehicle Criteria**	
PDR.3.1	Selection, Design, and Rationale of Launch Vehicle	Vehicle
PDR.3.1.1	Include unique mission statement, and mission success criteria	Vehicle
PDR.3.1.2	Review the design at a system level, going through each systems' alternative designs, and evaluating the pros and cons of each alternative	Vehicle
PDR.3.1.3	For each alternative, present research on why that alternative should or should not be chosen	Vehicle
PDR.3.1.4	After evaluating all alternatives, present a vehicle design with the current leading alternatives, and explain why they are the leading choices	Vehicle
PDR.3.1.4.1	Describe each subsystem, and the components within those subsystems	Vehicle
PDR.3.1.4.2	Provide a dimensional drawing using the leading design	Vehicle
PDR.3.1.4.3	Provide estimated masses for each subsystem	Vehicle
PDR.3.1.5	Review different motor alternatives, and present data on each alternative	Vehicle
PDR.3.2	**Recovery Subsystem**	
PDR.3.2.1	Review the design at a component level, going through each components' alternative designs, and evaluating the pros and cons of each alternative	Recovery
PDR.3.2.2	For each alternative, present research on why that alternative should or should not be chosen	Recovery

PDR.3.2.3	Using the estimated mass of the launch vehicle, performance preliminary analysis on parachute sizing, and what size is required for a safe descent	Recovery
PDR.3.2.4	Choose leading components amongst the alternatives, present them, and explain why they are the current leaders	Recovery
PDR.3.2.5	Prove the redundancy exists within the system	Recovery
PDR.3.3	Mission Performance Predictions	Recovery
PDR.3.3.1	Show flight profile simulations, altitude predictions with simulated vehicle data, component weights, and simulated motor thrust curve, and verify that they are robust enough to withstand the expected loads	Vehicle
PDR.3.3.2	Show stability margin simulated center of pressure (CP)/Center of Gravity (CG) relationship and locations	Vehicle
PDR.3.3.3	Calculate the kinetic energy at landing for each independent and tethered section of the launch vehicle	Recovery
PDR.3.3.4	Calculate the drift for each independent section of the launch vehicle from the launch pad for five different cases: no wind, 5-mph wind, 10-mph wind, 15-mph wind, and 20-mph wind. The drift calculations should be performed with the assumption that the rocket will be launch straight up (zero degree launch angle)	Recovery
PDR.4	**Safety**	
PDR.4.1	Demonstrate an understanding of all components needed to complete the project, and how risks/delays impact the project	Safety
PDR.4.2	Develop a preliminary checklist of final assembly and launch procedures	Launch Procedure
PDR.4.3	Provide a preliminary Personnel Hazard Analysis. The focus of the Hazard Analysis at PDR is identification of hazards, their causes, and the resulting effects. Preliminary mitigations and controls can be identified, but do not need to be implemented at this point unless they are specific to the construction and launching of the sub-scale rocket or are hazards to the success of the SL program (ie cost, schedule, personnel availability). Rank the risk of each Hazard for both likelihood and severity.	Safety

PDR.4.3.1	Include data indicating that the hazards have been researched (especially personnel).Examples: NAR regulations, operator's manuals, MSDS, etc.	Safety
PDR.4.4	Provide a preliminary Failure Modes and Effects Analysis (FMEA) of the proposed design of the rocket, payload, payload integration, launch support equipment, and launch operations. Again, the focus for PDR is identification of hazards, causes, effects, and proposed mitigations. Rank the risk of each Hazard for both likelihood and severity.	Safety
PDR.4.5	Discuss any environmental concerns using the same format as the personal hazard analysis and FMEA.	Safety
PDR.4.5.1	This should include how the vehicle affects the environment, and how the environment can affect the vehicle	Safety
PDR.4.6	Define the risks (time, resource, budget, scope/functionality, etc.) associated with the project. Assign a likelihood and impact value to each risk. Keep this part simple i.e. low, medium, high likelihood, and low, medium, high impact. Develop mitigation techniques for each risk. Start with the risks with higher likelihood and impact, and work down from there. If possible, quantify the mitigation and impact. For example; including extra hardware to increase safety will have a quantifiable impact on budget. Including this information in a table is highly encouraged.	General, VDS, Vehicle, Recovery, Payload
PDR.5	**Payload Criteria**	
PDR.5.1	Selection, Design, and Rationale of Launch Vehicle	Payload
PDR.5.1.1	Describe what the objective of the payload is, and what experiment it will perform. What results will qualify as a successful experiment	Payload
PDR.5.1.2	Review the design at a system level, going through each systems' alternative designs, and evaluating the pros and cons of each alternative.	Payload
PDR.5.1.3	For each alternative, present research on why that alternative should or should not be chosen	Payload
PDR.5.1.4	After evaluating all alternatives, present a payload design with the current leading alternatives, and explain why they are the leading choices.	Payload
PDR.5.1.5	Include drawings and electrical schematics for all elements of the preliminary payload	Payload
PDR.5.1.6	Describe the preliminary interfaces between the payload and launch vehicle	Payload & Vehicle

PDR.5.1.7	Determine the precision of instrumentation, repeatability of measurement, and recovery system	Recovery
PDR.6	**Project Plan**	
PDR.6.1	Requirements Compliance	N/A
PDR.6.1.1	Create a verification plan for every requirement from sections 1-5 in this handbook. Identify if test, analysis, demonstration, or inspection are required to verify the requirement. After identification, describe the associated plan needed for verification.	VDS, Vehicle, Recovery & Payload
PDR.6.1.2	Create a set of team derived requirements. These are a set of minimal requirements for mission success that are ideally beyond the minimum success requirements presented in this handbook. Like before, create a verification plan identifying whether test, analysis, demonstration, or inspection is required with an associated plan.	VDS, Vehicle, Recovery, & Payload
PDR.6.2	Budgeting and Timeline	Project Plan
PDR.6.2.1	Line item budget with market values for individual components	Budget
PDR.6.2.2	Funding plan describing sources of funding, and allocation of funds	Budget
PDR.6.2.3	Timeline including all team activities, and activity duration.	Project Plan, VDS, Vehicle, Recovery, & Payload