The Terra Nova Project
Critical Design Review

NASA University Student Launch Initiative

Florida State University - Panama City
4750 Collegiate Dr
Panama City, FL 32405

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Table of Contents

Table of Tables

Table of Figures

Acronym Bank

ATF – Bureau of Alcohol, Tabacco, Firearms and Explosives

A2G – Apogee to Ground

CAD – Computer Aided Design

CAM – Computer Aided Manufacturing

CDR – Critical Design Review

CFR – Code of Federal Regulations

CNC – Computer Numerical Control

DDS – Digital Design Studio

EMI – Electromagnetic Interference

FAA – Federal Aviation Administration

FAMU – Florida Agricultural and Mechanical University

FMEA – Failure Modes and Effects Analysis

FRR – Flight Readiness Review

FSU – Florida State University

GCSC – Gulf Coast State College

HAUS – Habitat for Agricultural Utilization Study

HOL – Holley Building

HPR – High Powered Rocketry

ICT – In-Circuit Testing

InSPIRE - Innovation, Sustainability, and Preeminent Research & Education

LRR – Launch Readiness Review

NAR – National Association of Rocketry

NASA – National Aeronautics and Space Administration

NFPA – National Fire Protection Association

OSHA – Occupational Safety and Health Administration

PC – Panama City

PCB – Printed Circuit Board

PDF – Payload Demonstration Flight

PDR – Preliminary Design Review

PHA – Preliminary Hazard Analysis

PhD – Doctor of Philosophy

PLA – Polylactic Acid

PLAR – Post Launch Analysis Report

POC – Point of Contact

PPE – Personal Protective Equipment

RAM – Rocketry & Mechatronics

RSO – Registered Student Organization

SDS – Safety Data Sheet

SEARS – SouthEast Alabama Rocketry Society

SAIC – Science Applications International Corporation

STEM – Science Technology Engineering and Math

TRA – Tripoli Rocketry Association

UAS – Unmanned Aircraft System

USLI – University Student Launch Initiative

4 Summary of CDR report

4.2 Team Summary

Team Summary	
The Terra Nova Project	4750 Collegiate Dr. Panama City, FL 32405
Elizabeth Gildersleeve – Mentor	NAR #: 119990 HPR Certification Level: Level 2 Email: Elizabeth.gildersleeve00@gmail.com Phone: (850) -775-8483 FSU PC Mechanical Engineering Alumni
Launch Week Plans:	Planning to attend NASA Launch Week

4.3 Launch Vehicle Summary

Launch Vehicle Summary				
Primary Motor	**Secondary Motor**	**HAUS Section Weight**	**Avionics Section Weight**	**Propulsion Section Weight**
L2200G-0	L1420P-R	16.375 lb.	16.438 lb.	Wet: 24.375 lb. Dry: 19.375 lb.
Drogue Parachute	**Main Parachute**	**Altimeter**	**Black Powder**	**CO_2 Canisters**
Fruity Chutes 15" Compact Elliptical16.375 Parachute	120" Fruity Chutes Iris Ultra 120" Compact Parachute	Egg Timer Proton	Estimated < 0.1g. Charges Located In Avionics Bay.	16 oz CO_2 Canisters. 2 for Each Separation, 2 for Redundancy.
Proj. Descent Rate (Landing)	**Drogue**	**Reefed Main**	**Full Main**	**Reefing-Line Cutter**
15.9 ft/s	Deployment at Apogee. 154 ft/s Descent Rate.	Deployment at 1500 ft.	Deployment at 750 ft.	No Energetics Used.
Rail Size: 12 ft 1515 rail				

4.4 Payload Summary

HAUS – Habitat for Agricultural Utilization Study (Terra Inquisitor)

The HAUS payload is a mechanically deployed, ground-operating experimental system designed to autonomously collect and analyze soil following launch and recovery. After landing, the payload separates from the launch vehicle using a non-energetic lead-screw-driven mechanism compliant with NASA safety requirements. Once deployed, the payload collects a soil sample and measures soil pH, storing all data onboard for post-flight analysis. The payload is fully self-contained and designed to remain functional after exposure to launch, recovery, and landing loads.

2 Changes made since PDR

2.2 Changes made to Vehicle

Since the Preliminary Design Review, the launch vehicle has undergone several design refinements driven by subscale flight testing, recovery system failures, and manufacturability considerations. The most significant changes include updates to recovery system architecture, material selections, and deployment energy management.

Following the Terra Parvis V1 subscale flight failure, the recovery system was substantially redesigned. Bulkhead materials were reselected from 3D-printed PETG to plywood for subscale flights to eliminate brittle failure modes while preserving the planned Inconel 718 bulkhead implementation for the full-scale vehicle. The airframe and transition were redesigned to be fully fiberglass rather than fiberglass-coated PETG to improve structural robustness and better represent the full-scale configuration.

Recovery system performance was improved through resizing of the main parachute from 72 inches to 48 inches to reduce deployment loads and eliminate excessive drag for the sub-scale. A reefing system was fully redesigned and validated through multiple ground and dynamic tests to enable staged main deployment and load reduction. CO_2-based separation systems were further characterized through repeated testing, leading to improved integration procedures and deployment reliability.

Avionics placement and recovery integration were refined based on observed mass distribution and impact energy concerns. These changes collectively improve structural survivability, recovery reliability, and alignment with NASA safety requirements while maintaining consistency with the original mission objectives.

2.3 Changes made to Payload

The payload design has evolved from the PDR configuration to improve reliability, simplify integration, and ensure compliance with safety and operational constraints identified during system-level development. The experimental scope was narrowed to focus exclusively on soil pH measurement, allowing the team to concentrate resources on robust execution rather than expanded sensing capabilities.

The payload separation mechanism was redesigned from an initial rack-and-pinion concept to a lead-screw-driven mechanical system. This change enables controlled, ground-based deployment without the use of energetics, improving safety and compliance with NASA requirements. The updated mechanism provides self-locking behavior.

Electrical and software architecture for the payload was refined to support autonomous sequencing, data logging, and time-stamped measurements within the required mission timeline. Mechanical interfaces between the payload and launch vehicle were clarified and standardized to ensure reliable separation and post-landing deployment.

These changes collectively increase payload reliability, simplify testing and verification, and improve system readiness for integrated flight operations.

2.4 Changes made to project plan

Several adjustments were made to the project plan in response to subscale flight test results, schedule constraints, and iterative design refinement. Following the partial failure of the Terra Parvis V1 subscale flight, the team reallocated effort toward recovery system redesign, additional ground testing, and structural material changes to address identified failure modes. This resulted in a revised subscale configuration (Terra Parvis V2).

Additionally, planned subscale flight tests were rescheduled due to weather-related launch cancellations. As a result, the second and third subscale flights were moved to January 10–11 in Samson, Alabama under SEARS operations. The project timeline was adjusted to prioritize verification-critical testing prior to FRR while maintaining compliance with NASA Student Launch requirements. Minor changes to the weekly objectives but mostly consistent with PDR.

3 Vehicle Criteria

3.2 Design and Verification of Launch Vehicle

3.2.1 Mission Statement

The mission of the Terra Nova Project is to design and manufacture a high-powered rocket able to successfully carry and deploy the HAUS payload, the Terra Inquisitor. The Terra Inquisitor will then collect and test a sample of soil. This will be achieved with commercially available and in-house manufactured components. The design of the recovery system is to safely reach ground level within 80 seconds of apogee. The safety of the landing is based on all individual rocket sections sustaining no more than 65 ft-lb of energy and not irreparable damage. The design and manufacturing of all recovery system components shall meet all specified safety criteria and NASA launch rules and guidelines. Custom bulkheads will be used to absorb forces imparted on the recovery system and manufactured using the new form of additive manufacturing, 3D metal printing.

3.2.2 Launch Vehicle Design and Rationale

The optimal design was created through the brainstorming of various design alternatives. These design alternatives were then compared based on the mission of the launch vehicle. The optimal designs were then selected and combined to form the Terra Nova Launch Vehicle.

The selection of the optimal component and/or material of construction of the rocket design was based on the employment of decision matrices. These matrices compare each material or component based on the particular selection criteria for each component and a weight factor associated with each criteria. The scoring was then created by multiplying the rating of the particular material for each criteria and its weight factor. All these scores are then summed up to produce the overall score. The material and/or component with the highest score will be selected for the construction of the rocket.

Technical Design Score Table 3.1	
Definition	Value
Poor	2
Fair	4
Good	6
Great	8
Excellent	10

Table 3-1 Technical Design Scoring

All designs were be based off design requirements (further outlined in 6.1 Derived Requirements) the usage of a primary ID of 6” and the usage of a Aerotech 75/5120 RMS (currently in supplies). This diameter was selected because any larger, weight and drag will become major concerns for a rocket motor that fits a 75/5120 RMS and is an L-class, any smaller and the payload design will be significantly constrained.

3.2.3 Rocket Vehicle Designs:

Nose Cone

Geometry

The primary considerations for the selection of the nose cone geometry are drag and availability. There are many different geometries to consider for the nose cone. The primary considerations include parabolic, elliptical, ogive, Von Karman, and conical. Overall, the drag forces associated do not show extreme variance for speeds below Mach 1 (13). However, there are minor differences that will affect the optimization of the rocket. These minor differences are not fully agreed upon by rocketeers. According to experimental data in "Drag of Nose Cones" by Ashley Milligan, the nose cone order from least drag to most drag is the long elliptical, parabolic, long cone, and ogive (a Von Karman was not included in the test). Though this experiment points to elliptical nose cones being the optimal design, it was limited in scope only testing each geometry at a speed of 39.28 MPH $\approx 57.6\,\text{fps}$. Thus, it is best to further consider other data and calculations before the determination of the nose cone geometry. According to calculations from "TR-11 Model Rocket Technical Report" by Dr Gregorek, the ogive nose cone would outperform the conical for velocities below Mach 1. The other nose cone shapes were not included in Dr Gregorek's report. Due to the variance in the data concerning the optimal nose cone geometry, the selection of the nose cone will be primarily based on commercial availability, weight, and cost. No elliptical nose cones were found commercially available for a 6" ID. Ogive and conical were available on multiple rocketry sites including Madcow Rocketry, Wildman Rocketry, and LOC 40 Precision. The Von Karman was only found available on Wildman Rocketry. Due to the larger availability and the direct calculations comparing conical to ogive conducted by Dr Gregorek, the ogive nose cone will be used for the design of the Terra Nova. The nose cone will have a fineness ration of 4:1 due to no commercially available smaller ratio.

Material of Construction and Cost

The decision matrix used to evaluate nose cone materials (Table 3-2) incorporated scores calculated by the multiplication of the criteria weight and the rating of the material. The ratings are based on a scale of 10 out of 10; the material that performs the best will receive a rating of 10. All other material scores for the particular criteria were rated relative to the best. In the event of all material ratings tie for a particular category, the rating for all will be 10.

4:1 Ogive Nose Cone			Fiberglass			Carbon Fiber		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.25	USD	191	10	2.5	239	7	1.75
Strength	.3	ksi	80.6	9	2.7	87.0	10	3
Density	.25	$\frac{\text{lb}}{\text{in}^3}$	.0650	9	2.25	.0578	10	2.5
Handling Safety	.2	Safety Score	6	10	2	6	10	2
Overall Score:					9.45			9.25

Table 3.2 Ogive Nose Cone Material Decision Matrix

Cost is the price of the component in USD. Cost was taken into consideration as a criteria due to the need to keep the overall cost of the vehicle low. Thus, the weighing factor for cost is 25%. Rating for this criteria was based on the lowest cost being optimal and so receiving a rating of 10.

Strength is the maximum strength in ksi the material can withstand in uniaxial tension. Strength was taken into consideration as a material criteria due the nose cone needing to withstand all the forces of flight and the impact of landing. If the nose cone material were to fail, the rocket would be considered unrecoverable. Hence, the weighing factor for material strength is 30%. Rating for this criteria was based on the highest strength material being considered optimal and thus receiving a score of 10.

Density is weight per unit volume of the material of construction in $\frac{\text{lb}}{\text{in}^3}$. Density was taken into consideration as a criteria due to the need to keep the overall weight of the vehicle low. This is especially true for all rocket components in the front most section of the rocket since the lower the weight the less restriction must be placed on the payload design without exceeding the impact energy requirement. Weight will also effect the center of gravity (CG) of the rocket and so effect rocket stability. Thus, the weight factor for density is 25%. Rating for this criteria was based on the lowest density being considered optimal and so receiving a rating of 10.

Boat-tailing

Justification

One of the greatest contributors to overall rocket drag is base drag. This drag is an imbalance of pressure forces at the base of the rocket that resist its motion. For a rocket the largest pressure differential is from the nose cone to the base of the rocket. This pressure differential is due to the flow separation at the aft of the rocket. This flow separation is a partial vacuum or vortex causing a wake at the base to be a low-pressure zone and thus an imbalance in pressure. Due to these factors, base drag is a function of the area at the base of the rocket and the pressure differential due to the sudden transition at the base of the rocket. According to research conducted by the authors of "SRT-2 Phoenix", base drag accounts for 35-40% of total rocket drag. They note that the incorporation of a diameter reduction has the potential to reduce this drag by nearly 60% (3). This is also corroborated in "TR-11 Model Rocket Technical Report" by Dr. Gerald Gregorek who notes that base drag can account for 30% of total rocket drag (4). Boat-tailing is a common method used to reduce the drag at the base of a rocket by reducing the diameter at the tail end of the rocket. This not only reduces the area the low-pressure wake acts upon, but also lessens flow separation and thus the decreases the pressure differential between the nose cone and base. Dr. Gregorek estimates the % reduction of drag through empirical correlations from experiments outlined in his report. Through this data, he created a figure outlining the percentage of original base drag compared to the ratio of the diameters with a transition angle less than 5 degrees (to ensure the flow will not separate along the boattail).

$$Diameter\ Ratio = \frac{diameter\ at\ end\ of\ boat - tail}{original\ diameter}$$

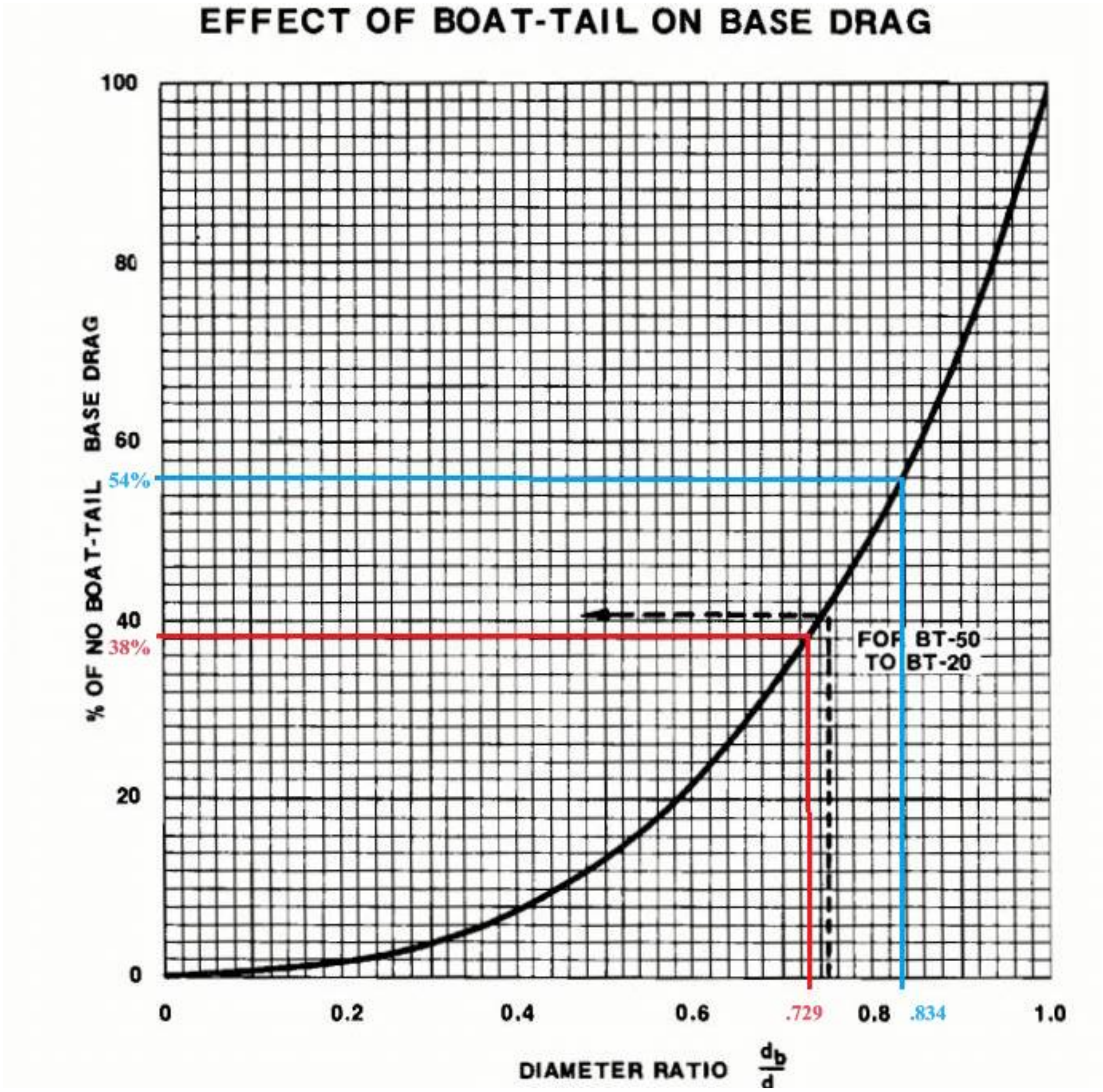


Figure 3-1 Effect of Boat-tail on Base Drag (4).

Using a boat-tail design with a of $Diameter\ ratio = \frac{5.15"}{6.17"} = .834$. According to Figure 3-1, when following the lines in blue lines, this diameter reduction equates to 54% of original base drag. Thus, by using this boat-tail design, the base drag is reduced by 46%. It is important to note that this this reduction of drag is true only for the coasting phases of rocket ascent (after motor burnout). This is due to the exhaust of the motor acting to break the low-pressure vortex attempting to form (3).

Disadvantages to boat-tailing are additional weight, cost, and fabrication time. Overall, with the significant reduction in base drag as well as the fire protection capabilities for the body of the rocket, a boattail will be incorporated in the design of both the full-scale and sub-scale rocket vehicles.

Material of Construction

Now that the inclusion of a boattail has been selected for the design of the Terra Nova's launch vehicle, the material of construction must be selected. The decision matrix used to evaluate nose cone materials (Table 3-3) incorporated scores calculated by the multiplication of the criteria weight and the rating of the material. The ratings are out of 10; the material that performs the best will receive 10 out of 10. All other material scores for the particular criteria were rated relative to the best. In the event of all material ratings tie for a particular category, the rating for all will be 10.

Boat-Tail			Fiberglass			Carbon Fiber		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.15	USD/Volume	1.10	10	1.5	11.7	4	.6
Tensile Strength	.1	ksi	80.6	5.5	.55	87.0	7	.7
Density	.25	$\frac{lb}{in^3}$	.0650	9	2.25	.0578	10	2.5
Manufacturing and Handling Safety	.15	Safety Score	6	6	.9	6	6	.9
Flame Retardant	.2	Flammability Class	B	6	1.2	B	6	1.2
Machinability/ Fabrication	.15	Experience/Available Equipment	3	3	.45	3	3	.45
Overall Score:			6.85				5.17	

Boat-Tail			Inconel 718			Aluminum 6061		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.15	USD/in^3	13.5	3	.45	1.90	9	1.35
Tensile Strength	.1	ksi	139	10	1	51.6	4	.4
Density	.25	$\frac{lb}{in^3}$	.296	3	.75	.0982	8	2
Manufacturing and Handling Safety	.15	Safety Score	3	10	1.5	3	10	1.5
Flame Retardant	.2	Flammability Class	A	10	2	A	10	2

Boat-Tail			Steel 1020		
Criteria	Weight	Parameter	Magnitude	Rating	Score
Cost	.15	USD/Volume	9.5	8	1.2
Tensile Strength	.1	ksi	81	6	.6
Density	.25	$\frac{\text{lb}}{\text{in}^3}$	.284	3.5	.875
Manufacturing and Handling Safety	.15	Safety Score	3	10	1.5
Flame Retardant	.2	Flammability Class	A	10	2
Machinability/ Formability	.15	Experience/Available Equipment	8	8	1.2
Overall Score:					7.375

Machinability/ Formability	.15	Experience/Available Equipment	7	7	1.05	10	10	1.5
Overall Score:					6.75			8.75

Table 3-3 Matrix Boat-tail Material Decision Matrix

Cost is the price of the material in USD/volume. Cost was taken into consideration as a criteria due to the need to keep the overall cost of the vehicle low. Thus, the weighing factor cost is 15%. Rating for this criteria was based on the lowest cost being optimal and so receiving a rating of 10.

Tensile Strength is the maximum strength the material can withstand in uniaxial tension in ksi. Strength was taken into consideration as a material criteria due the boat-tail needing to withstand all forces of flight and the impact of landing. If the boat-tail material were to fail during the recovery of the rocket, the rocket would be considered unrecoverable. However, it is not expected that the boat-tail will be subjected to extreme forces. The greater concern for the material construction is flammability. Hence, the weighing factor for material strength is 10%. Rating for this criteria was based on the highest strength material being considered optimal and thus receiving a score of 10.

Density is weight per unit volume of the material of construction in $\frac{\text{lb}}{\text{in}^3}$. Density was taken into consideration as a criteria due to the need to keep the overall weight of the launch vehicle low. This is true for the boat-tail as its primary function is to decrease base drag. Thus, the heavier the boat-tail is the less advantage is gained by the usage of a boat-tail. The weight will also effect the center of gravity (CG) of the rocket and so effect rocket stability. Since the boat-tail is at the base of the rocket it will lower the CG and thus reduce the stability of the rocket the heavier it is. Based on all these factors, the weight factor of density is 25%. Rating for this criteria was based on the lowest density being considered optimal and so receiving a rating of 10.

Manufacturing and Handling Safety is the safety hazards from the inherent toxicity of the materials of construction and the safety of the various machines required form the material. This is measured based on the Safety scores outlined in Section**Error! Reference source not found.**. Safety is always an important factor for the usage of any components or materials. Thus, the weighing factors 15%. The material with the lowest safety score is considered the safest material and will receive a rating of 10.

Flame Retardant is the flammability class of the material based on classes outline in Appendix C of "Material Selection in Mechanical Design" by Michael Ashby (**Error! Reference source not found.**).(11**Error! Reference source not found.**). Flammability of the material of the boat-tail is important due to the boat-tail being in direct contact with the exhaust of the rocket motor. Due the importance of this criteria to ensure the boat-tail does not receive permanent damage, it was weighted at 20%. The material that received the best class (A) received a rating of 10.

Machinability/Fabrication is measured by experience using the material and available equipment to form the material. This criteria was included due to the materials of construction either requiring the Terra Nova team to custom fabricate the boat-tail or to outsource the fabrication to the Terra Nova team's local machining shop, Maritech Machining Inc. Thus, the formability of the material based on previous experience, available equipment, and available equipment at Maritech Machining is important for the selection of the boat-tail material and so the weight factor is 15%. Rating for this criteria was based on the material that the Terra Nova team has the most experience with and/or Maritech is the best equipped to machine receiving a rating of 10.

Reducing Transition Airframe

Justification

There are many advantages and few disadvantages to the usage of a reducing transitional airframe. The primary disadvantage is availability and hence fabrication time. For the current set requirement of a primary ID of 6" (6.1.1 Derived Vehicle Requirements), there are no available transitions to any other diameter. Thus, the transition would need to be fabricated by the Terra Nova team.

The are four primary advantages to using a reducing transitional airframe weight reduction, streamlining/drag reduction, stability, and payload space. The weight reduction when using a transitional airframe can be calculated by summing the volumes of all body tubes from transition to the aft of the rocket and multiplying those volumes by the density of the material of construction.

According to our calculations, there is a 2.2 lb reduction when using the transition. This is a significant weight reduction, but on its own it does not fully justify the usage of a transition due to time required for custom-fabrication.

In addition to the weight reduction, the addition of a transition would reduce base drag similarly to boat-tailing. This is because the underlying principle of boat-tailing is the reduction of the area at the base of the rocket. By transitioning from a 6.17" OD to a 5.15" OD with a transitioning angle less than 5 degrees, the area at the base of the rocket would be reduced and so the base drag associated would be reduced. Due to the similarity between boat-tailing and reducing transitions, Dr Gregoreks diameter ratio vs % original base drag graph can be used to estimate the reduction of the base drag. Referencing the Figure 3.1 above, with a $\text{Diameter ratio} = \frac{5.15"}{6.17"} = .834$, the reduction in base drag would equate to 54% of the original drag or 46% reduction in base drag. Combining the transition and boat-tail would achieve a $\text{Diameter ratio} = \frac{4.5}{6.17} = .729$. Following the red lines on Figure 3-1 above with the new Diameter ratio, the new base drag will be 38% of the original drag or 62% less bases drag.

Material of Construction

Based in the base drag and weight reduction capabilities of a reducing transition, it will be included in the design of the Terra Nova's launch vehicle, thus its material of construction must be selected. The decision matrix used to evaluate nose cone materials (Table 3-4) incorporates scores calculated by the multiplication of the criteria weight and the rating of the material. The ratings are out of ten; the material that performs the best will receive 10 out of 10. All other material scores for the particular criteria were rated relative to the best. In the event of all material ratings tie for a particular category, the rating for all will be 10.

Reducing Transition			Fiberglass			Carbon Fiber		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.15	$\frac{USD}{in^3}$	1.10	7	1.05	11.7	2	.3
Strength	.35	ksi	80.6	9.75	3.41	87.0	10	3.5
Density	.25	$\frac{\text{lb}}{\text{in}^3}$	.0650	7.5	1.875	.0578	8	1.13
Handling Safety	.15	Safety Score	6	5	.75	6	5	.75
Formability	.10	Experience / Available Equipment	7	7	.7	5	5	.5

Overall Score:					7.89			6.18

Reducing Transition			Creality PETG			Creality PLA		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.15	$\frac{USD}{in^3}$	.387	9.75	1.46	.370	10	1.5
Strength	.35	ksi	7.25	3.5	1.225	7.69	3.75	1.31
Density	.25	$\frac{\text{lb}}{\text{in}^3}$	.0448	10	2.5	.0452	9	2.25
Handling Safety	.15	Safety Score	2	10	1.5	2	10	1.5
Formability	.10	Experience / Available Equipment	10	10	1.5	9.5	9.5	1.35
Overall Score:					7.69			6.66

Reducing Transition			Creality PETG & Fiberglass Composite		
Criteria	Weight	Parameter	Magnitude	Rating	Score
Cost	.15	$\frac{USD}{in^3}$	.637	8.5	1.28
Strength	.35	ksi	7.25	3.5	1.05
Density	.25	$\frac{\text{lb}}{\text{in}^3}$	.0512	8.5	2.13
Handling Safety	.15	Safety Score	6	5	.75
Formability	.10	Experience / Available Equipment	7	7	1.28
Overall Score:					6.49

~~*Table 3 Reducing Transition Material Decision Matrix*~~

Table 3-4 Reducing Transition Material Decision Matrix

Cost is the price of the material in USD/in^3. Cost was taken into consideration as a criteria due to the need to keep the overall cost of the launch vehicle as low as possible. Thus, the weighing factor cost is 15%. Rating for this criteria was based on the lowest cost being optimal and so receiving a rating of 10.

Strength is the maximum strength the material can withstand in uniaxial tension in ksi. Strength was taken into consideration as a material criteria due the transition to withstand the forces during of flight and the impact of landing. If the reducing transition material were to fail during the recovery or launch of the rocket, the rocket would be considered unrecoverable and have great safety concerns. As the transition is an inherent stress concentration, is custom fabricated, and is must be able to absorb considerable energy during all stages of the rocket flight, the material strength is crucial. Also, the overall design of the transition is very constrained to the OD's of both the connecting body tubes and couplers. Thus, the overall selected material must have considerable strength. Based on all these factors, the material strength was weighted as 35%. This value has been increased due to the results of the sub-scale flight to be detailed later. Rating for this criteria was based on the highest strength material being considered the optimal and thus receiving a score of 10.

Density is weight per unit volume of the material of construction in $\frac{\text{lb}}{\text{in}^3}$. Density was taken into consideration as a criteria due to the overall need to keep the weight of the launch vehicle as low as possible. This allowing room for the weight of other components. The weight of the transition will also effect the center of gravity (CG) of the rocket and so effect rocket stability. Thus, the weight factor density is 25%. Rating for this criteria was based on the lowest density being considered optimal and so receiving a rating of 10.

Handling Safety is the safety hazards from the inherent toxicity of the materials of construction. This is measured based on the Safety scores outlined in Section 5.4.4. Safety Data Sheets (SDS): Chemicals and Materials. Safety is always an important factor for the usage of any components or materials. Thus, the weighing factors 15%. The material with the lowest safety score is considered the safest material and will receive a rating of 10. For composite materials, the safety rating was assumed to be the rating of the least safe material of construction.

Formability is measured by experience using the material and available equipment to form the material. This criteria was included since the Terra Nova team will be custom fabricating this reducing transition. Thus, the formability of the material based on previous experience and available equipment is important for the selection of the transition material and so the weight factor is 10%. Rating for this criteria was based on the material that the Terra Nova team has the most experience with and the most equipment to form receiving a rating of 10.

Based on all these selection criteria, a Fiberglass will be used as the material of construction for the transition with the highest overall score of 7.89. This is due to the results of the failure of the transitional airframe decent. The composite of PETG and fiberglass resulted in a material of insufficient strength. In addition to this, the second model of the sub-scale proved the ability of the Terra Nova team to construct a completely fiberglass transition airframe using a printed mold.

Body Tubes and Couplers

No matter the rocket design, body tubes and couplers will be required. Therefore, the primary consideration for the body tubes and couplers will be the material of construction based on commercially available components. The material of construction will be assumed to be the same for both the couplers and the body tubes to ensure they are designed and fabricated to fit together. To effectively select a material of construction a decision matrix will be employed. The decision matrix will be evaluating the available components for a 6" ID of body tube. It is important to note there was no available phenolic, pre-glassed phenolic, or cardboard tubing for both the required IDs of the Terra Nova launch vehicle design. The decision matrix used to evaluate body tube and coupler materials (Table 3-5) incorporates scores calculated by the multiplication of the criteria weight and the rating of the material. The ratings are out of ten; the material that performs the best will receive 10 out of 10. All other material scores for the particular criteria were rated relative to the best. In the event of all material ratings tie for a particular category, the rating for all will be 10.

Body Tube and Coupler			Fiberglass			Carbon Fiber		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.35	$\frac{USD}{in}$	8.26	10	3.5	18.8	4	1.4
Strength	.30	ksi	80.6	8	2.4	87.0	10	3.0
Density	.25	$\frac{lb}{in^3}$	.0650	8.5	2.13	.0578	10	2.5
Handling Safety	.10	Safety Score	6	10	1.0	6	10	1.0
Overall Score:					9.03			7.9

~~Table 3– Body Tube and Coupler Material Decision Matrix~~

Table 3-5 Body Tube and Coupler Material Decision Matrix

Cost is the price of the component per linear inch or USD/in. Cost was taken into consideration as a criteria due to the need to keep the overall cost of the launch vehicle as low as possible. Since the primary components for the Terra Nova launch vehicle are the body tubes and the couplers. Minimizing the cost of the components will significantly reduce the overall cost of the launch vehicle and thus allow for the various costs of other components more niche to the design of the

Terra Nova Project. The cost magnitude for the Body Tube and couplers was calculated by summing the costs of both components after each was divided by their particular lengths. All costs were obtained from the Wildman Rocketry site for a consistent comparison. Due to all these considerations, the weighing factor cost is 35%. Rating for this criteria was based on the lowest cost being optimal and so receiving a rating of 10.

Strength is the maximum strength the material can withstand in uniaxial tension in ksi. Strength was taken into consideration as a material criteria due the all body sections and couplers needing to withstand the forces during of flight and recovery. If any body tube or coupler were to fail during the recovery or launch of the rocket, not only would the rocket be considered unrecoverable, there would also be significant safety concerns. As the body tubes and couplers are the required to be the primary absorbers of the considerable energy during all stages of the rocket flight, the material strength is crucial. Based on all these factors, the material strength was weighted as 30%. Rating for this criteria was based on the highest strength material being considered optimal and thus receiving a score of 10.

Density is weight per unit volume of the material of construction in $\frac{\text{lb}}{\text{in}^3}$. Density was taken into consideration as a criteria due to the overall need to keep the weight of the launch vehicle as low as possible. The weight of the body tubes and couplers are the primary contributors to the overall weight of the launch vehicle weight. Reducing the weight of the body and couplers allows room for the weight of other components. The weight of these components will also effect the center of gravity (CG) of the rocket and so effect rocket stability. Thus, the weight factor density is 25%. Rating for this criteria was based on the lowest density being considered optimal and so receiving a rating of 10.

Handling Safety is the safety hazards from the inherent toxicity of the materials of construction. This is measured based on the Safety scores outlined in Section 5.4.4. Safety Data Sheets (SDS): Chemicals and Materials. Safety is always an important factor for the usage of any components or materials. However, since the safety factors for both materials were exactly the same, the overall relative effect of this criteria on material selection is minimal. Thus, the weighing factor is 10%. The material with the lowest safety score is considered the safest material and will receive a rating of 10.

Based on all these selection criteria, fiberglass is the selected material for the design of the body tubes and couplers with the highest score of 9.03.

Bulkheads

Bulkheads will be required for any rocket design; the primary considerations are material of construction and the particular requirements set by the Terra Nova team. Before the material can be defined, the derived requirement by the Terra Nova team is to use 3D printed bulkheads. The particular form of 3D printing being considered is metal 3D printing. Metal 3D printing is a newer form of additive manufacturing. The only material accessible to the Terra Nova team is Inconel 718. Not only is this form of manufacturing available to the team, the material is also available through Florida State University Panama City's InSPIRE program. Through this program both the costs associated with manufacturing and inherent material costs are covered; there would be no incurred cost to the Terra Nova Project through the usage of this additive manufacturing. This design includes the testing of 3D printed Inconel 718 to ensure the material properties based the particular methodologies of printing employed by Maritech are sufficient for the Terra Nova. To further evaluate the usage of this material, a decision matrix will be employed comparing the various 3D printable materials available to the Terra Nova team. Then a comparison of the strength to weight characteristics between Inconel 718 and Aluminum 6061-T6 will be conducted to ensure the usage of Inconel does not increase the overall weight of the rocket as compared to a lighter weight metal. The decision matrix used to evaluate bulkhead materials (Table 3-6) incorporates scores calculated by the multiplication of the criteria weight and the rating of the material. The ratings are out of ten; the material that performs the best will receive 10 out of 10. All other material scores for the particular criteria were rated relative to the best. In the event of all material ratings tie for a particular category, the rating for all will be 10.

Bulkheads			Creality PETG			Creality PLA		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.25	USD/Volume	.387	2	.4	.370	2	.4
Tensile Strength	.3	ksi	2	3.5	1.05	2	3.75	1.05
Density	.3	$\frac{lb}{in^3}$	.0448	10	3	.0452	9	2.7
Handling Safety	.15	Safety Score	2	10	1.5	2	10	1.5
Overall Score:					5.95			5.65

Bulkheads			Inconel 718		
Criteria	Weight	Parameter	Magnitude	Rating	Score
Cost	.25	USD/in^3	N/A	10	2.5

Tensile Strength	.3	ksi	139	10	3
Density	.3	$\frac{\text{lb}}{\text{in}^3}$	.296	2	.6
Handling Safety	.15	Safety Score	3	8	1.2
Overall Score:					7.3

~~*Table 3 Bulkhead Material Decision Matrix*~~

Table 3-6 Bulkhead Material Decision Matrix

Cost is the price of the material per volume or USD/in^3 of material. Cost was taken into consideration as a criteria due to the need to keep the overall cost of the launch vehicle as low as possible. Due to this, the weighing factor cost is 25%. Rating for this criteria was based on the lowest cost incurred by the Terra Nova team being optimal and so receiving a rating of 10.

Strength is the maximum strength the material can withstand in uniaxial tension in ksi. Strength was taken into consideration as a criteria since the bulkheads are the primary component that must directly absorb the forces of recovery. If any bulkhead were to fail during the recovery or launch of the rocket, not only would the rocket be considered unrecoverable, there would also be critical safety concerns. Based on all these factors, the material strength was weighted as 30%. Rating for this criteria was based on the highest strength material being considered optimal and thus receiving a score of 10.

Density is weight per unit volume of the material of construction in $\frac{\text{lb}}{\text{in}^3}$. Density was taken into consideration as a criteria due to the overall need to keep the weight of the launch vehicle as low as possible. The weight bulkheads, depending on the material of construction, can become significant contributors to the overall weight of the launch vehicle weight. Reducing the weight of the bulkheads allows room for the weight of other components. The weight of these components will also effect the center of gravity (CG) of the rocket and so effect rocket stability. Thus, the weight factor density is 25%. Rating for this criteria was based on the lowest density being considered optimal and so receiving a rating of 10.

Handling Safety is the safety hazards from the inherent toxicity of the materials of construction. This is measured based on the Safety scores outlined in Section 5.4.4. Safety Data Sheets (SDS): Chemicals and Materials. Safety is always an important factor for the usage of any components or materials. However, since the safety factors for both materials were exactly the same, the overall relative effect of this criteria on material selection is minimal. Thus, the weighing factor is 10%.

The material with the lowest safety score is considered the safest material and will receive a rating of 10.

Based on all these selection criteria, Inconel 718 is the selected material for the design of the bulkheads with the highest score of 7.3.

Further justification of Inconel 718 must be provided by comparing this material to other metal lighter weight metals. This will ensure that usage of Inconel does not increase weight in comparison to the ensuring strength. Inconel 718, when un-heat-treated, has an ultimate tensile strength of 958 MPa (14), and a density of around .296 $\frac{\text{lb}}{\text{in}^3}$ (15). Aluminum 6061-T6 has an ultimate tensile strength of 260 MPa (16) and a density of around .0982 $\frac{\text{lb}}{\text{in}^3}$ (this density was calculated using the percentages of all alloying elements (16) and the densities of each element (17)). Assuming a 1in cube of material:

The strength to weight ratio of Inconel 718 = $\frac{958\ MPa\ x\ \frac{1\ ksi}{6.895\ MPa}}{.296\ \frac{\text{lb}}{\text{in}^3}\ x\ 1\ in^3} = 469\frac{ksi}{lb}$.

The strength to weight ratio of Aluminum 6061-T6 = $\frac{260\ MPa\ x\ \frac{1\ ksi}{6.895\ MPa}}{.0982\ \frac{\text{lb}}{\text{in}^3}\ \text{x 1 in}^3} = 384\frac{ksi}{lb}$.

The ratio between the two materials = $\frac{469\frac{ksi}{lb}}{384\frac{\text{ksi}}{\text{lb}}} = 1.22$.

Based on these calculations, Inconel 718 is 1.22 times stronger per pound than a heat-treated aluminum. Therefore, using Inconel 718 for our bulkheads would reduce the overall weight of the rocket without compromising its strength.

The last consideration for the usage of Inconel 718 is whether the material should be heat-treated. Heat-treating of this material is an important consideration because of the inherent porosity of 3D printed metals for power bed fusion. Heat-treating would greatly increase the strength and isotropy of the Inconel and thus less material would be required for the same strength. However, there is shrinkage of the material. According to Maritech the Inconel can be expected to shrink by 20%. Thus, dimensional accuracy of the printed material will be a consideration when selecting heat-treated vs non-heat-treated. However, this will depend on the experimental mechanical properties determined from the research of two of the Terra Nova's team, Safety Officer Jonathan Serbest and Payload Systems Officer Case Dyer. They are engaged in a separate project to determine the mechanical properties of Inconel 718 in uniaxial tension and the more application based properties of Inconel 718 in torsion. Further details regarding this project are included in **Error! Reference source not found.**.

Fin Design

Alternate Designs

The design and fabrication of the Terra Nova's fins is essential in ensuring a successful flight. The primary considerations for fin design is due to its effects on rocket stability, drag, and structural integrity. Stability is measured by the distance between the center of pressure (CP) and the center of gravity (CG) with the CP being below the CG for a stable rocket. It is required by HB Req. 2.3 that the stability of the rocket be above 2 Cal. There are many different fin shapes including ellipsoidal, delta, clipped delta, trapezoidal, swept, tapered swept, etc. The theoretical best fin shape for stability and lowest induced drag is the ellipsoidal (5). However, it was noted that after simulations combining the fins with a rocket nose cone and body, the ellipsoidal performed worse than trapezoidal and clipped delta (5). Structurally trapezoidal fins are superior to both the delta, clipped delta and swept with minimal loss in stability (6). Considerations of weight of the fins will primarily be due to material construction not selected geometry.

The number of fins used in model rocketry is generally 3-4 (6). More can be used but are often unnecessary. The number of fins will be determined by the desired stability margin and selected fin geometry. One primary consideration for the number of fins is pertaining to the fabrication and affixing the fins to the rocket. More fins requires more fins to be cut, post-processed, possibly additional weight, more fin slots (if fin slots are used), more time fixing fins to rocket body, etc. Thus, as the number of fins increases the more likely there will be variance fabricated fin shape and also the position of the fin on the body tubes.

With all these factors noted, the selected fin design is a "clipped" trapezoid. The overall trapezoidal shape was selected due to is simple geometry, structural integrity, and sufficient lift and drag affects. The simple geometry ensures the fin design is feasible using the facilities available (listed in Appendix A). The "clipping" of the trapezoidal geometry was to reduce the weight of the fins and due to the observed superior stability and drag performance based on simulation data in Open Rocket.

Material of Construction

Now that the fin shape is selected, the material of construction for the fins must be defined. To define the material of construction for the fins a decision matrix was employed. The decision matrix to evaluate the materials (Table 3-7) incorporates scores calculated by the multiplication of the criteria weight and the rating of the material. The ratings are out of ten; the material that performs the best will receive 10 out of 10. All other material scores for the particular criteria were rated relative to the best. In the event of all material ratings tie for a particular category, the rating for all will be 10.

Custom Fins			Fiberglass			Carbon Fiber		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.15	$\frac{USD}{in^3}$	1.10	7	1.05	11.7	2	.3
Strength	.40	ksi	80.6	9.75	3.9	87.0	10	4
Density	.20	$\frac{\text{lb}}{\text{in}^3}$	.0650	6	1.2	.0578	7	1.4
Handling Safety	.15	Safety Score	6	5	.75	6	5	.75
Formability	.10	Experience / Available Equipment	7.5	7.5	.75	7.5	7.5	.75
Overall Score:					7.65			7.2

Table 3-7 Material of Construction Matrix

Custom Fins			Creality PETG			Creality PLA		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.15	$\frac{USD}{in^3}$	.387	9.75	1.46	.370	10	1.5
Strength	.40	ksi	7.25	3.5	1.4	7.69	3.5	1.4
Density	.20	$\frac{\text{lb}}{\text{in}^3}$	.0448	10	2	.0452	9	1.8
Handling Safety	.15	Safety Score	2	10	1.5	2	10	1.5
Formability	.10	Experience / Available Equipment	10	10	1.0	9	9	.9
Overall Score:					7.36			7.1

~~*Table 3- Custom Fin Material Decision Matrix*~~

Table 3-8 Custom Fin Material Decision Matrix

Table 3-7 Material of Construction Matrix

Cost is the price of the material in USD/in^3. Cost was taken into consideration as a criteria due to the need to keep the overall cost of the fins as low as possible. However, there are only four fins thus the overall effect of the cost of the fins is less significant than other criteria. Thus, the weighing factor cost is 15%. Rating for this criterion was based on the lowest cost being optimal and so receiving a rating of 10.

Strength is the maximum strength the material can withstand in uniaxial tension in ksi. Strength was taken into consideration as a material criterion since the fins are a critical component of the rocket for stability. If the fins were to fail during the flight to apogee the flight of the rocket would become unpredictable and thus risk the success accuracy of the predicted rocket height. If the fins were to fail upon impact of the ground the rocket would be considered unrecoverable and thus fail its mission. Thus, the material strength for the fins was weighted as 40%. Rating for this criterion was based on the highest strength material being considered optimal and thus receiving a score of 10.

Density is weight per unit volume of the material of construction in $\frac{\text{lb}}{\text{in}^3}$. Density was taken into consideration as a criteria due to the overall need to keep the weight of the launch vehicle as low as possible. The weight of the fins will have a significant but not large effect on the weight of the rocket. Optimizing the weight of the fins allows room for the weight of other components. The weight of these components will also effect the center of gravity (CG) of the rocket and so have an additional effect rocket stability. Thus, the weight factor density is 20%. Rating for this criterion was based on the lowest density being considered optimal and so receiving a rating of 10.

Handling Safety is the safety hazard from the inherent toxicity of the materials of construction. This is measured based on the Safety scores outlined in Section 5.4.4. Safety Data Sheets (SDS): Chemicals and Materials. Safety is always an important factor for the usage of any components or materials. Thus, the weighing factors 15%. The material with the lowest safety score is considered the safest material and will receive a rating of 10.

Formability is measured by experience using the material and available equipment to form the material. This criterion was included due to the fins being custom fabricated by the Terra Nova team. Thus, the formability of the material based on previous experience and available equipment is important for the selection of the fin material. However, since the overall shape of the fins are not complex difficulty of forming the fins will not be a major challenge, hence, the weighing factor is 10%. Rating for these criteria was based on the material that the Terra Nova team has the most experience with and the most equipment to form receiving a rating of 10.

Based on all these selection criteria, fiberglass is the selected material for the design of the fins with the highest score of 7.65.

Recovery System Location

The avionics bay is the center of the recovery system with both separations happening at either end of the bay. Initially it was set to be in the middle of the rocket, just aft of the transition. This location is fine, but we have found an issue with the impact energy of the forward section and the only way we see to solve this is by rearranging the placement of mass in the body of the rocket. Doing this required the rearrangement of several components, but in terms of the avionics bay, it was moved slightly forward from just aft of the transition, to inside the transition. This change will effectively decrease the mass in the forward separating section enough to keep our ground impact energy within the constraints of the competition without increasing the mass of any other section beyond what is acceptable.

Centering Rings

The centering rings of the rocket are an important component. They are used to hold the motor tube for the RMS. The important consideration for the centering ring design is its material of construction. To evaluate which material should be used a decision matrix was employed. The decision matrix to evaluate the materials (Table 3-8) incorporates scores calculated by the multiplication of the criteria weight and the rating of the material. The ratings are out of ten; the material that performs the best will receive 10 out of 10. All other material scores for the particular criteria were rated relative to the best. In the event of all material ratings tie for a particular category, the rating for all will be 10.

Centering Rings			Fiberglass			Carbon Fiber		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.25	$\frac{USD}{in^3}$	1.10	7	1.75	11.7	2	.5
Strength	.20	ksi	80.6	9.75	1.95	87.0	10	2
Density	.30	$\frac{\text{lb}}{\text{in}^3}$	.0650	6	1.8	.0578	7	2.1
Handling Safety	.15	Safety Score	6	5	.75	6	5	.75
Formability	.1	Experience / Available Equipment	7.5	7.5	.75	7.5	7.5	.75
Overall Score:					7.0			6.1
Centering Rings			Creality PETG			Creality PLA		

Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.25	$\frac{USD}{in^3}$	.387	9.75	2.44	.370	10	2.5
Strength	.20	ksi	7.25	3.5	.7	7.69	3.5	.7
Density	.30	$\frac{\text{lb}}{\text{in}^3}$	.0448	10	3	.0452	9	2.7
Handling Safety	.15	Safety Score	2	10	1.5	2	10	1.5
Formability	.1	Experience / Available Equipment	10	10	1.0	9	9	1.0
Overall Score:					8.64			8.4

Table 3-1 Centering Ring Material Decision Matrix

Cost is the price of the material in USD/in^3. Cost was taken into consideration as a criteria due to the need to keep the overall cost of the centering rings as low as possible. Thus, the weighing factor cost is 15%. Rating for this criteria was based on the lowest cost being optimal and so receiving a rating of 10.

Strength is the maximum strength the material can withstand in uniaxial tension in ksi. Strength was taken into consideration as a material criteria since the ability for the centering rings to hold the motor tube in place is critical. However, the forces during flight and recovery acting on the centering rings is minimal since most of the forces will be exerted on the bulkheads and airframe. Thus, the material strength was weighted as 20%. Rating for this criteria was based on the highest strength material being considered optimal and thus receiving a score of 10.

Density is weight per unit volume of the material of construction in $\frac{\text{lb}}{\text{in}^3}$. Density was taken into consideration as a criteria due to the overall need to keep the weight of the launch vehicle as low as possible. The weight of the centering rings will have a significant but minimal effect on the weight of the rocket. Optimizing the weight of all the components allows room for the weight of other components. The weight of these components will also effect the center of gravity (CG) of the rocket and so have an additional effect rocket stability. Thus, the weight factor density is 30%. Rating for this criteria was based on the lowest density being considered optimal and so receiving a rating of 10.

Handling Safety is the safety hazards from the inherent toxicity of the materials of construction. This is measured based on the Safety scores outlined in Section 5.4.4. Safety Data Sheets (SDS): Chemicals and Materials. Safety is always an important factor for the usage of any components or materials. Thus, the weighing factor is 15%. The material with the lowest safety score is considered the safest material and will receive a rating of 10.

Formability is measured by experience using the material and available equipment to form the material. This criterion was included due to the centering rings being fabricated by the Terra Nova team. Thus, the formability of the material based on previous experience and available equipment is important for the selection of the centering ring material. However, since the overall geometry of the centering rings is not complex, forming the centering rings will not be a major challenge, hence, the weighing factor is 10%. Rating for these criteria was based on the material that the Terra Nova team has the most experience with and the most equipment to form receiving a rating of 10.

Based on all these selection criteria, PETG is the selected material for the design of the centering rings with the highest score of 8.64.

Motor Tubing

Similarly to the centering rings, the motor tube of the rocket primarily depends on material construction. The motor tube is important for the proper installation of the RMS. Hence, to effectively select the material of construction a decision matrix was employed based on commercially available components based on the 75/5120 RMS. The decision matrix to evaluate the materials (Table 3-9) incorporates scores calculated by the multiplication of the criteria weight and the rating of the material. The ratings are out of ten; the material that performs the best will receive 10 out of 10. All other material scores for the particular criteria were rated relative to the best. In the event of all material ratings tie for a particular category, the rating for all will be 10.

Motor Tube			Fiberglass			Cardboard		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.35	$\frac{USD}{in}$	2.10	4	1.4	.682	10	3.5
Strength	.20	ksi	25.4	10	2.0	4.21	4	.8
Density	.30	$\frac{\text{lb}}{\text{in}^3}$	.0650	6	1.8	.0450	10	3
Handling Safety	.15	Safety Score	6	6	.9	2	10	1.5
Overall Score:					6.1			8.8
Motor Tube			Phenolic					

Criteria	Weight	Parameter	Magnitude	Rating	Score
Cost	.25	$\frac{USD}{in^3}$	.716	9	2.25
Strength	.20	ksi	13.5	7.5	1.5
Density	.30	$\frac{\text{lb}}{\text{in}^3}$	.0450	10	3
Handling Safety	.15	Safety Score	2	10	1.5
Overall Score:					8.25

Table 3-2 Motor Tube Material Decision Matrix

Cost is the price of the component per linear unit in USD/in. Cost was taken into consideration as a criteria due to the need to keep the overall cost of the motor tube as low as possible. Due to this, cost was considered the most important criteria relative to the others with a weighing factor is 35%. Rating for this criteria was based on the lowest cost being optimal and so receiving a rating of 10.

Strength is the maximum compressive strength the material can withstand in uniaxial compression in ksi. Strength was taken into consideration as a material criteria since the ability for the motor tube to hold the RMS in place is critical. However, the motor tube is not the primary support for the RMS and so will experience minimal forces during flight. This is because the RMS will be supported against the Propulsion bulkhead. Thus, the material strength for the motor tube was weighted as 20%. Rating for this criteria was based on the highest strength material being considered optimal and thus receiving a score of 10.

Density is weight per unit volume of the material of construction in $\frac{\text{lb}}{\text{in}^3}$. Density was taken into consideration as a criteria due to the overall need to keep the weight of the launch vehicle as low as possible. The weight of the motor tube will have a significant but minimal effect on the weight of the rocket. The weight of the motor tube will also effect the center of gravity (CG) of the rocket and so have an additional effect rocket stability. Thus, the weight factor density is 30%. Rating for this criteria was based on the lowest density being considered optimal and so receiving a rating of 10.

Handling Safety is the safety hazards from the inherent toxicity of the materials of construction. This is measured based on the Safety scores outlined in Section 5.4.4. Safety Data Sheets (SDS): Chemicals and Materials. Safety is always an important factor for the usage of any components or materials. Thus, the weighing factors 15%. The component with the lowest safety score is considered the safest material and will receive a rating of 10.

Based on all these selection criteria, cardboard is the selected material for the design of the motor tube with the highest score of 8.8.

Separation Points

The Terra Nova Rocket will separate at 3 different locations: 2 during recovery and 1 after landing (for the payload, no pyrotechnics involved). Initially, these locations were aft of the transition and forward of the motor housing. With the potential changes listed above, the new separation points would be forward, rather than aft, of the transition, while the aft separation point would not change.

Location of Payload

The payload will be housed in Body Tube 1 and the HAUS coupler located above the reducing transition. This location of the payload was part of the justification for the usage of the reducing transition. This transition allows the rocket design to achieve its required apogee using the predetermined RMS without further limitations being imposed on the payload design.

Rail Buttons

Rail buttons are a critical component of the rocket to ensure the initial flight trajectory as close to desired as possible. The rail buttons are necessary protrusions from the body of the rocket and will be located beneath the CG of the rocket per Handbook Req. 2.13. Overall, there are many different possible locations for the rail buttons. The first rail button will be located a diameter's length (for the larger ID) below the CG as is commonly recommended by rocketeers. The second will be located within 4 inches of the last airframe of the rocket. Due to the usage of the reducing transition, the usage of standoffs will be necessary to offset the rail buttons from the airframe by half an inch. This will ensure the larger diameter airframe will not make significant contact with the launch rail and reduce the risk of both ripping off the rail buttons or strange flight trajectories.

Fasteners

Fasteners will be used throughout the entire rocket but are primarily relevant in attaching bulkheads and other primary components to the airframe. The highest exerted force on the fasteners will be during deployment, so the material selected must not come within a safety factor of rupture during recovery. The forces exerted during deployment of the reefed main, and the reefing line rupture are calculated in section 3.3 but will be refenced here for justification. The force created by the drogue deployment was neglected in calculations because the force created is significantly lower than either of those created by the main events.

$$F_{main,reefed} = 282.74\ lbs$$
$$F_{main,unreefed} = 643.58\ lbs$$

The highest force exerted during deployment is 643.58 lbs. Assuming the entire load is applied to a single bolt presents the absolute worst-case scenario. The selected fasteners will likely be 0.25" bolts, though a force analysis will allow specific material justification. Materials up for consideration are 2024 Al, 6061 Al, and stainless steel 316. With Mohr's Circle, it can be estimated that the maximum shear strength is half that of the maximum tensile strength

Fastener Material			2024 Aluminum			6061 Aluminum		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.20	$\frac{USD}{fastener}$	Quote Pending	6	1.2	Quote Pending	6	1.2
Strength	.40	ksi	34.0	7	2.8	22.5	5	2
Density	.25	$\frac{\text{lb}}{\text{in}^3}$	0.1	8	1.6	0.0975	10	2
Handling Safety	.15	Safety Score	3	10	1.5	3	10	1.5
Overall Score:					7.1			6.7

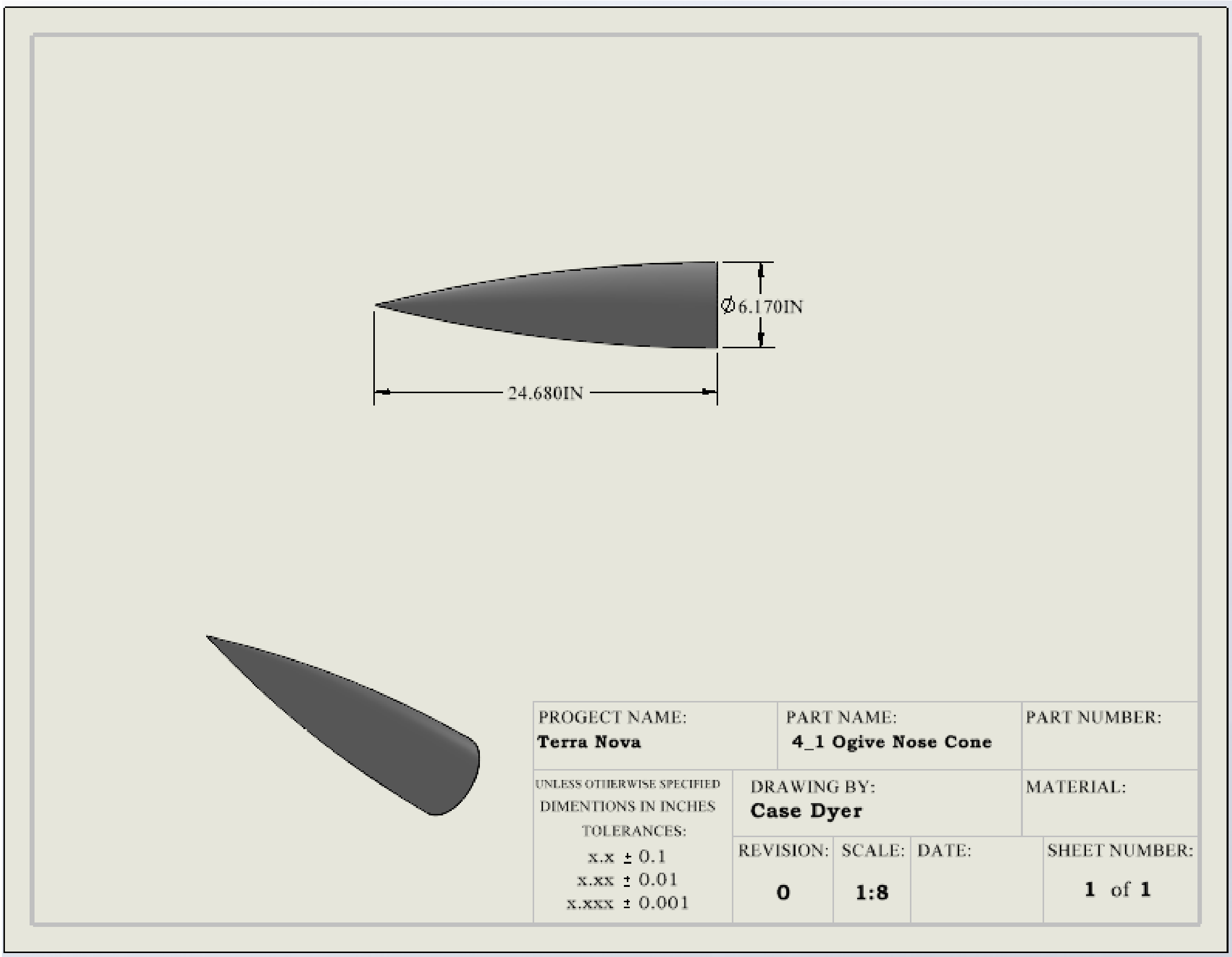
Fastener Material			Stainless Steel 316		
Criteria	Weight	Parameter	Magnitude	Rating	Score
Cost	.20	$\frac{USD}{fastener}$	$15.0	10	2
Strength	.40	ksi	42	10	4
Density	.25	$\frac{\text{lb}}{\text{in}^3}$	0.289	5	1.25
Handling Safety	.15	Safety Score	3 10	1.5	
Overall Score:					8.63

Table 3-3 Fastner Material

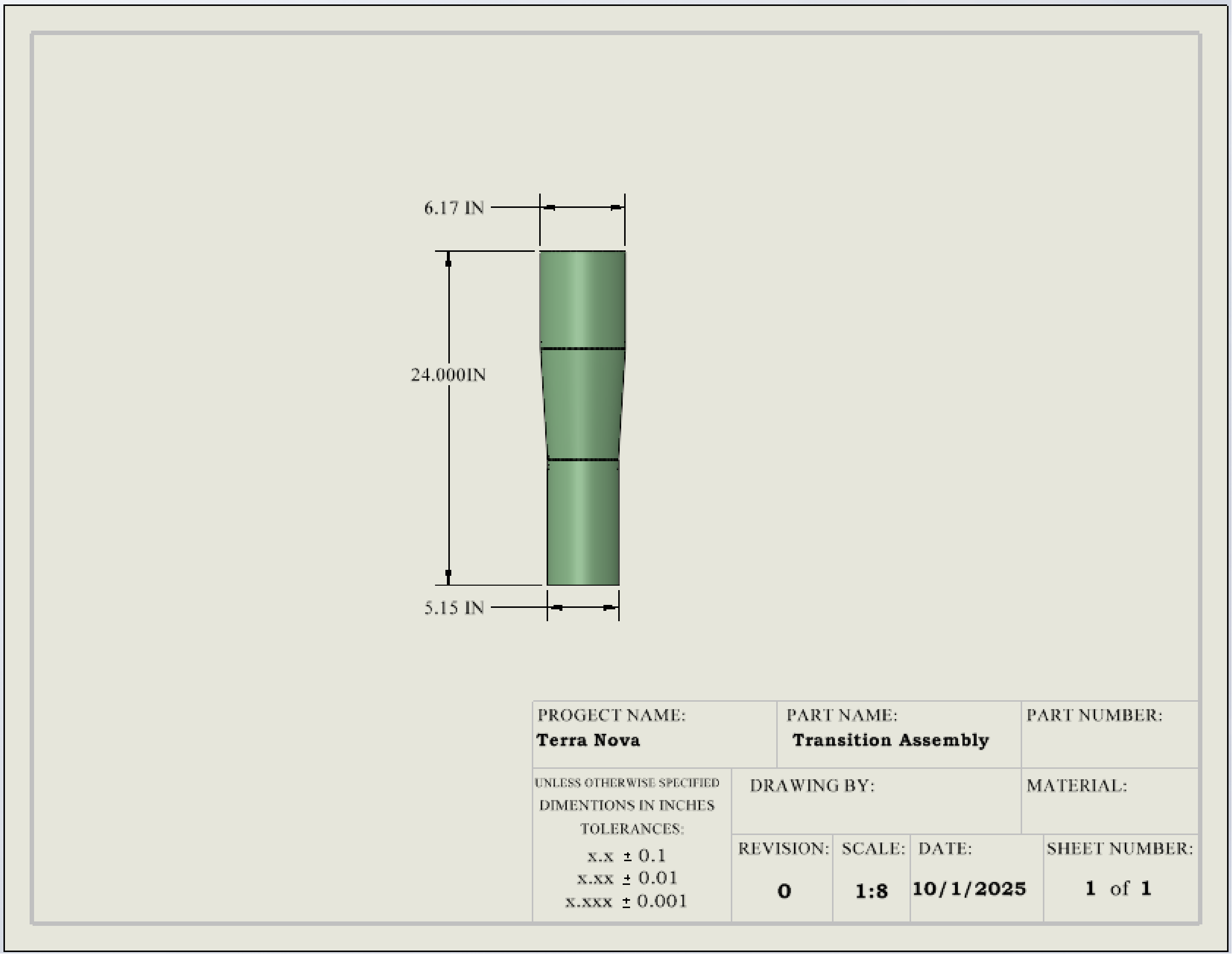
3.2.4 Vehicle Drawings & Dimensions

Below are the detailed drawing of the launch vehicle airframe and airframe related internal components.

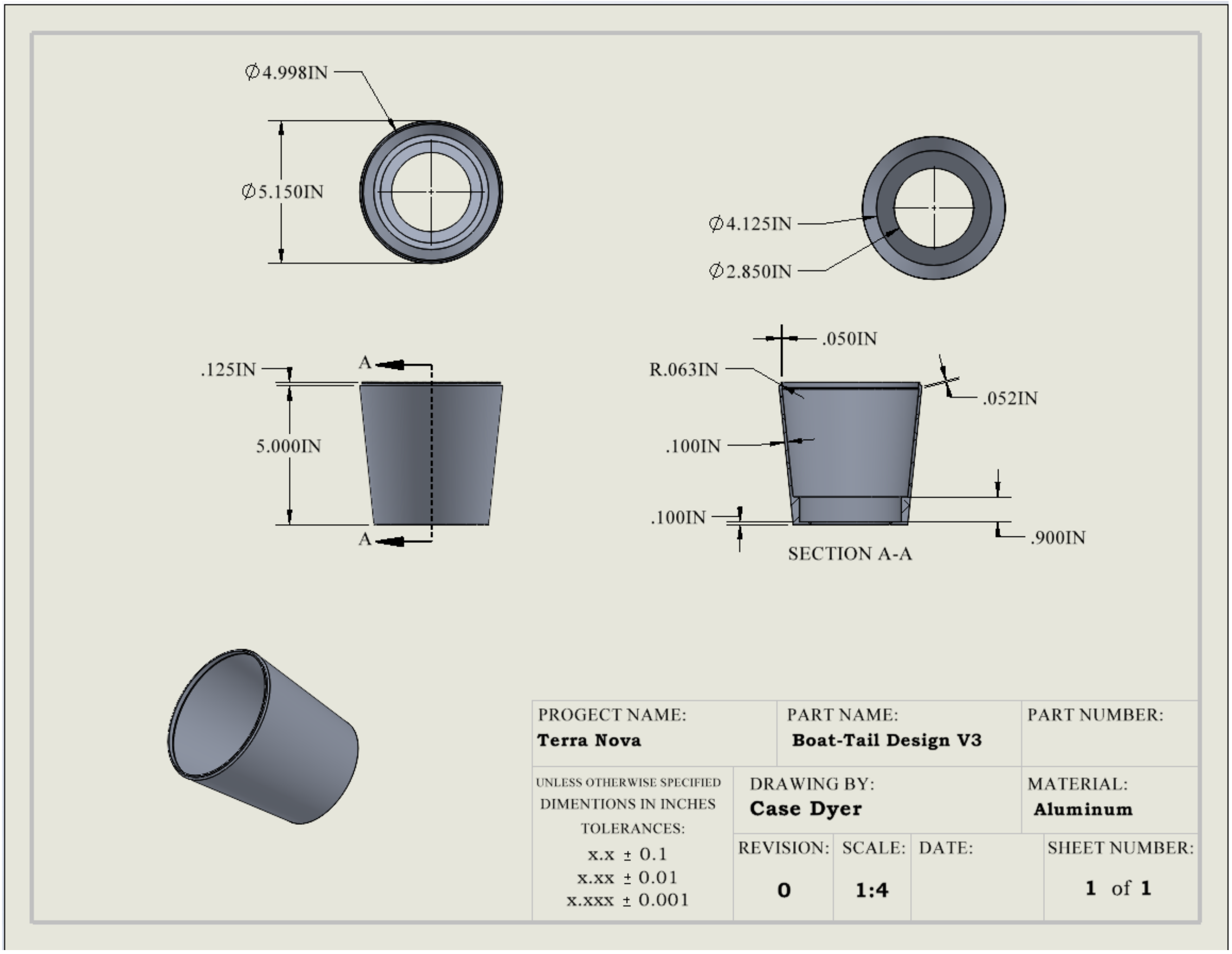
3.2.5 Nose Cone Drawings



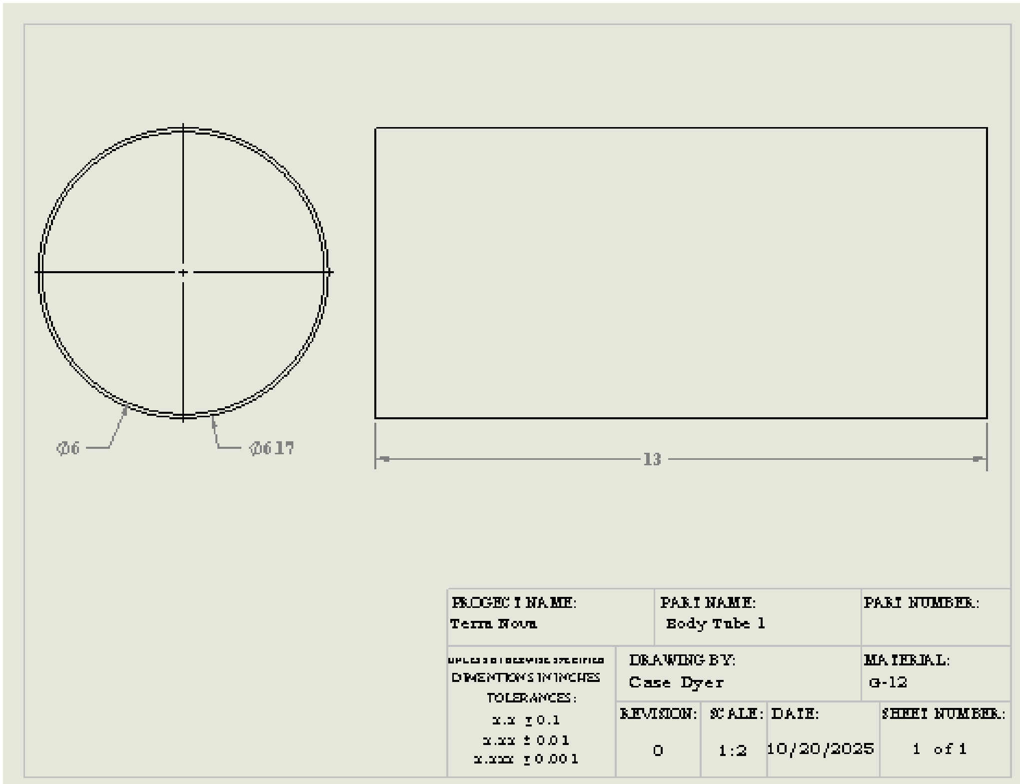
3.2.6 Transition

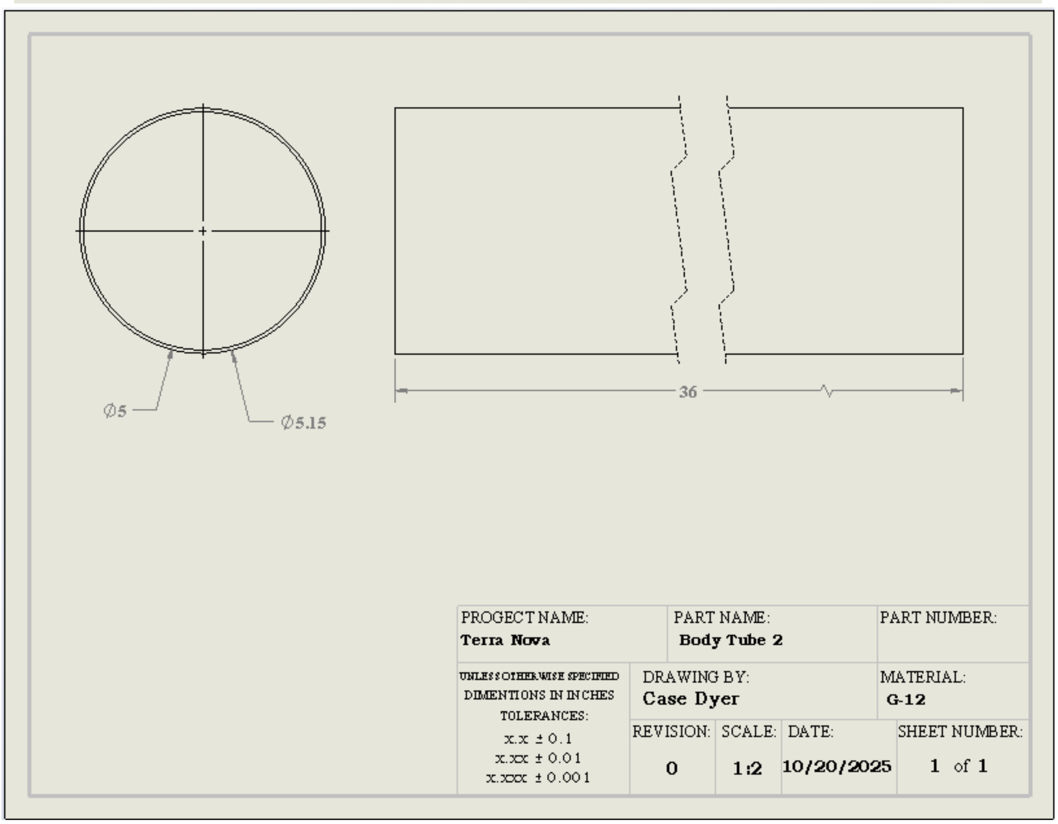


3.2.7 Boat-tail Drawings

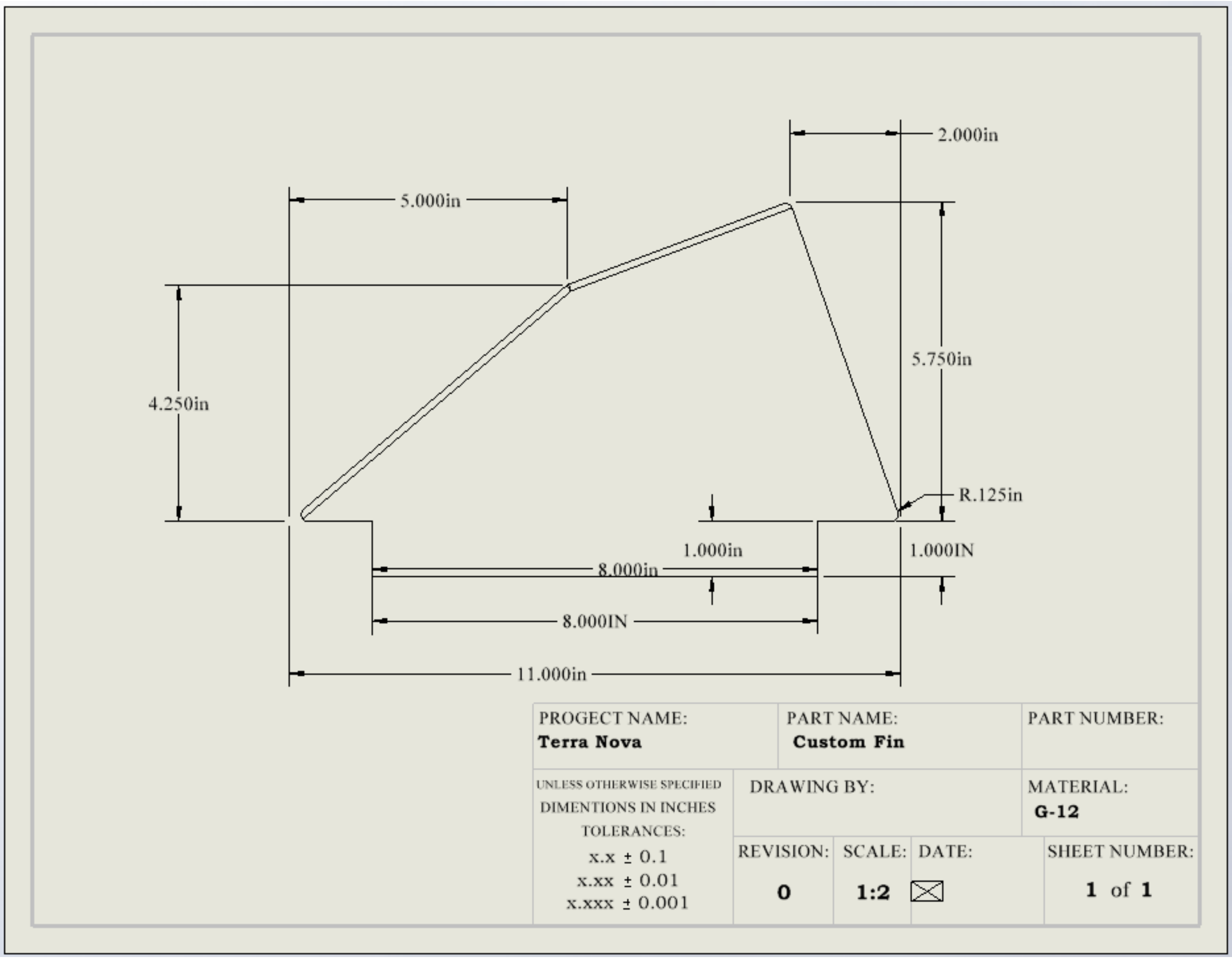


3.2.8 Body Tubes and Couplers

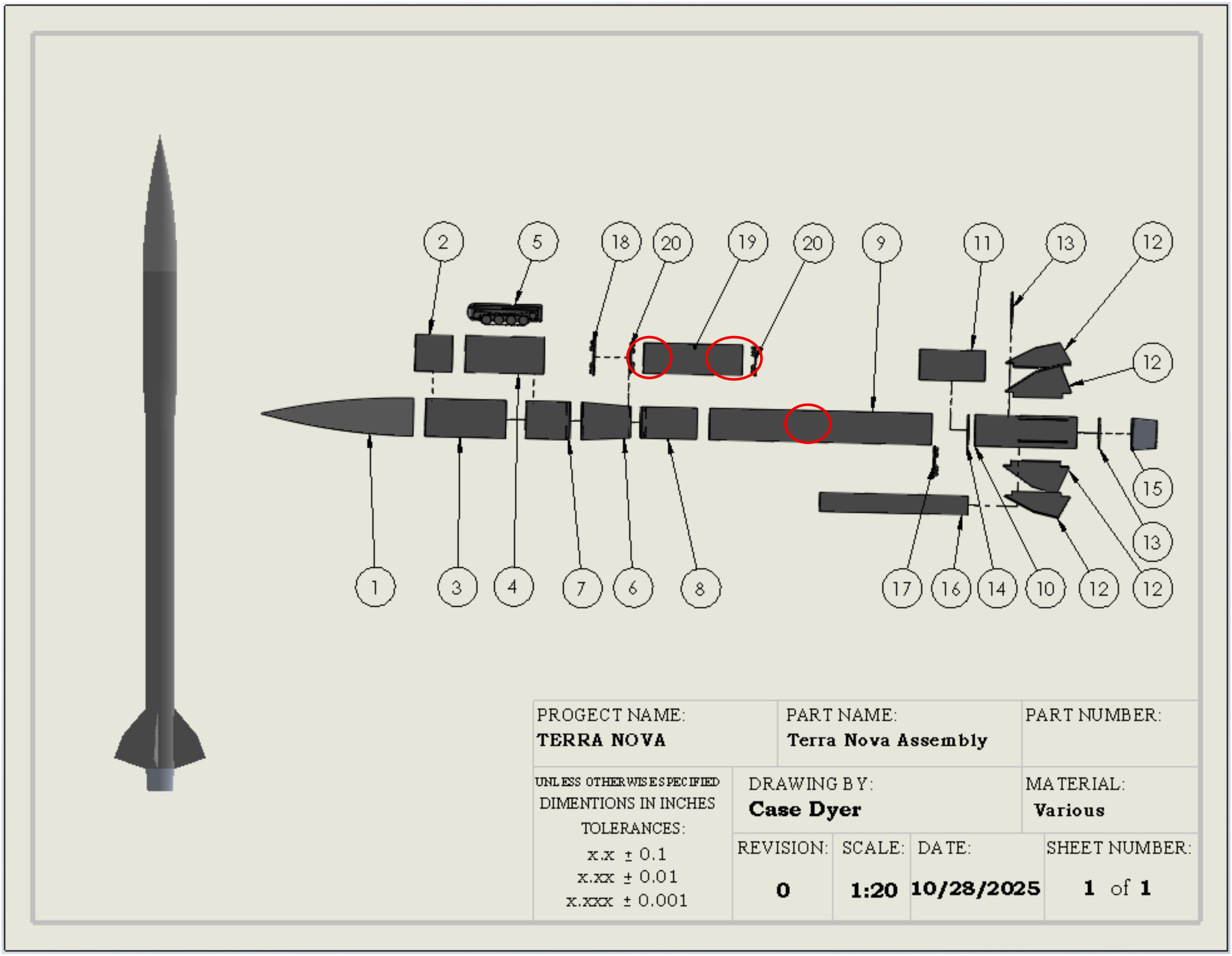




3.2.9 Fins


5.000in
2.000in
4.250in
5.750in
R.125in
1.000in
1.000IN
8.000in
8.000IN
11.000in
PROGECT NAME:
Terra Nova
PART NAME:
Custom Fin
PART NUMBER:
UNLESS OTHERWISE SPECIFIED
DIMENTIONS IN INCHES
TOLERANCES:
x.x ± 0.1
x.xx ± 0.01
x.xxx ± 0.001
DRAWING BY:
MATERIAL:
G-12
REVISION:
0
SCALE:
1:2
DATE:
SHEET NUMBER:
1 of 1

3.2.10 Full Assembly



= Energetics Locations

Terra Nova Launch Vehicle BOM		
Part Number	Part Name	Quantity
1	4:1 Ogive Nose Cone	1
2	Nose Cone Coupler	1
3	Body Tube 1	1
4	Payload Coupler	1
5	Terra Inquisitor	1
6	Reducing Transition	1
7	Upper Transition Airframe	1
8	Lower Transition Airframe	1
9	Body Tube 2	1
10	Body Tube 3	1

11	Propulsion Separation Coupler	1
12	Clipped Trapezoidal Fins	4
13	Motor Tube to Body Tube 3 Centering Rings	2
14	Motor Tube to Propulsion Coupler Centering Ring	1
15	Boat-Tail	1
16	Motor Tube	1
17	RMS Bulkhead	1
18	HAUS Section Bulkhead	1
19	Avionics Bay Coupler	1
20	Avionics Bay Bulkheads	2

3.3 Subscale Flight Results

3.3.1 Subscale Flight Overview

The flight of the Terra Parvis V1 failed its overall objective, though the recovered data paved the way for significant design improvements for its V2 counterpart as well as lessons learned that can be applied to the full-scale Terra Nova. The boost phase of the launch was a great success, though the recovery saw several consecutive failures including a failed drogue deployment, oversized main parachute, and failed bulkhead surfaces. These failures only served to force the team to double down, preparing a second subscale over the course of the following month, addressing all failure surfaces and preparing further for other possible failure modes. Improvements include using plywood for the V2 subscale bulkheads, constructing the entire airframe from fiberglass, resizing the main parachute to 48”, and removing the 3D printed mold transition entirely from the airframe by using it as a mold only and making the entirety of the finished transition from fiberglass rather than PETG and fiberglass. We initially planned to relaunch on the 01/03/26, before the CDR due date, though the launch was cancelled due to weather. We now plan to launch on Saturday 01/10/26, though this launch is likely to be rained out as well. If rain does cancel the launch, our local Launch Control Officer, John Hansel, has stated he is willing to host a rescheduled launch for USLI teams on 01/11/26.

3.3.2 Recorded Launch Data

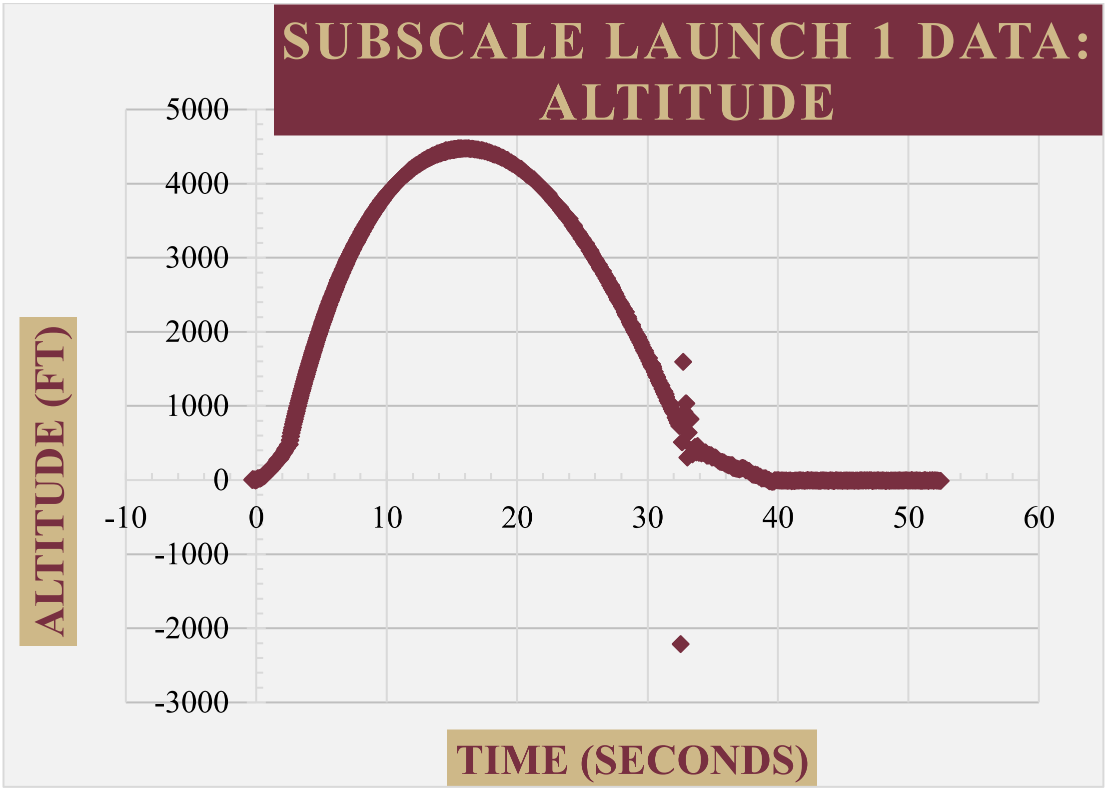


Figure 3-2 Altitude Data

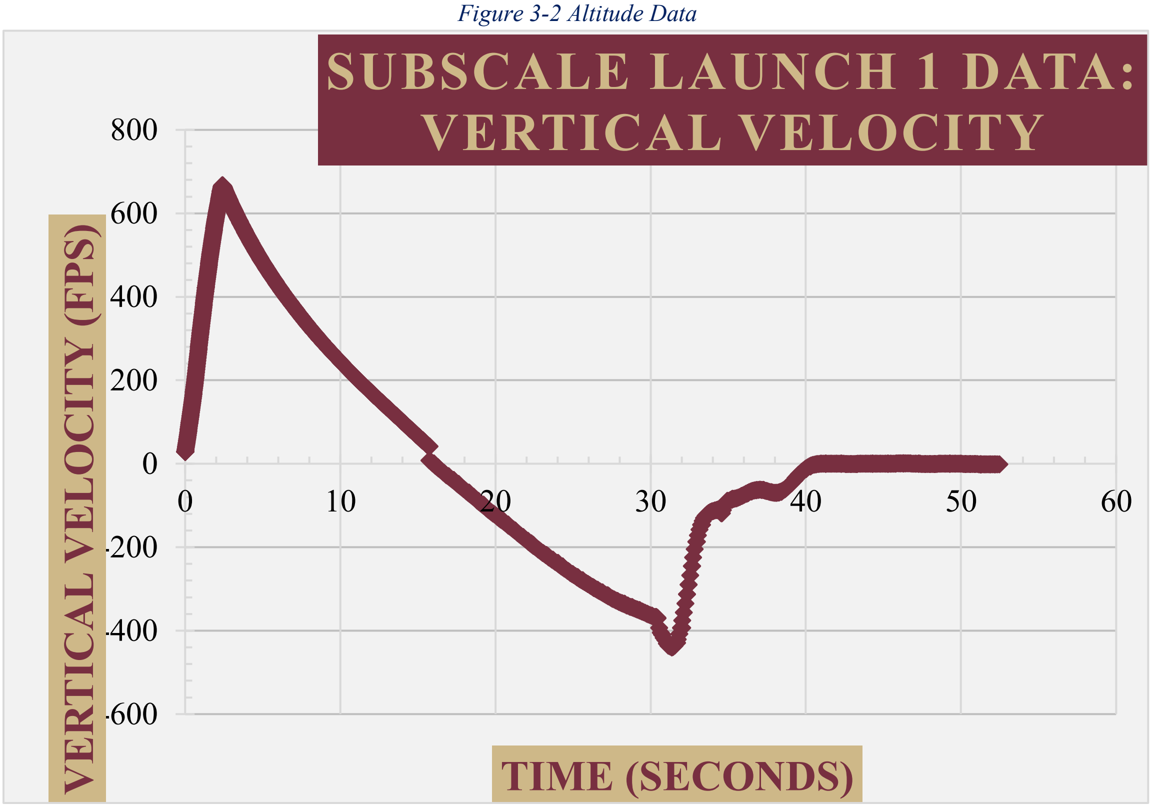


Figure 3-3 Vertical Velocity

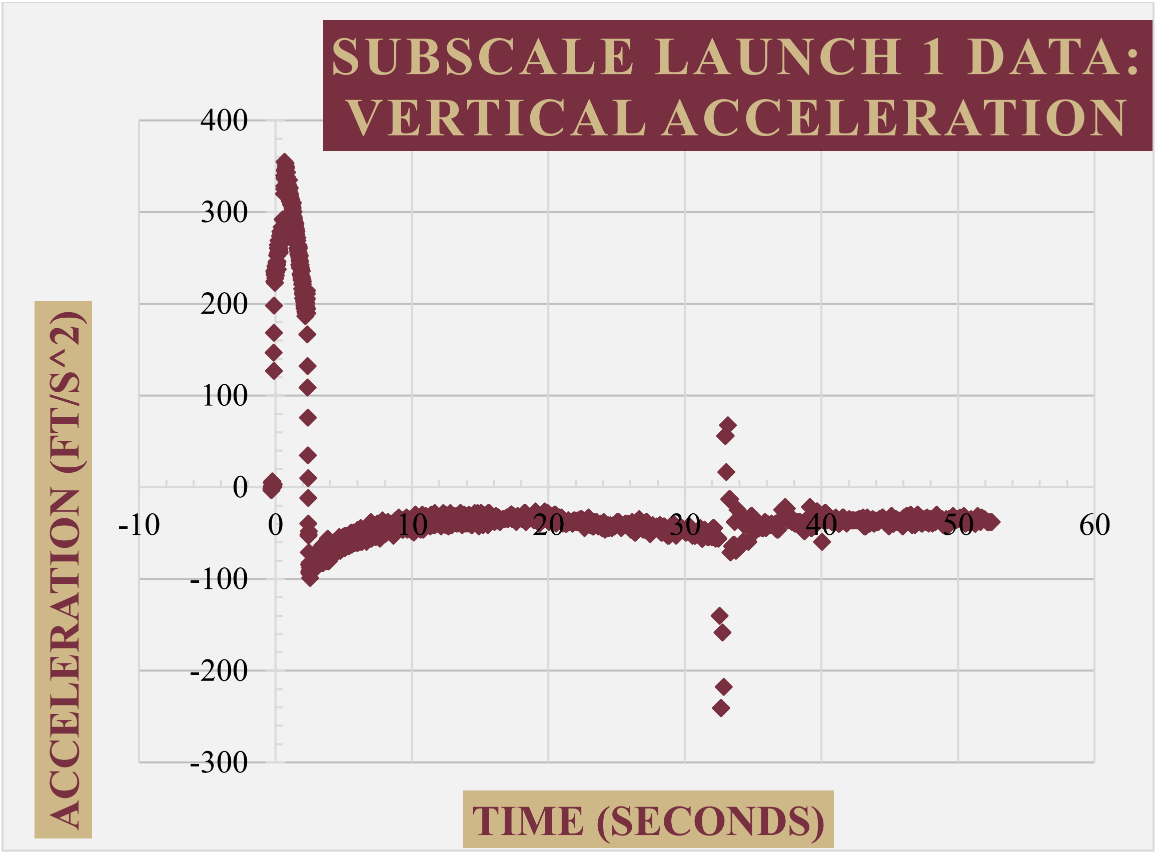


Figure 3-4 Vertical Acceleration

Recorded Launch Data						
Apogee	Max Ascent Velocity	Max Descent Velocity	Max Ascent Acceleration	Max Descent Acceleration	Total Flight Time	A2G time
4474.5 ft	667.1 ft/s	439.8 ft/s	354.7 ft/s^2	240.9 ft/s^2	54.5s	36.2s

3.3.3 Scaling Methodology

To test the design of the Terra Nova launch vehicle, the Terra Nova team produced a sub-scale model, the Terra Parvus. This rocket was scaled based on the principles of symmetry, Geometric, Dynamic, and Kinematic. Geometric similarity is when a model's shape is the exact same as the design but scaled by a constant factor. Dynamic symmetry is when all forces in the model flow are scaled by a constant factor that corresponds to the full-scale prototype. Kinematic symmetry is when velocity at any point in the body flow is proportional by the scaling factor to the velocity of the full-scale prototype. Complete symmetry is achieved when the model has symmetry for all three of the forms geometric, dynamic, and kinematic. Achieving complete similarity is challenging but it was the goal of the Terra Nova team to achieve as much similarity as possible to ensure a valid sub-scale test for the full-scale design. To design the sub-scale a constant scaling factor had to be selected. Per NASA Handbook Req 2.16 the subscale can be no larger than 75% full-scale. Based on this requirement, and the largest available airframe components that fit tis maximum criteria are 4” body tubes. Geometric similarity was the primary scaling method used in designing the subscale. While perfect geometric similarity was not achieved, the design was satisfactory to represent the flight of the full scale. The coefficient of drag can be estimated using the following formula and compared to the full scale Terra Nova.

$$ma = -mg - \frac{1}{2}\rho C_D A v \mid v \mid$$

$$C_D = \frac{2(-ma - mg)}{\rho A v |v|}$$

$$C_D = 9.81$$

3.3.4 Launch Predictions, Performance, & Analysis

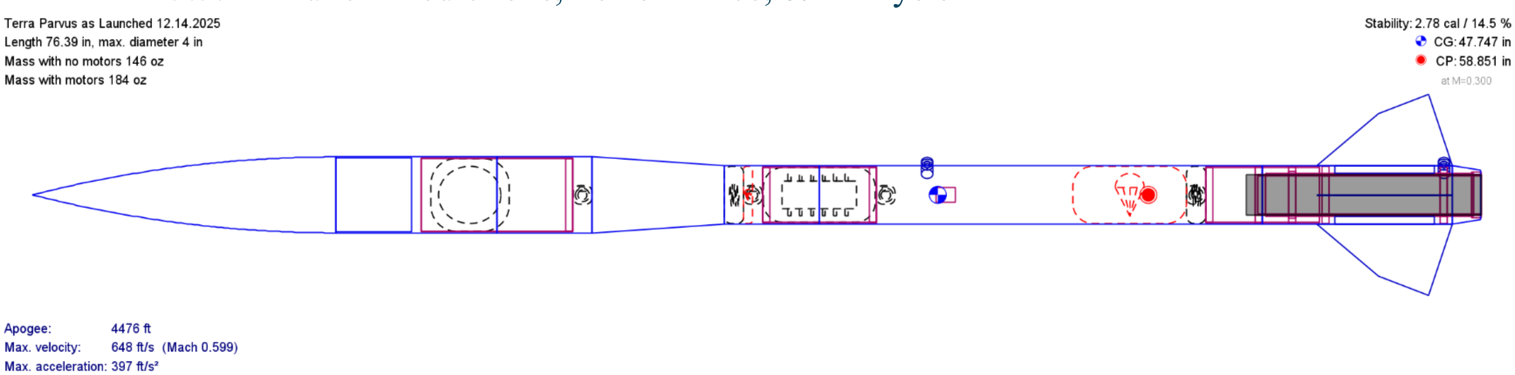


Figure 3-5 Terra Parvus as Launched December 14th 2025

The Terra Parvis V1 was launched on 12/07/2025 in Samson AL. While at the site, our flight computer recorded an average temperature of 57.3°F and average pressure of 13.9psi. Along with the length and angle of the launch rod, these metrics were input to our pre-launch simulations to determine important metrics to compare results to. This simulation provided the following key metrics.

Pre-Launch Simulation						
Apogee	Max Ascent Velocity	Max Descent Velocity	Max Ascent Acceleration	Max Descent Acceleration	Total Flight Time	A2G time
4698 ft	610.38 ft/s	92.68 ft/s	394.62 ft/s^2	1702 ft/s^2	121s	104.4s

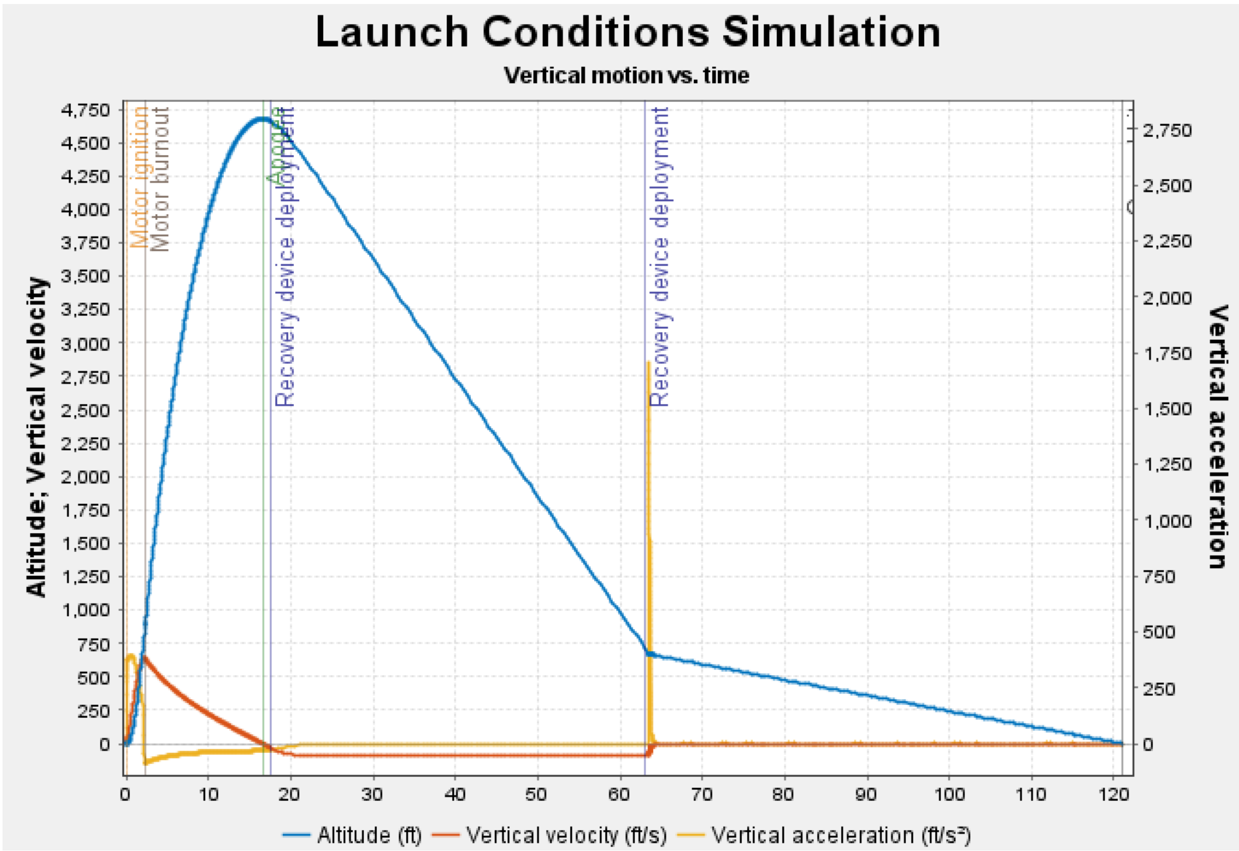


Figure 3-6 Terra Parvus Launch Simulation

Comparing these simulations with the recorded launch data from section 3.2.2 tells the whole story. The actual apogee reached was very close to the projected apogee, reaching just 223.5 ft short of the projected value. Acceleration and Velocity values are even closer during the ascent phase, 39.92 ft/s^2 below and 56.72 ft/s above projected values, respectively. While the ascent of the rocket was simulated accurately and performed as intended, the descent phase saw several subsequent catastrophic failures. First, the drogue separation did not occur. Upon recovery, the CO_2 ejection canisters were analyzed, and it was clearly noted that the apogee casing did not fire at all. The black powder & e-matches inside had not been spent, and the 8g cartridge inside was unpunctured. This failure to eject was related to an electrical failure, though the specific cause was not found. Prior to launch, the team was experiencing issues with properly configuring the Telemetrum flight computer, so it could have been improper programming just as easily as it could have been a disconnected/improperly connected e-match. This failure allowed the rocket to hold a steady 1g acceleration throughout the entire drogue phase and increase speed in a free fall state until the opening of the main. This continuous acceleration greatly increased the speed at main deployment from a projection of 88 ft/s to an incredible 439 ft/s. This major difference in velocity led to an even greater difference in the main deployment force. Despite this the acceleration recorded was lower than the projected values, measured at 240.9 ft/s^2 vs a simulated 1702 ft/s^2. In this case both values are terrible representations of a successful flight. The projected value 1702 ft/s^2 creates a 482.77 lb. deployment force, which is within the strength rating of the shock cords and attachment hardware, but is unnecessarily high given the relatively low weight of the rocket. The actual recorded value, 240.9 ft/s^2 is not a true representation of the force created. At a velocity of 440 ft/s, the drag force created by the parachute was roughly 13487.7 lbs., which is significantly higher than any single material used is specified to endure. While the actual pseudoforce was unbearable, even the predicted value points directly to the first design oversight of this launch. The 72” parachute was significantly oversized, almost double the size of the 48” parachute selected for the heavier fiberglass body of the V2. The 72” parachute was projected to reach terminal velocity at 11.9 ft/s, while a velocity of 20.7 ft/s would have been acceptable to keep the impact energy under 60 ft-lbs, well below the maximum specified by NASA (75 ft-lbs). The actual landing velocity is unknown, as the velocity reported by the flight computer is not consistent with the acceleration data. The acceleration maintains a consistent range between 33-43 ft/s^2, while the velocity damps out around 0 ft/s. In this case, the proper selection should have had a minimum diameter of 39.8” or been a 48” parachute. This error can only be attributed to improper preparation and hasty decision making which contributed to launch failure. It should also be noted that the reefing system (described in section 3.3.3) was still in early development and did not fly with the V1 subscale. The other major design oversight and contributor to failure was the bulkhead design. To more accurately model how we plan to 3D print Inconel bulkheads on the full scale, we attempted to 3D print our subscale bulkheads as well. Though 3D filament is granular and inherently brittle, there are cases where rocketeers have successfully used this fabrication method; however, most successful designs involved a minimum of 80% infill, upwards to 100%. We set this design to be 80% infill, though when the print file was transferred between teammates to be printed, it reset to the base setting of 20% infill. This oversight saw another improperly specified component fly without any testing or strength validation. Ultimately, the large force caused by the main after the failed drogue deployment caused both shock cords (Body Tube 1 – Body Tube 2 & Body Tube 2 – Body Tube 3) to rip out of the bulkheads, bringing almost the exact cross section of PETG encased within the eyebolt washer along with them. The rocket was not successfully recovered at the day of the launch, only the shock cords and eyebolts they were connected to. While unfortunately, no picture of the recovered cords was taken, we did record a photo of the bulkheads after recovering the rocket body the next day.

Figure 3-7 AV Bulkheads (Top), FWD & Aft Bulkhead (Bottom)

Figure: AV Bulkheads (Top), Fwd & Aft Bulkheads (Bottom)

After recovering the rocket, it was in much better shape than expected. Body Tubes 1 & 3 were almost completely intact, except for two fins, and the AV bay was fully recovered so that flight data could be extracted. Body Tube 2 saw much different results, as the transition was ripped straight down the side of the airframe. All recovered sections can be visualized as landed below.

Figure 3-8 As Launched Configuration Body Tube 1 (Left), Body Tube 2 (Right), Body Tube 3 (Bottom)

It was deduced that the monstrous deployment force yanked the shock cord hard enough to rip through the airframe, though why this did not happen to the carboard airframe (the side of the body tube the main parachute was housed in) was unclear. After further analysis, we recalled that the transition was initially 3D printed as well, and the fiberglass only served as a very thin coating to the mold. It is most likely that the thin wall of fiberglass was not enough reinforcement to account for the weakness of PETG.

3.3.5 Changes Since

Several changes to the subscale design have been implemented, as well as lessons learned to be applied to the full scale Terra Nova. The subscale design improvements include changing bulkhead material from PETG to ¼” Plywood, purchasing fiberglass airframes and improving the transition manufacturing, resizing the subscale main parachute, and overclocking the reef system development.

3.3.5.1 Separation Testing

In response to the failed apogee deployment, the Peregrine CO_2 ejection system went through thorough testing with the Terra Parvis V2 airframe. Though the primary purpose was to ensure a successful V2 launch, this series of tests helped us to prove the consistency of the method. In total, 7 full deployment tests were conducted, with 6 of them coming after the launch of the V1.

The first test demonstrated the main separation between Body Tubes 2 & 3, initiated by a 12g cartridge. After this test, it was essentially assumed that the 8g cartridge would work on the other side as well. This decision was made due to time & material constraints just before launch. Though further testing proved this to be accurate, the lack of persistence in validating systems cannot go unstated. Further testing could have helped us develop more thought-out procedures that could have prevented the drogue deployment failure during the launch. After the V1 launch, the next 5 tests were much more meaningful.

The 2nd test failed to separate, though a specific preventable cause was found. The AV Bay responded to test initiation, though the CO_2 ejected from the canister slowly, filling the airframe but not causing separation. The assembly instructions for the device called for guiding pieces to go around the nozzle of the CO_2 canister, though these pieces caused problems. They prevented the canister from making solid contact with the puncturing pin, as well as making it harder to load the device (these guiding pins were not used during subscale launch). We noticed the fracture surface of the canister was much smaller than other successful deployments. This noticing led to further realizations about the device that were strengthened after further testing.

A third deployment test was conducted directly after this failed test without the guiding pieces and separated with significant force. While the distance travelled by the separating body in this test was not measured, it was visually consistent with another test measured to travel 23’ after separation.

The 4th deployment test was very similar to the third in results, though it was done with all parachutes and recovery hardware packed inside the airframe. After this test it was also visually noted that the packed airframe ejected with even more force than an empty airframe.

The 5th, 6th, and 7th tests were conducted by only separating Body Tubes 1 & 2 with the specific goal of measuring opening force in mind. A small computer assembly was programmed with an accelerometer to measure the maximum velocity it travels after detecting a “launch” event. The theory was that this data, along with the mass of the object and the distance it travels, could be used to find an applied force by relating the energy equation to the work equation. The results were deemed inconclusive because the calculated forces based on recorded data were noticeably lower than the minimum force required to break the shear pins. Errors in the experiment were attributed to friction and drag forces skewing the distance travelled from the initial applied force. Instead, a more logical method was used in section 3.3.11 to size the CO_2 canisters on the full-scale. Another unique issue with this test involved another malfunctioned deployment. The 5th test saw a traditionally powerful ejection that travelled 23' from the ejection point. The other two ejection tests saw a malfunction caused by another small puncture surface on the CO_2 canisters, though both events caused separation of the airframe. The malfunction in this test was attributed to improper lubrication of the device. The added friction within the housing could have led to a less aggressive firing of the canister. The difference between puncture surfaces can be visualized in section 3.3.11. Conducting these tests has allowed us as a team to become more familiar with these CO_2 devices and identify what causes failure in order to prevent it during launch scenarios. The tests also paved the way for us to more accurately size canisters for the full-scale design

3.3.5.2 Main Parachute Resizing

As was discussed in section 3.2.4, the Terra Parvis V1's main parachute was significantly oversized. This design overlook has no excuse other than hasty & unchecked calculations. The equation is simple, and after realizing the issue it was not difficult to properly size the parachute for the V2 launch. The initial selection was a 72" Rocketman parachute with a C_d of 2.2, which has been resized to a 48" Rocketman parachute with a C_d of 2.2. This new selected parachute will keep the subscale under impact energy requirements while contributing to a faster descent that helps achieve both our requirements and derived goals. While these goals are set for the full-scale, it is important to show a proper demonstration of current design in order to gather data that can be used to improve the design further.

3.3.5.3 Reefing System Development

The reefing system did not fly in the Terra Parvis V1, though if it had, it would have made little difference in the outcome. Development was delayed throughout the semester and crammed into the short time we had between finals and subscale launch. At the day of launch, the reefing system still employed the paracord loops (described in section 3.3.3) that caused an asymmetric opening. The cord logistics for the reefing line saw the paracord run from the bulkhead, around the parachute, and back down to the bulkhead. This amount of traffic filled the reefing loops and caused so much friction that the parachute would not open after severing the reefing line.

Following the launch, the reefing design was completely overhauled and several tests were conducted to demonstrate the design and make improvements where needed. The first following test saw the improved reefing design, using brass grommets reinforced by fire blanket strips along the skirt of the parachute that a single reefing line can thread through. The Mako line cutter was not used as the objective of the test was to prove the design would hold under load. While real deployment scenarios could not be replicated, a teammate's vehicle was used to deploy the parachute in 30mph winds. The design held as intended and moved on to the next phase.

The following test was intended to be a full demonstration of the reefing mechanism along with the line cutter, and an electronics casing was printed to hold the system that controls event initiation. This casing was wired to an armed mako and ready to cut the line, though the electronics were unresponsive when the test was initiated. The vehicle was decelerated to the minimum speed the parachute would open at, and the reefing line was cut by hand with a knife. Though it was not a full demo, the parachute opened as intended after the line was severed.

The final two tests conducted were two consecutive successful demonstrations of the full reefing mechanism. The parachutes were put under load using the same vehicle method as prior test, and the line cutter was initiated on command with the electronics in test configuration. Videos/photos of all tests were recorded and can be referenced HERE. This system is under continued design and is subject to change, though the current selections are to be considered final unless a better method is presented. The development of this system should have started earlier and consistently progressed, but the failed subscale launch provided proper motivation to create quality designs and demonstrations. For the Terra Parvis V2 launch, two motors were purchases so we can fly once with and once without the reefing system. Though the team has full confidence in its design, we want to ensure a successful subscale launch in order to continue progressing in the competition.

3.3.5.4 Subscale Bulkhead Material Reselection

The Subscale Bulkhead design was 3D printed with PETG filament and intended to emulate the 3D printed Inconel bulkheads that will be used in the full-scale rocket. A lack of infill is what led to failure (excluding other failure modes) if it would have been able to withstand recovery at all. Rather than attempting to flesh out the PETG design, we opted to rebuild using ¼” plywood bulkheads since they are somewhat of a standard in model rocketry. The FSUPC RAM team from the previous year used plywood bulkheads for both their sub & full-scale designs, and rocketry forums repeatedly encouraged the use of plywood bulkheads. While the unique Inconel bulkheads will be used in the full scale, plywood will serve as a “keep it simple stupid” option for the Terra Parvis V2.

3.3.5.5 Subscale Airframe Material Reselection

The Terra Parvis V1 was constructed primarily from cardboard, with the transition being a fiberglass coated PETG print. While the cardboard did not fail, the printed & glassed transition failed decisively. The team decided to rebuild the subscale’s airframe and coupler completely from fiberglass to more accurately model the full-scale’s fiberglass airframe as well as decisively prevent future airframe failure.

3.4 Recovery Subsystem

3.4.1 Recovery CONOPS

The Recovery system for the Terra Nova rocket will have two separation events: one at apogee and the other at 1000 ft. Both events will be initiated via CO_2 ejections, with the drogue event using a 25g cartridge and the main event a 38g cartridge. Both events will use 38g cartridges for redundancy. The drogue deployment will use a 15” fruity chutes parachute with a C_d of 1.5. This chute will reach a projected terminal velocity of 142.79 ft/s. After the main event, a 120” Rocketman parachute with a C_d of 2.2 will be deployed, reefed to a 30” diameter. This phase is projected to fall at a rate of 58.95 ft/s. The reefing line will be attached to a Mako line cutter, with redundant matches wired to two separate altimeters. At 650 ft, these devices will initiate, severing the reef line and allowing the parachute to expand to its full 120” diameter. In this phase of recovery, the vehicle is projected to descend at 14.91 ft/s, which will impact the ground with less than 65 ft-lbs of energy. This meets both the NASA requirement and our derived goal. Additionally, the descent time from apogee to ground projects at 79.15s, also meeting both the NASA requirement and our derived goal of 80s. The purpose of the reefing system is to create a stable transition between the small drogue and large main without causing accelerations large enough to damage the airframe or other load bearing components.

3.4.2 Drogue Parachute

The selected drogue parachute for the Terra Nova rocket is the Fruity Chutes’ 15” Elliptical Parachute, attached directly to the bulkhead on Body Tube 1. It will require no additional protection since the CO_2 release does not heat the pressurized volume or pose a risk to the integrity of the parachute. This item is in saturated stock on the company’s website and is one of the few parachutes we could find at a 15” diameter. With a Coefficient of Drag (C_d) of 1.5, it will provide an estimated terminal velocity of 142 ft/s. A high descent velocity is key for our Apogee to Ground (A2G) goal of 80s. The descent velocity under this drogue parachute was hand calculated based on the drag force equation:

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

Which was compared to OpenRocket simulations. Velocities varied between 141-147 ft/s across simulations. The drag force (F_d) was approximated to be the weight of the rocket using our most updated OpenRocket model. The Weight force after burnout is projected to be to be 675 oz (46.19 lbs.), and is related to the drag force of the parachute in a relatively static frame.

Figure 3-9 Fruity Chutes 15" Elliptical Parachute

https://shop.fruitychutes.com/collections/rocketry/products/15-elliptical-parachute-0-8lb-20fps

3.4.3 Main Parachute

The selected main parachute will be Rocketman Parachutes' 120" High Performance parachute, attached to the bulkhead on Body Tube 3. While again no additional protection is required due to the clean nature of the CO_2 deployment, a parachute bag will be used to ensure smooth deployment of the complex reefing system (section 3.3.3). This item is in stock on the company's website, and we have purchased parachutes from this supplier before. The parachute is advertised to have a C_d of 2.2. Using the same weight defined above, the drogue and main parachutes deployed together have a terminal velocity of 15.37 ft/s. The most updated model of the Terra Nova Rocket projects the heaviest separating section to be Body Tube 1 at 17.94 lbs, greater than Body Tubes 2 & 3 at 16.13 & 9.94 lbs, respectively. This is relevant because the primary consideration when selecting the main parachute is landing velocity. Defining the maximum ground impact energy to be 65 ft-lbs and the mass of the heaviest section (Body Tube 1), a maximum landing velocity is defined to be:

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

$$65\ ft - lbs = \frac{1}{2}\left(\frac{17.94 lbs}{32.17\frac{lbs}{slug}}\right)v_{land,max}^2$$

$$v_{land,max} = 15.27\frac{ft}{s}$$

Based on the parameters of this parachute and US standard atmospheric pressures, its terminal velocity can be projected via the drag force equation, combining the effects of both the main and drogue parachutes:

$$W = F_d = \frac{1}{2}\rho v^2 (C_{d,d}A_d + C_{d,m}A_m)$$

$$42.19 = \frac{1}{2}(20.48 * 10^{-4})v^2[(1.5)\left(\frac{\pi * \frac{15}{12}^2}{4}\right) + (2.2)(\frac{\pi * 10^2}{4})]$$

$$v = 14.91\ ft/s$$

Calculations were again compared to OpenRocket Simulations, which provided similar landing velocities averaging at 15.1 ft/s across simulations. This shows this parachute will land safely and exceed the parameters set by NASA, as well as our own derived requirement.

Figure 3-10 Rocketman 120” Hight Performance Parachute

https://www.the-rocketman.com/products/rocketman-high-performance-cd-2-2-parachutes?variant=42195940048990

3.4.4 Main Parachute Reefing System

The Main Parachute Reefing System is what will make it possible to hit our defined A2G goal of 80 seconds while landing safely under 65 ft-lbs of energy. The A2G goal requires a very fast descent (small drogue), while the Impact Energy goal requires a soft landing (large parachute). Transitioning from such high speeds (defined estimates of 152 – 15.37 ft/s) creates acceleration values upwards of 2800 ft/s^2 (simulated in OpenRocket). This creates a force that is completely unreasonable and would likely immediately destroy the weakest load-bearing portion of the rocket.

To solve this problem, the reefing system was introduced as a replacement to the initial staged recovery concept. This system was deemed more reasonable, as the staged system would require a system to deploy two parachutes in series without any line tangling between separate parachutes. At 1200 ft, the Reefed Main parachute will be deployed with the reefing line choking the skirt of the parachute to a 30” diameter, before the line is cut allowing the parachute to open to its full 120” diameter. The reefing line, reefing loops, and line cutting device are the three main subsystems required to reef the main parachute, which will be explained separately. The reefing mechanism is currently designed to work for the subscale and will serve as a prototype/demonstration for the full scale design.

Figure 3-11 Reefed Parachute Prototype

The most current design of the reefing line involves the Supplementary Cord (defined later in this section to be Paracord 550) running in a loop around the skirt of the parachute. Since the parachute is to be reefed to a specific diameter, the circumference of the related circle defines the length of the cord. A 30” diameter reef would require a reef length of 94.25”.

Several attachment methods have been considered thus far, including paracord loops around the shroud lines, stitching paracord to the shroud lines, and puncturing the skirt of the parachute itself to run the line through. The paracord loops concept was ruled out because the loops could slide up and down the shroud lines which caused the parachute to open asymmetrically during tests, while stitching directly to the shroud lines could jeopardize the integrity of the parachute. While the same integrity concerns were present when puncturing the parachute, the most recent design provides a solution. A Nylon Fire blanket that was on hand in our facility was cut into strips, stitched along the edges, and sewn to the skirt of the parachutes with Kevlar thread, midpointing each shroud line. These strips of fabric serve to reinforce the ½" Brass grommets that were installed to secure the loop for the reefing line. This design was tested up to 40mph winds on a 72" parachute with a C_d of 2.2. This value presents a confirmed strength exceeding 240lbs of drag force and is subject to further testing and improvement. Future design improvements include a Nylon/Fiberglass Fire Blanket as use for fabric reinforcement and continued practice in manufacturing processes.


Figure 3-12 Reefing Loops (Left) & Stitched Fire Blanket Strip (Left)

The final portion of the reefing mechanism is the line cutting device. Early concepts involved fabricating the device in house, though a better option was found commercially available in Tinder Rocketry's Mako Line cutter. This device is a stainless-steel cylinder with openings for the reefing line and a recovery attachment line. It requires an e-match to fire a small black powder charge that will push a cylindrical "firing pin" to cut the rope cleanly. This device has been thoroughly tested and the blast is completely contained inside the device. The most updated design requires that this device be placed on the reefing line at the skirt of the parachute and the later described "supplementary cord" used to attach the line cutter to its bulkhead. A separate flight computer is required to operate this device since it is outside the Avionics Bay, so a redundant system using a telemetrum & a computer developed in house will be encased in a protective box and held inside the airframe via the eyebolt on Body Tube 3. E-matches will run from the casing to the Mako, and the supplementary cord between the casing and reefing line must be shorter than the e-match cable so that the match does not take on any load and risk disconnect. The prototype for the casing was

3D printed and tested in environments up to 40 mph, though plywood is the current choice for the final product.

Figure 3-13 Mako Line Cutting Device

Figure 3-14 Reefing E-Casing, Exposed (left), Enclosed (Right)

Shown above is an image of the prototype system. While it is designed not to take on much load, deploying electronics still presents a massive risk of failure. The final design will involve the electronics casing to be attached directly to the bulkhead on Body Tube 3, with an e-match running up to the device. The electronics inside the casing will not likely change, as all the device needs to do is read altitude and deploy at the designated height.

Due to the ambitious nature of this design, we are also preparing for a traditional deployment in the event it is deemed unsafe for use, though two successful demonstrations of the parachute have been documented.

A series of tests were conducted, first to ensure the integrity of the parachute, then to demonstrate the opening of the device. The Parachute was attached via a quick link to the bed of a teammate's truck and guided into the wind by hand. There were not many options to test with the wind speed needed to demonstrate the design, so safety had to be taken very seriously during these tests. To prioritize safety, no tests were conducted on roads populated with cars, precise communication was used between teammates, and it was ensured that we did not exceed the speed limit (45mph) on the road used for testing. The final test showed a reefed parachute opening, then an initiation of the line cutter, finally followed by the full opening of the parachute.

Figure 3-15 Reefed Parachute (Left), Open Parachute (Right)

3.4.5 Shock Cord

The shock cord selected is a ¼" diameter PARAMAX paracord. This cord is cheap, easy to work with, and boast a 1200lb tensile strength that is more than sufficient for the forces of recovery. This can be proven by a force analysis of the full opening of the main parachute, since it will cause the greatest acceleration during recovery. The simulated acceleration of our most updated model during the opening of the main averages out to 408 ft/s^2 across simulations. Using the calculation weight of 48.75 lbs, a mass of 1.52 slugs can be used to find a force:

$$F = ma$$

$$F_{max} = (1.52\ slug)(408\frac{ft}{s^2})$$

$$F_{max} = 617.9\ lbs$$

This cord provides a safety factor of 1.94 against the largest deployment force, justifying it for use. Initially a Kevlar-Elastic hybrid cord was proposed to be used, though it would be extremely difficult to fabricate and would require thorough testing before it could be flown. PARAMAX presented a similarly strong, purchasable material that does not require additional help with shock loading.

For shock cord sizing, the 4x rocket length rule of thumb will be used in reference to the larger 6" diameter. This method sizes 40' of cord for both deployments.

Figure 3-16 Paramax Paracord

The shock cord will be attached to all hardpoints via Double-Davey knots. These knots were used when both Kevlar and PARAMAX cord were tested in tension past 1000lbs, and seemed to only get tighter as load was applied. The first subscale flight also saw the use of these knots, and they held firm despite an unsuccessful recovery.

Figure 3-17 Paramax Double Davey Knot

3.4.6 Supplementary Cord

A supplementary cord was selected for roles that either required smaller cord than PARAMAX or were not expected to take on much loading. Paracord 550 with a 5/32" Diameter was selected for this role. In the first subscale flight it was used to connect the drogue parachute to its bulkhead, and in future flights it will be used for the reefing line & the line cutting device, while the full-scale drogue will be connected via quick link. All uses require loop knots, so again the same Double-Davey knot is used again and has not unintentionally failed in any scenarios.

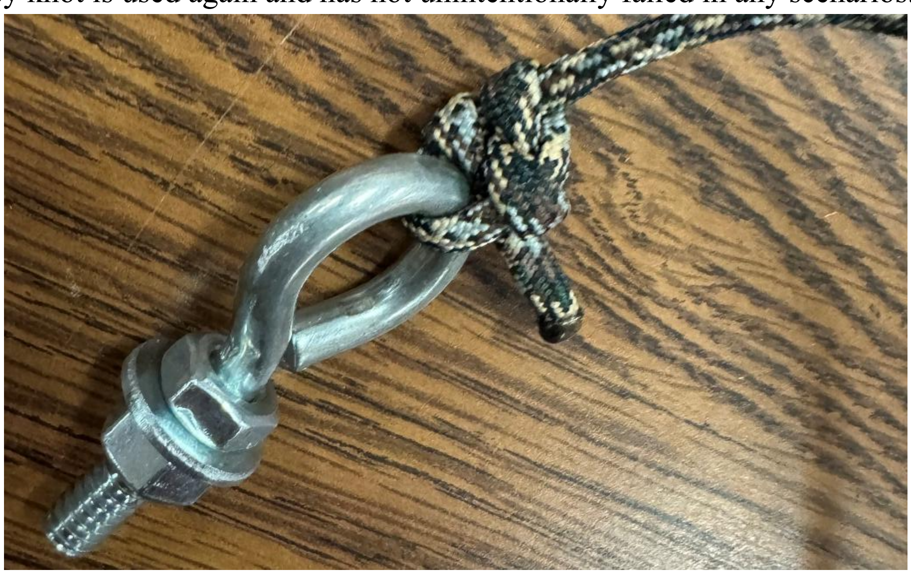


Figure 3-18 Paracord 550 Double Davey Knot

3.4.7 Attachment Hardware

All hardpoints (bulkheads) will employ the same attachment hardware, including a ½” eyebolt attached to the frame and a ¼” quick link to attach the parachute. US Rigging has both of these pieces in stainless steel 316 and the eyebolt and quick link in the size specified above are rated to 3000lbs and 1430lbs, respectively. These values are well above the simulated peak force of 620lbs, and thus meet the requirements needed to withstand recovery.

Figure 3-19 SS 316 ½” Eyebolt (left), & SS 316 1/4 “ Quick Like (right)

https://usrigging.com/

The quick link will be used to quickly attach and detach parachutes and shock cords from the airframe. Any uses of either the shock or supplementary cord will be tied via a Double-Davey knot to the quick link, while most parachutes already have a ring that can slip around the link. This assembly will be used at all four hardpoints, with two more eyebolts to connect lines to the reef’s electronic casing. While shock cords will run between bulkheads, the drogue parachute will be directly attached to the aft bulkhead on body tube 1, and the main & other assemblies will be connected to the forward bulkhead on body tube 3.

3.4.8 Shear Pins

Before a specific shear pin can be selected, the forces it must withstand must first be understood. When the rocket is assembled at ground level, the internal pressure inside the body will be equal to the atmospheric pressure around it; however, as the rocket increases altitude, the atmospheric pressure will decrease while the rocket's internal pressure will stay mostly constant. This increasing pressure differential will create a force that grows in magnitude as the rocket ascends. The weight of the rocket's components will also attempt to separate the different sections of the rocket, while friction between the body tubes and couplers will act to prevent this action. Based on hand solved calculations, the worst-case scenario of these forces will happen when the rocket is perfectly vertical, just as it hits apogee.

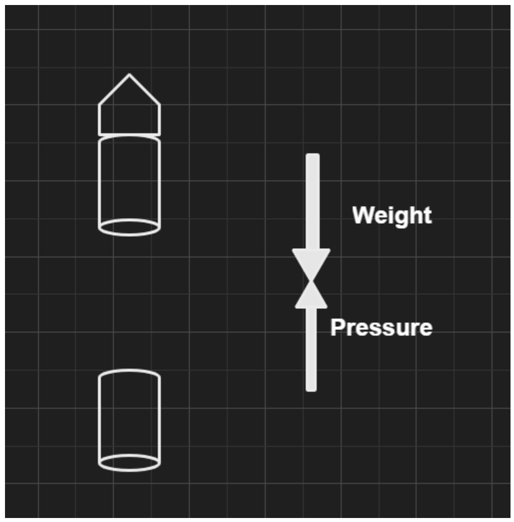


Figure 3-20 Nosecone FBD.

This free body diagram demonstrates the forces acting to separate the nose cone (Section 1) from sections 2&3. Simulating the worst-case scenario, friction is neglected, thus providing no assistance to the shear pins to keep the rocket together. With the air pressure at ground level set to 14.696 psi, and the air pressure at apogee (roughly 5500 ft, more than +600 of the simulated 4848.7 ft to account for Huntsville's average ground level of 600 ft above sea level) is 12.016 psi. This creates a pressure differential:

$$\Delta P_{apogee} = P_{ground,inside} - P_{air,apogee}$$
$$\Delta P_{apogee} = 2.68\, psi$$

With this estimated pressure differential, a force is created:

$$F_{P,apogee} = \Delta P_{apogee} * A_c$$

Since the nosecone separation occurs at the top of the transition, the diameter of the rocket this force will be acting on at its greatest is 6". Thus:

$$F_{P,apogee} = (2.68psi) * (\frac{\pi(6)^2}{4})$$
$$F_{P,apogee} = 75.775\ lbs$$

The weight of sections 2 & 3 of the rocket will also contribute to a natural separation in this exercise. Sections 2 & 3 weigh 16.4375 lbs and 13.875 lbs, respectively. The minimum force the shear pins must withstand can be modeled by the maximum force created in this scenario. Thus:

$$F_{pin} = F_{P,apogee} + W_{2,3}$$
$$F_{pin} = (106.09lbs) + (16.4375lbs + 13.875lbs)$$
$$F_{pin} = \ 106.09\ lbs$$

A study found on Apogee Rockets and published on rocketmaterials.org showed the required ejection force for 2-56 and 4-40 shear pins. For this project, 4-40 shear pins will be selected, but the amount used will reference this chart.

#2 Nylon Screws				#4 Nylon Screws		
# of Pins	**Peak Load (lbs)**	**Peak Load (Each Pin)**		**# of Pins**	**Peak Load (lbs)**	**Peak Load (Each Pin)**
2	53.123	26.56		2	81.304	40.65
2	45.952	22.98		2	85.148	42.57
2	50.848	25.42		2	75.944	37.57
2	51.799	25.90		2	80.391	40.20
2	47.924	23.96		2	80.908	40.45
Avgs	***49.93***	***24.64***			***80.75***	***40.30***
3	62.637	20.88		3	119.273	39.76
3	60.569	20.19		3	110.999	37.00
3	64.395	21.47		3	99.969	33.32
3	62.413	20.80		3	113.554	37.85
3	68.760	22.92		3	116.208	38.73
3	66.643	22.21		3	121.121	40.37
				3	123.689	41.23
Avgs	***64.24***	***21.41***			***114.97***	***38.32***
4	86.699	21.67		4	143.405	35.85
4	86.855	21.71		4	152.368	38.09
4	78.771	19.69		4	142.026	35.51
4	84.269	21.07		4	160.489	40.12
4	85.617	21.40		4	153.302	38.33
4	90.402	22.60		4	154.766	38.69
				4	160.368	40.09
Avgs	***84.44***	***21.36***			***152.38***	***38.21***

Figure 3-21 Shear Pin Strength Chart

https://web.archive.org/web/20131026023457/http://www.rocketmaterials.org/datastore/cord/Shear_Pins/index.php

Given the required minimum strength to prevent unintentional separation, most trials using the 3 shear pin configuration meets our derived requirement, though some trials fell short. To add an extra factor of safety, we will use 4, 4-40 shear pins to connect separation points in the Terra Nova rocket, providing up to 54lbs of strength above the estimated requirement.

3.4.9 Bulkheads

The bulkheads for the Terra Nova rocket will be constructed from Inconel 718 and be 3/16" thick. This will be achieved with help from Meritech Machining, a local industry partner. In exchange for conducting materials testing on their new additive manufacturing techniques, they will supply us with the printed parts we need for the full scale. While tensile testing has not yet been conducted, torsional properties have been tested and converted to tensile properties using Von Meses theory. A deducted 283,000 psi can be assumed to be the material's tensile yield strength for calculations. Given the larger bulkhead area of 28.27 in^2, the force required to rupture would be just over 8,000,000 pounds. This selection is certainly overkill, though it will complete our derived goal of using additive manufacturing in our design. It also removes bulkhead failure from the picture completely, since other components would fail long before the bulkheads even yield.

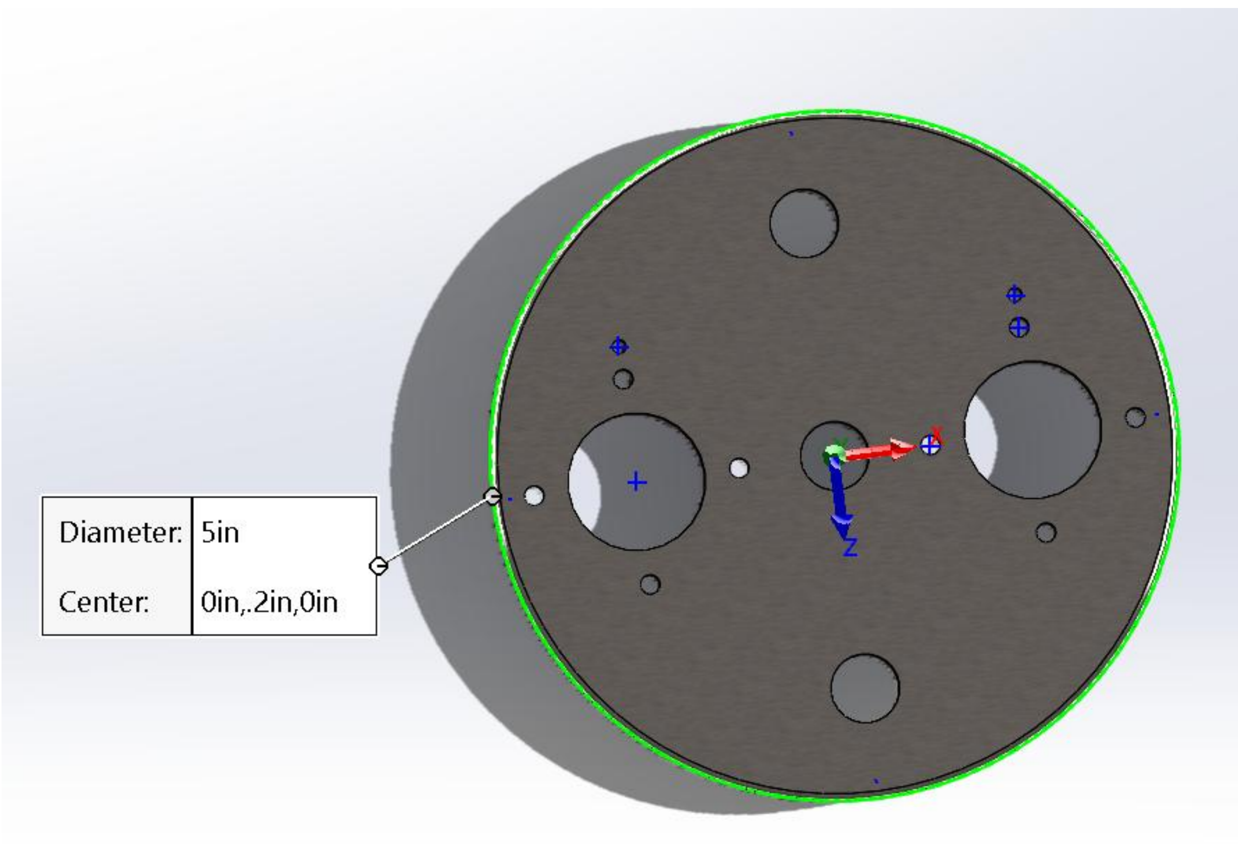


Figure 3-22 Avionics Bulkhead

3.4.10 Avionics Bay

The avionics bay will use a dual redundant system with two flight computers directing 4 CO_2 Ejection systems. The use of the ejection systems will be described further in the next section, while this will focus primarily on the components & logic of the AV (avionics) bay. The AV Bay will be in the center of body tube 2, just aft of the transition. As an assembly it will record flight data and initiate separation events, with the reefing subsystem will operate independently of this system. A Telemetrum and Easy Mini have been selected as the preferred commercially redundant flight computers, and they will both be wired up to a screw switch. This switch will be aligned with the pressure port to allow the system to be armed on the launch pad without a system that protrudes from the airframe. All equipment will be powered via a 9V LiPo Batteries. Each side of the AV bay will hold two Eagle devices; each wired to a different computer in the event of either one failing. These ejection devices will initiate separation after the flight computer reads the defined target altitude/stage via an e-match, with the redundant systems following a short delay. While the primary role of the AV bay is to initiate separation, the flight data it records is very useful in flight analysis/vehicle improvement upon recovery.

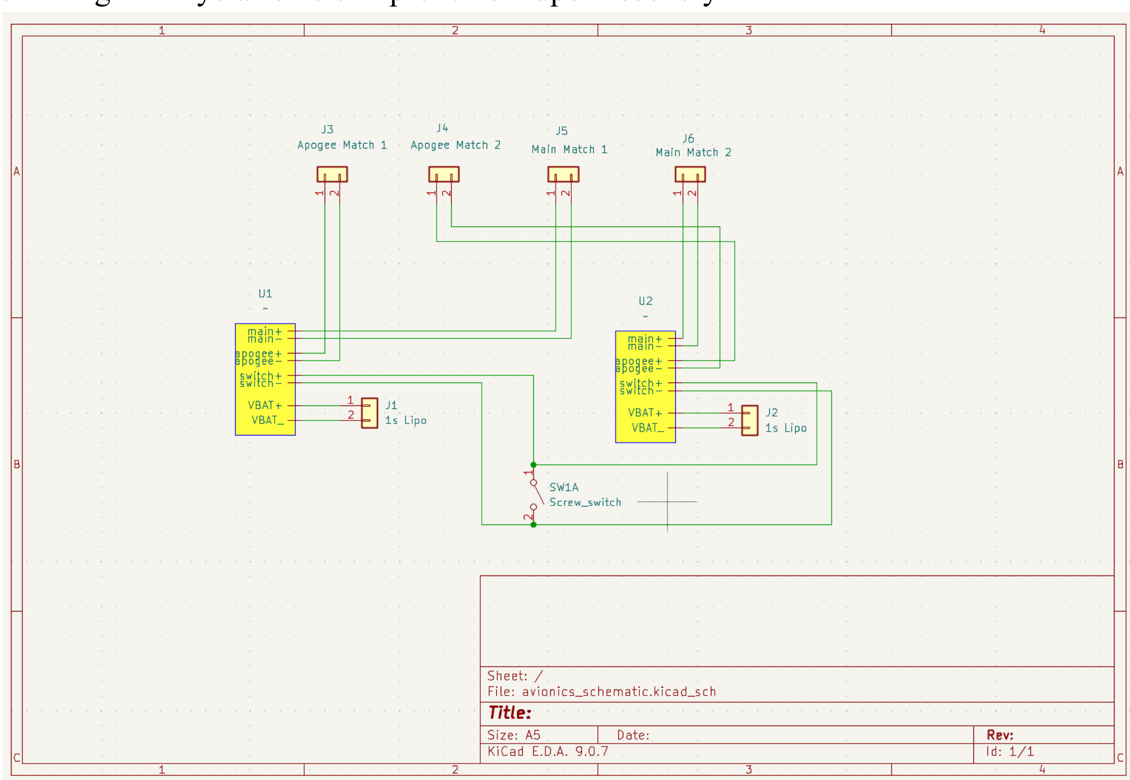


Figure 3-23 AV Bay Logic Diagram

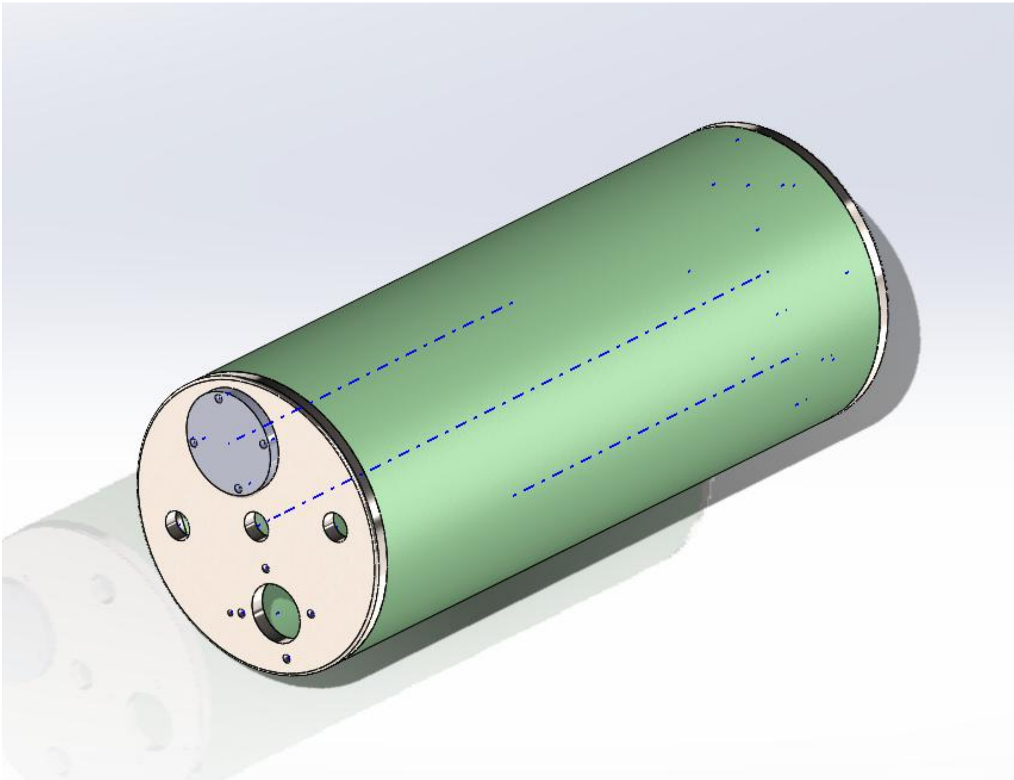


Figure 3-24 Avionics Bay Assembly

3.4.11 Eagle CO_2 Ejection System

The Eagle CO_2 Ejection System is the selected separation method for the Terra Nova Rocket. This method uses minimal black powder, is totally reusable, and doesn't pose a burning risk to parachutes. The system uses a small, contained black powder charge to force a pointed "firing pin" to puncture a CO_2 cartridge. These systems are loaded in accordance with the manufacturer's instructions and are fired via e-match, making for standard deployment in every way except the delivery method.

Figure 3-25 Eagle CO_2 System

https://www.tinderrocketry.com/the-eagle-co2-ejection

3.4.12 CO_2 Cartridge Sizing

The Eagle system can be adapted to hold cartridge sizes from 16g-86g, so there are plenty of options to pick the right size for the Terra Nova's mission. Initial attempts to determine the amount of CO_2 required involved the use of the ideal gas law.

$$PV = nRT$$

Using the volume of the container to be pressurized, a force (pressure/area) required to break the shear pins, the gas constant, and an estimated temperature, it was idealized that the number of moles and thus mass of CO_2 could be determined. These assumptions netted low amounts of CO_2 to pressurize the rocket (2.48g for drogue separation & 8.99g for main separation). To confirm these calculations, we turned to testing.

A series of deployment tests were conducted and can be referenced in HERE. Across 6 total separation tests, only one failed and had a specific cause of failure. The kit instructions use a plastic piece that guides the CO_2 cartridge to the firing pin, though it instead prevents the cartridge from making solid contact and was the cause for the failed separation test. Two more tests malfunctioned due to improper device lubrication; thus, the cartridge was not properly punctured but still caused separation. A figure describing the different puncture surfaces on the CO_2 cartridge can be visualized below, with two successful deployments on the left and two malfunctioned (but still separating) deployments on the right:

Figure 3-26 CO_2 Puncture Surfaces

Two different volumes were pressurized, with the forward section using an 8g cartridge and the aft section using a 12g cartridge. Since the full scale rocket will have similar geometry to the subscale, and it has been confirmed that these sizes will cause separation when properly prepared, a relationship between the sub and full-scale rockets can be developed.

The drogue separation for the subscale used 8g of CO_2 to pressurize the forward section of the transition. The volume of this section can be calculated using the formula for a conical frustrum:

$$V = \frac{\pi L}{12}(D_1^2 + D_1 D_2 + D_2^2)$$

$$V = \frac{\pi(6")}{12}((4")^2 + (4")(3") + (3")^2$$

$$V = 62.96in^3$$

Using this volume and the CO_2 used, it can be stated that this geometry will successfully separate at a CO_2/Volume ratio of $\frac{1g\ CO2}{7.87in^3}$. Since the full scale forward separating volume uses the same geometry, this ratio can be applied to the enlarged 190.59in^3 volume of this same section on the full scale rocket. Using the ratio just developed, 24.22g would be the mass required to separate the full scale, so a 25g canister can be selected with a 38g canister added for redundancy.

For the aft deployment, the same rules can be applied. Due to similar scaled geometry, it can be assumed that a ratio between volume and mass of CO_2 can be developed to size the full scale cartridges. The volume of the aft pressurized section on the subscale can be found using the formula for the volume of a cylinder:

$$V = \frac{\pi L D^2}{4}$$

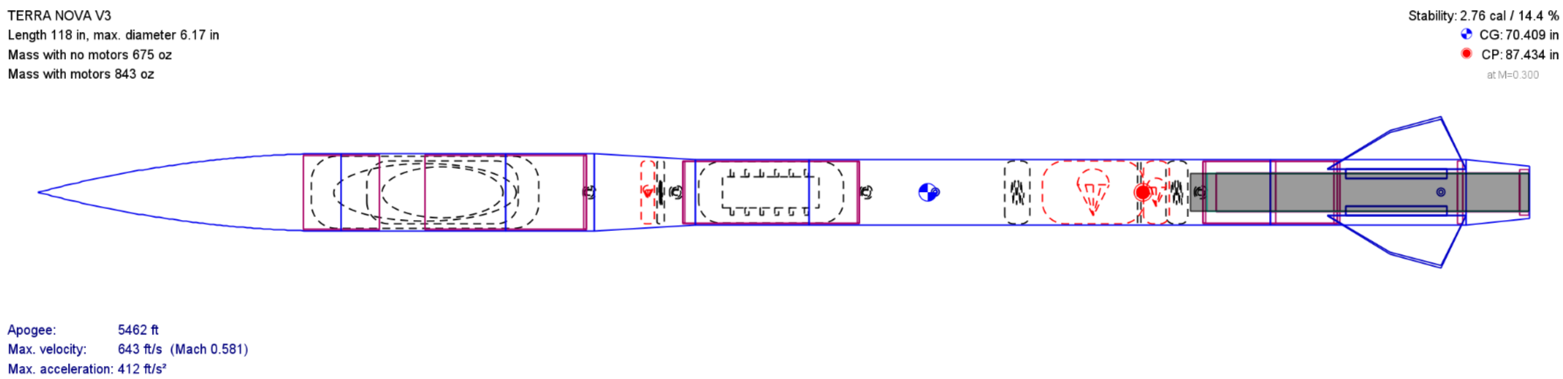
$$V = \frac{\pi(33)(4")^2}{4}$$

$$V = 276.46in^3$$

Given that consistent separation has been proven for this volume using a 12g Cartridge, a CO_2/Volume ratio of $\frac{1g\ CO2}{23.04in^3}$ can be developed for this geometry. Applying this to the full scale volume for this section (647.95in^3), a minimum of 28g of CO_2 must be used. Based on these calculations, a 38g CO_2 cartridge can be sized, with another 38g as redundancy. Given a 10g increase over the minimum, using the same size for redundancy is acceptable as the primary deployment device should separate properly with a factor of safety. These numbers will be verified through deployment testing as construction of the full scale commences.

3.5 Launch Vehicle Predictions

3.5.1 Simulated Rocket Vehicle Overview



Primary Vehicle Predictions								
Apogee	Descent Time	CP	CG	Max Deployment Accel	Velocity off Rod	Max Velocity	Weight (Motor)	Weight (Burnout)
5462 ft	78.9 s	87.434”	70.409”	607 ft/s^2	80.8 ft/s	643 ft/s	52.69 lbs.	42.19 lbs.

Calculation Discrepancies

Values while boost phase metrics were modeled solely using OpenRocket, descent phase values were all calculated throughout the CDR document. Comparisons to predicted data were made along with these calculations, and proved consistent.

Thrust

The predicted thrust of the L2200G is shown in Figure 3-29. The maximum anticipated thrust of the motor is 3099 N with a total impulse of 5104 Ns. The average thrust of the motor is 2243 N. Motor burnout occurs after 2.4 seconds using 2518 g of propellent. The thrust to weight ratio for the Terra Nova design and the L2200G is 8.94:1.0 exceeding the minimum requirement of 5.0:1.0 weight ratio per Handbook Req. 2.12.

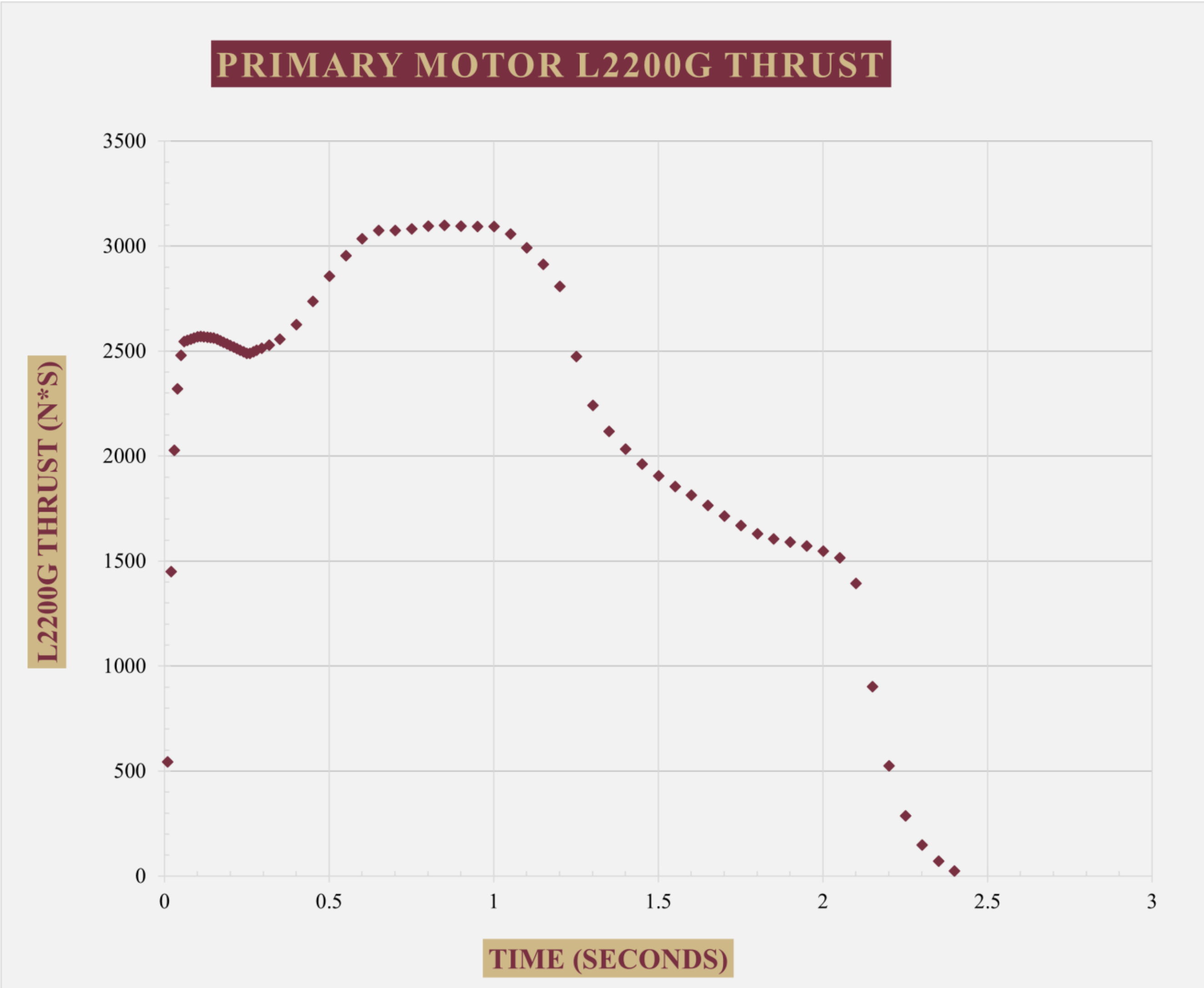


Figure 3-27 Thrust Curve

Stability

The static stability of the Terra Nova is required per Handbook Req. 2.11 to be above 2 calibers. The Terra Nova's static stability on the launch pad is 2.71 calibers meeting the requirement of 2. The stability of the rocket at rail exit is 2.8 calibers. However, the stability of the rocket changes with velocity, weight, and angle of attack. Figure 3-30 details the expected stability of the launch vehicle over the course of its flight to apogee.

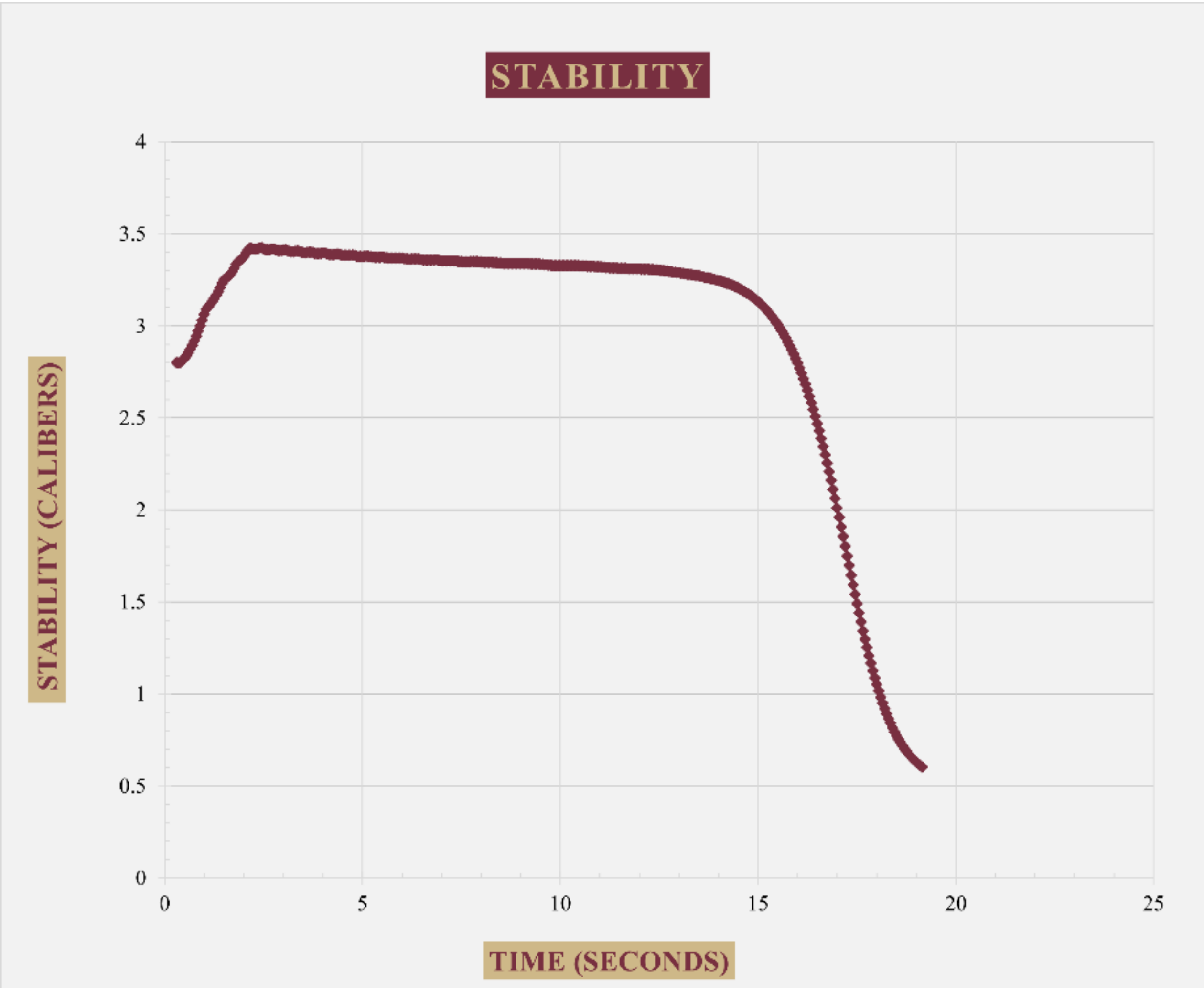


Figure 3-28 Stability Curve

3.5.2 Primary Motor – L2200G-0

Flight Simulation

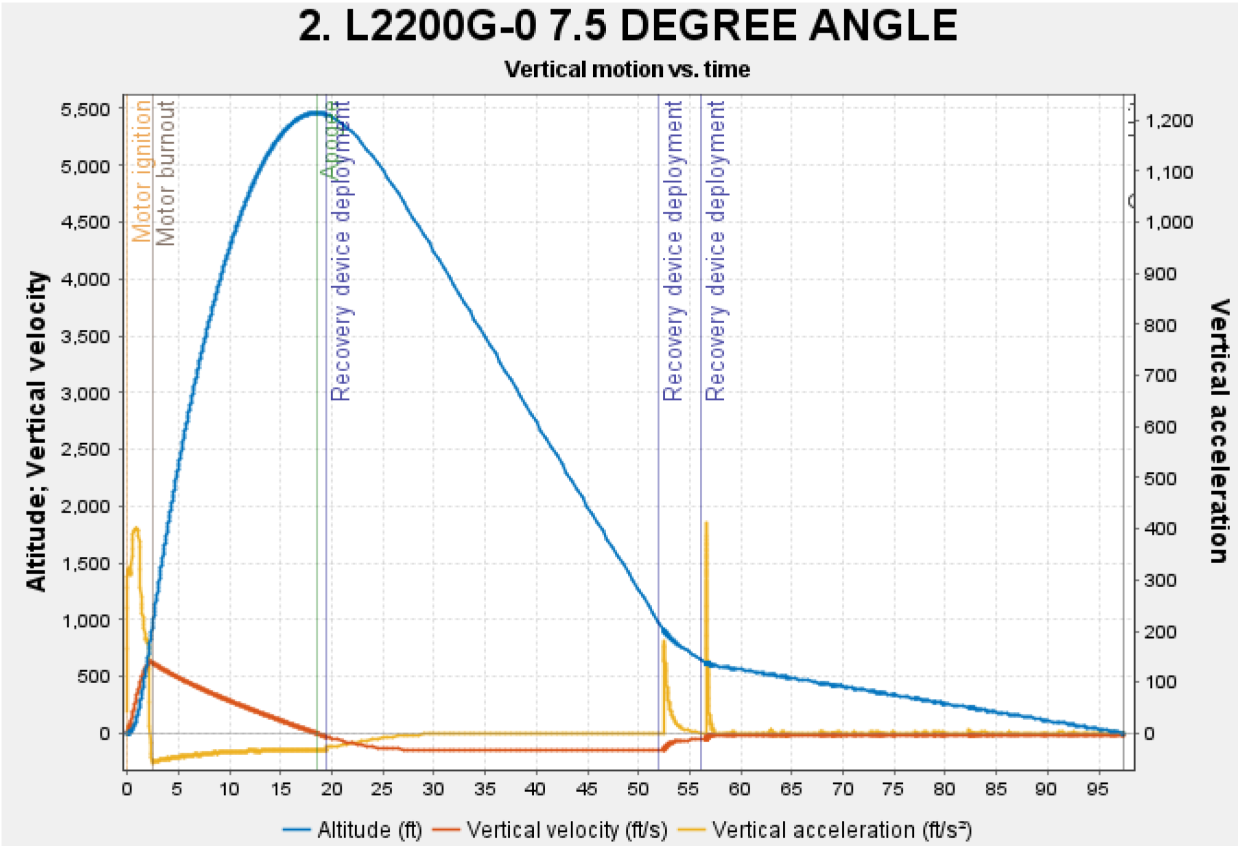


Figure 3-29 Primary Motor Flight Simulation Data

The chart above displays simulated flight data with our primary motor choice. The peak acceleration of 407 lbs is within load constraints defined in section 3.3

3.5.3 Secondary Motor – L1420R-P

Flight Simulation

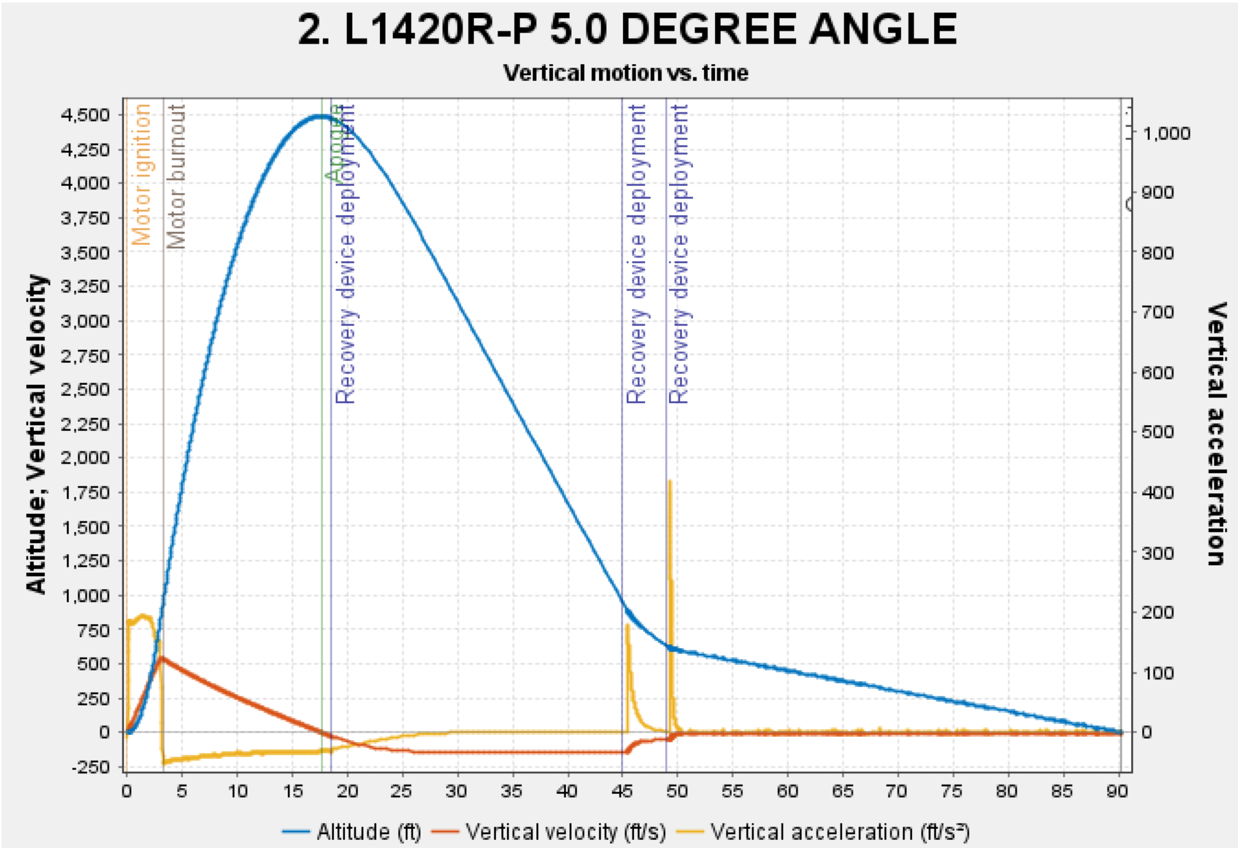


Figure 3-30 Secondary Motor Flight Simulation Data

The chart above displays simulated flight data with our secondary motor choice. The peak acceleration of 416 lbs is within load constraints defined in section 3.3

3.5.4 Tertiary Motor L1365M-0

Tertiary Motor Need Justification

While NASA requires a primary and secondary motor to be selected, and this selection is not subject to change unless the selected motors are unavailable, both primary and secondary motors are currently out of stock with all suppliers we have communicated with. We are on a call list with Chris's Rocketry and Wildman Rocketry as they are the two vendors set to receive the next resupply. In the event these motors are not available, the L1365-M is compatible with our RMS system and presents a similar flight profile to the L2200G-0

Flight Simulation

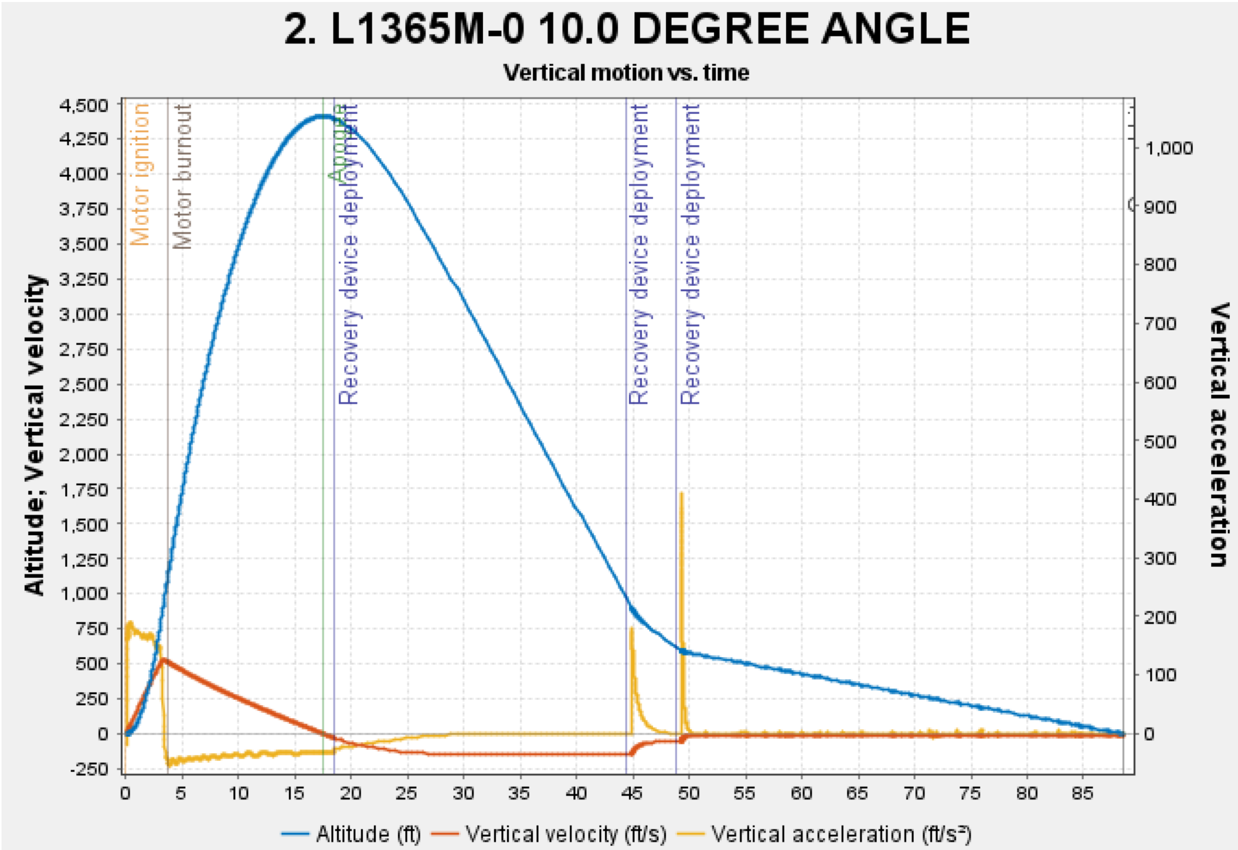


Figure 3-31 Tertiary Motor Flight Simulation Data

The chart above displays simulated flight data with tertiary motor choice. The peak acceleration of 410 lbs is within load constraints defined in section 3.3

3.5.5 Launch Recovery Predictions

Impact Energy

The impact energy requirement is the most important constraint on the recovery system because it sets the standard for safe recovery and constrains the main parachute size. Our derived goal for impact energy is 65 ft-lbs of energy upon recovery. This topic was discussed briefly in section 3.3.2, though the process will be reiterated here. Body Tubes 1, 2, and 3 weigh 17.94, 16.13, and 9.94 lbs, respectively. Taking Body Tube 1 to be the heaviest section at a mass of 0.558 slug, a Maximum landing velocity to stay within both NASA and derived requirements can be specified.

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

$$65\ ft - lbs = \frac{1}{2}(0.558\ slug)v^2$$

$$v = 15.27\frac{ft}{s}$$

Compared to OpenRocket, the Terra Nova in its described configuration is projected to land at an average velocity of 15.1 ft/s across simulations. Hand calculations have also provided a value of 14.91 ft/s, meeting this requirement.

Descent Time

Descent time can be modeled by extrapolating the parachute velocities over the distance they fall, assuming a static reference frame and a transient period of less than 1 second. The drag force equation is used to model these velocities:

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

$$v_{drogue} = 142.79\ ft/s$$

$$v_{main,reefed} = 58.95\ ft/s$$

$$v_{main,open} = 14.91\ ft/s$$

OpenRocket simulations have been used to fine tune the deployment heights of the main sequencing to the reefed main deployment at 1000 ft and the reefing line being cut at 650 ft. Initially, there were concerns about how low the main sequences could be initiated though tests showed that the parachute transitions well to opening states. Given our average simulated apogee of 5472.67 ft, the times in each deployment phase are as follows:

$$t_{drogue} = 31.32s$$

$$t_{main,reefed} = 4.24s$$

$$t_{main,unreefed} = 43.59s$$

Adding these values sums to the total descent time

$$t_{Descent} = 79.15s$$

This meets both the NASA requirement of 90s and our derived requirement of 80s. This calculation is consistent with OpenRocket simulation data, with descent time averaging at 77.9s across simulations

Wind Drift

Once descent time is calculated, wind drift can be easily modeled with two key assumptions. The first is that the rocket reaches apogee directly above the launch pad, and the second is that the rocket descends in a steady state drift with the wind as it descends. This calculation will be tabulated for several wind velocities using the following formula:

$$Drift = v_{wind} * t_{Descent}$$

Wind Drift Calculations							
Wind Velocity (ft/s)	0	5	10	15	20	25	30
Horizontal Drift (ft)	0	156.6	313.2	469.8	626.4	783.0	939.6

4 Payload Criteria

4.1 Payload Objective

The objective of the Terra Nova's HAUS payload (the Terra Inquisitor) is to autonomously collect and test a 60 mL sample of soil for pH level within 12 minutes of landing. The test data from the soil sample will be time-stamped and stored in a SD card on the Inquisitor. The Inquisitor will be a rover like vehicle capable handling various terrains and obstacles. It will initiate its autonomous soil testing protocol after landing.

4.2 Payload Systems

4.2.1 Mechanical Separation

The first stage of the payload is its deployment. This will be achieved without using energetics and after the rocket has safely landed; thus, this design will not eject in the air as an unmanned aircraft system (UAS). To achieve this separation at ground level, a mechanism must be designed to overcome various frictional and weight loads and separate the HAUS Section of the rocket airframe for the deployment of the Terra Inquisitor. For the optimal design of such a mechanism, these loads need to be quantified, and various concepts must be considered.

The force required to separate can be simplified to multiple forces acting at a single point on the HAUS Section of the rocket. This single point will be assumed to be at the location of separation. The forces acting at this point include the force needed to separate (F_{min}), the weight of the section (W_{HS}), friction due to ground-airframe contact ($f_{S,GA}$), friction due to coupler-airframe contact ($f_{S,CA}$), and the normal forces acting between the coupler-airframe and the airframe-ground which would be equal (F_N). The magnitude of this force will be the derived requirement for the minimum force that a mechanism must be able to exert. This is illustrated in Figure 4-1.

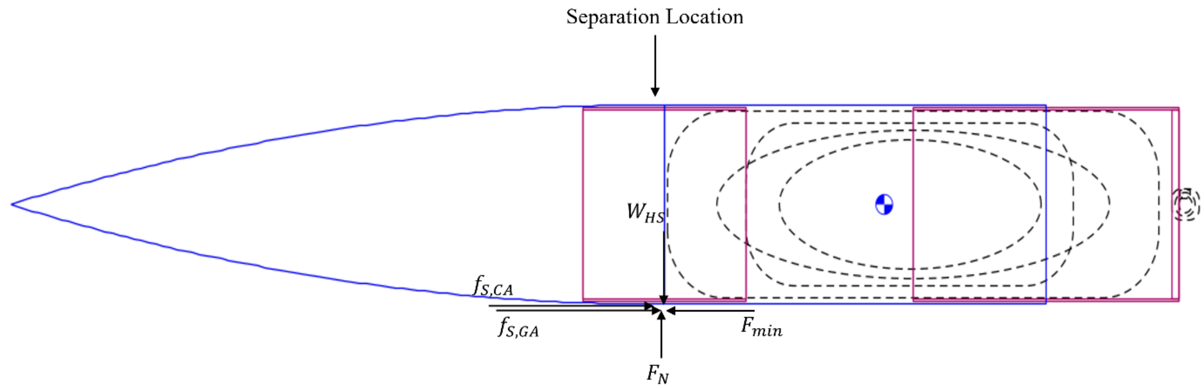


Figure 4-1 Separation FBD when θ = 0°

However, it must also be considered that the HAUS section may land on a slope, thus it is possible that there would be another orientation at which more force would be required to separate. Figure 4-2 below illustrates this alternative.

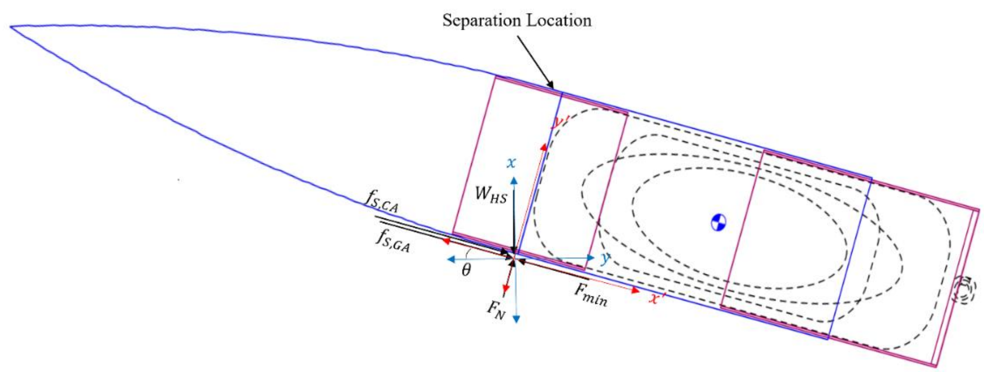


Figure 4-2 Separation FBD when θ ≠ 0°

This possibility can be simulated in MATLAB for all angles from 0-90 degrees to find the worst possible orientation of landing that requires the most force to separate. To affectively quantify a worst case-scenario landing, it will be assumed that the static coefficient of friction between the soil and the airframe is $\mu_{S,GA} = 1$ and that the static coefficient of friction between the fiberglass coupler and airframe is also $\mu_{S,CA} = 1$. These assumptions are certainly inflated, however, more accurate static coefficients of friction between these materials were not found; hence, to quantify a worst-case scenario for the required separation force, these assumptions were implemented. Using this worst case-required force, the design of the mechanical separator will be provided with sufficient redundancy to overcome a variety of other obstacles that cannot be quantified. The weight of the HAUS Section will be assumed to be maximum allowable weight per the derived requirement for the HAUS section $W_{HS} = 250\ \text{oz}$ (further outlined in 6.1.3 Derived Payload Requirements). Figure 4-3 below shows that the maximum force required based on the assumptions outlined is 34.9 lb when the HAUS section lands and remains on a slope of 27°. Note: Since the frictional forces due to the normal force are both equal to one, the effective force due to the sum of these forces is double the normal force for any particular landing orientation.

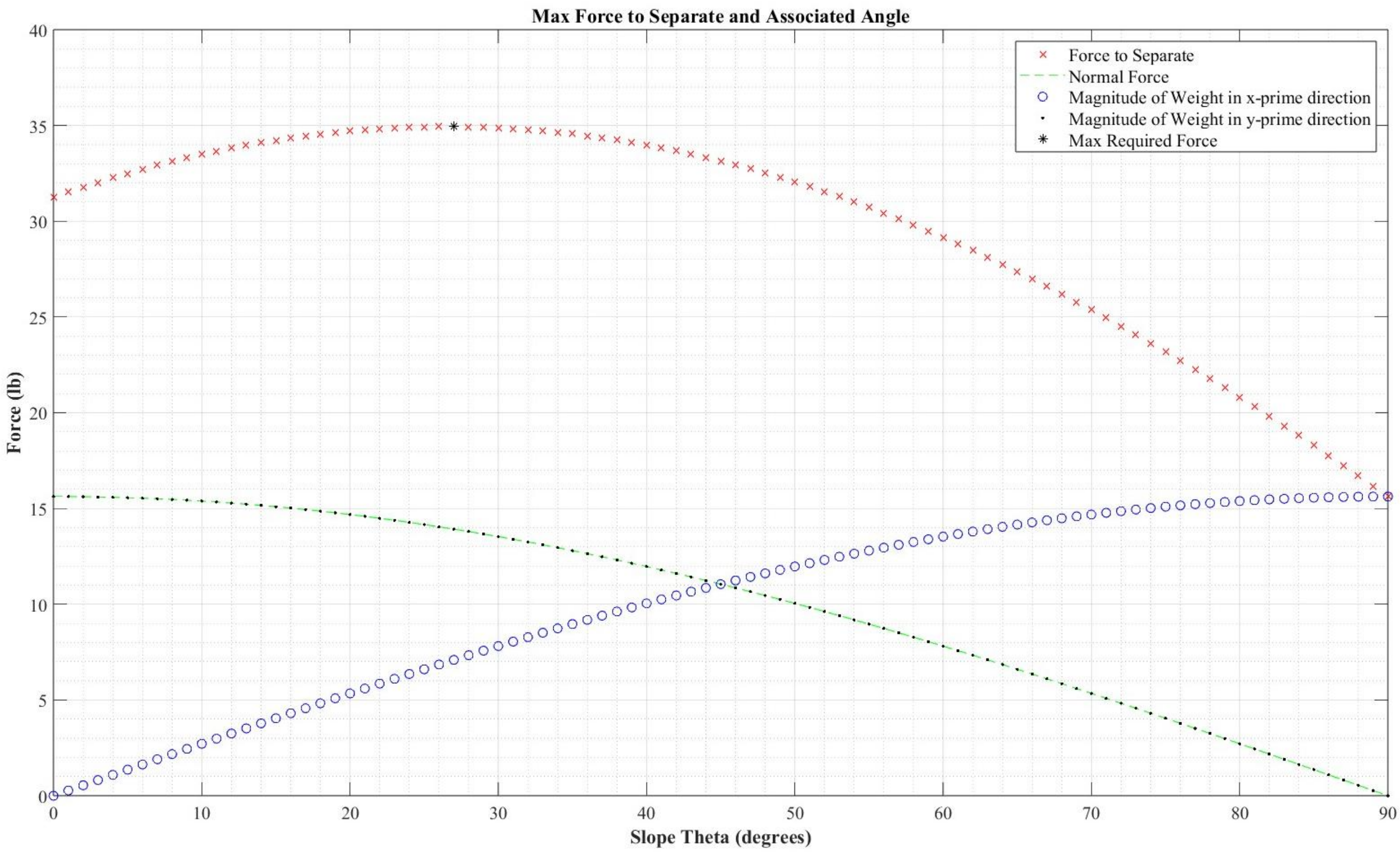


Figure 4-3 Max Force to Separate and Associated Angle

Though there are many limitations to this force estimation, it is overall, an inflated estimation of the required force to separate which is a good basis to move forward from. This is an inflated estimation due to the assumptions of static coefficient of friction for both the soil and coupler to the airframe being equal to one. This is most likely not the case since the rocket airframe will be smoothly painted and if necessary, there are methods to smoothen the contacting surfaces between the coupler and airframe. Also, by analyzing the CG in both figures above, it is clear that the assumption of the weight is acting fully at this point of separation is inaccurate. However, this is also an indication that the approximation of the maximum force required is an inflated value and hence is a good reference point for the design of the mechanical separation. By using this force as the minimum force that must be achieved by the mechanical separation mechanism or MSM, there is built in redundancy in the case of other factors that cannot be quantified or predicted.

Now that the derived requirement of minimum force that a MSM must achieve has been formulated, various mechanisms must be considered. Since the initial reference for this design is separation after landing and thus usage of energetics is prohibited per HB Rule 4.6, there are two primary separation mechanism for consideration. One is a rail and geared rack system. The other is a power screw and nut system. Both concepts are similar, they use the mechanical advantages of geared type systems to produce linear motion from rotary motion of a motor. One advantage of the geared rack and rail is the ability for a large variety of input pinion gears. However, there are many complications with the usage of such a system. Primarily, the method of supporting the pinion gear motor assembly to maintain contact with the pinion. There are high risks of the pinion gear torquing its supporting frame till it loses contact with the rack. One major disadvantage of the rack and pinion are they do not amplify the output force to the extent that a power screw can. Thus, either a more complex gearing system would need to be implemented, or a stronger motor would be required, which would increase costs and also weight. Both systems involve some complications for the proper mounting configurations to provide sufficient separation distance for the deployment of the Terra Inquisitor. On the other hand, the power screw would not be able to slip from the separation nut and so there would be no risk of loss of contact. An example of the forcing abilities of power screws is the scissor jack. This jack is often used for car spare tire kits due to its ability to lift a portion of a car and hold its position with only the required input torque of the average person. The particular ability to be “self-locking” is a major advantage to the power screw. A self-locking screw is a screw that due to frictional forces will not rotate till those frictional forces are overcome by the input torque. A self-locking screw would ensure the HAUS Section would be held fast under the tensile or compressive forces exerted during flight without the need to implement a further means of retention. Below is a decision chart to characterize various component criteria for the furthering of an effective selection between the mechanisms. Both mechanism were compared off of commercial components found on McMaster Carr.

Mechanical Separation			Geared Rack Mech			Power Screw Mech		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.25	USD/ft	9.24	5.5	1.375	4.88	10	2.5
Tensile Strength	.10	ksi	78.3	10	1	63.8	8	.8
Weight	.30	lb/ft	.175	7	2.1	.131	10	3
Handling Safety	.10	Safety Score	3	10	1	3	10	1
Implementation	.25	Experience	7	7	1.75	10	109	2.5
Component # on McMaster Carr			6295K233			98935A205		
Overall Score:					7.225			9.8

Table 4-1 Mechanical Decision Matrix

Cost is the price of the components in USD/ft. Cost was taken into consideration as a criteria due to the need to keep the overall cost of MSM low. Thus, the weighing factor cost is 25%. Rating for this criteria was based on the lowest cost being optimal and so receiving a rating of 10.

Tensile Strength is the maximum strength the material can withstand in uniaxial tension in ksi. Strength was taken into consideration as the commercially available materials of construction must be strong enough to withstand forces from flight and separation. If these components were to fail, the payload would not be able to be deployed and hence the Terra Nova would fail its mission. However, since the commercially available materials are both steels the relative importance of the strength of the materials is less significant to the comparison of the two mechanism. Hence, the weighing factor for material strength is 10%. Rating for this criteria was based on the highest strength material considered the optimal and thus receiving a score of 10.

Weight is weight per length of the components lb/ft. Weight was taken into consideration as a criteria due to the need to keep the overall weight of the MSM as low as possible to allow more room for the design of the payload. Weight of the system will also effect the center of gravity (CG) of the rocket and so effect rocket stability. Thus, the weighing factor for the weight of the component is 25%. Rating for this criteria was based on the lightest component being considered optimal and so receiving a rating of 10.

Handling Safety is the safety hazards from the inherent toxicity of the materials of construction. This is measured based on the Safety scores outlined in Section 5.4.4. Safety Data Sheets (SDS):

Chemicals and Materials. Safety is always an important factor for the usage of any components or materials. However, since both materials are very similar in both being low-carbon steels, there is not a significant difference in handling safety. In addition, since these are commercially available components, there are no manufacturing hazards for the Terra Nova team. Thus, the weighing factors 10%. The material with the lowest safety score is considered the safest material and will receive a rating of 10.

Implementation is measure of how easy it will be to create the MSM using a particular component. It is measured by experience with these systems derived from understanding how they work and what is needed to effectively and reliably integrate them to the rest of the MSM. Due to its overall importance in the selection of the system, the weighing factor is 25%. The component that is considered to be the most effective, reliable, and easy to integrate will receive a score of 10.

Based on all these considerations, the Terra Nova team will implement a power screw as the primary means of mechanical separation.

Now that the power screw is the selected mechanism, the minimum torque that a motor must provide to supply the force of 35 lb required to separate must be determined. The equation 4.1 relates the type of power screw, input torque, and output force.

Equation 4.1: $T_{in} = \frac{F_{out} * d_m}{2} * \left(\frac{l + \pi * \mu_s * d_m * sec(alpha)}{\pi * d_m - \mu_s * l * sec(alpha)}\right)$

$$T_{in} = \text{Torque input}$$
$$F_{out} = \text{Force output}$$
$$OD = \text{Outermost Screw Diameter}$$
$$p = \text{Screw Pitch} = \frac{1}{\#\text{ of threads per inch}} \text{ for a ACME screw}$$
$$l = \text{Screw Lead} = p \text{ for ACME screw}$$
$$\alpha = \text{Lead Angle}$$
$$d_m = \text{Mean Screw Diameter} = OD - \frac{p}{2}$$
$$\mu_s = \text{coefficient of static friction}$$

The separation system incorporates this same McMaster Carr power screw in the analysis above, all the properties are outlined in Table 4-2. The coefficient of static friction was assumed to be .5 for dry and clean steel-to-steel friction (9).

Power Screw Dimensions Table 4-2		
Dimension	Unit	Value
F_{out}	lbf	35
OD	in	1/4
Threads	# threads/in	16
p	in	0.0625
l	in	0.0625
α	°	14.5
d_m	in	0.219
μ_s	dimensionless	0.5
Part	#	98935A205

Table 4-2 Power Screw Dimensions

$$T_{min} = \frac{35\text{lb} * 0.219}{2} * \left(\frac{.0625 + \pi * .5 * .219 * sec(14.5°)}{\pi * .219 - .5 * .0625 * sec(14.5°)}\right) = 2.44\ in * lb = 39\ in * oz$$

Based on this calculation, the minimum required torque that will be required of a motor using this power screw is 39 in-oz.

4.2.2 Mechanical Separation Design

The payload is made to only deploy after landing to fulfill the HAUS mission objective while putting safety and rule compliance first. We created a purely mechanical separation and deployment system because energetics are not allowed for payload deployment. The system creates exit clearance, removes the nose cone, and positions the payload for release in a controlled, repeatable way using a motor-driven lead screw assembly.

A geared lead screw system is powered by a stepper motor, which translates rotational motion into linear translation. The HAUS payload is retained and supported until deployment while the nose cone is pushed forward off the airframe by this linear translation.

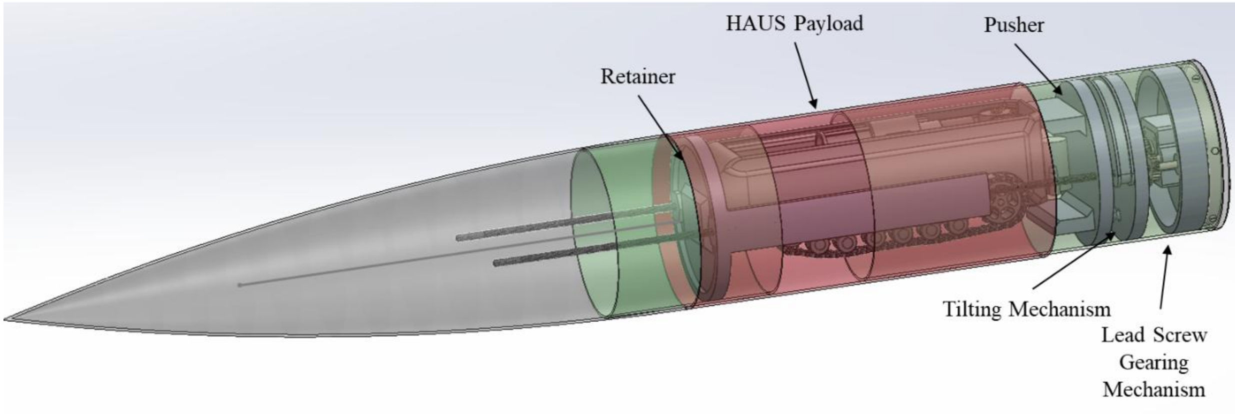


Figure 4-4 Separation Mechanism Assembly

<u>Retainer and Pusher Assembly</u>

Function: Nose cone removal + payload retention and positioning

The retainer is rigidly attached to the nose cone, while the pusher is located just aft of the payload. As the lead screws advance, the retainer and pusher translate together, generating a controlled axial force that:

1. Pushes the nose cone off the body tube to open the forward bay, and

2. Maintains positive retention of the HAUS payload until deployment is initiated.

Material: PETG was selected for its ease of fabrication and sufficient stiffness

<u>Tilting Mechanism</u>

Function: Create clearance for payload exit

Once the nose cone is removed, the payload must have enough clearance to exit the airframe reliably. The tilting mechanism is designed to open the payload bay to approximately 45°, ensuring a clear deployment path without requiring high deployment forces.

Operation:

The tilting mechanism is held in the closed position while constrained by the body tube. After separation, the mechanism becomes unconstrained and spring-loaded arms rotate outward, driving the system to the designed deployment angle. The arms are geometrically arranged such that their overlap and travel limits consistently drive the mechanism to the correct final position regardless of which way the system is positioned post-deployment.

Spring selection rationale:

The team selected nickel-plated extension springs (spring steel construction) with a maximum safe force of 11.5 lbf and a maximum deflection of 2.4 in. This provides sufficient stored energy to rotate the arms through the required travel range (approximately 180° for the small arm and 210° for the large arm), achieving repeatable deployment to the target angle while also being commercially viable.

Design impact:

If testing shows under-rotation, binding, or inconsistent deployment angle, design changes include spring rate adjustment, hinge friction reduction, arm geometry revisions, or addition of mechanical hard-stops to lock the final angle.

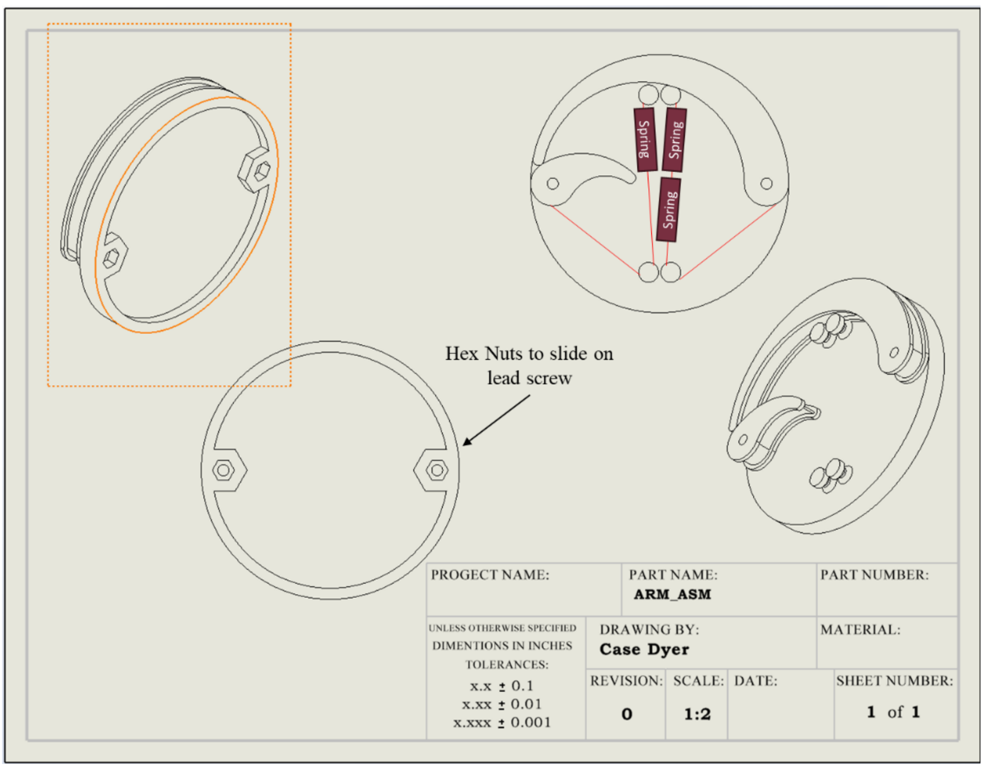


Figure 4-5 Tilting Mechanism

<u>Lead Screw Gearing Mechanism</u>

Function: Provide controlled, repeatable actuation force

The lead screw mechanism is powered by a NEMA 17 stepper motor with a rated torque of 2.65 in*lbf. The motor drives a mirrored geartrain that synchronizes the lead screws and increases available torque to the actuation system.

Geartrain behavior:

The gear layout is configured such that the input gear drives an output gear with approximately 2:1 torque multiplication (output gear pitch diameter ≈ 2× input pitch diameter), with the center gear serving as an idler so that we can keep our lead screws .5 in from the outside coupler. This results in increased torque at the lead screws, improving reliability under friction and misalignment loads while preserving the stepper motor's controllability.

Design impact:

If deployment force is insufficient or motion is inconsistent, the team can adjust gear ratio, motor current limits, screw pitch, or introduce additional alignment constraints (guides/bushings).

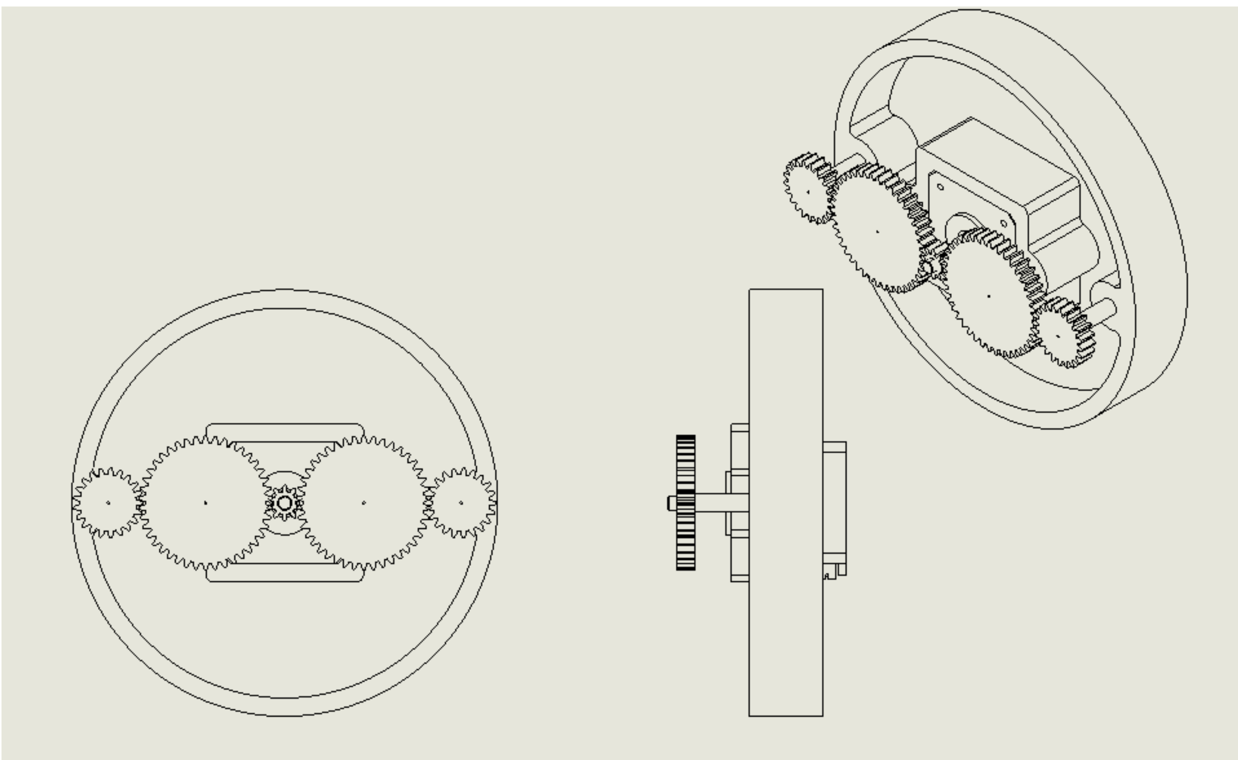


Figure 4-6 Lead Screw Gearing Mechanism

4.2.3 Chassis

Overall, the design of the chassis relied on the optimization of the existing design for weight reduction and durability. There were many possible chassis designs, hence a weight to size design was selected as the design of the chassis. The size of the chassis is limited by the diameter of the coupler the Terra Inquisitor will be housed in during flight. This coupler has a inner diameter of

5.82”; thus setting the largest size of the chassis. In addition to this there needs to be space for the mechanical separation device.

One of the primary considerations was the material of construction. To compare various materials, a decision matrix will be implemented. The decision matrix Table 4-3 incorporates scores calculated by the multiplication of the criteria weight and the rating of the material. The ratings are out of ten; the material that performs the best will receive 10 out of 10. All other material scores for the particular criteria were rated relative to the best. In the event that all material ratings tie for a particular category, the rating for all will be 10.

Chassis			Fiberglass			Carbon Fiber		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.20	$\frac{USD}{in^3}$	1.10	7	1.4	11.7	2	.4
Strength	.15	ksi	80.6	9.75	1.63	87.0	10	1.5
Density	.25	$\frac{\text{lb}}{\text{in}^3}$	.0650	6	1.5	.0578	7	1.75
Handling Safety	.15	Safety Score	6	5	.75	6	5	.75
Formability	.25	Experience / Available Equipment	5	5	1.5	5	5	1.5
Overall Score:					6.78			5.9
Chassis			Creality PETG			Creality PLA		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	.20	$\frac{USD}{in^3}$	.387	9.75	1.95	.370	10	2
Strength	.15	ksi	7.25	4	.6	7.69	4	.6
Density	.25	$\frac{\text{lb}}{\text{in}^3}$	.0448	10	2.5	.0452	9	2.25
Handling Safety	.15	Safety Score	2	10	1.5	2	10	1.5
Formability	.25	Experience / Available Equipment	10	10	2.5	9	9	2.25
Overall Score:					9.05			8.6

Table 4-3 Terra Inquisitor Chassis Decision Matrix

Cost is the price of the material in USD/in^3. Cost was taken into consideration as a criteria due to the need to keep the overall cost of the chassis being low as possible. Thus, the weighing factor cost is 20%. Rating for this criteria was based on the lowest cost being optimal and so receiving a rating of 10.

Strength is the maximum strength the material can withstand in uniaxial tension in ksi. Strength was taken into consideration as a material criteria due the payload chassis needing to withstand the forces during of flight and the impact of landing. If the chassis of the payload were to fail it would render the mission of the rocket a failure. However, since the chassis of the payload will not be the primary absorber of all these forces, the material strength was weighted at 15%. Rating for this criteria was based on the highest strength material being considered the optimal and thus receiving a score of 10.

Density is weight per unit volume of the material of construction in $\frac{\text{lb}}{\text{in}^3}$. Density was taken into consideration as a criteria due to the need to keep the overall weight of the payload chassis as low as possible. This allows room for the weight of other components in the payload to keep the overall weight of the HAUS Section below the maximum 250 oz. The weight of the payload will also effect the center of gravity (CG) of the rocket and so effect rocket stability. Thus, the weight factor for density is 25%. Rating for this criteria was based on the lowest density being considered optimal and so receiving a rating of 10.

Handling Safety is the safety hazards from the inherent toxicity of the materials of construction. This is measured based on the Safety scores outlined in Section 5.4.4. Safety Data Sheets (SDS): Chemicals and Materials. Safety is always an important factor for the usage of any components or materials. Thus, the weighing factors 15%. The material with the lowest safety score is considered the safest material and will receive a rating of 10.

Formability is measured by experience using the material and available equipment to form the material. This criteria was included due to the chassis being custom fabricated by the Terra Nova team. Thus, the formability of the material based on previous experience and available equipment is important for the selection of the chassis material and so the weight factor is 25%. Rating for this criteria was based on the material that the Terra Nova team has the most experience with and the most equipment to form receiving a rating of 10.

Based on all these selection criteria, Creality PETG was the selected material for the design of the chassis with the highest score of 9.05.

4.2.4 Chassis Design

Taking into consideration these factors, the design of the chassis can be observed in FIGURES

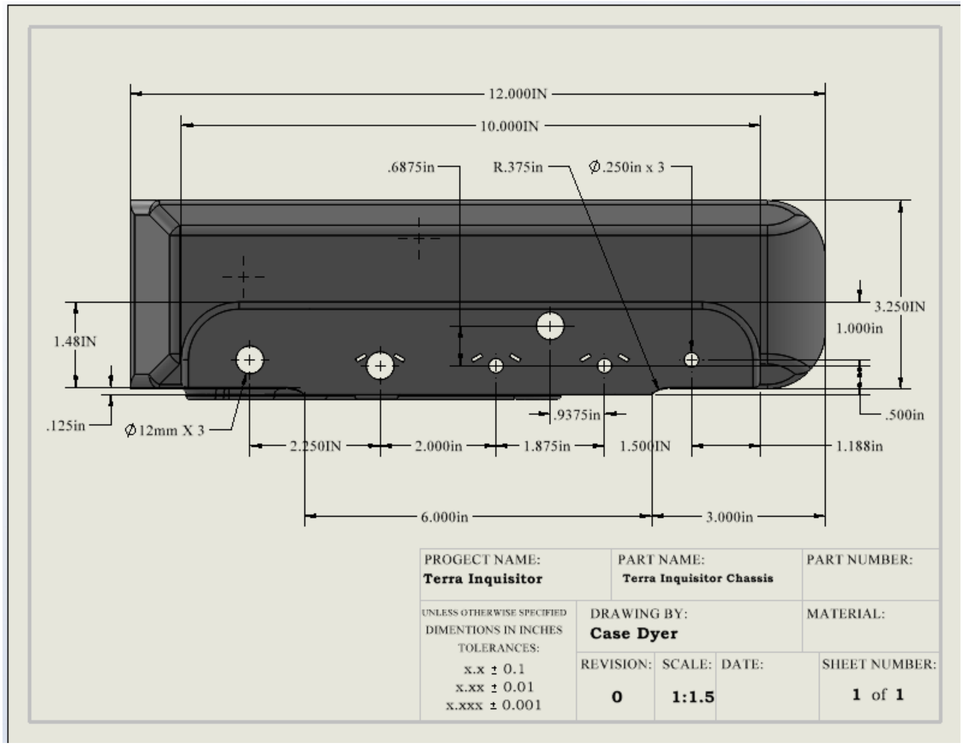


Figure 4–7 Terra Inquisitor Side View

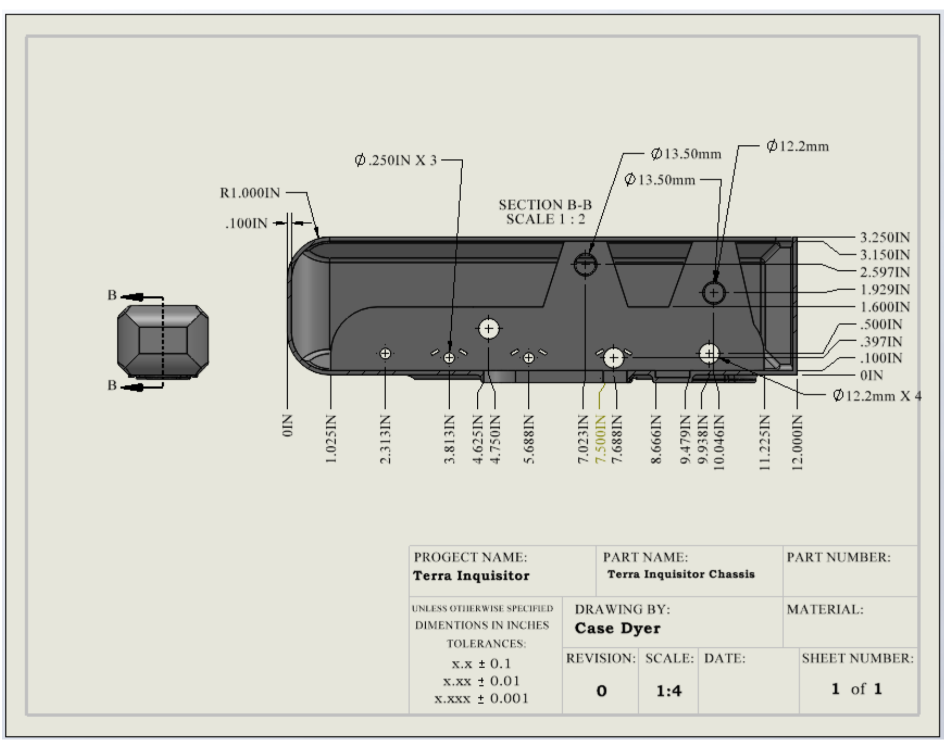


Figure 4–8 Terra Inquisitor Side Cross-Sectional View

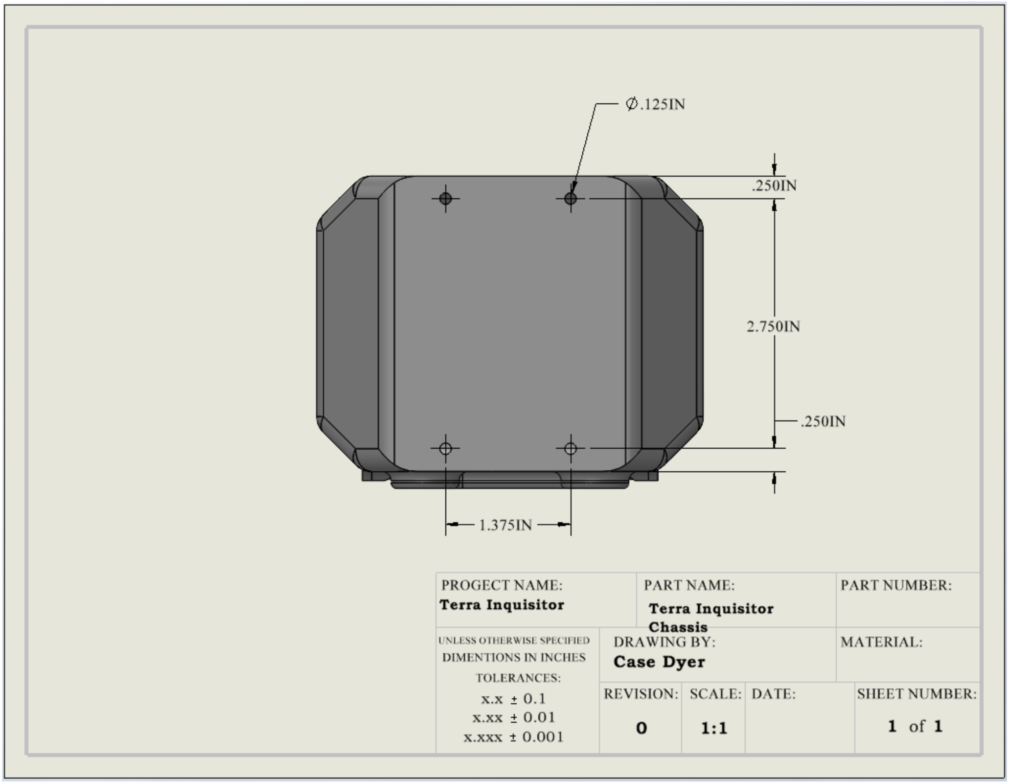


Figure 4–9 Terra Inquisitor Rear View

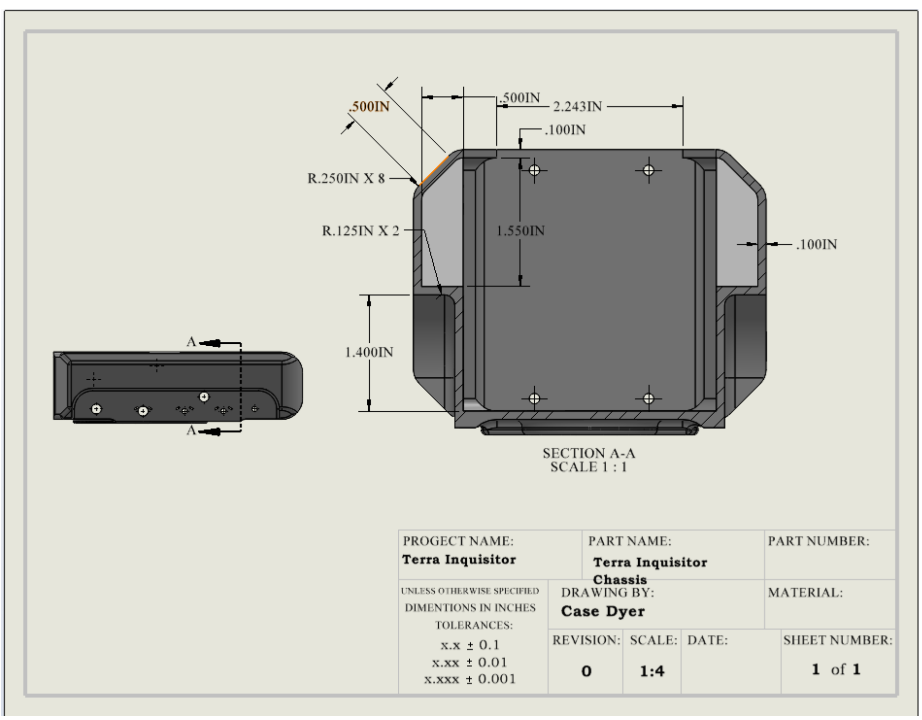


Figure 4–10 Terra Inquisitor Front Cross-Sectional View

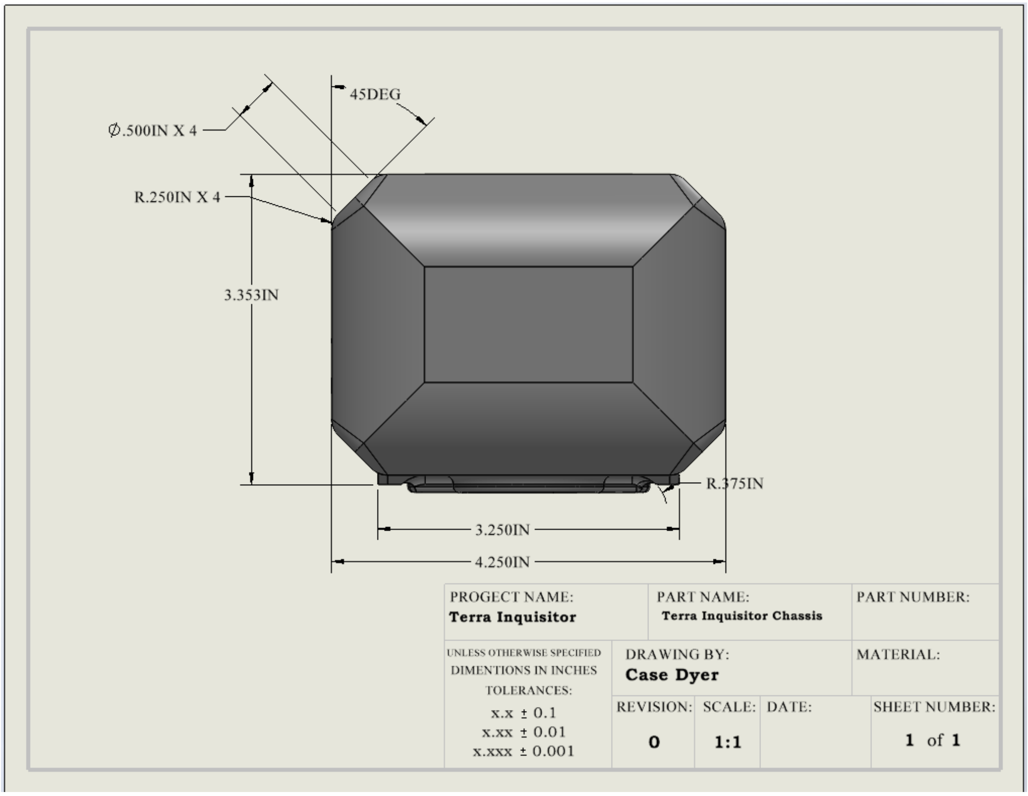


Figure 4–11 Terra Inquisitor Front View

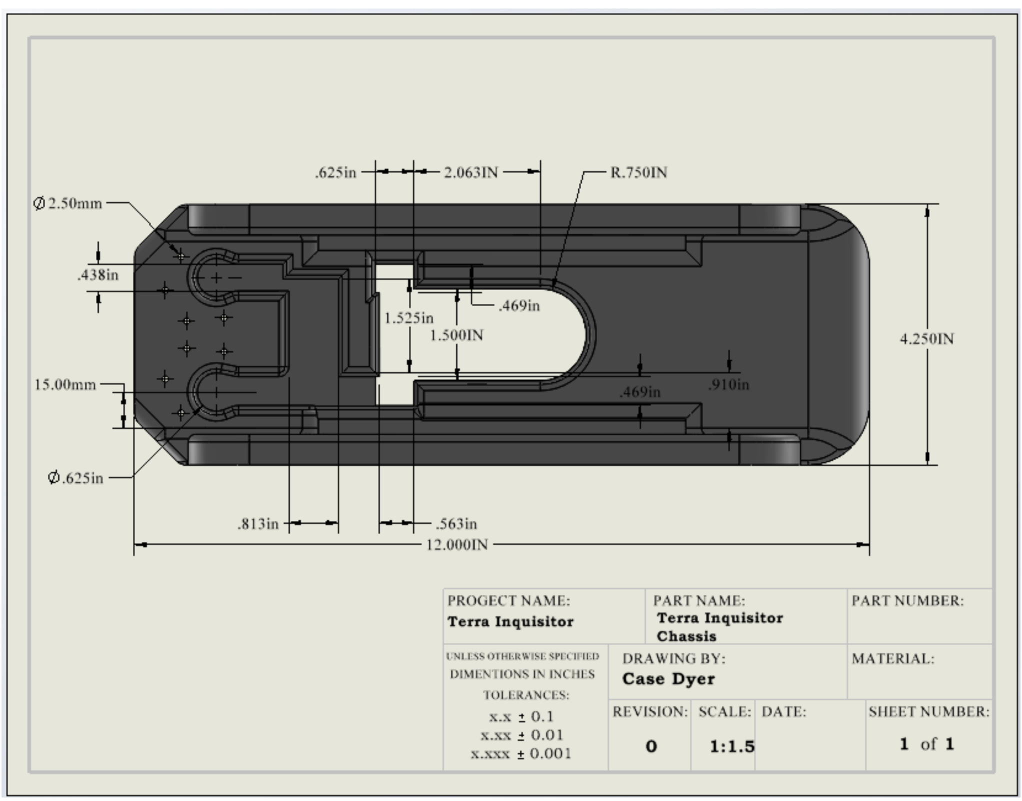


Figure 4–12 Terra Inquisitor Bottom View

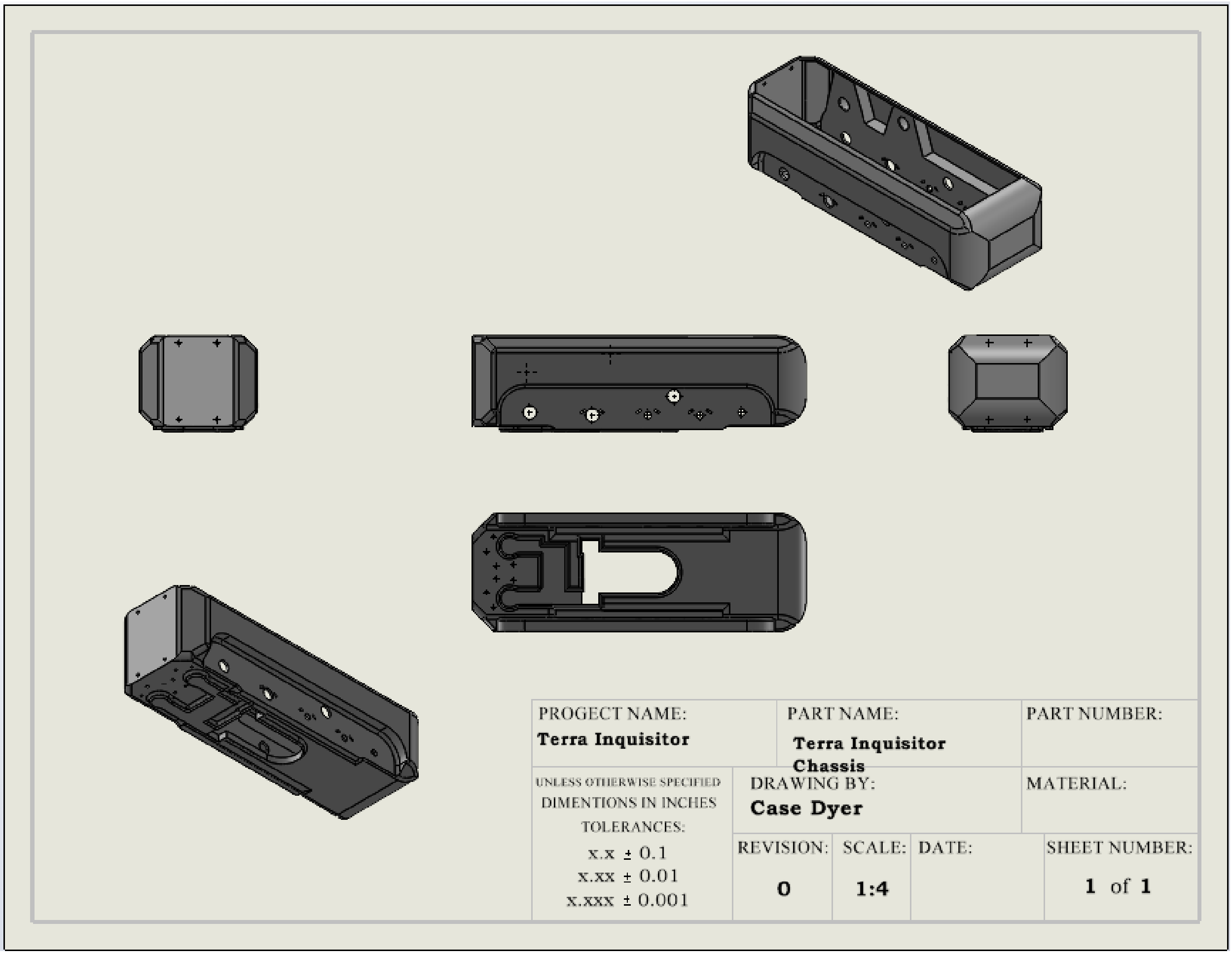


Figure 4–13 Terra Inquisitor Various Views

4.2.5 Propulsion System

Based on the overall design and material of construction for the chassis selection, the force the chassis will be required to withstand for the payload propulsion system and also the required driving force ($F_{d,min}$) from the propulsion were defined. This driving force was then be used in conjunction with the radius of the system to determine the required torque (T_{min}) to be provided by the electrical motors. This was defined by using a similar methodology to the separation force. All forces were assumed to be acting at a single point. The forces the payload propulsion system were be required to overcome are the rolling resistance (f_{RR}) of the ground contact system (such as a track system) and the weight of the Terra Inquisitor (W_{TI}) when at angles $\theta \neq 0°$. Two FBDs (Figures 4-4 and 4-5 on the next page) illustrate this simplified system.

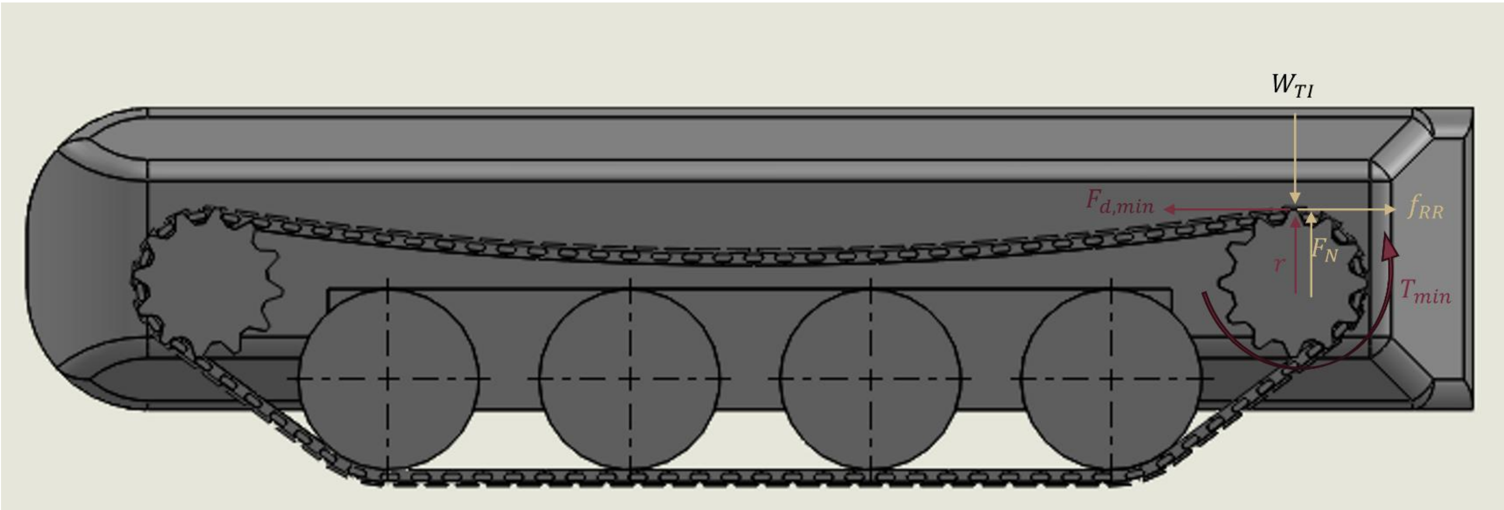


Figure 4-14 Driving FBD when θ ≠ 0

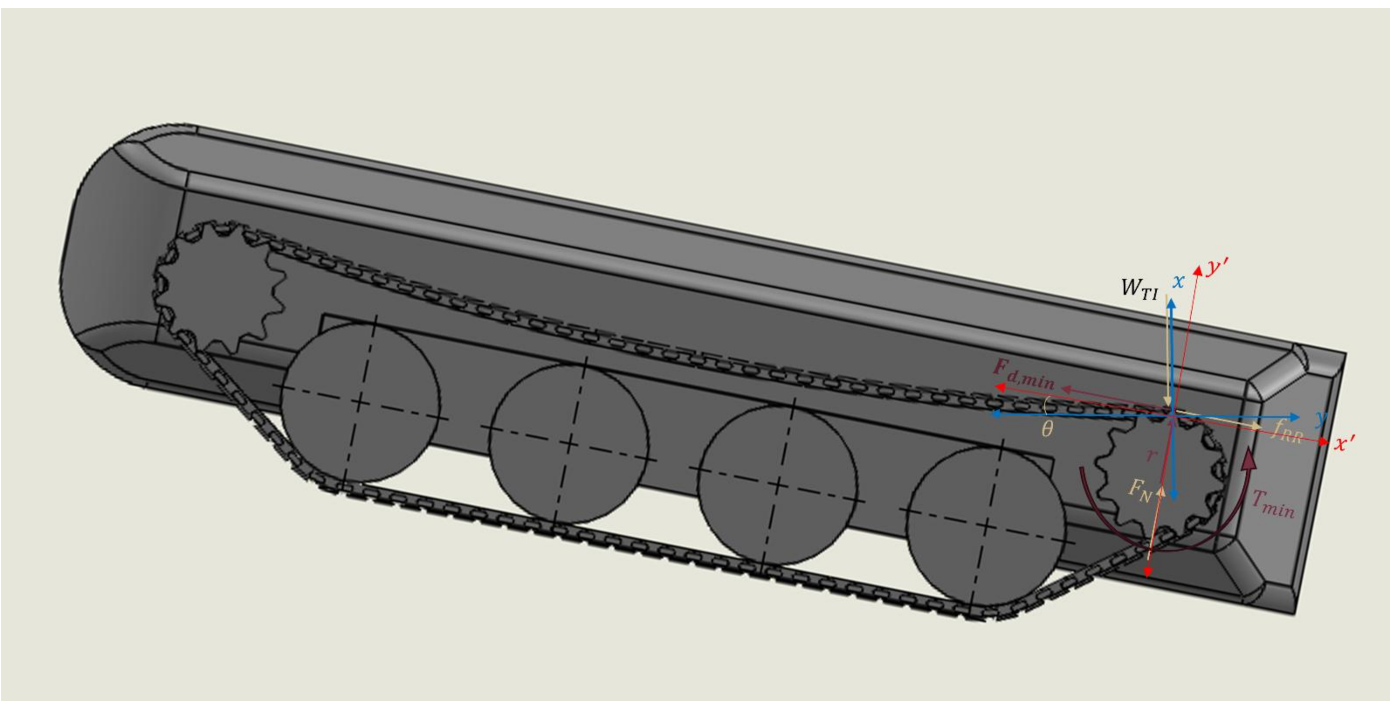


Figure 4-15 Driving FBD when θ ≠ 0°

These possibilities were simulated in MATLAB for all angles from 0-90 degrees to find the maximum force (and thus torque for a particular radius) required for this system and at what orientation it occurred. To effectively quantify this maximum force, it was assumed that the rolling resistance of ground contact system is .4 for tires in soft sand (9). Using this worst-case required force, the design of the Inquisitor propulsion system designed is designed with sufficient redundancy to overcome a variety of other obstacles that cannot be quantified. The weight of the

Terra Inquisitor was assumed to be maximum weight allowed for the system of 96 oz. This is based on the maximum allowable weight per the derived requirement for the HAUS section $W_{HS} = 250\text{ oz}$ (further outlined in 6.1.3 Derived Payload Requirements) minus the weight of all the airframe components and the mechanical separation mechanism. Figure 4-6 below shows that the maximum force required based on the assumptions outlined is 6.5 lb when the Inquisitor drives on a slope of 68°. Since rolling resistance is a function of the coefficient of rolling resistance and the normal force, the number of tires or tracks will not affect the total rolling resistance since all the normal forces will always sum to the magnitude of the weight of the payload in the y-prime direction.

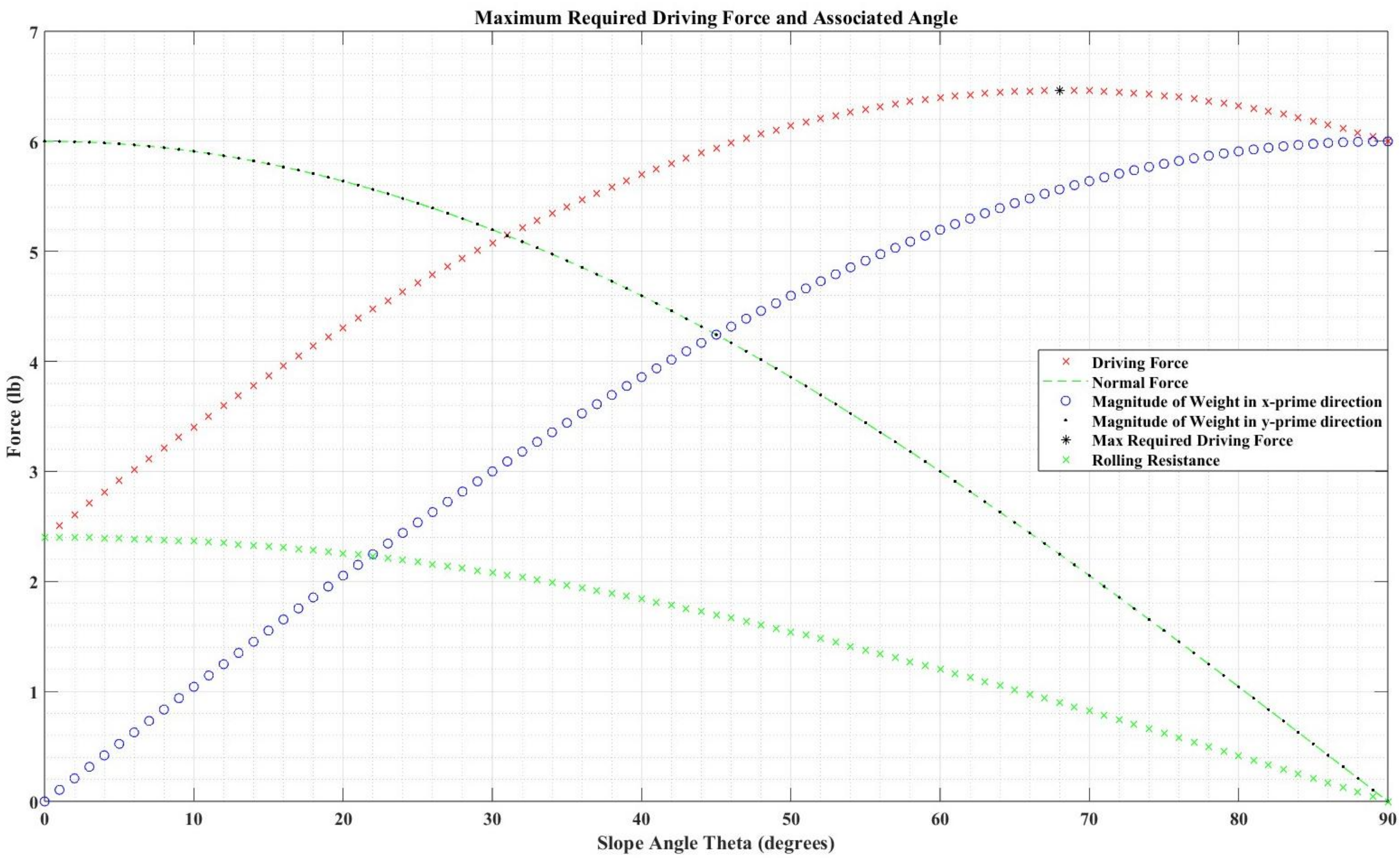


Figure 4-16 Maximum Required Driving Force and Associated Angle

Based on this maximum force, the output torque required by a motor was determined based on the ground traction system. There were two primary forms of ground traction systems considered, wheels and tracks. Both had many advantages and disadvantages and incorporated their own unique challenges to the payload design. A tire system would either require some sort of steering system or the risky usage of tank steering due to lower traction capabilities. In addition to this, the wheel size cannot easily be increased to improve traction due to the size constraints to fit within the coupler. To increase the tire size would necessitate the reduction in distance between the two front tires or the two rear tires thus increasing instability. The track system would require multiple road or roller wheels, proper sizing of the track, proper sizing of the sprockets, and a track tension system. However, the tracks reduce the pressure acting between the track and the ground thus reducing sinkage of the vehicle into the various terrains it will encounter. This reduction in interaction pressure will be critical as the Terra Inquisitor will need to be able to handle both soft sand and mud. Tracks have superior traction and therefore are much more suitable for tank steering navigation. The traction increase of the tracks also does not come at the same size cost as tires. According to experimental results completed in “The Tractive Performance of Rubber Tracks”, the rubber tracks tested were found to always have less slip than tires. Their rolling resistance was largely independent of the type and more depended on track tension and inherent power losses in track systems. The authors noted that the tires tested always proved to have deeper slots dug into the ground and their rolling resistance was very dependent on tire-soil interactions (8). Based on the analysis of this research and the need for the Terra Inquisitor’s to be able to handle various terrains, a track system was the preferred system and thus implemented into its propulsion system as can be seen in FIGURES. Implementing a track system will reduce the dependency of the reliability of on soil conditions based on the findings of the earlier research. This will ensure the payload design will be sufficient for any soil conditions it may experience. It is also notable that the experimental data for the tracks proved that the largest measured coefficient of rolling resistance was .24. This further proves that the previous simulations used an over estimated rolling resistance and thus estimate a greater input force than can be expected. However, the previous resistance will still be used to allow for the various interaction is the driving of the track system that will add to the losses of the system. This further allows for redundancy in the design of the propulsion system in the event of other unquantifiable obstacles and losses.

The next consideration was the number of motors necessary to drive the system. At minimum two motors are necessary to drive the left and right sides of the track system independently to allow track steering. Every additional motor implemented involves more wiring, power consumption, motor drivers, proper motor mounting, more software to control each motor and to ensure proper timing for all desired motions, etc. Based on this, the less motors required the more optimal the design. Using the required minimum output force determined earlier, the required torque to drive the systems was estimated. The design of the track system uses a reference diameter of 1.625 inch driving sprocket, due to size constraints. The total required torque is $\mathrm{T} = \mathrm{F} * \mathrm{r} = 6.5\,\mathrm{lb} * (1.625\mathrm{in}/2) =$ in-lb. $=$ 60.5 kg-mm or 84 oz-in. This torque can be provided by a single motor. Hence only two motors will be used to reduce complexity of the design and complexity of implementation. Using two motors, the required minimum torque that must be supplied by each motor is half the total torque, $\mathrm{T}_{single\ motor} = 30.2$ kg-mm or 42.0 oz-in.

Multiple systems were employed to form an effective propulsion system. These systems are detailed in the Figures below.

4.2.6 Propulsion System Design

The propulsion system of the payload incorporates multiple systems, a motor system, idler wheel assemblies, the tracks on either side, tensioner wheel, etc. Below is the an exploded view of the system.

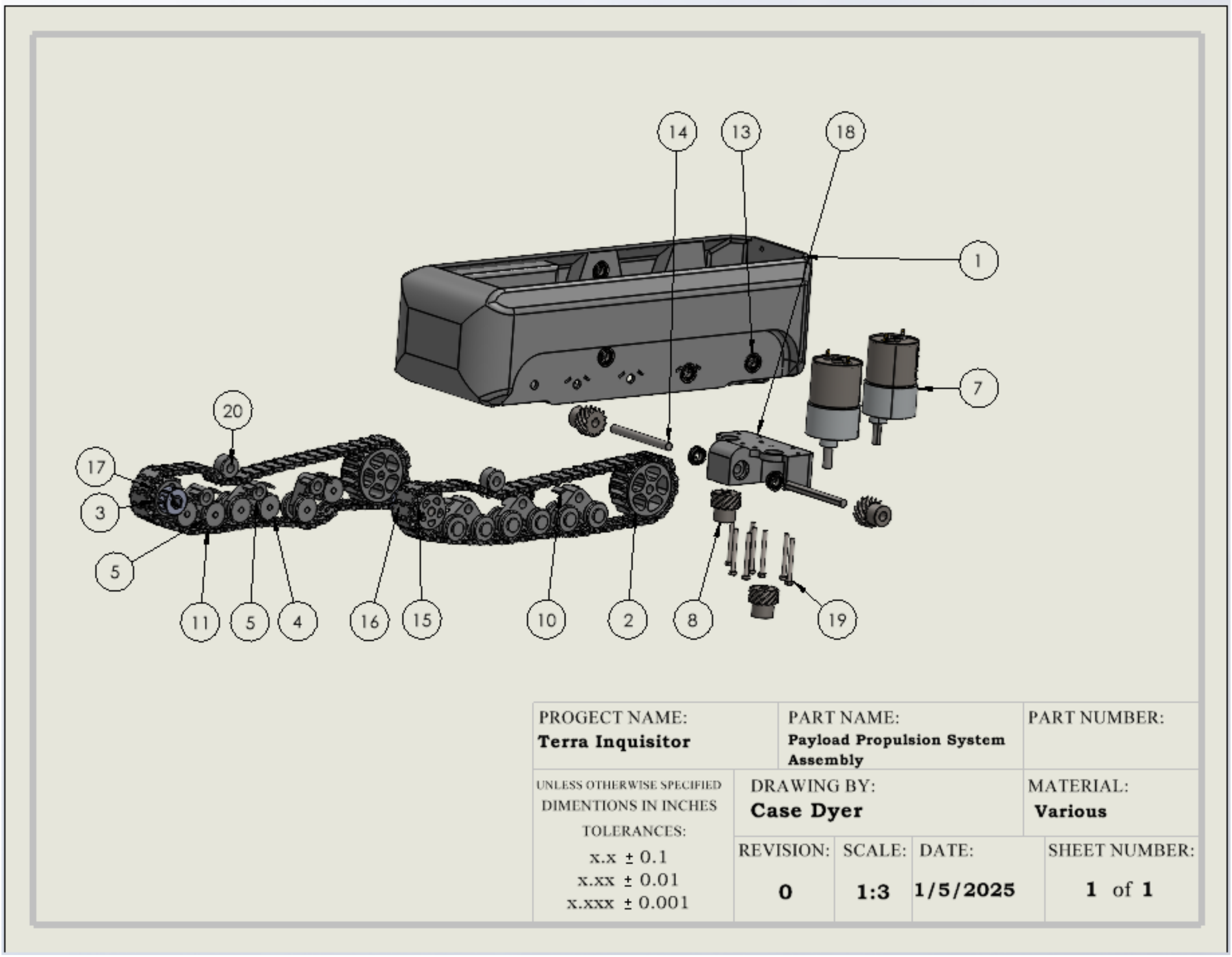


Figure 4-17 Payload Propulsion System Assembly

Payload Components List		
Item Number	Component Name	Quantity
1	Terra Inquisitor Chassis	1
2	Rear Sprocket	2
3	Front Sprocket	2
4	Idler Wheel	24
5	Wheel Rotator	6
6	Wheel Rotator Pin	12
7	Pololu 4741 Motor	2
8	3598N291	4
9	Lipo S3 Battery	1
10	Idler Wheel Assembly Guide	12
11	Track Chain	110
12	Track Pin	110
13	6mm ID Flanged Ball Bearing	14
14	6mm D-Shaft (Stainless Steel, 60mm Length)	2
15	2910-0919-0006.STEP	2
16	2800-0004-0010_90128A207.STEP	2
17	6mm D-Shaft (Stainless Steel, 30mm Length)	2
18	Rear Motor Mount	1
19	B18.6.7M - M3 x 0.5 x 30 Indented HHMS --30N	8
20	Tensioner Wheel	2
21	4mm ID Non-Flanged Ball Bearing (10mm OD, 4mm Thickness)	24

4.2.7 Soil Collection and Containment

The soil collection system for the Terra Inquisitor is a crucial system to the overall mission of the payload to collect and test soil. There were various systems considered for the collection of soil. All the designs considered assumed that the system would be implementing a form of drilling by an auger. This was assumed as very common form of soil collection and the most versatile for the various soil conditions. It has also been implemented by NASA for various lunar and Mars missions such as the Perseverance Rover, equipped with multiple percussive rotary drills. Due to the limited space available in the Terra Nova, the drill had to be compact and moved from its horizontal storage location during flight to a vertical stage for soil collection. Two primary methods were considered by the Terra Nova team, a linkage system that can achieve a fluid motion for the drilling system or a mechanism that will be rotated into each of phase. The primary problems for such a mechanism was the requirement of many different motors to achieve all desired phases of motion in addition to the motor for the drilling. Every additional motor requires more space, increases electrical power consumption, increases cost, and weight. Linkages can be more challenging to synthesize; however, to ensure an optimized design, they were still considered. A linkage would only require one motor to actuate the full motion of drilling and then one motor to power the drill. The linkage synthesized could provide inherent mechanical advantage as well as support for the drilling assembly during flight and recovery. The primary challenges to the linkage were the space required, fulfillment of desired motion, linkage joining methods, ensuring no interference between the links, the required fabrication of all links implemented, and link material of construction. Though there are many challenges to the linkage design, it was considered the superior design based on reduction in electrical power consumption thus requiring less batteries which were found to be a significant contributor to the overall weight of the payload. It also reduces coding complexity by reducing the number of motors that must act in harmoniously to provide the desired motion. Lastly, the inherent mechanical advantage of the system further reduces the necessary power input. Many different linkage combinations were synthesized and evaluated to determine a linkage capable of the desire motion, mechanical advantage, and space requirements. This synthesis was achieved by the selection of a particular coupler curve from a symmetric four-bar coupler curve atlas in "Design of Machinery" by Roert Norton. The primary initial coupler curve selection was based on coupler curves with two straight portions roughly 90° apart. After examining the various coupler curves, the coupler curve in Figure NUMBER was selected for further consideration and analysis. This analysis was achieved through MATLAB simulations of the position of each linkage based on the input angle, the position of the desired output point P, Coupler Curve of point P, angular velocity of all links in initial four-bar configuration, angular acceleration, force in each joint for the primary four bar, and the mechanical advantage achieved at desired point P.

Figures BLANK AND BLANK are the simulations for the first linkage considered in extensive depth. The linkage was designed using link ratios based on the crank length ($\mathrm{L_2}$) (equal to 1.5 units in this case) which are $\mathrm{L_1}\ (\text{Ground Link}) = \mathrm{L_3}\ (\text{Coupler Link}) = \mathrm{L_4}(\text{Rocker Link}) = \mathrm{BP}$ (line segment connecting the joint of coupler and crank with the point of interest P) = 2.0 * L_2 with an angle gamma from the L_3 to BP of 225°. Since these links are proportional to the size of the crank link, the crank size can be determined, and all other links will be scaled accordingly.

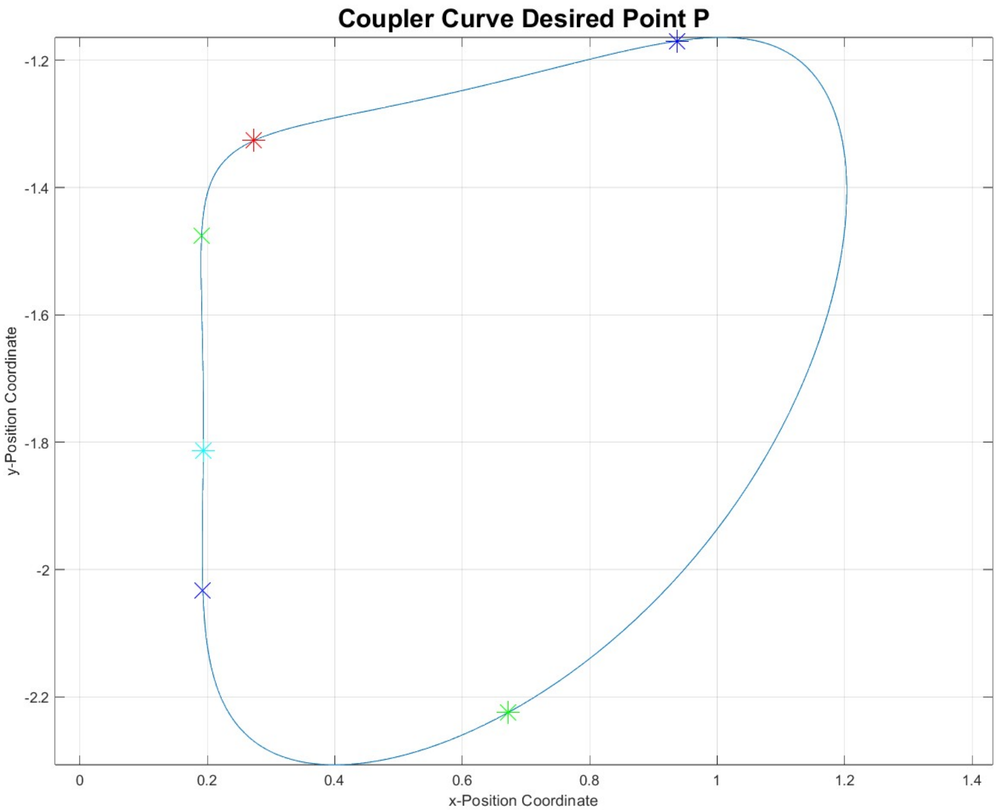


Figure 4-18 Initial Drill Linkage Coupler Curve

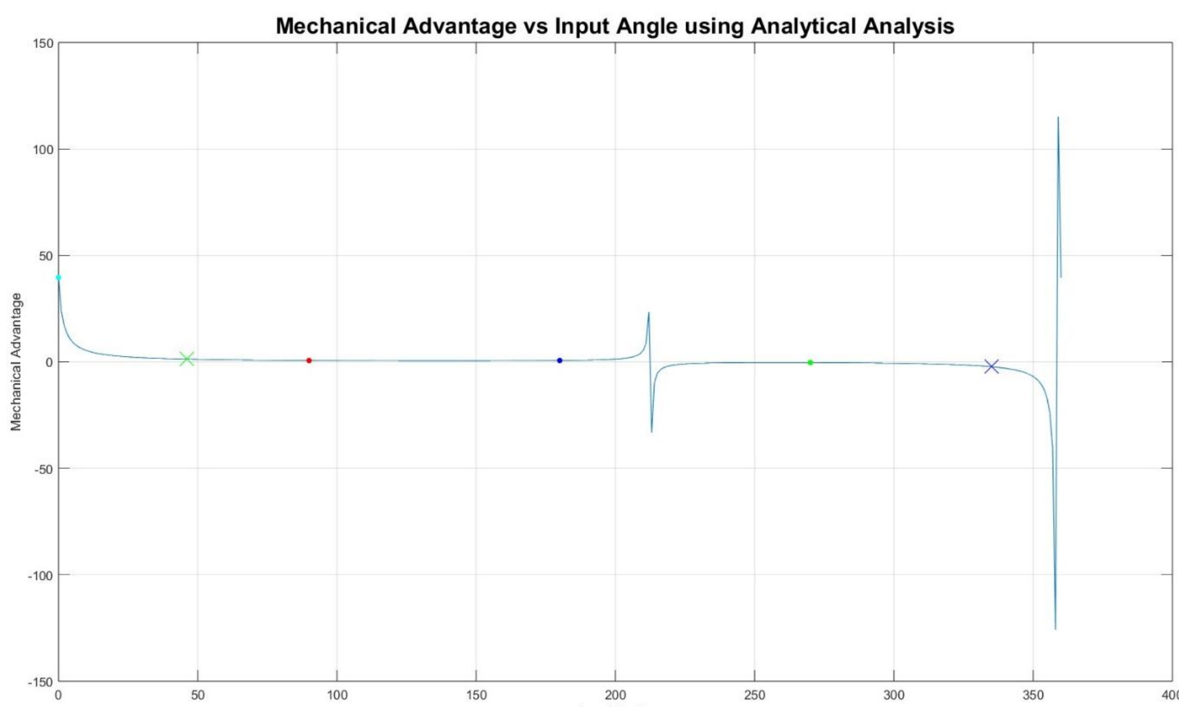


Figure 4-19 Initial Drill Linkage Coupler Curve

This linkage achieved a vertical motion of .6” when properly size. It exhibited excellent mechanical advantage characteristics for the vertical drilling phase with an average mechanical advantage of 17. However, in spite of, the obtained motion range motion is insufficient for the an effective drilling motion. This system also required significant amount of space. In addition the space requirements and insufficient motion, it was observed that the system would cause the multiple linkage to contact the ground and so not allow the drill to achieve its proper orientation. The Figure blank below exhibits these effects.

Due to this, multiple new designs were created. The linkage below provided excellent motion while remaining comparatively compact. Initially the mechanical advantage of the system was promising with a peak mechanical advantage of 57 and an average of 3. However, this mechanical advantage dropped to 0.02 in the most crucial range of motion, the vertical motion drilling phase. This indicates that the system would have to be capable of an input force 50 times that of the output force. This is unfeasible for any motor able to fit within the launch vehicle, within the budget, and with the power requirements to achieve. In addition to this, the vehicle chassis and the links have not been designed to withstand such magnitudes of forces and would thus pose significant safety hazards when in operation. Hazards to both the vehicle and its systems as well as the team members operating the systems. In addition to this, the extreme compactness of the system posed a challenge for the joining of all the links. Thus, this linkage was not selected.

The final linkage considered achieved a vertical range of 1.3”. Nearly double that of the first linkage, with a significant decrease in the special requirements. The system also is oriented in such a way as providing room for various link joining methods that will be outlined later. The desired range of motion follows the red star to the blue “x” on the figure below.

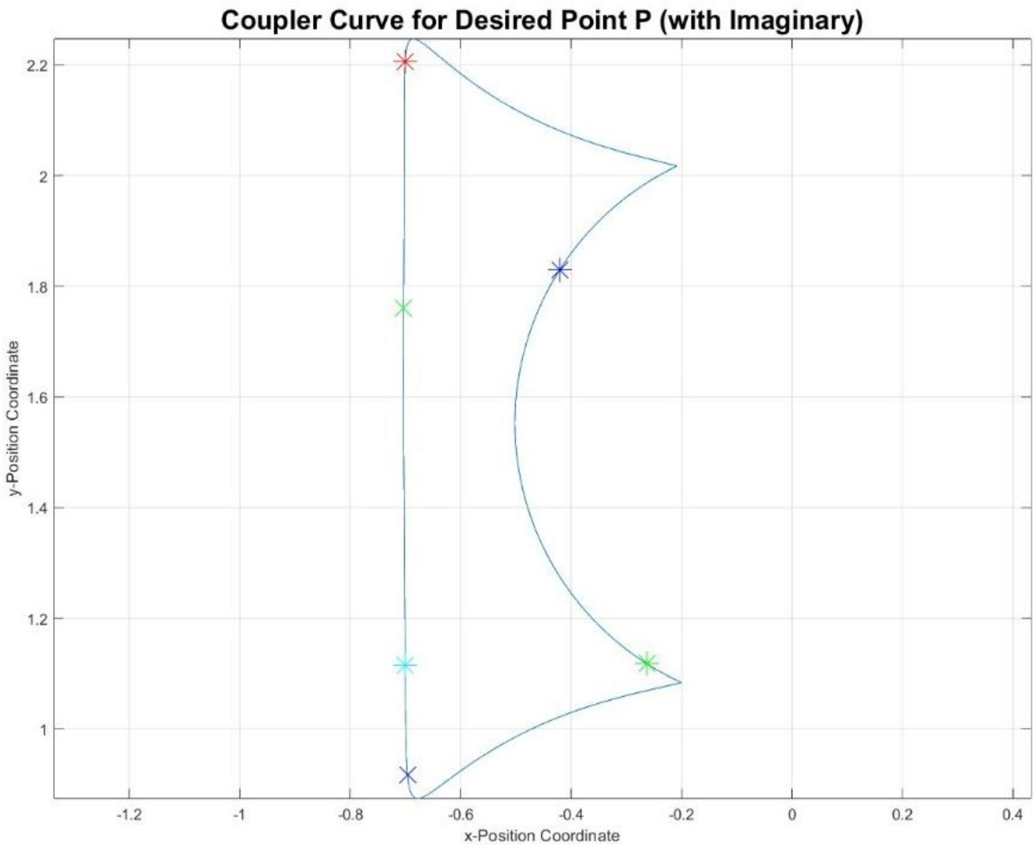


Figure 4-20 Final Drill Linkage Coupler Curve

The mechanical advantage the linkage provides is a peak of 66, an average of 4.4, and a minimum of 1.5 during the vertical range of motion. This can be observed in the figure below. The range of motion follows the specified point of 90° traveling clockwise (backwards) to 335°. These two input angle positions correspond to the red star and the blue "x" in the above coupler curve respectively.

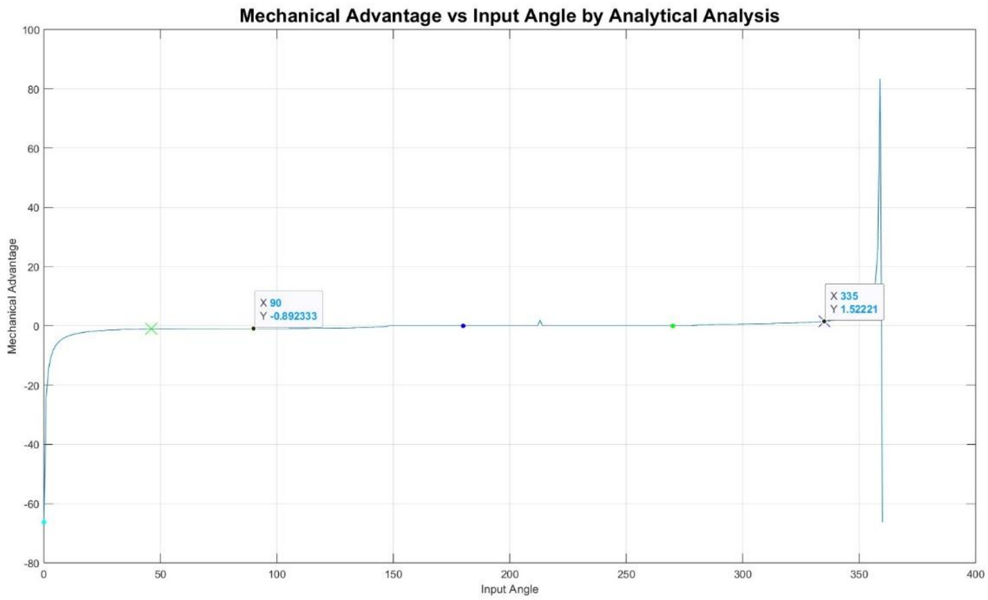


Figure 4-21 Final Drill Linkage Mechanical Advantage

4.2.8 Soil Collection Design

Below are the detailed drawings of each individual linkage, the linkage system outside the payload, and then incorporated into the payload systems.

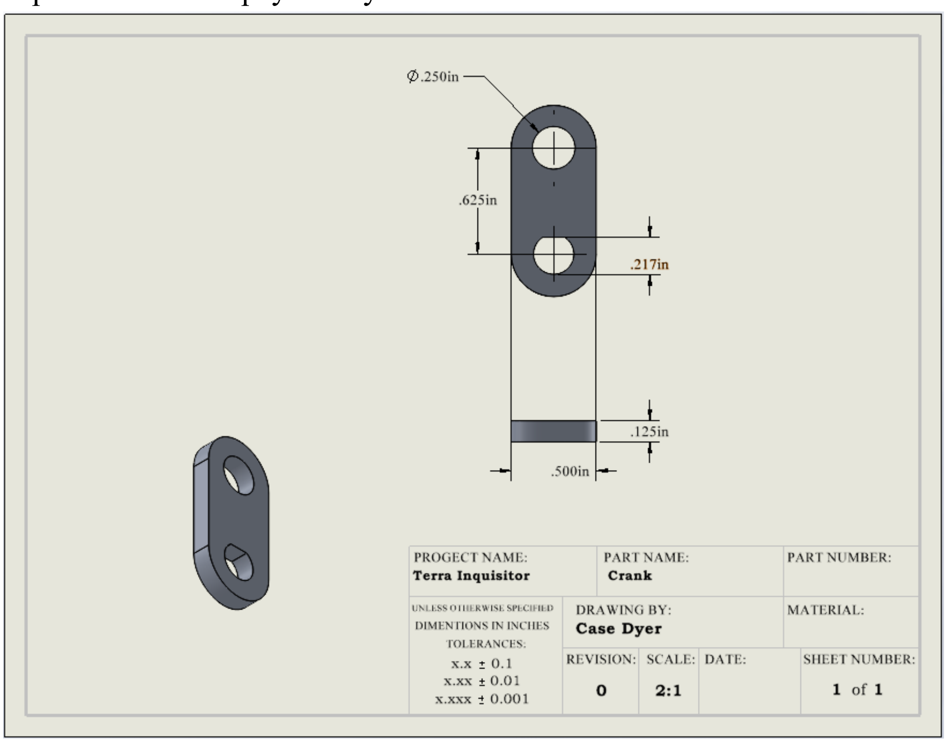


Figure 4–22 Crank (L1)

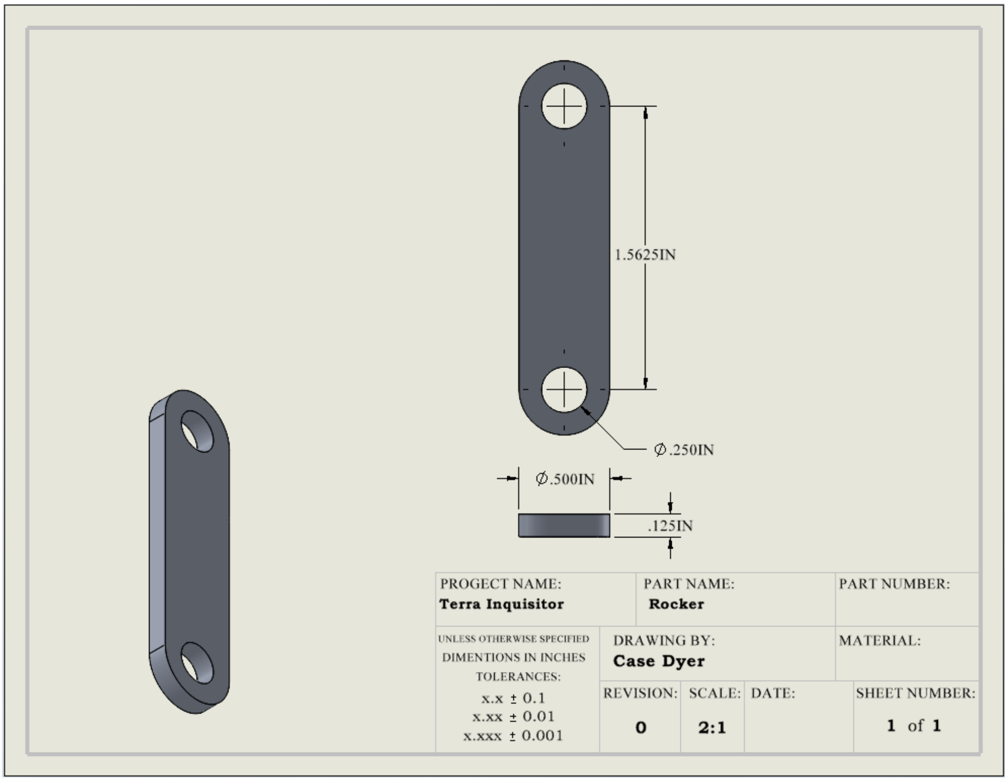


Figure 4–23 Rocker (L2)

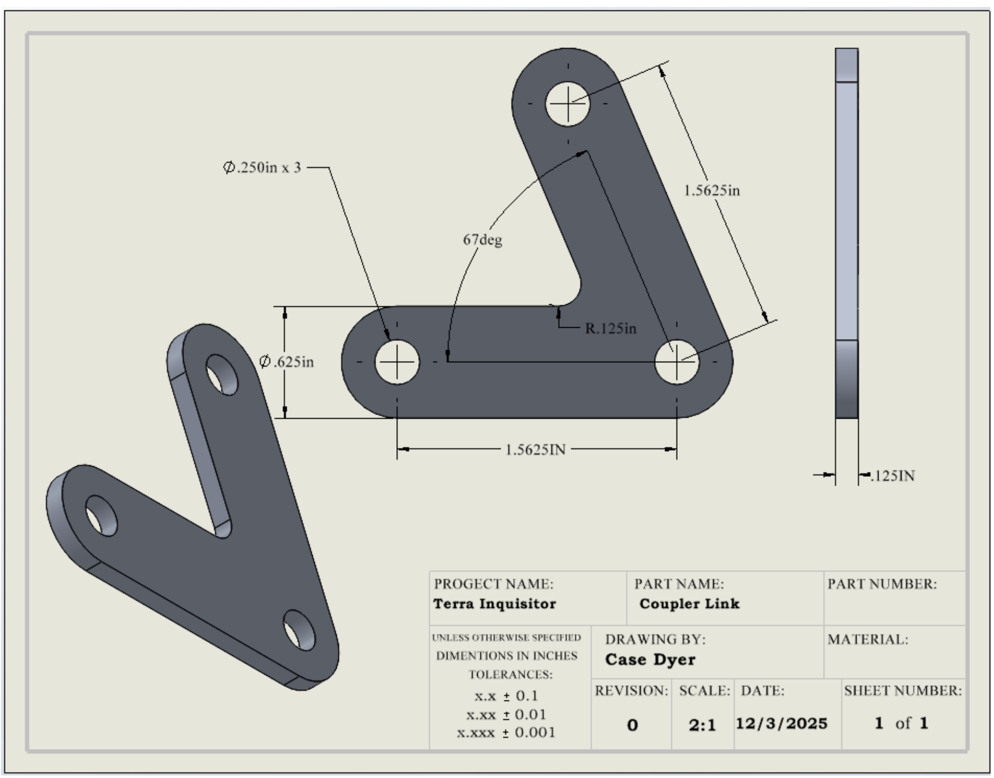


Figure 4-24 Coupler Link (L3)

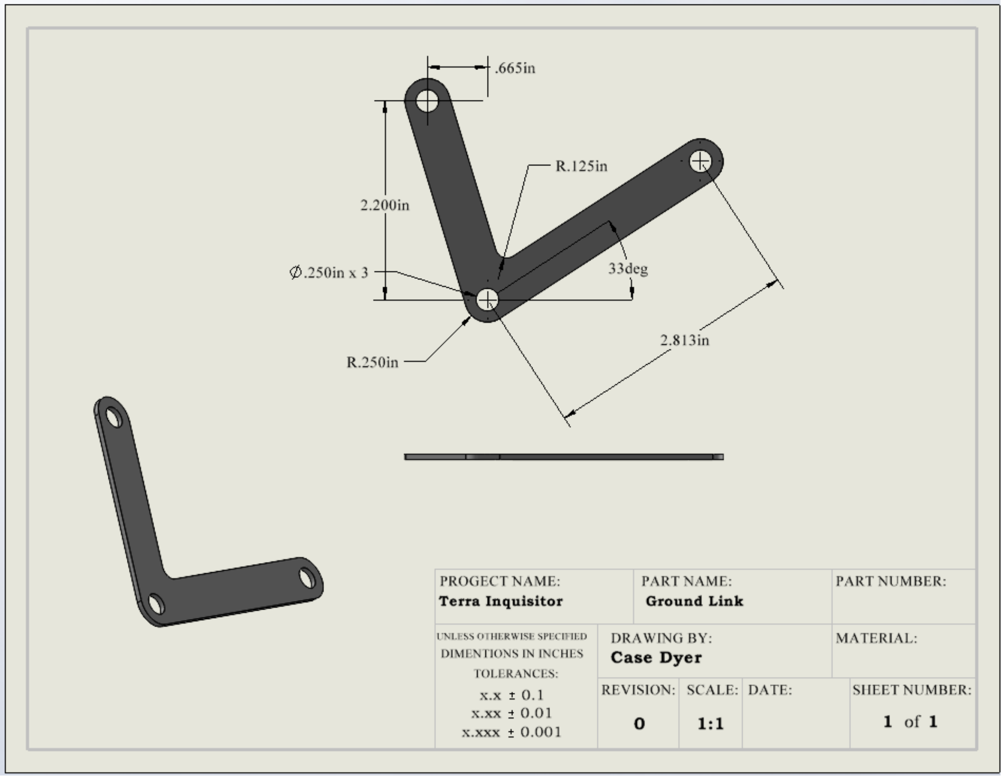


Figure 4-25 Ground Link Representation (L4)

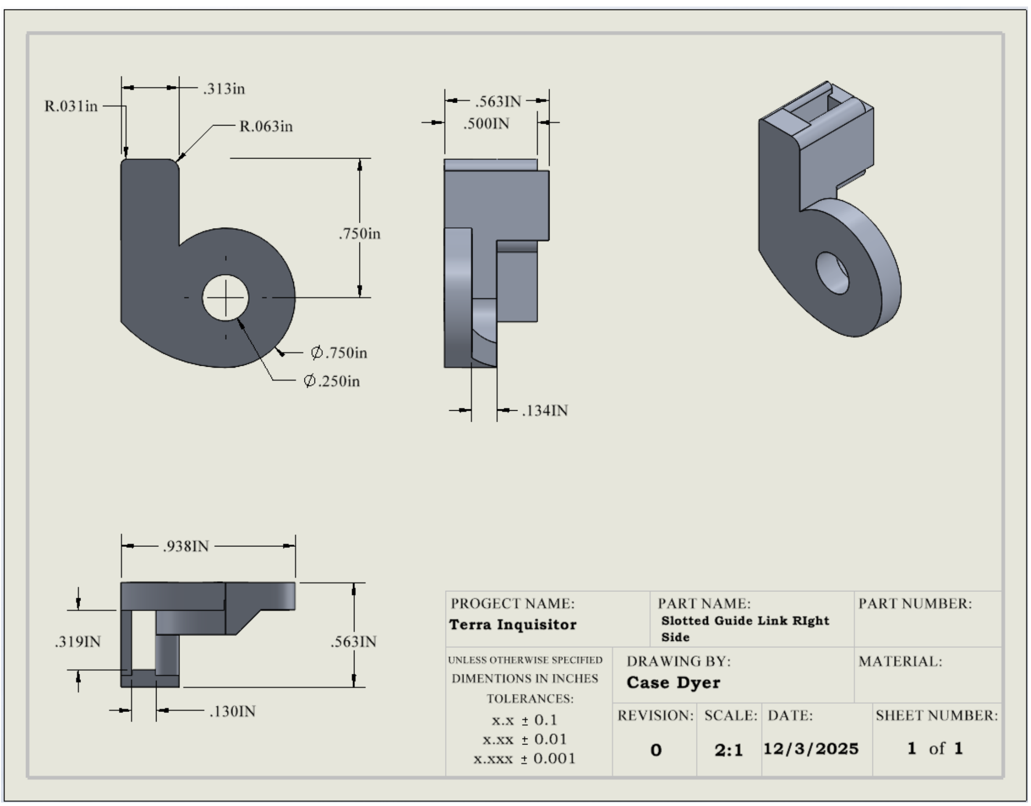


Figure 4-26 Slider Guide Link (L6)

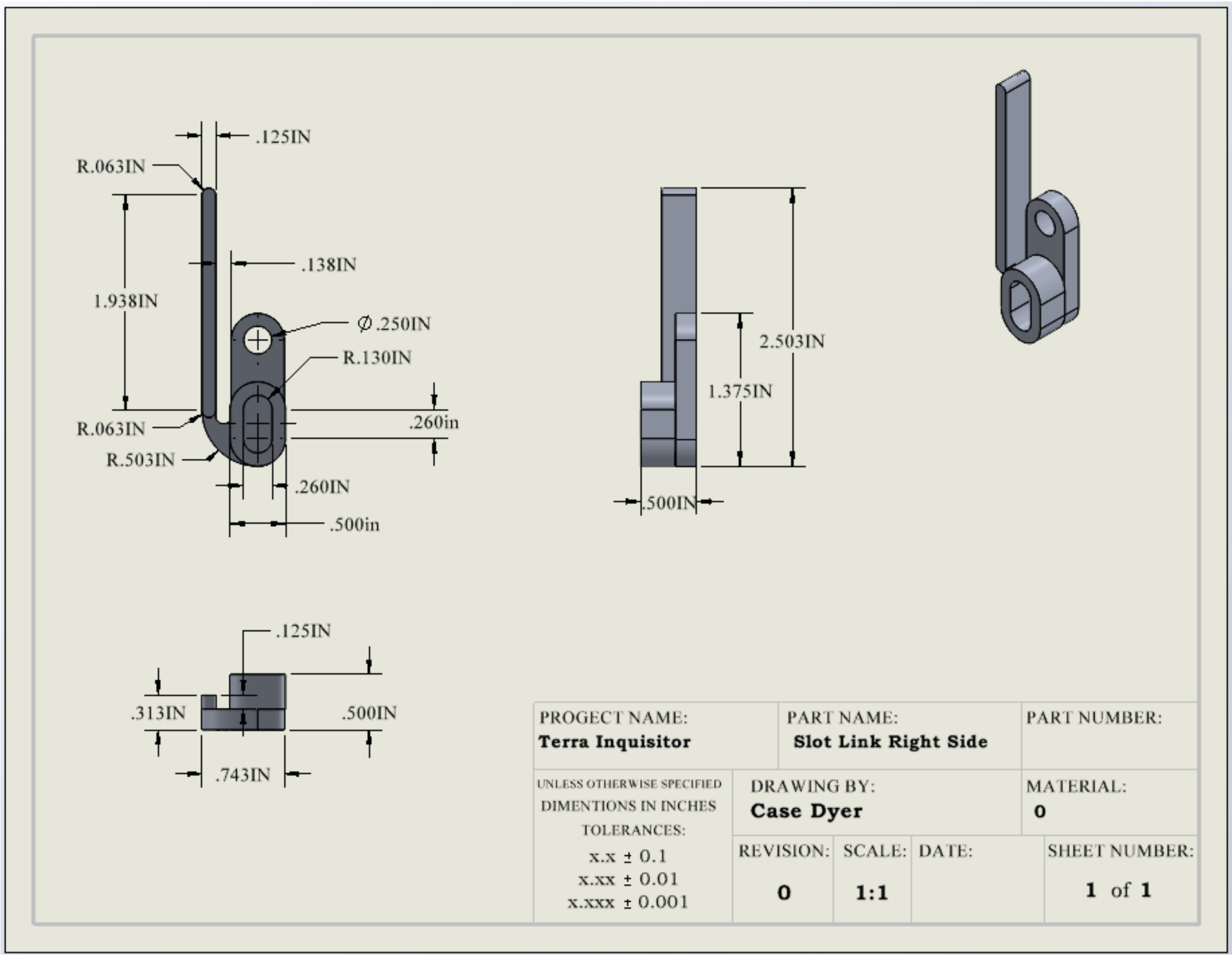


Figure 4-27 Slider Link (L5)

The above linkage designs were exactly mirrored for the left side of the linkage system. By using the two linkages in parallel, the loads will be shared by the two sides and thus roughly split between the two. It is also important to note the ground link detailed above in Figure BLANK, is for illustration purposes and estimation of the vertical space required by the system when stored in the launch vehicle. The mounting locations on the payload will exactly match the measurements outlined by the ground link design.

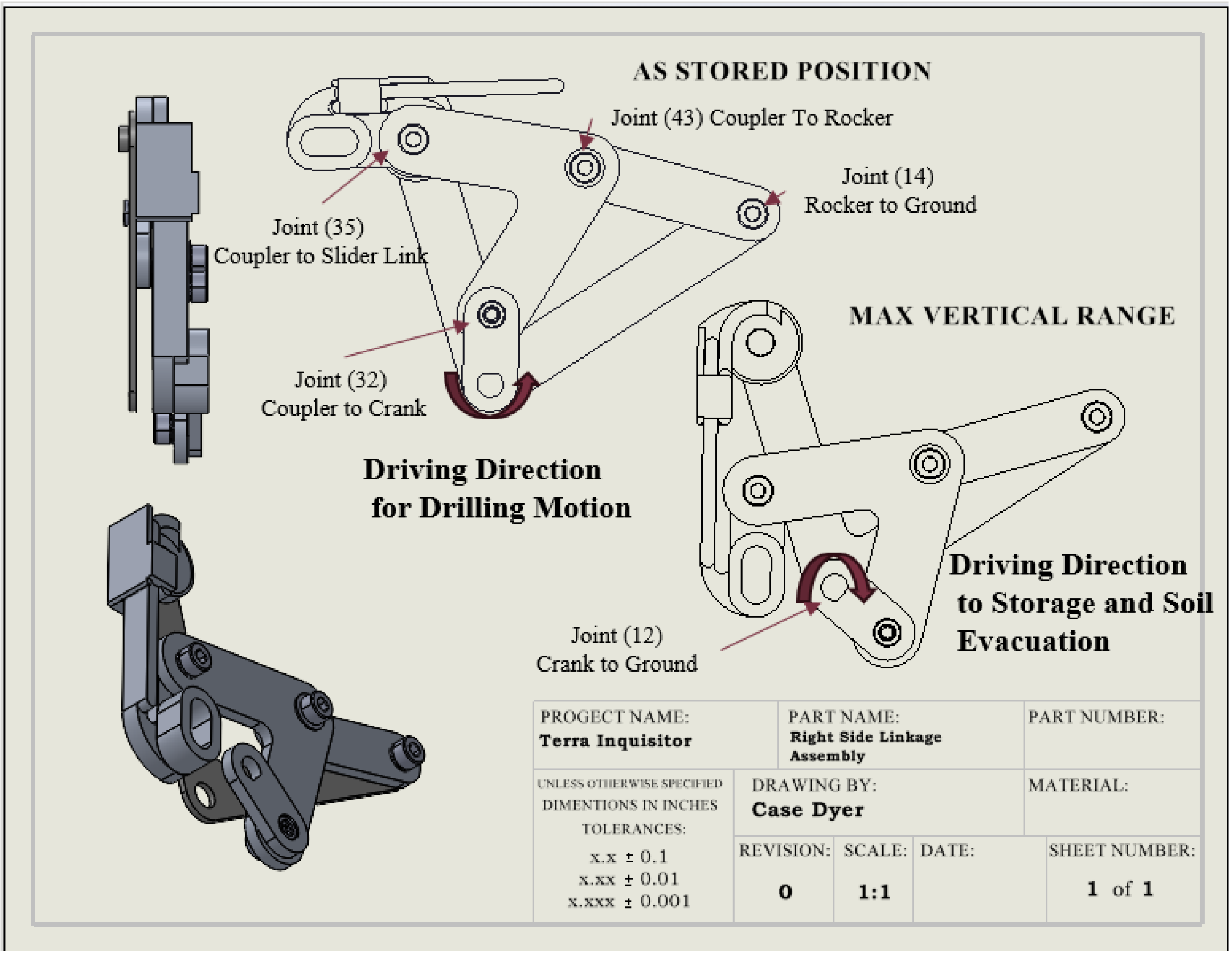


Figure 4-28 Drill Linkage Assembly

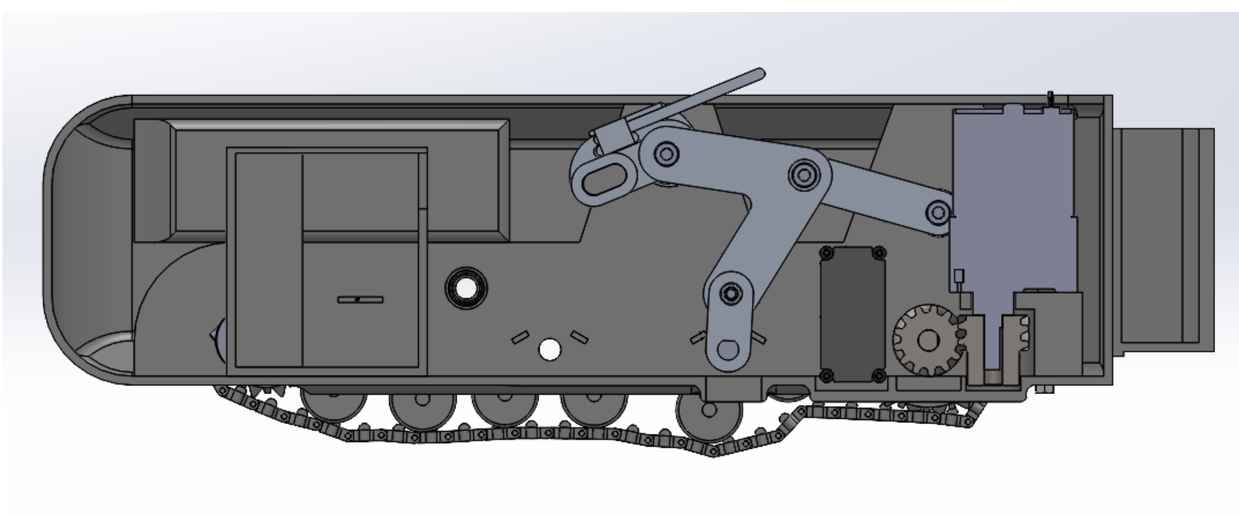


Figure 4–29 Drill Linkage as Joined to the Chassis

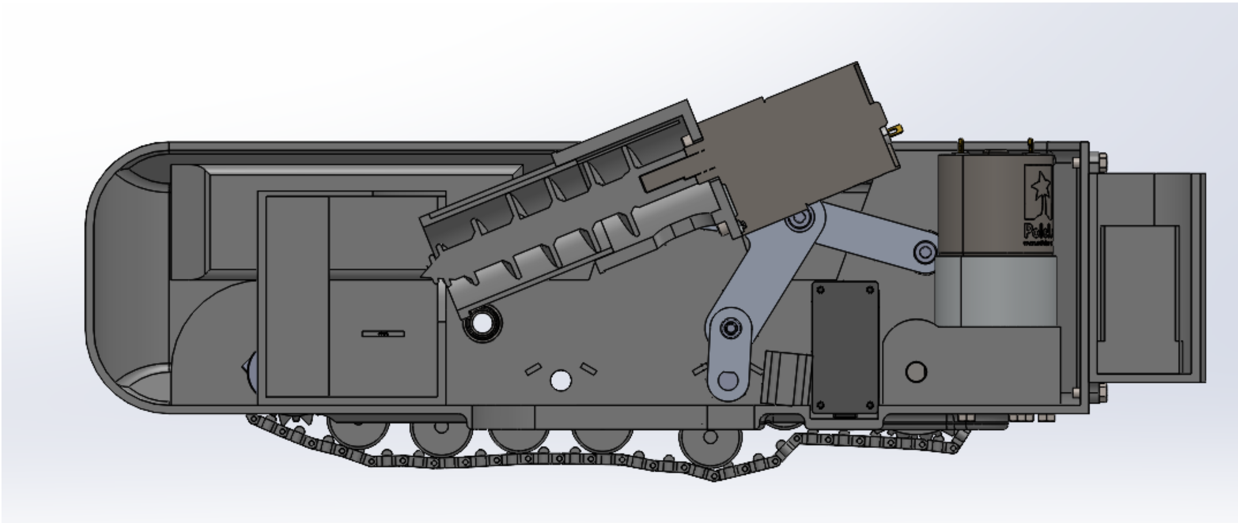


Figure 4–30 Drill Linkage with Drilling Head Assembly

Not only does this linkage system create the desired motion, storing the drill horizontally while not drilling and rotating the linkage to its vertical position to drill, it provides mechanical advantage to the motion thus pushing the drill into the soil for improved soil collection.

With the selection of the linkage finalized, sizing of the joints and max required torque estimations were conducted. These estimations were conducted using dynamic loading and an assumed constant vertical loading of 3.52 lb. This force includes the weight of drilling assembly (.52lb) and the desired maximum output force of 6 lb. This is estimated based on the maximum allowable weight of the Terra Inquisitor of 6 lb. It is undesirable that the drilling system lift the payload off the ground when in its drilling phase and thus the output force needs to be less than the weight of the payload. This max output force of 6 lb. is disturbed between both parallel links (the right and left side) and thus each side must withstand at least 3 lb. due to the output force. With the constant loading defined, the system was simulated in MATLAB software to incorporate the dynamic loading based on the angular and linear accelerations, inertial moments, and linkage weights to estimate the force in each joint. Joints are labeled in Figure BLANK above. Based on the simulations in MATLAB the required torque for each crank angle is displayed in Figure BLANK below. The maximum force experience throughout the linkage system is 5.24 lb. at the ground to crank joint, see Figure BLANK below.

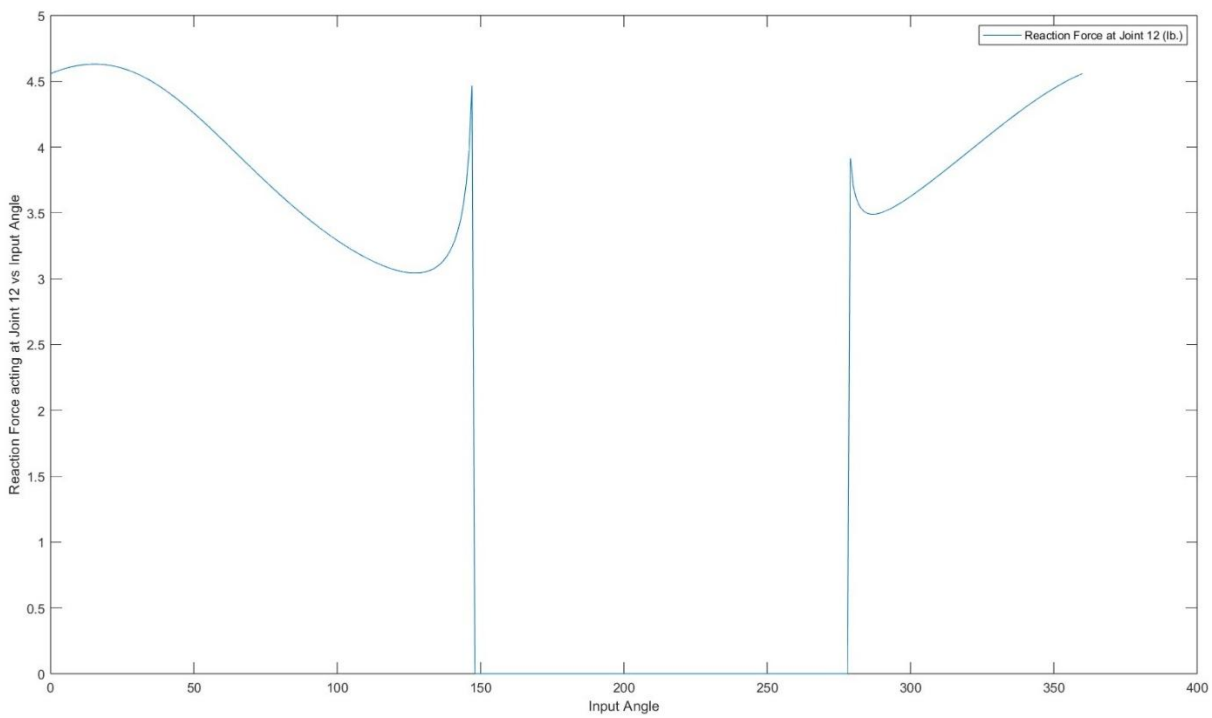


Figure 4–31 Force at Ground to Crank Joint (Joint 12)

With the maximum force experience by any linkage in the system, the shaft size was then sized by considering the primary cause of failure in the joints to be due to straight shear. This was assumed due to the primary load-bearing joints having minimal distance between them. Only distance enough to ensure the links will not contact anywhere else except the joints. The joint with the largest distance (for the rocker and the coupler) has the second smallest experienced force in the system at 3.75 lb. A 6mm diameter shaft of steel was selected based on its common usage in RC part companies. Based on this diameter, and an assumption that the steel specified was a 304 Stainless Steel per (19) with a yield strength of 17.3 ksi, this shaft diameter has a Safety Factor of 25.6. This was estimated using a $\tau = \frac{V*Q}{I*t}$ which for a circular cross-section simplifies to $\tau = \frac{16*V}{3\pi D^3} = 676\, psi$. $SF = \frac{17.3*10^3 psi}{676\, psi} = 25.6$. Even though this safety factor is significantly above what is necessary for the system to account for the many unpredictable loads. This diameter was the smallest diameter compatible across the many different components and was thus selected.

The torque required by the system was also estimated in the MATLAB simulation as can be seen in the figure below.

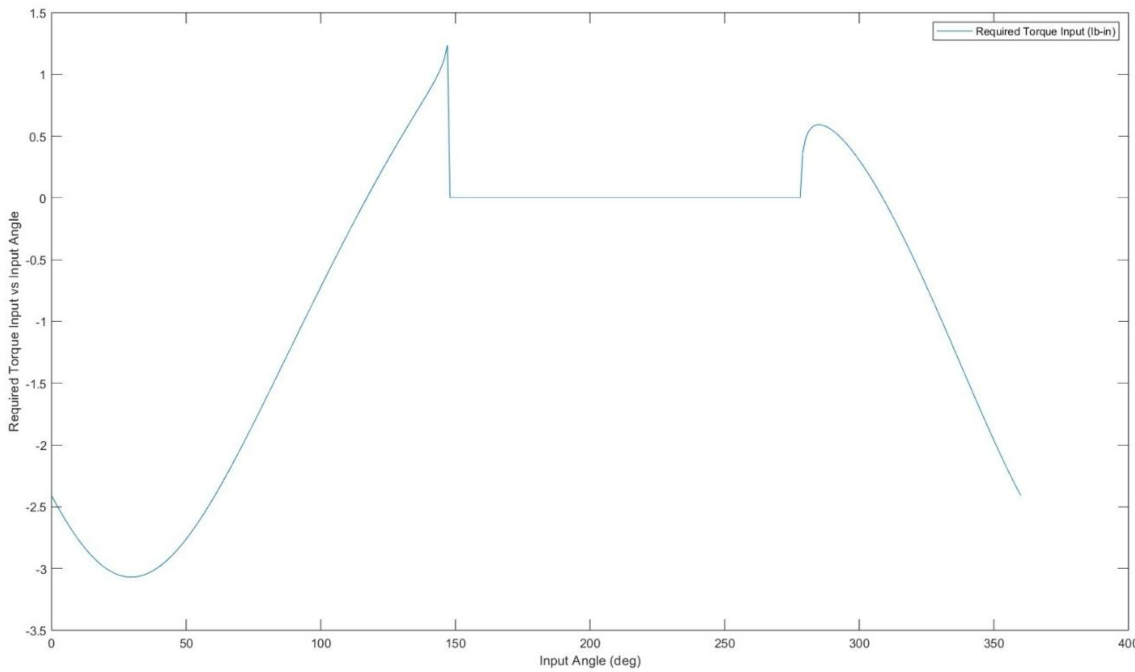


Figure 4-32 Torque input vs input angle

The maximum required torque for a single side of the linkage system was estimated to be 3.1 lb-in. Assuming the torque requirement would doubles to power both sides, the minimum output torque of the system must be 6.2 lb. Based on this, 2000 Series Dual Mode Servo was selected to power the system providing 350 oz-in or 21.9 lb-in. This servo motor was also selected due to the provider also being the supplier of all the gears, shafts, and bearings of the Terra Inquisitor Systems and thus the servo motor have commercially available components for effective integration.

Last portion to affectively quantify is the drilling assembly itself and soil collecting capabilities of system. The auger system show below was physical tested to determine how many times (if more than one) would be required to fully collect the required 60 ml of soil. Based on the testing completed, it can require up to 5 successive drillings to obtain around 60 ml.

The soil is contained by the container fixed in the front of the payload. The system works by the linkage system rotating the drilling head assembly into drilling position and drilling soil up into the lower cover of the drilling assembly. The linkage is then reversed to bring all the soil within the lower cover of the drilling assembly into the payload. The auger is then reversed to expel the soil into the collection container, which the quantity of soil is then continuously weighted to provide real time information of the required number of drilling runs.

The drilling assembly in its entirety is show below.

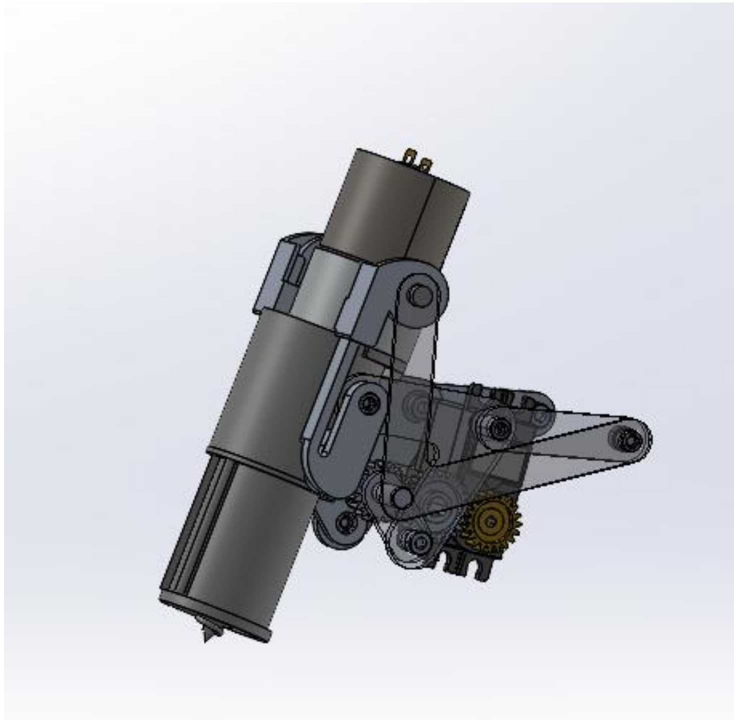


Figure 4-33 Drill Assembly and Linkage

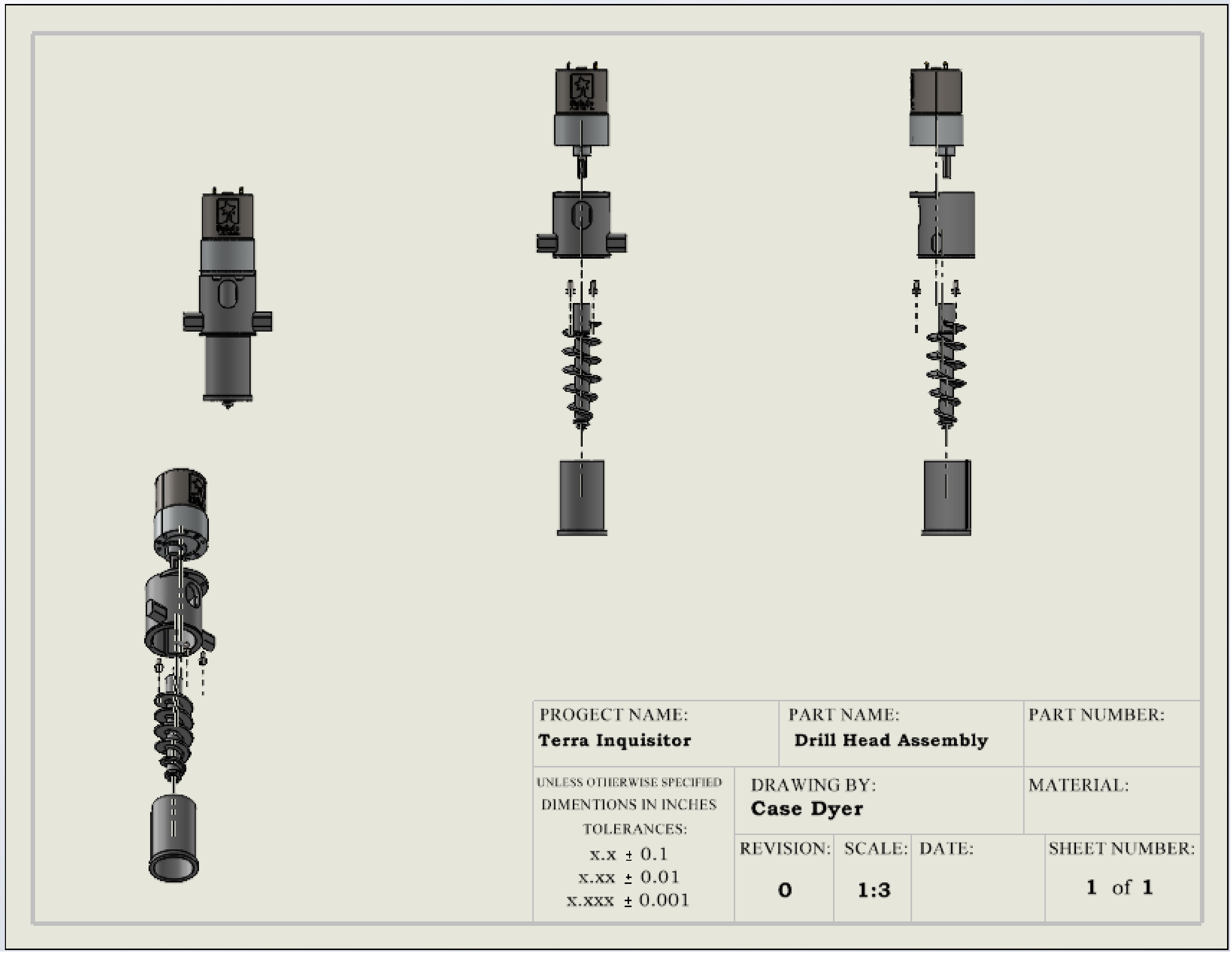


Figure 4-34 Drill Head Assembly

4.3 Payload Component List

Payload Components List		
Item Number	Component Name	Quantity
1	Terra Inquisitor Chassis	1
2	Rear Sprocket	2
3	Front Sprocket	2
4	Idler Wheel	24
5	Wheel Rotator	6
6	Wheel Rotator Pin	12
7	4mm ID Non-Flanged Ball Bearing (10mm OD, 4mm Thickness)	24

8	Pololu 4741 Motor	3
9	3598N291	4
10	Lipo S3 Battery	1
11	Idler Wheel Assembly Guide	12
12	Track Chain	110
13	Track Pin	110
14	6mm ID Flanged Ball Bearing	15
15	6mm D-Shaft (Stainless Steel, 60mm Length)	2
16	6mm D-Shaft (Stainless Steel, 30mm Length)	2
17	Rear Motor Mount	1
18	B18.6.7M - M3 x 0.5 x 30 Indented HHMS --30N	8
19	Tensioner Wheel	2
20	Soil Container	1
21	Adfruit 4547 Submersible Pump	1
22	ENV-45-pH	1
23	Ground Link	2
24	Crank	2
25	Coupler Link	2
26	Rocker	2
27	Slotted Guide Link Right Side	1
28	Slot Link Right Side	1
29	Slot Guide to Ground Joint	2
30	1501 Series M4 x 0.7mm Standoff (6mm OD, 12mm Length)	2
31	M4 x 0.7mm Zinc-Plated Socket Head Screw (6mm Length)	8
32	2807 Series Stainless Steel Shim (6mm ID x 9mm OD, 0.50mm Thickness)	2
33	1501 Series M4 x 0.7mm Standoff (6mm OD, 8mm Length)	2
34	2807 Series Stainless Steel Shim (4mm ID x 7mm OD, 0.50mm Thickness)	6
35	1501 Series M4 x 0.7mm Standoff (6mm OD, 6mm Length)	4
36	Slotted Guide Link Left Side	1
37	Slot Link Left Side	1
38	Drill Bit Lower Cover	1
39	Auger Drill Bit	1
40	Drill Bit Upper Cover	1
41	B18.3.1M - 3 x 0.5 x 6 Hex SHCS -- 6NHX	4
42	6mm D-Shaft (Stainless Steel, 90mm Length)	1
43	2305 Series Brass, MOD 0.8, Servo Gear (25 Tooth Spline, 20 Tooth)	1
44	Servo Center Gear Block	1
45	Water Tank	1
46	B18.6.7M - M3 x 0.5 x 8 Indented HHMS --8N	4

4.4 Electronics and Power

The Terra Inquisitor's electrical system will implement a PCB to simplify communication and power distribution across the electrical system. This PCB will be designed using KiCad 9.0, a free PCB design software. Subsystem schematics are divided into different sheets to better display power distribution and communication buses.

To design the power system behind the Terra Inquisitor, the electrical system was broken into five subsystems. These subsystems include voltage regulation, microcontroller, sensors, motor control, and outputs. Designing around these subsystems allowed for a clear understanding of the power requirements for each system. As expected, the logic behind the Terra Inquisitor operates on 5V logic, while systems like the motor control require a higher voltage to operate. In the final design, the Terra Inquisitor's voltage regulation system outputs both 6V and 5V power rails. This will be covered in further detail below.

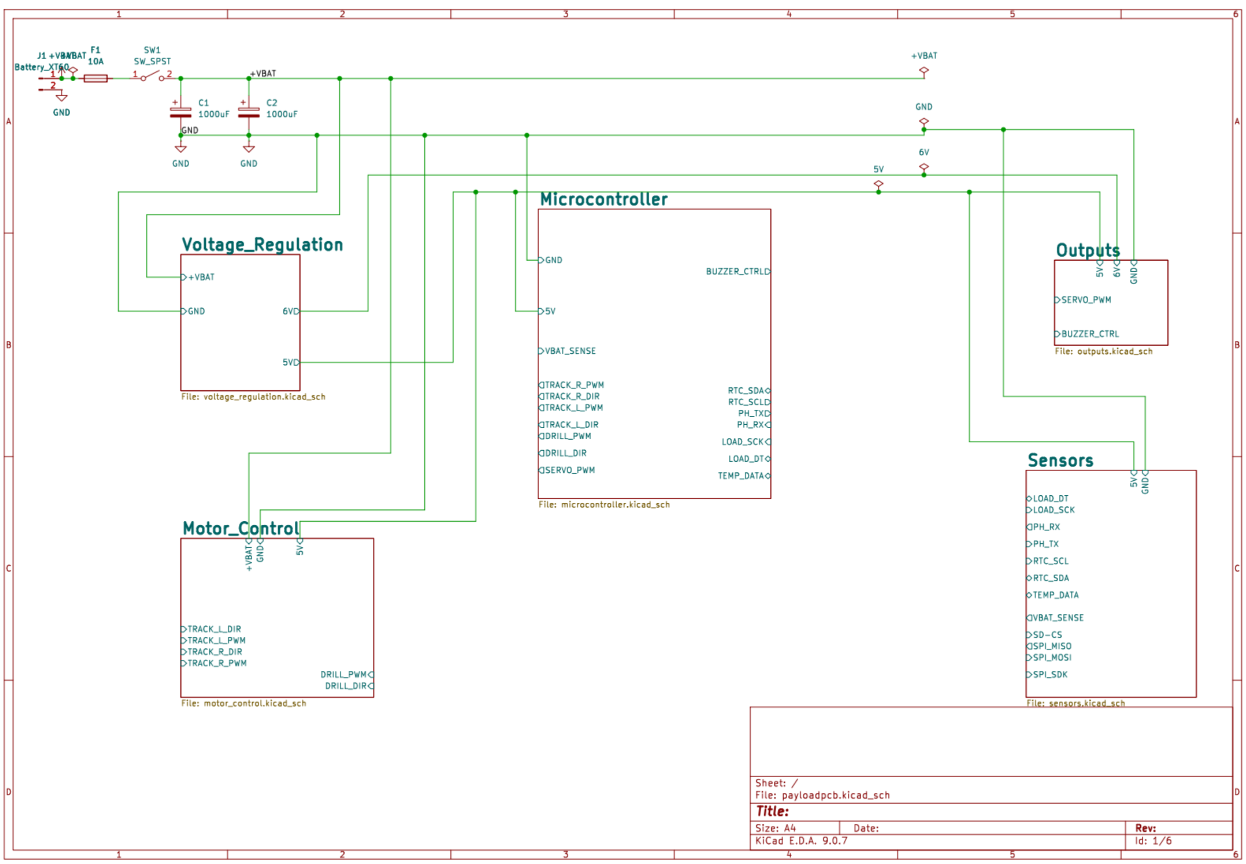


Figure 4–35 *Terra Inquisitor Power Distribution Schematic*

4.4.1 Power

The Terra Inquisitor requires a versatile power source. The mission of the rover, while not exceedingly long, is power intensive. The mission will require various electric motors that facilitate the movement and experimental procedure of the Terra Inquisitor.

There are a few requirements that a power source for this vehicle must meet. The overall weight of the Terra Inquisitor is very important; therefore, a lightweight power source is crucial. The power source must be able to meet the demand of the electrical components and provide the voltage level that each element requires. Lithium batteries (LiPo) are the best fit for a power source on the Terra Inquisitor. LiPo power sources are commonly used in similar applications where lightweight, yet intense power is needed, especially with electric motors. To select a LiPo power source, a decision matrix will be used to determine the best fit for the Terra Inquisitor.

Power Source			LiPo 3s			LiPo 4s		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	0.15	USD	$20-$40	10	1.5	$30-$65	7	1.05
Weight	0.35	grams(g)	380	10	3.5	470	6	2.1
Capacity	0.25	mAh	5200	10	2.5	5200	10	2.5
Discharge Rate	0.25	C	80	7	1.75	120	10	2.5
Overall Score:					9.25			8.15

Table 4-4 Terra Inquisitor Power Source Decision Matrix

Cost was measured in USD. Cost was evaluated because of the need to keep the overall cost of the Terra Inquisitor's power system low. As shown above, the selection with the lowest cost received a ranking of 10. The magnitudes were selected after researching current price rates for LiPo batteries with similar specifications to the ones shown here. Cost was weighed the least as the cost of power sources does not range widely to any value that would make cost significantly important in this comparison.

Weight was measured in grams (g). Weight was evaluated because of the need to keep the overall weight of the Terra Inquisitor low. As shown above, the selection with the lowest weight received the ranking of 10. The magnitudes were selected after researching LiPo batteries with similar specifications to the ones shown here. Weight was weighed significantly higher than the rest of the criteria as it affects the outcome of not only the Terra Inquisitors mission, but also the vehicle launch and recovery as well. Saving any weight from the Terra Inquisitor, and therefore the Terra Nova, is beneficial to every system.

Capacity was measured in milliamp hours (mAh). This was evaluated because of the power requirements of the Terra Inquisitor's electrical system. As shown above, the magnitudes are the exact same. This is because the capacity was used as a reference point so that the other criteria might be related to each other. The compared batteries were related by having the same capacity. The capacity listed in the decision matrix does not indicate the capacity required for the electrical system's selected design. It stands only for the sake of comparison.

Discharge rate was measured in C. This was evaluated because of the need for intense power discharge to the electrical system during the mission. As shown above, the rating for the higher discharge rate received a 10.

4.4.2 Voltage Regulation

The 3s LiPo batteries have a nominal voltage level of 11.1 volts and a fully charged voltage of 12.6 volts. This further increases the need for a voltage regulation system. This voltage swing will be safe for the motor control system, but with systems like many of the sensors and the microcontroller that operate at 5V, the voltage regulation system will safely provide power and ensure the microcontroller is safe from damage by supplying too much current. As mentioned before, the finalized design requires power in the form of 11.1 V (directly from the battery), 5V (common for sensors and logic devices) and a 6V power rail.

These parameters present a unique situation for the voltage regulation system. The 6V rail will meet the optimal voltage of a servo motor that will be used. This 6V will be provided by a switching (buck) converter. This will provide the necessary power requirements with minimal power loss, as the switching converter operates at 90% efficiency. While this meets the 6V voltage requirement, the 5V requirement will be met using a linear regulator. This is a significant component of the regulation system, as it will provide the sensors with power free of any switching noise that could interfere with their readings.

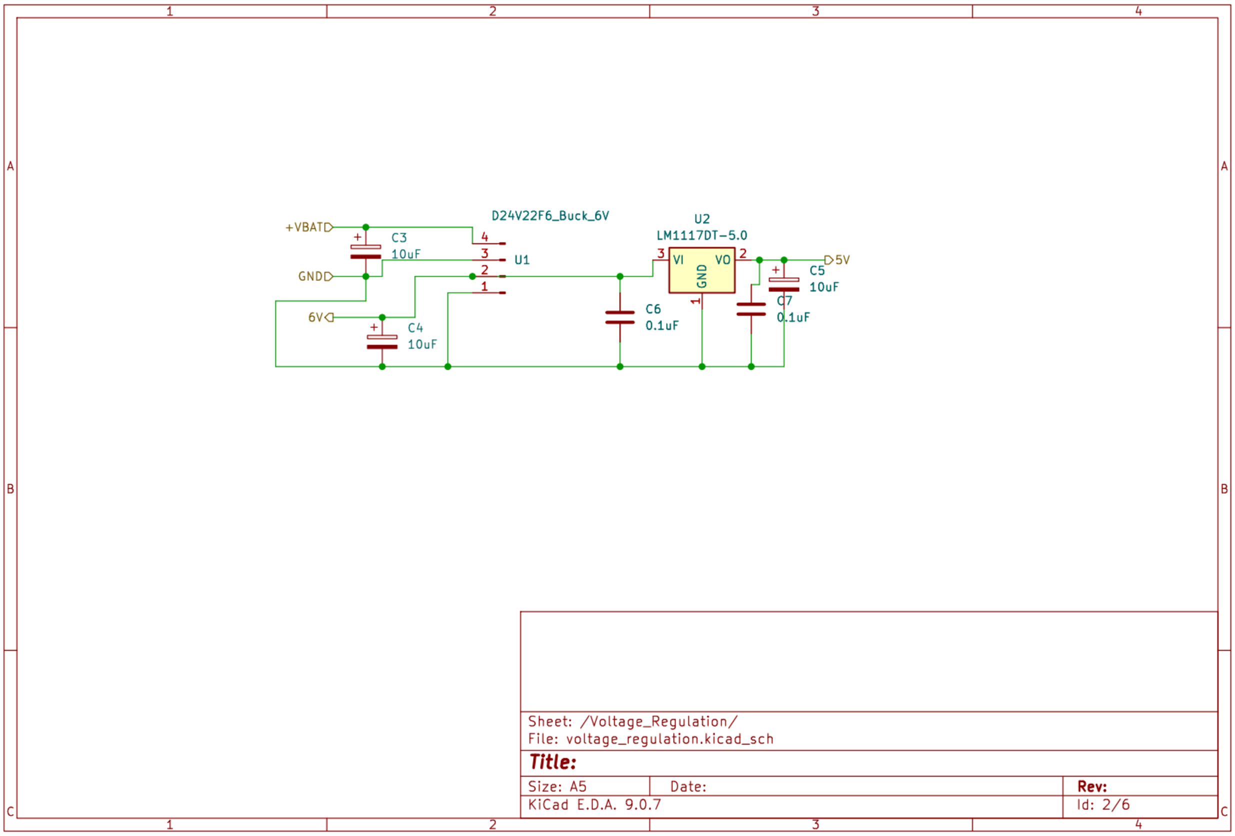


Figure 4–36 *Voltage Regulation System*

4.4.3 Automation

The Terra Inquisitor will be automated by an MCU on a commercial development board. Important criteria include accepting power from a 12-volt source, obtaining many pins, including pins capable of reading analog input signals, outputting pulse width modulation signals, and pins capable of I2C and SPI communication protocols. These are essential to the operation of the Terra Inquisitor in terms of movement, experimental procedures, and data logging. Preferred criteria also include having a simple software interface and large amount of community support for programming. The MCU will control any movement of the rover, the pH sensor, the temperature sensor, and the experimental data logging.

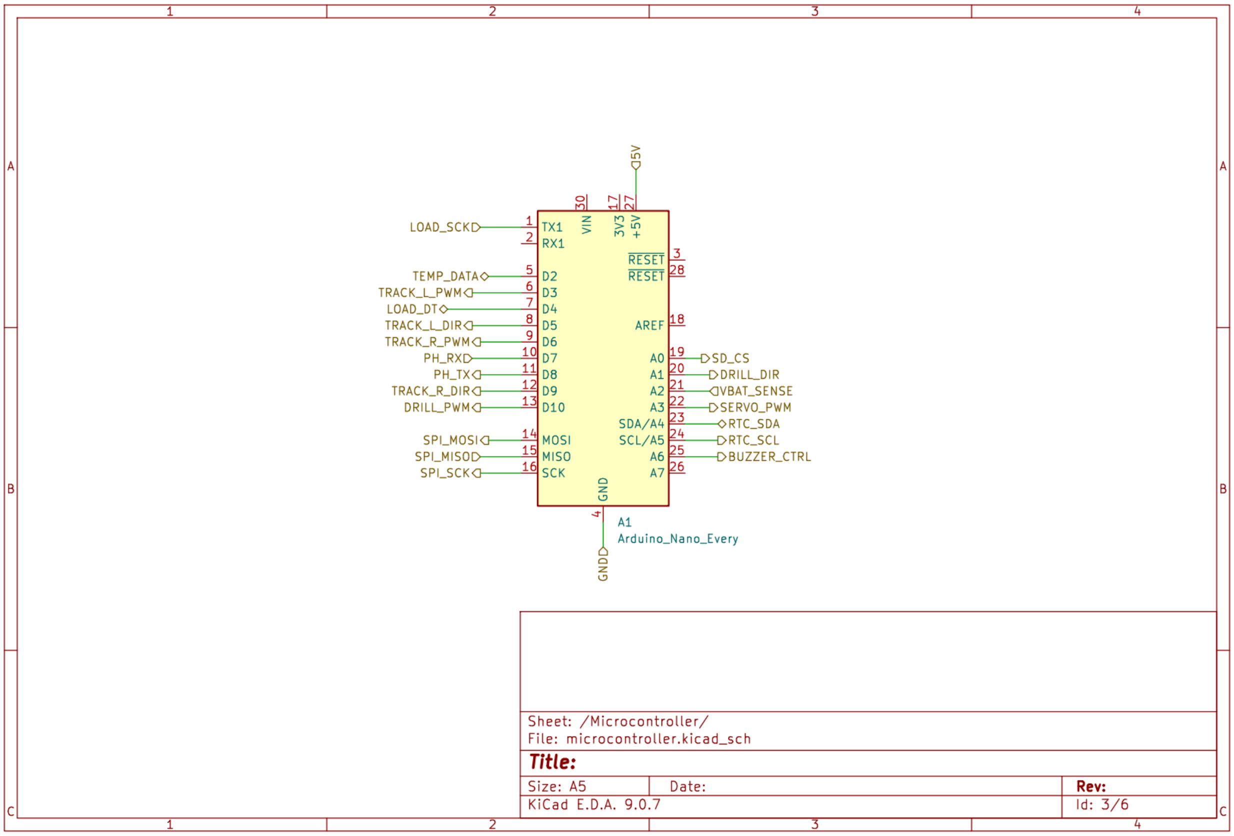


Figure *4–37 Terra Inquisitor Automation Schematic*

Figure 4-15 displays the pinout requirements of the development board in control of the Terra Inquisitor. These pin requirements can be met by many commercial development boards. A development board capable of being programmed with the Arduino IDE will be selected because of the ease of implementation.

The Terra Inquisitor will be controlled using Arduino Nano Every. This development board includes all ideal components required by the Terra Inquisitor's electrical system. As shown in the schematic above, both the SPI and the I2C communication protocols will be utilized on the board. The on-board ADC will be used for reading sensor data by several pins. The board will be powered though the 5V pin, which bypasses the on-board voltage regulator. The Nano Every was the ideal choice for the Terra Inquisitor with its small size, being only 45 mm in length and only 18 mm in width. The development board also only weighs around 7 grams, ideal for conserving weight within the launch vehicle.

There will also be a commercial development board to control the mechanical separation of the launch vehicle. This board must include an ADC to read an input signal from an altimeter. This signal will initiate the mechanical separation process. In its entirety, the mechanical separation electrical system will include its own power source, development board, altimeter, and two electric motors.

4.4.4 Motor Control

The movement of Terra Inquisitor will be conducted using three DC Motors and one servo motor. These motors are responsible for turning the tracks, Adjusting the position of the drill linkage, and turning the drill itself. One important aspect of electrical design is the how these motors will be controlled. Mentioned in the power section, the servo motors require a unique input voltage of 6V, while the DC motors are power directly from the battery.

The three DC motors will be controlled using two TB6612FNG modules. These are dual H-Bridge motor drivers capable of handling current peaks of up to 3.2A. These modules are ideal for our system due to their current rating and relatively small size. These modules take input power directly from 3s battery, also requiring 5V power to conduct logic. In the case of the Terra Inquisitor, one of these dual motor drivers will handle two DC motors (the left and right track motors), and the other will only handle one (the drill motor). Displayed below is the schematic for the DC motor control system.

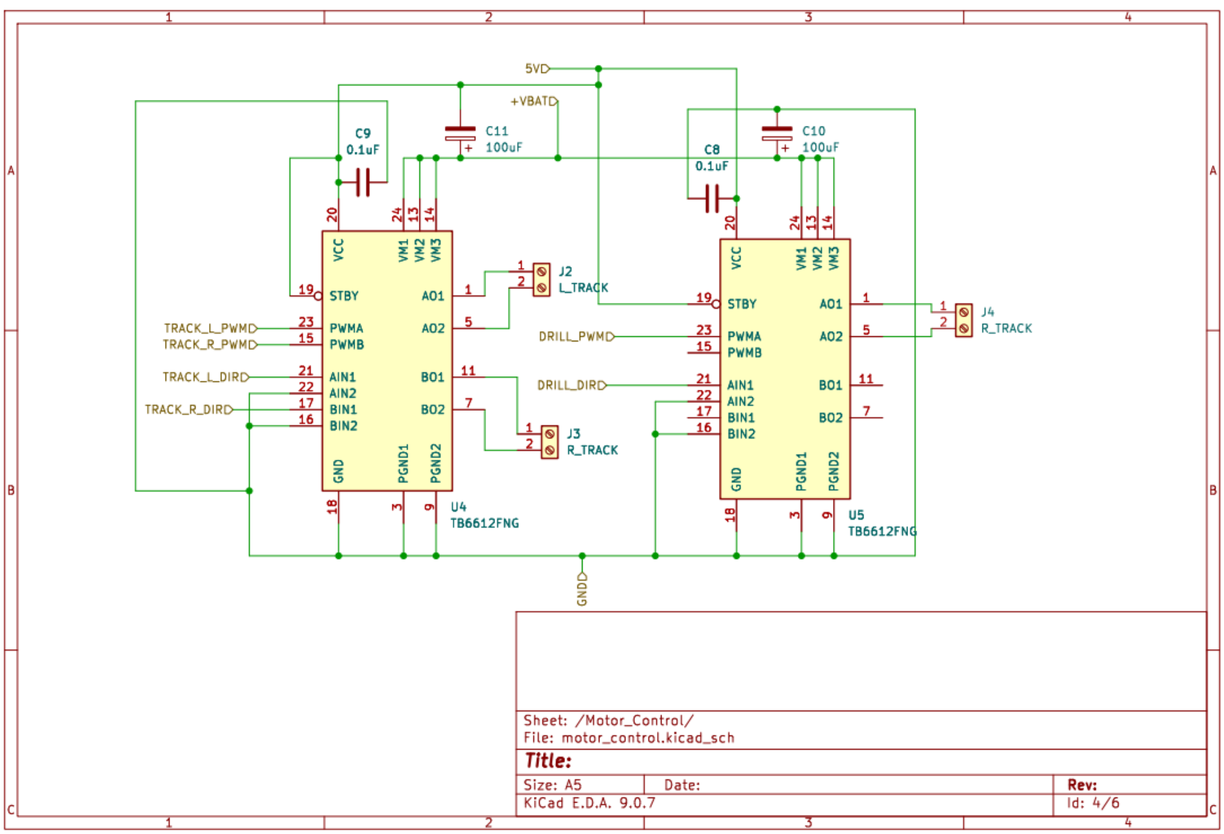


Figure 4–38 *DC Motor Control System*

The servo motor will be controlled directly from the microcontroller. The Arduino "Servo" library will be utilized.

4.4.5 Soil Testing

The mission experiment of the Terra Inquisitor is to conduct a pH test of the soil where the launch vehicle lands. The action of taking a pH test has many requirements. The payload rover must contain a containment vessel for a soil sample, means to obtain a sample, means to obtain accurate pH data, and a way to timestamp and store the data within the payload. The latter two goals require significant electrical systems to operate.

The electrical system design responsible for obtaining accurate pH sensor data can be broken into two systems: sample preparation and sensors. The most important electrical sensor to the mission of the Terra Nova is the pH probe. A pH probe is by far the most idea method for obtaining a soil pH reading. A probe must be selected that meets the requirements of the Terra Inquisitor's experiment. These requirements relate to the potentially harsh conditions the rover will be subject to throughout the launch process and the nature of the experimental procedure itself. Taking these into account, a decision matrix will be implemented below to choose between two viable options for the pH probe. Probe 1 is the Atlas Scientific Spear Tip/Soil pH Sensor. This probe is designed to be used in soil and is therefore an excellent fit for this system. Probe 2 is the Gravity Lab Grade Analog pH Sensor. This sensor is a more cost-effective option that can be easily operated using Arduino.

pH Probe			Probe 1			Probe 2		
Criteria	Weight	Parameter	Magnitude	Rating	Score	Magnitude	Rating	Score
Cost	0.25	USD	$169.99	3	0.75	29.99	10	2.5
Durability	0.25	scale (1-10)	8	10	2.5	4	5	1.25
Size (Length)	0.3	millimeters	127.7	10	3	177.7	6	1.8
Accuracy	0.15	pH	0.002	10	1.5	0.1	7	1.05
Overall Score:					7.75			6.6

Table 4-5 Terra Inquisitor pH Probe Decision Matrix

The purpose of this decision matrix was to compare two very different probe devices. These devices were selected as best for the Terra Inquisitor in their own respective classes. The decision being made here is mostly a budgetary decision. We decided to spend more money for better accuracy in our system, which was more ideal for the Terra Inquisitor.

Cost was measured in USD. Cost was evaluated because of the need to keep the overall cost of the Terra Inquisitor low. As shown above, the selection with the lowest cost received a ranking of 10.

The magnitudes were selected based on the sale price provided by the website. Cost was weighed at 25% due to the wide price range for pH probes.

Durability was measured on a scale of 1-10, with 1 being the least durable and 10 being the most durable. These rankings were based on information presented on the website pages for each probe. This includes structural design and material.

Size was measured in millimeters. Size was evaluated because of the need to conserve space within the payload. With a limited amount of space available for the Terra Inquisitor within the launch vehicle, size was weighed the highest among the sensor criteria.

Accuracy was measured in the potential error of the returned pH value. Accuracy was evaluated because of the need to carry out a successful experiment. It was weighed the least among the criteria because of the small potential error for the less accurate selection. Both of the probes return relatively accurate readings.

PH probes are passive sensors, requiring signal conditioning and offset modules. The Terra Inquisitor’s electrical system will use the Atlas Scientific EZO module for this purpose. This module will be incorporated into the PCB where it will connect to a BNC connected, for connection to the probe itself.

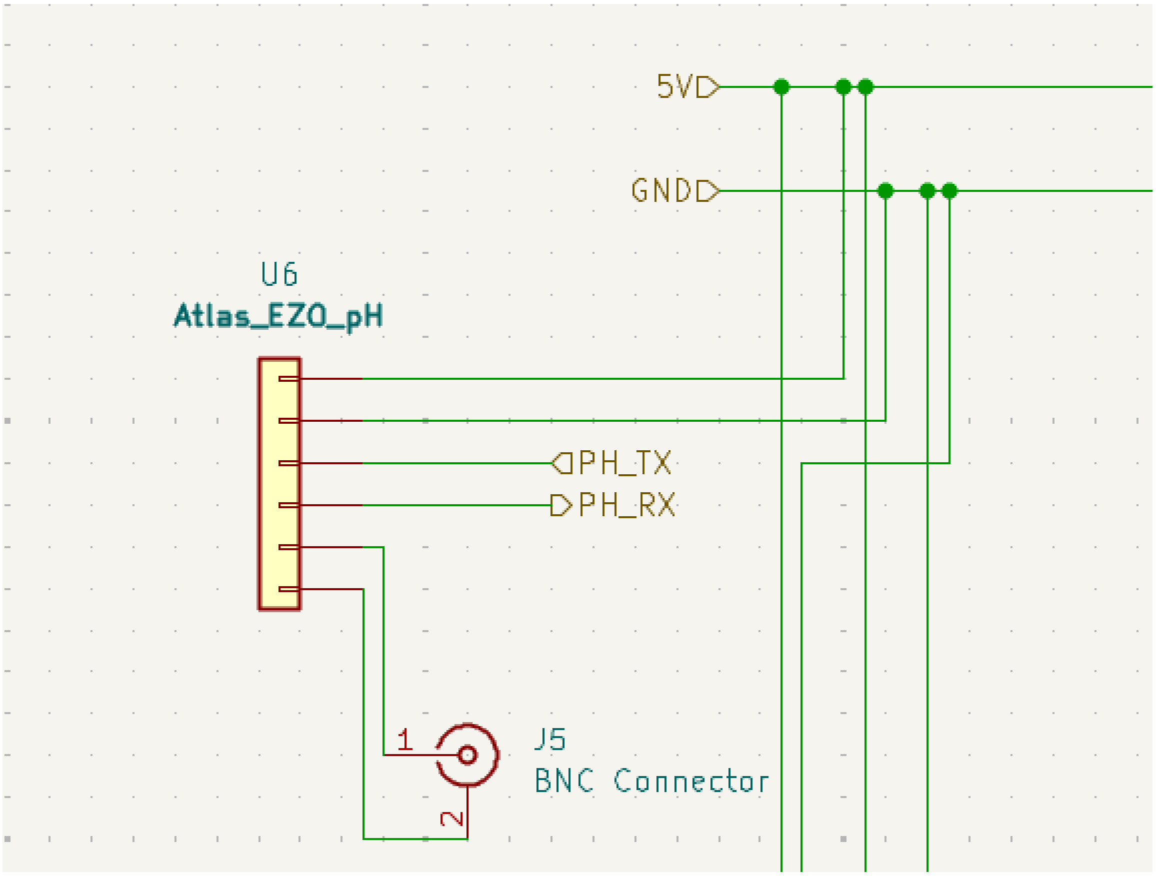


Figure 4–39 *pH Probe Module and Connector Schematic*

Another aspect of obtaining accurate pH readings is the incorporation of a temperature sensor. The accuracy of pH sensors by themselves is known to vary with temperature. This can be mitigated by including a temperature sensor in the design of the Terra Inquisitor. This sensor will take the temperature of the soil sample and include it with the other timestamped data. This will allow for post-mission temperature correction using software such as Excel. The equation for correcting the pH values is:

$$pH_{corrected} = pH_{measured} + (\Delta T * k)$$

Where $pH_{measured}$ is the pH value provided by the sensor, ΔT is the calculated change in temperature from calibration, and k is ≈ -0.03 pH/°C deviation from initial temperature.

The sample preparation requirement is another aspect of the Terra Inquisitor's design that plays a large role in the electrical system. The sample preparation will include weighing the sample to ensure enough has been obtained and flushing the sample with water to ensure accurate pH results. To find the weight, a load cell will be utilized combined with an ADC/amplifier module. This cell will be calibrated using soil from the launch site in order to calculate the density of the soil that will be measured. This is how the Terra Inquisitor will know when it has gathered a large enough sample. The module that will be used is the HX711 amplifier module. The entire breakout module will be soldered directly onto the PCB. The module requires 5V power from the linear regulator and communicates with the microcontroller through GPIO pins.

As previously mentioned, the soil sample must be flushed with water for the pH probe to work correctly. Water is required for the soil solution to “react” with the probe, allowing for a voltage across the membrane of the probe. For this to occur within the Terra Inquisitor, a system must be incorporated so that water can be pumped into the sample container. This will be accomplished using a DC water pump. The water pump that will be used is the Adafruit Submersible 3V DC Water Pump. This water pump will be the means that the soil sample is prepared for testing. This pump will be powered and controlled using a simple transistor driving circuit. PWM signal will be sent from the microcontroller to the base of an NPN BJT transistor. The negative terminal of the DC pump will be connected to the collector of the transistor, while the positive terminal is connected to 5V. The emitter of the transistor will then be grounded. This configuration will allow the microcontroller to control the flow of power to the pump, without powering the pump itself, as the pump would draw more current than the microcontroller can safely supply.

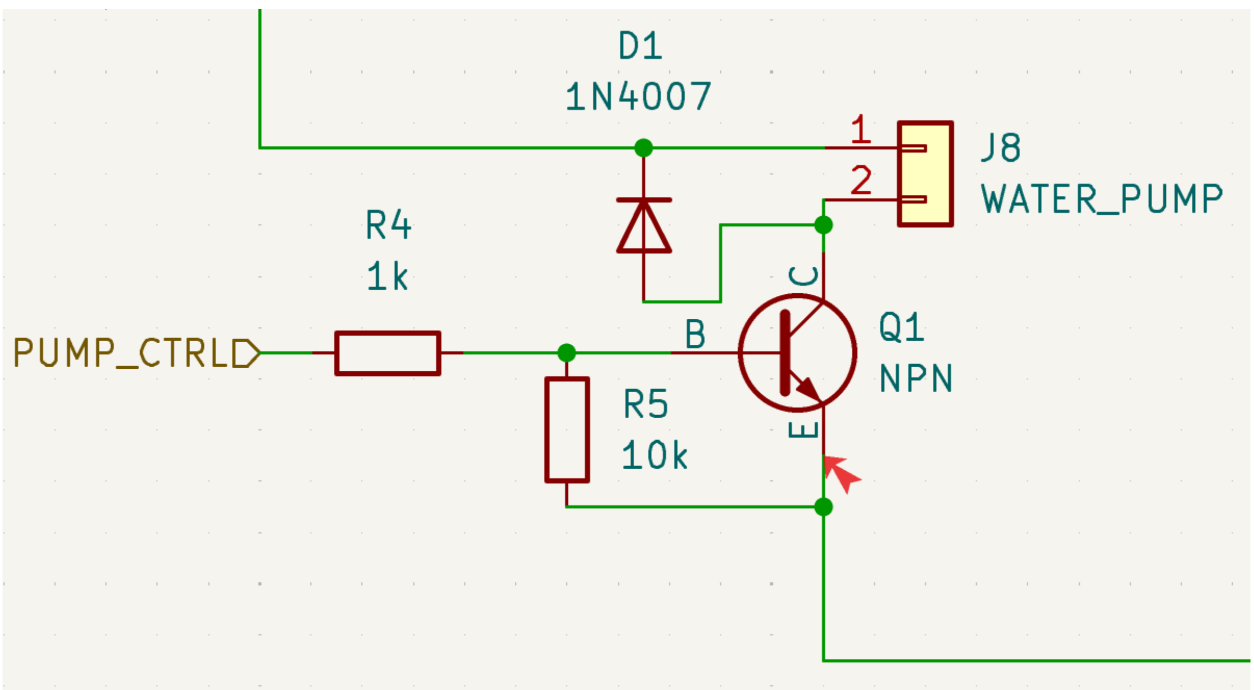


Figure 4–40 *DC Water Pump Driving Circuit Schematic*

Timestamping and logging the data collected by these sensors is an equally important aspect of the electrical system. To timestamp the data, an RTC (Real Time Clock) module will be attached to the PCB. The current design of the Terra Inquisitor uses a DS3231 RTC module to accomplish this task. This module, like sensors on the Terra Inquisitor, requires 5V power. This module will use the I2C communication protocol to communicate with the microcontroller gathering data from the sensors. This communication occurs over the SDA and SCL pins on Arduino Nano Every.

After the data has been collected and timestamped, the Terra Inquisitor must be able to store the data to be analyzed post-mission. This task requires a data storage component, along with the means to communicate the data to the device. The Terra Inquisitor will do this using a microSD along with a microSD card module. This way, a microSD card can be easily placed and removed from the Terra Inquisitor. This module will be powered off the 5V rail. The microcontroller will communicate with the module using SPI communication protocol. This protocol utilizes the CS, MOSI (Master Out Slave In), MISO (Master In Slave Out), and SDK pins on the Arduino Nano Every. Decoupling capacitors are used near the 5V and GND pins to filter out high frequency noise.

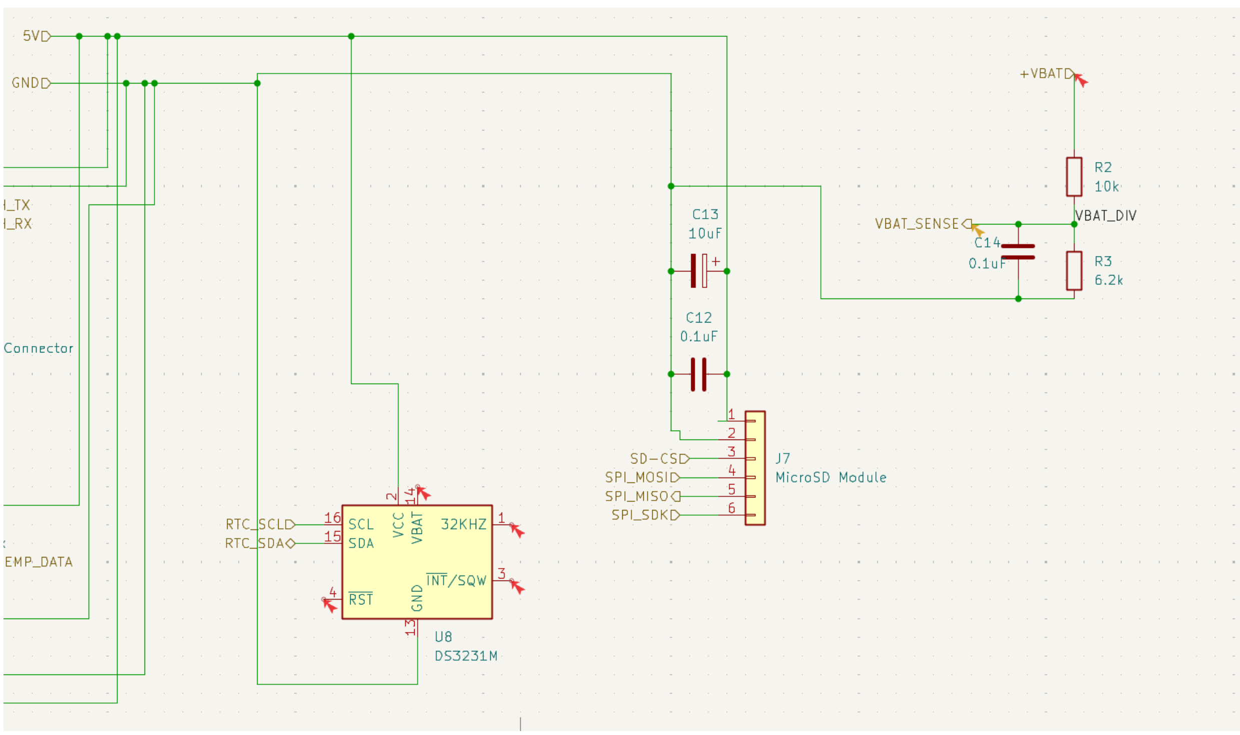


Figure 4–41 *RTC, MicroSD Module, and Battery Level Detection Schematic*

4.4.6 Miscellaneous Electronics

There are a few other aspects of the Terra Inquisitor's electrical system that are not directly responsible for obtaining soil pH data, but will be of value in the development, testing, and deployment of the Terra Inquisitor. Among these are a buzzer and battery level sensor.

A buzzer will be included to alert of the current state of the payload. This is ideal in a multitude of situations. In the development and testing scenario, knowing what state of operation the Terra Inquisitor is in will be crucial to any debugging that will need to take place. This is especially evident as the serial pins on the Arduino Nano Every are being utilized as GPIO pins in the current design. Audible state verification will also be ideal for assurance on the launch pad. When the launch vehicle is in the position on the pad, a buzzer would allow for the knowledge that the payload is powered on and in the correct state internally.

This buzzer will be powered off the 5V rail and controlled using a BJT NPN transistor like that of the DC water pump. The microcontroller will send a control signal to the base of the transistor which will control the flow of power to the buzzer. A flyback diode will be installed in parallel to the buzzer, with the cathode (band) towards the positive (5V) terminal. A decoupling capacitor is included to reduce any high frequency noise. These can be seen in the schematic included below.

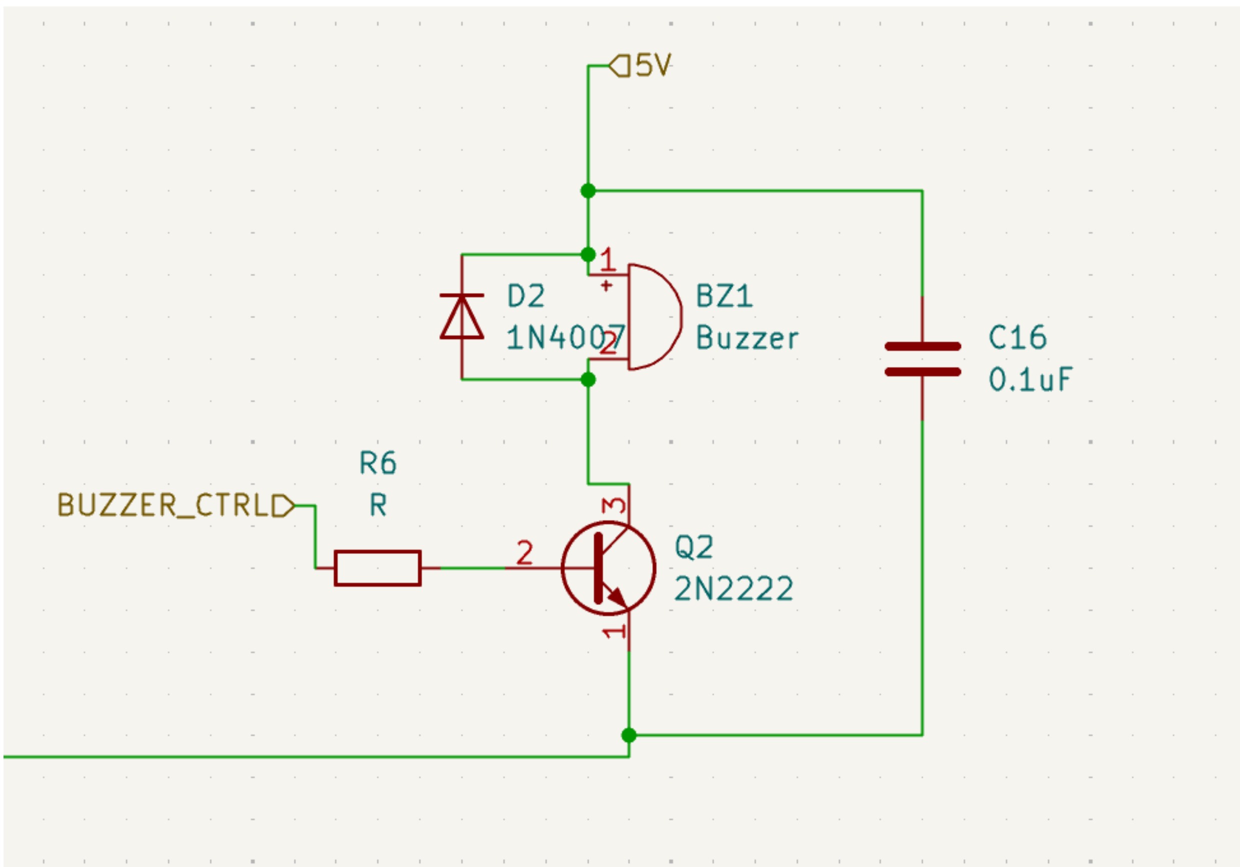


Figure 4–42 *Buzzer Power and Control Schematic*

The other valuable electrical circuitry included in the design is the battery level detection. Having knowledge of the battery charge level can be beneficial in the testing stages by enabling a low charge shutdown protocol in the Terra Inquisitor's software. This is a necessary step in ensuring the safety of the electronics. This voltage sensing circuitry is a high resistance voltage divider circuit that capitalizes on the voltage swing of the 3s lithium battery. These batteries have a fully charged voltage of 12.6 and a depleted voltage of 9V, with a nominal voltage of 11.1V. With this swing, a voltage divider will be able to scale down the voltage coming from the battery to a level that can be safely measured by an analog pin on the Arduino Nano Every. A decoupling capacitor will be installed between the voltage being reported to the ADC and GND to reduce any high frequency noise. A schematic of the design is shown below.

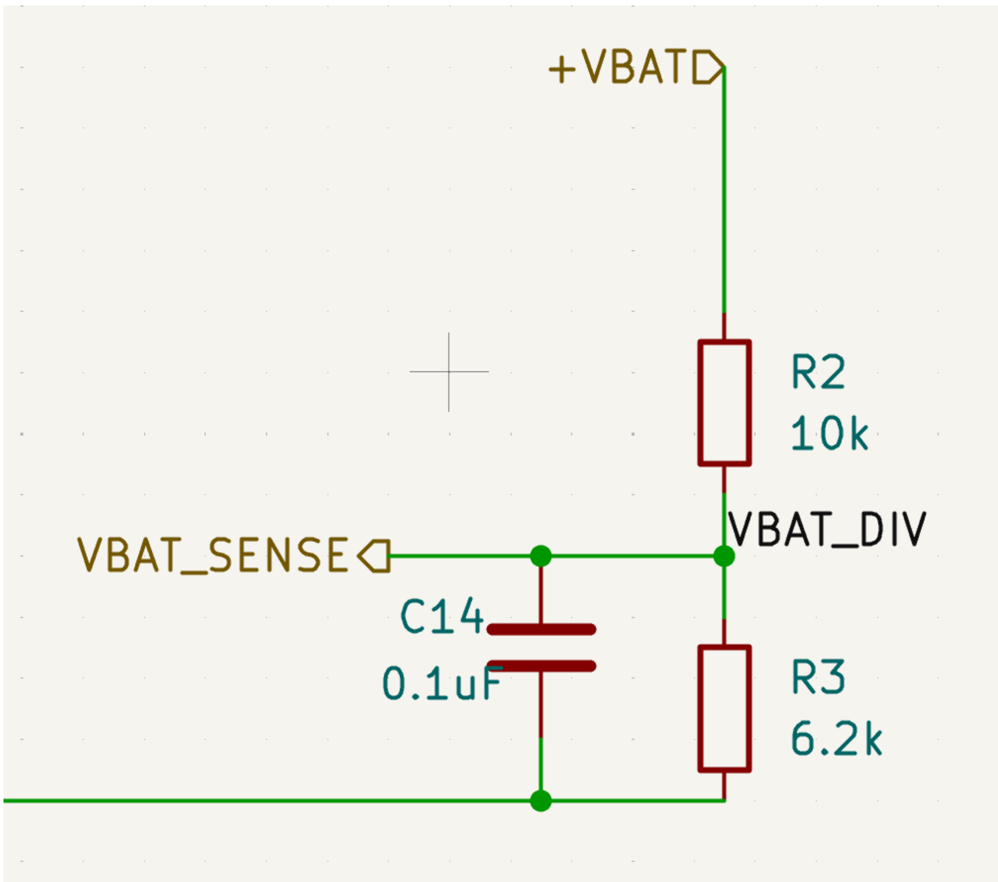


Figure 4-18 Voltage Sensing Divider Schematic

4.5 Experiment Success Criteria

The evaluation of the Terra Inquisitor's experiment is based on the successful deployment, collection, and logging of soil pH test data with proper time stamps. Successful separation is defined as the MSM autonomously separating the HAUS section after reaching ground level from apogee. Successful deployment of the Terra Inquisitor is defined as its autonomously exiting the HAUS coupler. The Terra Inquisitor's success will be based on its ability to follow its autonomous routine, and collection of at minimum 60ml of soil. The success of the soil testing will then be based on relative accuracy of the pH readings to a commercially available handheld pH reader with a known accuracy.

4.6 Payload and Launch Vehicle Interfaces

Based on the vehicle technical design detailed earlier, the Terra Inquisitor will be housed in the HAUS Section of the rocket above the transition. The payload will not be in direct contact with the airframe of the rocket but will be secured to the MSM such that after landing and separation the payload can be deployed. The MSM system will be mounted to the airframe of the rocket nose cone coupler and nose cone.

5 Safety

5.1 Team Safety Culture

To ensure the safety of team members and all involved during the full progression of this project, a safety plan must be developed and implemented. The Safety plan will show a team-wide comprehension of the guidelines and expectations that must be upheld. As specified by the 2026 Student Launch Handbook, section 5.2, each team is required to have a safety officer. The responsibilities of the safety officers are also described in section 5.2 and give an essential understanding of expectations and requirements. The team safety officer will support the team leader in completing all tasks safely and incident-free. This will be done with adherence to all governing authorities included in 5.2 Compliance with Laws, Codes, and Regulations. During all launch procedures, NAR launch procedures and safety regulations will be used for flight design and operations. NAR/TRA team mentor and the Range Safety Officer (RSO) will oversee the launch operation and motor handling and have ultimate authority over launch readiness and viability. While facilitating and implementing the safety plan is the main responsibility of Jonathan Serbest, our safety officer, all team members acknowledge the individual requirements of understanding and compliance of the plan outlined below. This has been done through a safety briefing and provides confirmation of safety plan competency and pledge to plan adherence.

4.3 Procedure and Preparation Verification

4.3.1 Pre-Launch

4.3.1.1 Recovery and Respective Electronics Preparation with Reefing System and Verification Procedure

Authority Responsible:
Lead: Austin Zaal (Recovery Officer)

GENERAL PPE REQUIRED:

- Closed-toe shoes
- Safety glasses
- Gloves

The recovery system is extremely important in ensuring the safe launch and landing of the rocket. All high-powered rockets require a recovery system capable of returning the rocket and all components safely to the ground. This will be done while preserving the rocket's original condition and at a non-hazardous ground high velocity. Since the motor selected has an impulse over 2560 N-sec, this requires the implementation and use of an electronic actuated recovery system. Our rocket will satisfy both of these requirements in accordance with NFPA 1127 (4.10.1 & 4.10.2).

1. **Disarmed Electronics Verification** – Ensure no voltages or currents are present in avionics bay when systems are unarmed. *This must be done to prevent the possibility of premature detonation and possible injury.* This includes:
    a. Verifying there is no power to either flight computer.
    b. Verifying energetic circuits are open in two places (open switch and unarmed flight computer).
    c. Ensure solid soldered connections among energic circuits.
    d. Verify two fully charged power sources are present.
2. **Ensure Energetics are Disabled** – Ensure black powder/CO2 cartridges are unarmed and placed in avionics bay housing. *This must be done to prevent the possibility of premature detonation and possible injury.*
3. **Reefing System Connection** – Ensure reefing system is disconnected from power by opening the box and checking for red or green lights. If no lights are visible, connect e-match to reefing system by inserting wire through small hole in box, insert individual e-match leads into screw terminals, and secure using screwdriver.
4. **Verify Arming Key Switch is Properly Installed** – Ensure arming system is working correctly. *Switch Key must be used to activate electronics on-board only after the rocket is on the pad to prevent premature separation while handling.* This is done by:
    a. Remove both power sources from the avionics bay.
    b. Connect multimeter to the terminals of the arming switch.
    c. Turn the key switch to the "armed" position.
    d. With the multimeter in the "Continuity" mode, verify the terminals are properly connected.
    e. Turn the key switch to the unarmed mode.
    f. Using the multimeter, verify the connection is open.
    g. Replace the power sources back into the avionics bay.
5. **Verify Means to Fire Energetics are Present –** Visually verify electronics matches have been secured in proper position and have secure connections to logic electrics and power.
6. **Check Reefing System Electronics –** Visually inspect the electronics within the reefing system electrical box. Ensure all components are seated securely within the box. Using multimeter, ensure lithium battery is charged to at least 4.0V.

7. **Reefing System Power Verification -** Connect battery to JST connector. Verify one red light appears on the ESP-32 as well as one green light for each of the sensors (Three lights total). Close and secure reefing electrical box.
8. **Check Bulkhead Rigidity** – Ensure all bulkhead connection points are secure.
9. **Prep Chutes and Components** – Properly fold & bag drogue, pilot, and main parachutes respectively.
10. **Prep Shock Cords** – Properly bundle shock cords and connect shock cord to bulkheads via quick links.
11. **Insert Chutes and Components** – Carefully pack reefing electronics, cords and parachute bags into their respective housings. Parachute bags will be flame-resistant bags since black powder is being utilized for separation. *Ensuring the reefing parachute is not tanked and properly folded limits the chances of reefing malfunction. These malfunctions can lead to a fast ground-hit-velocity and possess a greater hazard to the personnel present on site*. NFPA 1127 (4.10.3)

4.3.1.2 Payload Preparation, Mechanical Separation, and Verification Procedure

Authority Responsible:
Lead: Case Dyer (Payload Officer)

GENERAL PPE REQUIRED:
- Closed-toe shoes
- Safety glasses
- Gloves

- ☐ Verify the mechanical separation assembly is correctly assembled, with all fasteners properly installed, torqued, and secured.
- ☐ Verify all payload fasteners are tight and safety checked.

1. **Confirm the retention system is engaged –** Ensure that the retention system and microcontroller are connected and power is present.

- ☐ Verify the water applicator system is secured and leak-free.
- ☐ Verify the payload is installed into the forward body tube and oriented correctly.

- [ ] Verify payload wiring is secured, strain-relieved, and free of pinch points.
- [ ] Verify the forward body tube is properly sealed.
- [ ] Verify payload batteries are charged to greater than 90%.
- [ ] Verify all electronic connections and solder joints are intact.

2. **Power on the payload and verify successful boot of the payload computer** – Communication between the mechanical and electrical are paramount in ensuring satisfaction.

- [ ] Verify voltage and current levels are within nominal operating limits.
- [ ] Verify data logging is enabled.
- [ ] Verify tracking system functionality.
- [ ] Verify the mechanical linkage actuates correctly.

3. **Cycle the drill mechanism through its full range of motion** – This is to make sure that drill is capable of expected movement to critical positions. This can be done by physically operating the linkage through its entire expected path of movement.

- [ ] Verify drill motor direction and operation.

4. **Perform a no-load test of separation motors –** To verify that separation system is in working order before launch, engage motor drivers to spin all motors.

5. **Initiate mechanical separation test** –
   a. Verify solenoid unlocks as commanded.
   b. Verify orientation stepper motor rotates the assembly to the correct deployment position.
   c. Verify separation stepper motors extend to the designed deployment length.

- [ ] Verify the mechanical separation system functions as designed.

6. **Inspect all components** - Looking for signs of overheating, abnormal wear, or damage.

- ☐ Re-verify battery state of charge, voltage, and current draw.
- ☐ Verify no abnormal power consumption is present.
- ☐ Verify mechanical separation assembly is securely mounted to the rocket airframe.

7. **Re-initiate mechanical separation test –** to confirm system readiness for flight.

4.3.1.3 Rocket Assembly Preparation and Verification Procedure

Authority Responsible:
Lead: Krieg Conrad (Project Manager)

GENERAL PPE REQUIRED:

- Safety glasses
- Nitrile or latex gloves
- Closed-toe shoes

1. **Ensure inclusion of all major subsystems –**

- ☐ Ensure avionics bay is installed and secured
- ☐ Ensure payload assembly is installed and properly oriented
- ☐ Ensure recovery systems are attached to avionics bay bulkheads

2. **Install recovery systems into airframe –**

- ☐ Properly fold and pack drogue and main parachutes
- ☐ Properly stow shock cords and recovery lines
- ☐ Ensure no twisting, pinching, or binding is present

3. **Insert transition section into airframe –**
   - ☐ Insert transition coupler into the aft opening of the forward body section
   - ☐ Ensure full insertion and proper alignment

4. **Install first set of shear pins –**
   - ☐ Install shear pins at all designated joints
   - ☐ Verify pins are fully seated and secure

5. **Install avionics and aft airframe sections –**
   - ☐ Insert avionics bay and aft body coupler into the forward airframe
   - ☐ Ensure mechanical engagement at all interfaces

6. **Install second set of shear pins –**
   - ☐ Install shear pins at all remaining designated joints
   - ☐ Verify proper shear pin count and placement

7. **Install nose cone –**
   - ☐ Insert nose cone into forward body tube
   - ☐ Verify retention system is engaged

8. **Locate and mark CP and CG –**
   - ☐ Measure and mark center of pressure (CP)
   - ☐ Measure and mark center of gravity (CG)

9. **Inspect vent ports –**
   - ☐ Ensure all vent ports are unobstructed
   - ☐ Verify vent holes align with avionics requirements

10. **Perform final airframe inspection –**
    - ☐ Inspect entire vehicle for surface defects or damage
    - ☐ Verify all fasteners are present and secure
    - ☐ Ensure no foreign objects remain inside the rocket

4.3.1.4 *Motor Preparation*

Note: Reloadable motor kit shall not be removed from manufacturer packaging until team is ready to install the motor into the reloadable motor system (RMS) and ready to proceed with launch upon competition. During the process of motor preparation and loading, the motor shall be handled by the NAR/TRA team mentor, Elizebeth Gildersleeve and accompanied by the team Safety Officer, Jonathan Serbest.

GENERAL PPE REQUIRED:

- Safety glasses
- Nitrile or latex gloves
- Closed-toe shoes

PRELIMINARY PROCEDURE:

- **Ensure Locational Safety** – Before continuing further, the nature of the following steps includes highly explosive material. In accordance with NAR/TRA, NFPA 1127, and ATF 27 CFR, motor preparation shall not be within "50 feet of any smoking, matches, open flames, and spark producing devices" (27 CFR 555.212).
- **RMS Inspection** – RMS must be fully capable of housing the specified motor. This means that the RMS must be clean and without any internal impedance, blocking the motor from being installed. If inspection reveals a defect, the motor shall not be installed until the issue is resolved.
- **Motor Kit Inspection –** Remove motor from manufacturer sealed packaging and begin inspecting all components of the kit. The kit must pass a detailed inspection before use. Possible defects include but are not limited to indentions, incisions, tears, missing components, and leaking solid propellant.

Failure to complete the preliminary procedure will result in the stoppage advancing to the procedure until satisfied.

PROCEDURE:

1. **Load Motor into RMS –** Using the instructions included in the Kit, install the motor into the RMS. During this time, the team safety officer must survey the area, continuing to make sure that the area utilized is an ignition free zone.
2. **Load RMS into Rocket** – Once the team mentor has successfully installed the motor and additional components into the RMS, insert the RMS into the bottom of the rocket and push until the top of the RMS is pressed firm against the bottom of the lower bulkhead.
3. **Secure the RMS into Rocket** – Once the RMS is pressed firmly into place, screw the boattail onto the threaded end of the aft. When fully screwed on, the boat tail presses against the bottom of the RMS and secures it in place.
    - **RMS and Boat-tail Inspection** – Make sure that both RMS and Boattail are ridged and mated together

Do not insert igniter. This will be done once the rocket is attached to the launch rod and resting in its vertical position on the launch pad.

4.3.1.5 Setup on the launch pad

GENERAL PPE REQUIRED:

- Safety glasses
- Nitrile or latex gloves
- Closed-toe shoes

1. **Disarmed Electronics Verification** – Look to step one of the *Recovery and Respective Electronics Preparation* for procedure information.
2. **Ensure Electronic are disabled** - Ensure that any onboard energetics and firing circuits of any kind are to be disabled until the rocket is firmly fixed to the launch rod and in its vertical orientation on the launch pad. NFPA 1127 (4.13.7)
   This applies to:
   a. Avionics Bay
   b. Reefing system (if applicable)
   c. Payload Vehicle
   d. Payload Mechanical Separation
3. **Rocket Transport** – Once the rocket is prepared for launch, 3 team members, similar in shoulder height, will begin to carry the rocket to the designated launch rod.
4. **Securing the Rocket to Launch Rod –** The rail buttons are located on the outside of the rocket air frame and will slide into the launch rod grove. This groove runs up the entirety of the rod. Once inserted, secure the rocket at rest, to the bottom of the pad.
5. **Initiate Payload and Separation Power Systems** – Power to payload rover and mechanical separation systems must be initiated using mechanical switches. These systems then wait for activation protocol within the software to be carried out.
6. **Arm Energetics and Firing Systems** – Using key cylinder switch, all electronic within the avionics bay must be armed. During the arming of all onboard energetics and firing circuits, there shall be no participants or spectators of any kind near the launch pad other than authorized personnel tending to the arming and disarming operations.
7. **Installing the Electrical Motor Igniters** – Once the rocket is in the correct orientation, install the motor igniters into the motor. Please see the "*Igniter installation*" section for information on proper installation procedure. A safety interlock is required to be wired in series, and the launch switch shall not be installed until the rocket is ready for launch.

4.3.1.6 Igniter Installation

Authority Responsible:
Lead: Elizabeth Gildersleeve (Team Mentor)

GENERAL PPE REQUIRED:

- Closed-toe shoes
- Safety Glasses

1. Rocket Transport – The safety officer will carry the motor igniter as the rocket is being transported to the launch pad. Once the rocket is attached to the launch rod and in place, igniter installation can begin. Diagram below is provided by ESTES.

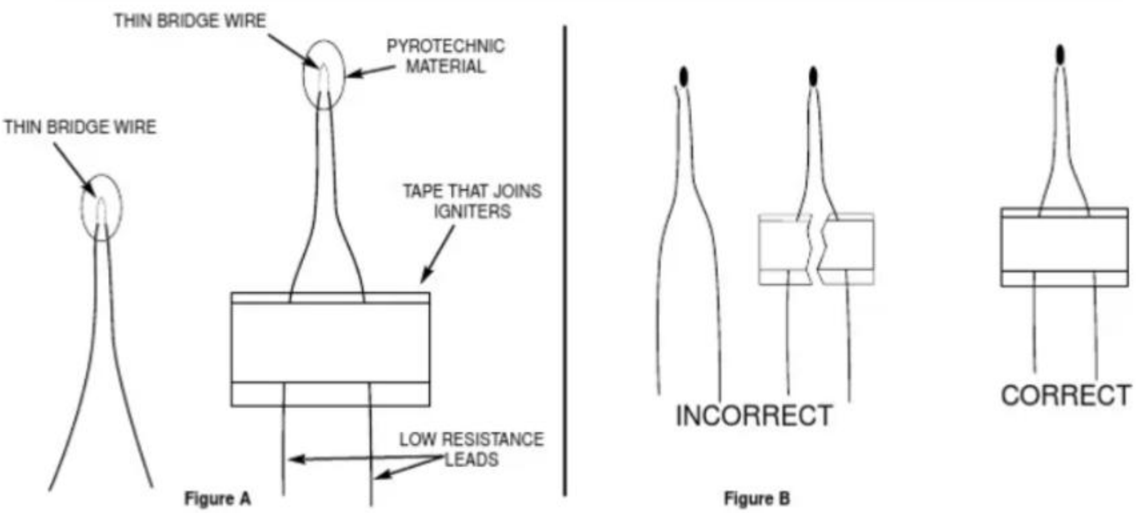


Figure 5-13 Igniter Installation Guide

1. **Igniter Installation** – Insert the igniter end into the base of the rocket motor. This should be inserted until the igniter reaches the top of the motor opening.
2. **Tape Leads** – Leads must be separated from each other and taped to prevent contact. This is to prevent pre-detonation.
3. **Tape Leads to Rocket Base** – To keep the igniter pressed to the top of the motor, bend the ends of the exposed wire over the base of the rocket and tape to the air frame.
4. **Lead Connection** – Connect each lead to separate alligator clips and the alligator clip wires to the ignition system.

4.3.2 Launch

At launch, rocket is in place and inspection has deemed the rocket adequate for flight through the use of the defined procedure/checklists and verified through respective documentation and AHJ confirmation including the team mentor, Elizebeth Gildersleeve and RSO.

Note: All sanctioned launch areas by NAR / TRA are to have proper safety accommodation based on all AHJ.

- **Confirmation of area safety and shutdown to launch** – Area is deemed safe when location of all participants and spectators meet the minimum distance table, provided by NAR High Power Rocket Safety Code highlighted below.

MINIMUM DISTANCE TABLE

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

Note: A Complex rocket is one that is multi-staged or that is propelled by two or more rocket motors
Revision of August 2012

Table 5 Minimum Distance

1

After the area is deemed to be safe for launch, a five second countdown will be used before launch initiation.

- **Verify Rocket Stability** –Verify rocket stability on the launch pad and firmly coupled to the launch rod. If stability is inadequate, launch of the rocket must not proceed and must be cancelled until future notice.

1. **Watch Rocket** – It is crucial that the team and nearby spectators are always informed of the rocket's location throughout the length of its flight. Pointing to its location will be employed to inform the location of the rocket.
2. **Stay clear of rocket until it has hit the ground** – There shall not be an attempt to catch the rocket while descending or retrieving the rocket in areas of hazard like powerlines or on top of trees. If the rocket lands on a powerline, local AHJ shall be called. NFPA 1127 (4.10.4-6).

4.3.3 Troubleshooting

No continuity detected during testing:

GENERAL PPE REQUIRED:

- Safety glasses
- Nitrile or latex gloves
- Closed-toe shoes

1. Take apart avionics bay where electronics are visible.
2. Visibly inspect e-matches and connections.
   a. Verify match wire are completely intact.
   b. Verify match wires are secure within screw terminals.
   c. Verify match wires are in correct screw terminals.

No connection between TeleBT and TeleMetrum for Testing

1. Unplug TeleBT from computer and close AltOS.
   a. Plug TeleBT back into computer.
   b. Verify battery is plugged into TeleBT
   c. Try connection again using the "Monitor Idle" menu option.
   d. Wait 60 seconds for connection.
2. Remove electronics from avionics bay
   a. Verify power is delivered to flight computer by ensuring light is on.
   b. Visually inspect antenna solder joint.
   c. Place electronics back in avionics bay.

4.3.4 Post-Launch

GENERAL PPE REQUIRED:

- Safety glasses
- Nitrile or latex gloves
- Closed-toe shoes

3. Confirm recovery location –

- Establish visual contact with the vehicle
- Acquire and verify GPS signal

4. Recover vehicle –

- Approach vehicle cautiously from upwind

• Verify no visible hazards (smoke, heat, damaged components)
• Secure the vehicle

5. **Safe all systems –**
• Power down all avionics and payload electronics
• Verify no energetic devices are armed

6. **Inspect energetics and hardware –**
• Inspect for residual black powder or ejection charge debris
• Verify no unintended ignition or damage occurred

7. **Retrieve flight data –**
• Remove data storage devices
• Verify successful data retrieval

4.4 Safety and Environment

4.4.1 Personal and Environmental Hazard Analysis

4.4.1.1 Hazard Analysis Methodology

Risk identification is necessary as this will be the backbone of our safety procedure moving forward. Identifying the possible risks in and around the progression of our rocket helps all involved to suppress the chance of a major hazard happening. To begin, the criteria on how the analysis of probability is weighted is in Table 3-1 below.

Probability Levels		
Definition	Value	Probability of Occurrence
Unlikely	1	<1%
Uncommon	2	1% to 15%

Occasional	3	15% to 50%
Probable	4	50% to 80%
Likely	5	Above 80%

Table 5 Probability Levels

Now that the metrics for our probabilities are defined, the criteria for assessing risks and hazards will be handled through both the severity of the hazard and its likelihood. Table 3-2 below illustrates this.

Hazard Severity Criteria					
Description	**Value**	**Personnel Health**	**Equipment Health**	**Launch Vehicle**	**Mission Health**
Negligible	1	No injury	Insignificant damage	Insignificant damage/ minor reconstruction	Mission success not impacted
Moderate	2	Minor injury	Moderate damage	Moderate reconstruction	Possible mission success
Critical	3	Severe injury	Major damage	Major reconstruction required	Partial mission success
Disastrous	4	Life-threatening injuries	Complete destruction	Complete loss	Mission failure

Table 5 Hazard Severity Criteria

With the hazard severity visually defined, the point hierarchy from both graphs combine to give us a comprehensive understanding of the risk analysis. Defined in color and number notation, is the risk level. The risk level is found by multiplying the corresponding number rating of x-plane identifier with the intersecting y-plan identifier. This gives us the ability to score risks across our two identifiers. Note that ranking of these hazards will be made under the assumption that none of the mitigations have been instituted yet.

Risk and Hazard Assessment Matrix					
		Severity			
		Negligible (1)	Moderate (2)	Critical (3)	Disastrous (4)
Probability	Unlikely (1)	1	2	3	4
	Uncommon (2)	2	4	6	8
	Occasional (3)	3	6	9	12
	Probable (4)	4	8	12	16
	Likely (5)	5	10	15	20

Table 5-3 Risk and Hazard Assessment Matrix

The Consolidated Hazard Assessment Matrix simplifies the numerical results of the hazard analysis by grouping the full range of values into four categories. Each category provides clear guidance on risk level, assists the team in making informed decisions per the severity of hazards, the intensity of required mitigations, and overall safety measures necessary for project activities.

Consolidated Hazard Assessment	
Definition	Value
Minimal	1-5
Tolerable	6-10
Immediate Attention	11-15
Failure	16-20

Table 5-4 Consolidated Hazard Assessment Matrix

List and meaning of hazard groups:

Minimal (1–6):

Hazards of this caliber are small and can be easily avoided by using standard safety procedures.

Tolerable (6–10):

Hazards of this caliber require the implementation of mitigation measures as the average severity for these cases is moderate. Mitigations like training, proper PPE, spatial awareness, etc. must be constantly enforced.

Immediate Attention (11–15):

Hazards of this caliber must be dealt with swift and rigorous mitigation. Before proceeding, the Safety Officer must be notified and holds the authority to stop activities if deemed necessary.

Failure (16–20):

Hazards of this caliber are unacceptable and will not be attempted. The Safety Officer must be notified, and a new technique, design, safety mitigation, etc., must be implemented to lower the risk level before proceeding.

4.4.1.2 Construction and Manufacturing Hazard

Construction Hazard Table

Hazard	P'	S'	Value	Cause	Effects	Mitigation
Acetone Chemical Exposure	3	2	6	Teammate can be exposed to gas inhalation and skin contact during prepping of the rocket airframe for painting.	Perspiratory irritation or difficultly breathing.	• Use spray paint in an open and well-ventilated area. • Spray paint in a controlled fashion to limit the airborne VOC at one time. • Ensure that others are wearing proper PPE or are not in the area of effect. PPE: Gloves, safety glasses, and a face mask capable of filtering chemical and partial inhalants shall be used.
Chop Saw contact or Amputation Injury	1	4	4	• Chop saw blade jumps and jitters when cutting with a dull blade. • Improper maintenance or cleaning	Cuts, lacerations, or even lib amputation.	• Make sure that blade shield is in place. • All wires shall be inspected before use. • Storage of tools safely away from heavy occupied zones and safe to hold in their stored state. • Don't work with dull blade. PPE: Heavy-duty gloves, Safety glasses, earplugs, muffs, and a face mask capable of filtering chemical and partial inhalants shall be used.
Drilling Tool Injury	1	4	4	• Unclean or oily hands while working • Negligent handling • Leaving energized tools unattended • Open wires	Cuts and lacerations	• Use proper technique for drilling. • Utilize the drill for its intended use. PPE: Heavy-duty gloves and Safety glasses
Epoxy inhalation	3	2	6	While applying and sanding epoxy, VOCs are emitted that are toxic to breathe.	Perspiratory irritation or difficultly breathing.	• Operate tools capable of sanding epoxy in a well-ventilated area. • Ensure that others are wearing proper PPE or are not in the area of effect. PPE: Gloves, Safety glasses, and a face mask capable of filtering chemical and partial inhalants shall be used.
Excessive Noise Exposure	3	2	6	Environments with power tools used for extended periods of time.	Hearing loss	Use earmuffs or plugs when using tools that are capable of 85 decibels or more.
Fiberglass Dust Inhalation	3	3	9	• Fiberglass airframe of rocket will be subject to cutting and sanding causing airborne debris.	Perspiratory irritation or difficultly breathing.	• Operate tools capable of sanding and cutting fiberglass in a well-ventilated area. • Ensure that others are wearing proper PPE or are not in the area of effect. PPE: Gloves, Safety glasses, and a face mask capable of filtering chemical and partial inhalants shall be used.

Hand Saw contact	1	3	3	• Cutting with excessive force can cause the slipping of jumping of the blade. • Unclean or oily hands while working	Cuts and lacerations.	• Make sure that gloves are clean when working • Cut away from yourself PPE: Heavy-duty gloves and safety glasses
Hot Solder Contact injury	2	2	4	• The soldering iron head is extremely hot. • Leaving the iron energized and unattended	Burns varying in degree depending on severity.	• Soldering iron mount should be used utilized when not using. • Do not leave the iron powered with not using. • Well ventilated PPE: Safety glasses, latex gloves, face mask capable of filtering chemical and partial inhalants shall be used.
Hot Solder Gas Inhalation	2	2	4	Gases created when soldering metal flux.	Perspiratory irritation or difficultly breathing.	
Rotary Sander Injury	3	3	9	• High RPM rotary head can be unstable and hard to control when improperly sanding a surface. • Leaving energized tools unattended. • Negligent handling • Rough, tool head. • Exposed wires	Skin abrasion and lib entanglement, and inhaling debris	• Operate the sander head parallel to the surface of interest. • Use new sandpaper when starting needed • Ensure that others are wearing proper PPE or are not in the area of effect. PPE: Heavy-duty gloves, Safety glasses, ear plugs, muffs, and a face mask capable of filtering chemical and partial inhalants shall be used.
Spray Paint Fume Inhalation	3	2	6	Airbourne paint particles when painting rocket components or airframe.	Perspiratory irritation or difficultly breathing.	• Use spray paint in an open and well-ventilated area. • Spray paint in a controlled fashion to limit the airborne VOC at one time. • Ensure that others are wearing proper PPE or are not in the area of effect. PPE: Gloves, Safety glasses, and a face mask capable of filtering chemical and partial inhalants shall be used.

Table 5-5 Construction and Manufacturing Hazard Table

4.4.1.3 Launch Preparation Hazards

Pre-Launch Hazards						
Hazard	P'	S'	Value	Cause	Effects	Mitigation
Battery runaway	1	3	3	• Negligence in charging before preparing for launch. • Cell Defects/Degradation	• Electric shock • 1st, 2nd, and 3rd Burns • Puncture of battery	• Disconnect from the charger before leaving and store in a secure place. • Disconnect battery from charger when full charged PPE: Gloves
Electrical Short	1	2	2	• Damaged or Worn Wiring • Loose connections • Moisture from the environment • Water reservoir leaks water onto the microcontroller • Excessive current	• Electric shock • Minor burns	• Inspect all wires before events like storage, testing, and launch. • Inspect all terminal connections for loose wires or broken connections PPE: Gloves
Injury using power tools	3	3	3	• Unclean or oily hands while working • Negligent handling • Leaving energized tools unattended • Open wires	• Cuts • Lacerations • Punctures	• Utilize the power tools for their intended use. PPE: Heavy-duty gloves and Safety glasses
Trip or falling hazards	3	1	3	Unclean workstation	• Bruises • Cuts • Broken bones	• Ensure that there is a wide path of obstruction-free space for foot traffic • Keep the workstation clean and tidy PPE: Closed-toed shoes
Unintentional ignition of black powder or energetic devices	1	4	4	Electrical current is run through the system prematurely	• 1st, 2nd, and 3rd Burns • Bleeding • Cuts	• Black powder must be stored in dry, ignition-free areas. • Possession and loading of the motor shall be done by the team mentor. • Avionics bay, including CO2 canisters, black powder, and E-matches, shall be handled while deactivated and unarmed. • Stored in accordance with NFPA standard. • Transported in accordance with ATF PPE: Gloves, safety glasses

Table 5-6 Pre-launch Hazards

4.4.1.4 Launch Hazards

Launch Hazards						
Hazard	P'	S'	Value	Cause	Effects	Mitigation
Debris Impact	1	4	4	Rocket fails to deploy in a controlled manor and breaks apart in flight Rocket safely deploys and is falling back down to the ground	Injuries like blunt force trauma, bleeding, puncture wounds, or even death	• Verify that the simulation data shows a controlled descent after all components are weighted at the launch site. • When the rocket is launched, point to the rocket until safely on the ground. This is to continually make the rocket's location known to the surrounding personnel
Igniter misfire	1	4	4	Electrical current is present while or after connecting motor igniter	Severe injuries, including burns, hearing loss, and blindness, may occur if personnel are in the immediate area of the rocket when the igniter fires.	• Ensure that the electrical circuit is closed before connecting the ignitor • Only required personnel should be near the rocket when ignitor is installed
Late Main deployment	3	2	6	• Improper folding • Loss of connection between the airframe and the parachutes during deployment • Forgetting to attach the parachute system to the eye bolt of the bulkhead • Reef does not fully open at its expected altitude. • Reef lines are tangled	Rocket does not slow to a manageable velocity before hitting the ground, and contact with someone could be fatal	• Extensive testing of the reefing system before launch. • Inspect the parachute to look for preexisting weakness before launch. • Ensure that all components are assembled correctly before launching. • Verify that the simulation data shows a controlled descent after all components are weighted at the launch site.
Premature separation	2	3	6	• Higher than expected jerk during chute deployment or improper shear pin sizing for the size of the rocket. • Premature detonation • Electrical current is run through the e-match prematurely	The motor impulse remains in effect while the airframe separates, potentially causing a catastrophic failure and resulting in the loss of the rocket. This could result in falling debris of varying velocities.	• Inspect the microcontroller and ensure that e-matches are utilizing the proper output ports. • Ensure that the flight computer is disabled to prevent premature puncture while assembling and to prevent possible injury to the team and surrounding persons • Verify that the simulation data shows a controlled descent after all components are weighted at the launch site.

Vehicle instability	3	3	9	• Wind or rail departure anomaly • Inaccurate measurement of weights • Recovery structures or airframe geometry is compromised	The motor impulse remains in effect while the airframe separates, potentially causing a catastrophic failure and resulting in the loss of the rocket. This could result in falling debris of varying velocities.	• Verify that the simulation data shows a controlled descent after all components are weighted at the launch site. • Ensure that the wind on site is below 20 mph.

Table 5-7 Launch Hazards

4.4.1.5 Environmental Hazards

Environmental Hazards						
Hazards	P'	S'	Value	Causes	Mitigations	Verification
Potential Effects and Hazards to Environment						
Material Waste	5	1	5	• PLA filament waste due to supports and shaping. • Packaging (cardboard, Styrofoam, bubble wrap) • Excess epoxy.	• Recycle as much material as possible. • Use as much as needed	• A designated recycling bin will be located in the major area of waste production such as the Senior Design Lab.
Chemical Aerosol	3	2	6	Chemical reaction producing a aerosol by-product.	• Were proper PPE when using any chemical • Ventilated room or open air is required.	• Proper PPE for according to manufacturer specification
Energetic by-product (post launch)	5	1	5	Ash, smoke, dust, and soot.	• Do not approach rocket launch pad or rocket until by-products have clearly dissipated.	• Post-Launch check log shall be implemented to verify completion of debris removal.
Field Debris	5	1	5	Broken fins, launch preparation disposables, snack trash and water bottles.	• Continue to clean throughout the day. • Bring plenty of trash bags and dispose of the trash properly with any recyclable trash be handled accordingly.	• Post-Launch check log shall be implemented to verify completion of debris removal.
Batteries	5	2	10	Improper disposal of batteries or any device using lead-acid or AGM technology.	• Dispose of batteries per the manufacture specification.	• Log of all batteries being disposed of and when.

Potential Environmental Effects and Hazards on the Rocket						
Dense pockets of air and cross wind	3	3	9	Early deployment due to avionics bay air density verification.	• The avionics bay will be equipped with multiple pitot tubes for ample air circulation and barometric pressure accuracy	• Pre-flight check for any pitot tubes obstructions. • Flight computer and avionics will be calibrated to launch site elevation. • Pre-Launch check log shall be implemented to verify completion of calibration and obstruction check.
Typical Barometric Pressure Change	5	1	5	• Impacts accuracy of data collection.	The avionics bay will be equipped with multiple pitot tubes for ample air circulation and barometric pressure accuracy	• Pre-flight check for any pitot tubes obstructions. • Flight computer and avionics will be calibrated to launch site elevation. • Pre-Launch check log shall be implemented to verify completion of calibration and obstruction check.
Terrain and Vegetation (including water landing)	3	1	3	• Harsh landing to rocket body. • Incapable of rocket recovery. • Risks to payload deployment	• Air frame shall be made with a safety factor ≥ 2 of forces expected at ground hit velocity. • Avionics bay will house a remote GPS • Payload shall be designed to traverse a multitude of terrains.	• Testing of material strength prior to launch. • Testing of Avionics Bay GPS will occur prior to launch. • Payload will be tested for its ability to operate in extreme terrains during its design and building processes. • Pre-Launch check log shall be implemented to verify completion of GPS calibration check.
Temperature and Humidity Extremes	4	1	4	Affect to burn time, apogee, ground hit velocity, and time to ground.	• Rocket preparation shall be done in shade for as long as possible. • Survey weather patterns thought the day.	• Ensure there are sufficient tent coverings to work under.
Solar Exposure and Thermal Gradients	4	1	4	Weakness adhesives	• Rocket preparation shall be done in shade for as long as possible. • Survey weather patterns thought the day. • Check epoxy fillets for structural rigidity.	• Check material rigidity • Pre-Launch check log shall be implemented to document confirmation of material rigidity verification.
Lightning or Severe Weather	3	2	6	Safety of mission is jeopardized as it cause severe concerns for equipment and personnel well-being.	• Do not flight until instructed at further notice by the RSO	• Verbal confirmation my mentor and RSO before resuming launch. • Pre-Launch check log shall be implemented to document confirmation of both parties.

Table 5-8 Environmental Hazards

4.5 Failure Modes and Effects Analysis

The failure mode and Effects analysis Provides a detailed list of possible failures and mitigation strategy. Potential failure modes for payload, recovery, and all of its supporting systems are described along with their causes and the resulting effects. For the following tables, the FMEA is performed by finding all causes to the failure defined and its respective effects at three levels: component, sub-scale, and system levels. A failure that appears minor at a component level can become a larger hazard as the effects continue to propagate to higher levels of the system. Each component onboard the rocket serves a specific and critical role, and failure at any of these levels has the potential to influence the overall mission success. By examining failures across the board of level hierarchy, the FMEA ensures that both localized and system-wide consequences are properly identified and mitigated. The FMEA ensures the safety of personnel, mission success, and a launch compliant with the NASA USLI expectations. FMEA risk and hazard rankings are determined using probability and severity criteria, with consideration given to impacts on personnel safety, hardware integrity, and mission success.

4.5.1 Recovery Structures and Component FMEA

Recovery Structures and Components										
Item	P	S	Value	Use	Failure Mode	Failure Cause	Effects			Mitigations
							Component-Level	Subsystem-Level	System-Level	
Mechanical										
Air frame body tubes	1	3	3	Provides the geometry of the structure and the majority of rigidity. Contains payload, avionics system, motor, and respective hardware.	Cracking or bent (Depending on material composition)	Damage while transporting or in-transit while in package.	Cracks, breaks, and fiberglass fraying during construction or assembly.	Incapable of containing all components and sub-systems or unstable.	Does not meet safety protocol and standard; disqualifying it from launch.	• Sand down the surface of the airframe after cutting to length. • Inspect all body tubes before construction and assembly.
Air frame body tubes	1	3	3	Provides the geometry of the structure and the majority of rigidity. Contains payload, avionics system, motor, and respective hardware.	Cracking or bent (Depending on material composition)	Damage during deployment of parachutes.	Cracks or breaks during construction or assembly.	Incapable of containing all components and sub-systems or unstable.	Does not meet safety protocol and standard; disqualifying it from launch	Verify that the simulation data shows a controlled descent after all components are weighted at the launch site.
Bulkhead	2	4	8	Use primary to subdivide portions of the rocket into smaller compartments, which provides structural rigidity and, in some cases, allows tie-off areas for parachutes.	Epoxy fails	Failure due to high instant acceleration during chute deployment or lack of adhesion between airframe and centering rings.	Epoxy fails and bulkheads fall out of place.	The bulkhead is unable to contain the contents within the sub-section or possible loss of parachute	Contents like the payload fail to deploy or achieve task successfully or catastrophic loss of the vehicle if the chute is lost during deployment.	• Properly mix the epoxy and apply generously to the edges of the bulkhead • Allow ample time for curing of epoxy before launch.
Bulkhead	2	4	8	Use primary to subdivide portions of the rocket into smaller compartments, which provides structural rigidity and, in some cases, allows tie-off areas for parachutes.	Eye bolt fails	Eye bolt shears out of the bulkhead	Loss of parachute from bulkhead	Rocket is incapable of falling in a controlled and safe manner.	The rocket hits the ground at high velocity and sustains catastrophic damage.	• Ensure that the eye bolt is firmly secured to the bulkhead • Verify that the simulation data shows a controlled descent after all components are weighted at the launch site.

Centering rings	1	3	3	Located in the lower unit of the rocket. Centering rings provide stability and proper fitment of the motor throughout the flight of the vehicle.	Geometry fails	Cracking or splitting due to high jerk during chute deployment. Typically, at the location of chute eye bolt due to local stress concentration.	Rings fail and fall out of place.	Failure to provide motor support causes the motor shifts inside the body tube of the rocket.	Flight is unstable, erratic, or catastrophically fails.	Inspect the centering rings before installing into the aft of the rocket
	1	3	3	Located in the lower unit of the rocket. Centering rings provide stability and proper fitment of the motor throughout the fight of the vehicle.	Epoxy fails	Failure due to high jerk during chute deployment or lack of adhesion between airframe and centering rings.	Epoxy fails and ring fall out of place.	Failure to provide motor support causes the motor shifts inside the body tube of the rocket.	Flight is unstable, erratic, or catastrophically fails.	• Properly mix the epoxy and apply generously to the edges of the bulkhead • Allow ample time for curing of epoxy before launch.
	1	3	3	Located in the lower unit of the rocket. Centering rings provide stability and proper fitment of the motor throughout the fight of the vehicle.	Epoxy fails	Improper application, due to a lack of adhesion. Improper mix of two-step solution, or insufficient amount used.	Epoxy fails and ring fall out of place.	Failure to provide motor support causes the motor shifts inside the body tube of the rocket.	Flight is unstable, erratic, or catastrophically fails.	• Properly mix the epoxy and apply generously to the edges of the bulkhead • Allow ample time for curing of epoxy before launch.
Couplers	1	3	3	Positioned between two body tubes and provides support between sections.	Geometry fails	Cracking or splitting due to high jerk during chute deployment. These sites of failure would typically be located at the screws that fix the coupler and the airframe.	Coupler cracks or breaks during flight.	Coupler isn't fixed to the body tube	Stability of the rocket or its geometry is compromised.	Inspect the couplers for preexisting weaknesses before installing them throughout the rocket during the construction and assembly.
	1	3	3	Positioned between two body tubes and provides support between sections.	Epoxy fails	Improper application, mostly likely due to a lack of adhesion , improper mix of two step solution, or insufficient amount used.	Coupler epoxy fails during flight.	Coupler isn't fixed to the body tube	Stability of the rocket or its geometry is compromised.	• Properly mix the epoxy and apply generously to the edges of the bulkhead • Allow ample time for curing of epoxy before launch.
CO2 Canisters	1	4	4	Utilized as an energetic source for airframe separation after apogee is reached.	Premature puncture of the CO2 canister	E-match is set off prematurely	CO2 canister is punctured and gas is expelled into the airframe	Airframe separates	The motor impulse remains in effect while the airframe separates, potentially causing a catastrophic failure and resulting in the loss of the rocket.	• Inspect the microcontroller and ensure that e-matches are utilizing the proper output ports. • Ensure that the flight computer is disabled to prevent premature puncture while assembling and to prevent possible injury to the team and surrounding persons.
E-match	1	4	4	Used to send an electrical charge from the microcontroller to a black powder charge. This charge is what sets the firing pin into motion, and ultimately, the CO2 canister puncturing	Electrical current is run through the e-match prematurely	E-match is set off prematurely	CO2 canister is punctured and gas is expelled into the airframe	Airframe separates	The motor impulse remains in effect while the airframe separates, potentially causing a catastrophic failure and resulting in the loss of the rocket.	

Fins	1	3	3	Primary provides stability to the rocket and proper trajectory. These are located at the base of the rocket.	Geometry fails	Cracking or splitting when the rocket hits the ground after flight.	Fins cracks or breaks when the rocket hits the ground at the end of flight.	Fin separates from the airframe.	Does not meet the NASA USLI protocol for reusability and deemed as an unsuccessful flight.	Inspect the surface of fins to ensure that there are no preexisting weaknesses before launch.
	1	3	3	Primary provides stability to the rocket and proper trajectory. These are located at the base of the rocket.	Epoxy fails	Improper application, mostly likely due to a lack of adhesion , improper mix of two step solution, or insufficient amount used.	Epoxy cracks or breaks when the rocket hits the ground at the end of flight.	Fin separates from the airframe.	Does not meet the NASA USLI protocol for reusability and deemed as an unsuccessful flight.	• Properly mix the epoxy and apply generously to the edges of the bulkhead • Allow ample time for curing of epoxy before launch.
Firing Pins	1	4	4	A small metal pin that punctures CO2 canisters when a controlled explosion like black powder or Pyrodex occurs.	Premature detonation	E-match is set off prematurely	CO2 canister is punctured and gas is expelled into the airframe	Airframe separates	The motor impulse remains in effect while the airframe separates, potentially causing a catastrophic failure and resulting in the loss of the rocket.	• Inspect the microcontroller and ensure that e-matches are utilizing the proper output ports. • Ensure that the flight computer is disabled to prevent premature puncture while assembling and to prevent possible injury to the team and surrounding persons
Motor	1	2	2	Delivers trust to the rocket.	Motor doesn't ignite.	Damage while transporting or in-transit while in package or manufacture defect.	Failure to ignite from the launch pad.	Lack of trust provided to the rocket	The launch is scrapped, and the motor must be inspected to determine the issue before continuing.	• Inspect the motor for any defects to the motor. • Team Mentor shall be present and in charge or motor inspection and loading.
	2	1	2	Delivers trust to the rocket.	Motor doesn't ignite.	Improper or faulty motor igniter installation.	Failure to ignite from the launch pad.	Lack of trust provided to the rocket	The launch is scrapped, and the motor must be inspected to determine the issue before continuing.	• Ensure that the igniter wire legs are not touching or crossing. • Ensure that the igniter has hit the top of the motor before completing the circuit.
	3	1	3	Delivers trust to the rocket.	Premature burnout	Increased wind resistance or unaccounted additional weight.	Motor burns out before the expected apogee simulated.	Rocket begins to fall prematurely.	Premature initiation of the recovery process with the onset of drogue deployment.	Verify that the simulation data shows a controlled descent after all components are weighted at the launch site.
Paracord	1	4	4	Connects the parachute to the bulkhead	Severing	Higher than expected jerk during chute deployment	Paracord fails	Chute connection to airframe is severed.	The rocket begins to speed up, and a catastrophic impact is imminent.	Testing of paracord strength is employed to ensure strength in the worst situations

Parachute	3	4	12	Ensures the safe descent of the rocket after apogee is reached. This is done by slowing the rocket down to a manageable ground hit velocity.	The Drogue parachute does not open	• Improper folding • Loss of connection between the airframe and the parachutes during deployment • Forgetting to attach the parachute system to the eye bolt of the bulkhead	• Failure to deploy parachute	• Rocket does not slow to a manageable velocity before the main parachute is deployed	Upon main deployment, Main parachute is destroyed, and a catastrophic impact is imminent.	• Inspect the parachute to look for preexisting weakness before launch. • Ensure that all components are assembled correctly before launching. • Verify that the simulation data shows a controlled descent after all components are weighted at the launch site.
	3	4	12	Ensures the safe descent of the rocket after apogee is reached. This is done by slowing the rocket down to a manageable ground hit velocity.	Main Parachute does not open	• Improper folding • Loss of connection between the airframe and the parachutes during deployment • Forgetting to attach the parachute system to the eye bolt of the bulkhead • Reef does not fully open at its expected altitude. • Reef lines are tangled	Failure to deploy parachute	Rocket does not slow to a manageable velocity before hitting the ground	The rocket is likely to incur irreversible damage and flight is deemed unsuccessful due to an inability to relaunch in its final state.	• Extensive testing of the reefing system before launch. • Inspect the parachute to look for preexisting weakness before launch. • Ensure that all components are assembled correctly before launching. • Verify that the simulation data shows a controlled descent after all components are weighted at the launch site.
	2	4	8	Ensures the safe descent of the rocket after apogee is reached. This is done by slowing the rocket down to a manageable ground hit velocity	Parachute integrity fails	• Shroud lines sever • Reefing patches tear from the canopy skirt • Canopy tears • Higher than expected jerk during chute deployment	• Parachute geometry fails	Rocket does not slow to a manageable velocity before hitting the ground	The rocket is likely to incur irreversible damage and flight is deemed unsuccessful due to an inability to relaunch in its final state.	
Screw Switch	1	1	1	Used to arm the rocket once secured to the launch pad and ready for flight.	Screw switch fails to arm the avionics bay and reefing electronics	• Loss of connectivity between the switch and the electrical components • The electrical circuit is stuck closed. • Frayed or severed wires	• Screw switch fails	• Lack of power in the avionics bay	Unable to launch	• Bring multiple switches to the launch site • Test the system with the key switch before launch day
Shear pins	1	4	4	Allow for selective separation of rocket airframe.	Premature separations	Higher than expected jerk during chute deployment or improper shear pin sizing for the size of the rocket.	Premature shearing	Premature separation between coupler and airframe components	The motor impulse remains in effect while the airframe separates, potentially causing a catastrophic failure and resulting in the loss of the rocket.	Verify that the simulation data shows a controlled descent after all components are weighted at the launch site.

Electrical										
Antenna (TeleTB)	2	2	6	Provides wireless connection to the rocket avionics bay	Failure to configure	Lack of connection between Avionics Bay.	Data transfer between the electronics on board and the antenna is nonexistent	Real time data is not recorded during flight	Lack of data required for comparing with the projected flight data from Openrocket simulation	• Test system before launch • Ensure connectivity before launch
Battery	2	3	6	Provides necessary power to effectively operate all electronic components in the payload vehicle.	Battery insufficient charge	• Negligence in charging before preparing for launch. • Cell Defects/Degradation	Battery doesn't have the potential expected	Lack of power needed to effectively use the electronic onboard the payload	Payload is unable to complete its mission.	• Charge batteries after reach round of usage. • Carry battery charger and portable battery pack to launch site • Bring extra batteries to launch site
	1	3	3	Provides necessary power to effectively operate all electronic components in the payload vehicle	Battery thermal runaway	• Over charging • Over-discharging • Cell Defects/Degradation • External Short Circuit • Overheating	Battery explodes	• Fire • Lack of power needed to effectively use electronic onboard the payload	• Payload is unable to complete its mission • VOCs are released in the immediate area • Potential to cause chemical burns from battery acid	• Disconnect from the charger before leaving and store in a secure place. • Disconnect battery from charger when full charged • Bring extra batteries to launch site
Electrical Circuit	3	3	9	Delivers power through the recovery system and	Electrical short	• Damaged or Worn Wiring • Loose connections • Moisture from the environment • Water reservoir leaks water onto microcontroller • Excessive current	Loss of power	Microcontroller is incapable of communicating with internal electrical components within the payload. Lack of power or communication between components to effectively use the electronic onboard the payload.	Payload is unable to complete its mission	• Inspect all wires before an events like storage, testing, and launch. • Inspect all terminal connections for loose wires or broken connections • Test water pump and reservoir for leaks or areas of concern
Flight computer (Telemetrum)	2	3	6	Responsible for deployment events of drogue and main parachutes by means of airframe separation	The flight computer fails to separate airframe and deploy parachutes	• Lack of power • Loose connection • Frayed or broke wires	Chutes do not deploy	Rocket does not slow to a manageable velocity before hitting the ground	The rocket is likely to incur irreversible damage, and flight is deemed unsuccessful due to an inability to relaunch in its final state.	• Inspect the flight computer for chips, missing components, and other preexisting weakness before launch. • Ensure that all components are assembled correctly before launching. • Ensure that battery is charged and connected to the flight computer • Ensure that the screw switch allows power to flow to flight computer once engaged.
Reefing system	3	3	9	Allows for the main Parachute to have a small surface area for a faster ground descent time and open to the full capacity of the parachute at a specified altitude.	Reefing electronic system fails to open the main parachute when specified	• Lack of power • Loose connection • Frayed or broke wires	The reefing system is not open main fully	Rocket is falling at a fast velocity than expected.	The rocket is likely to incur irreversible damage, and flight is deemed unsuccessful due to an inability to relaunch in its final state.	• Inspect all electrical components for chips, missing components, and other preexisting weaknesses before launch. • Ensure that all components are assembled correctly before launching. • Ensure that the battery is charged and connected to the flight computer • Ensure that the screw switch allows power to flow to the flight computer once engaged.

Software										
AltOS	1	2	2	Open-source software and compatible with firmware for hardware products from Altus Metrum, the AltOS source tree also includes AltosUI, which provides for flight monitoring, post-flight analysis, device configuration and firmware updating	Failure to configure	Lack of connection between Avionics Bay.	Data transfer between the electronics on board and the antenna is nonexistent	Real time data is not recorded during flight	Lack of data required for comparing with the projected flight data from Openrocket simulation	• Test system before launch • Ensure connectivity before launch
Arduino IDE	3	3	9	IDE used to code the reefing flight computer.	Failure to configure Unexpected deployment of the full main parachute	Bugs	Premature or late deployment of main parachute and reefing	Unexpected descent velocity and time-to-ground.	• The rocket is likely to incur irreversible damage if the velocity is too high, and flight is deemed unsuccessful due to an inability to relaunch in its final state. • Has the possibility of missing the 80 sec window for time to ground that is projected	• Extensive tests to ensure the reefing system is bug-free
Open rocket	3	2	6	Software that provides Simulation data capable of giving us proper altitude deployment based off many parameters like weight, wind velocity, site altitude, and more. This is used when confirming altitude for main and reefing deployments.	Inaccurate simulation	• Inaccurate weight of the rocket at the launch site • Unexpected wind speeds that are unaccounted for in the program	Premature or late deployment of main parachute and reefing	Unexpected descent velocity and time-to-ground	• The rocket is likely to incur irreversible damage if the velocity is too high, and flight is deemed unsuccessful due to an inability to relaunch in its final state. • Has the possibility of missing the 80 sec window for time to ground that is projected	• Compare data from launch with simulation to find error

Table 5-9 Recovery Structures and Components FMEA

4.5.2 Payload FMEA

FMEA Payload										
Item	P	S	Value	Use	Failure Mode	Failure Cause	Effects			Mitigations
							Component-Level	Subsystem-Level	System-Level	
Mechanical										
Linkage assembly	3	3	9	Used in delivering the auger head between multiple orientations during the sampling process.	Drill head jams before full deployment	Drill head clearance between its motion travel from its vertical to horizontal orientation and the ground is restricted due to geometry constrains within the airframe.	The drill head is unable to reach a perpendicular orientation to the ground	Drill cannot gather the ideal volume of soil in the correct orientation	Sample collection will take longer than project to harvest the adequate amount of soil	Test payload on multiple types of terrain to determine if clearance is an issue
	3	3	9	Used in delivering the auger head between multiple orientations during the sampling process.	Mechanical linkage binds during deployment	• Tight tolerances while manufacturing assembly components • Lack of cleaning or dirt and grime buildup while completing the soil extraction, sampling, and testing.	Hindered movement between links and joints	The assembly is unable to move under its own power	Incapable of completing the task and failure of mission.	Test linkage assembly under load to ensure that all components work smoothly
	2	3	6	Used in delivering the auger head between multiple orientations during the sampling process.	Linkage structural failure under load	The torque of the servo motor exceeds the strength of the linkage assembly.	• Cracking, splintering, or snapping links • Linkage joints sheer	• The assembly is unable to move	Incapable of completing the task and failure of mission.	Test motor mounts under highest simulated forces do verify structural integrate under load.
Motor mount	2	3	6	Mounts provide stability and rigid support of the motor while under load.	Motor mount failure (torque or flight loads	• The torque of the servo motor exceeds the strength of the motor mount. • Damages due to excessive vibration or turbulence during flight or at landing	• Motor mount breaks or cracks	• The motor mount is unable to support the weight of the motor. • Incapable of providing support to the motor under load. • Incapable of providing support to the motor under load.	• Erratic movement or large amounts of jerk between assembly and motor. • Possible incapacitation of assembly and motor cooperation	Test motor mounts under highest simulated forces do verify structural integrate under load.
Electrical										
Battery	2	3	6	Provides necessary power to effectively operate all electronic components in the payload vehicle.	Battery insufficient charge	Negligence while preparing for launch.	Battery doesn't have the potential expected.	Lack of power needed to effectively use the electronic onboard the payload	Payload is unable to complete its mission.	• Charge batteries after reach round of usage. • Carry battery charger and portable battery pack to launch site
	1	3	3	Provides necessary power to effectively operate all electronic components in the payload vehicle.	Battery thermal runaway	• Over charging • Over-discharging • Cell Defects/Degradation • External Short Circuit • Overheating	Battery explodes	• Fire • Lack of power needed to effectively use electronic onboard the payload	• Payload is unable to complete its mission • VOCs are released in the immediate area • Potential to cause chemical burns from battery acid	• Disconnect from the charger before leaving and store in a secure place. • Disconnect battery from charger when full charged
Electrical circuit	3	4	12	Provides power distribution and communication between all electrical components	Electrical short	• Damaged or Worn Wiring • Loose connections • Moisture from the environment • Water reservoir leaks water onto microcontroller • Excessive current	Loss of power	Microcontroller is incapable of communicating with internal electrical components within the payload. Lack of power or communication between components to effectively use the electronic onboard the payload.	Payload is unable to complete its mission	• Inspect all wires before an events like storage, testing, and launch. • Inspect all terminal connections for loose wires or broken connections • Test water pump and reservoir for leaks or areas of concern
DC motor	1	3	3	Utilized in the operation of vehicle movement by driving tracks and operates the auger head during soil collection.	DC motor failure	• Mechanical wear • Loose connections • Poor ventilation • Overloading • Environmental factors such as dirt, moisture • Vibration experienced during flight	• Overheating • Lack of movement	• Payload track is unable to operate • Drill head is unable to operate	Lack of movement needed to effectively use the electronic onboard the payload	• Inspect component for dirt and grime or everyday use wear. • Inspect all wires before an events like storage, testing, and launch. • Inspect all terminal connections for loose wires or broken connections
Servo Motor	1	3	3	Utilized in the movement of the linkage assembly	Servo motor failure	• Mechanical wear • Poor ventilation • Overloading • Environmental factors such as dirt, moisture • Vibration experienced during flight	• Overheating • Lack of movement	Linkage assembly is unable to operate	Lack of movement needed to effectively use the electronic onboard the payload	
Motor drivers	1	3	3	Electrical interface between the motor and the microcontroller and controls the speed, direction, and torque of motors.	Motor driver failure	• Mechanical wear • Poor ventilation • Overloading • Environmental factors such as dirt, moisture • Vibration experienced during flight	• Overheating • Lack of movement	• Payload track is unable to operate • Drill head is unable to operate • Linkage assembly is unable to operate	Lack of movement needed to effectively use the electronic onboard the payload	

Software										
Multi-Motor Communication	3	3	9	The variety of motors all have specific roles but work together to produce successful payload operation.	Motor synchronization failure	• Code discrepancy • Issues with communications between microcontrollers and motors • Inadequate current to operate all motors for the duration of the mission	• Unwanted independence between each motor.	• Communication between multiple servo and DC motors do not work in harmony.	• Erradic movement of the payload vehicle • Possible failure of the mission.	• Ensure proper sizing of the battery. • Ensure that the battery is fully charged • De-bug and troubleshoot code with payload testing
Data Interpretation	4	3	12	Data interpretation is needed to effectively utilize sensor data for procedure implementation which in code.	Incorrect sensor data interpretation	Human error when interpreting data received from sensors.	Inaccurate data constants or data analysis.	Hinders benchmark and sub-system testing	Inadequate amounts of sample material tested	De-bug and troubleshoot code with payload testing
Servo motor movement	3	3	9	Utilized in the movement of the linkage assembly	Servo motor range insufficient or excessive	• Code discrepancy in rotation range. • Inadequate torque for assembly movement. .	• The servo motor doesn't turn the full range expected • The servo has a larger range than expected	• Too little range in servo motor rotation prevents linkage from reaching its vertical orientation • Too much range in servo motor rotation causes linkage to go into a toggle position	• Sample collection will take longer than project to harvest the adequate amount of soil • Binding of assembly causing incapacitation and mission failure	De-bug and troubleshoot code and verify motor sizing with payload testing

Table 5-10 Payload FMEA

5 Project Plan

5.2 Testing

5.2.1 Testing Philosophy and Overview

Launch vehicle testing is conducted as an integrated verification effort intended to validate structural integrity, recovery system reliability, avionics functionality, and overall flight performance. Rather than treating testing as a set of isolated experiments, verification is performed through a combination of bench testing, ground demonstrations, inspection, analysis, and multiple subscale and full-scale flight tests.

Flight testing is treated as a system-level verification activity from which multiple test objectives are derived. Data collected from flight instrumentation, combined with post-flight inspection and analytical comparison, is used to assess performance and identify necessary design changes. The test program intentionally includes iterative flight testing, including a failed flight test followed by corrective design actions and subsequent verification flights.

5.2.2 Launch Vehicle Testing

Launch vehicle testing encompasses all verification activities related to structural components, recovery systems, avionics, and integrated flight behavior. Testing includes material-level verification of critical components, functional testing of deployment systems, arming and continuity verification, and flight-derived validation of performance metrics.

5.2.2.1 Launch Vehicle – Required Test Matrix

A. Material & Component Qualification (Pre-Flight)

Test ID	Test Name	Objective	Variable(s)	Method	Success Criteria	Status
LV-MAT-01	Inconel 718 Material Testing	Verify Material Properties are sufficient	Post-Processing, Load	Tension and Torsion Tests	Data gives us enough information to make technical design choices	Partially Complete
LV-MAT-02	Cord Material Test	Verify shock cord withstands deployment loads	Cord type, Load	Tension tests	Data gives us enough information to make technical design choices	Complete

B. Recovery & Avionics Functional Testing (Pre-Flight)

Test ID	Test Name	Objective	Variable(s)	Method	Success Criteria	Status
LV-DEP-01	Full-Scale Drogue Deployment Verification	Verify reliable drogue deployment	CO2 Canister sizing	Ground deployment Test	Clean separation	Incomplete
LV-DEP-02	Full-Scale Main Deployment Verification	Verify main parachute deployment	CO2 Canister sizing	Ground deployment Test	Clean Separation	Incomplete
LV-DEP-03	Reefing Line Cutter Actuation Test	Verify cutter reliability under load	Cutter charge mass	Bench / ground test	Successful cut under tension	Complete
LV-DEP-04	Sub-Scale Drogue Deployment Verification	Verify reliable drogue deployment	CO2 Canister sizing	Ground deployment test	Clean separation	Complete

LV-DEP-05	Sub-Scale Main Deployment Verification	Verify main parachute deployment	CO2 Canister sizing	Ground deployment test	Clean Separation	Complete
LV-AV-01	Avionics Arming & Continuity Test	Verify arming logic and wiring continuity	Arming state	Bench test & inspection	Correct arming indication, full continuity	Complete
LV-AV-02	Deployment Channel Output Verification	Verify altimeter firing channels	Channel configuration	Bench test	Correct output at programmed event	Planned
LV-AV-03	Battery Endurance Verification	Verify power margin for mission duration	Load profile	Bench test	≥150% mission duration	Planned

C. Subscale Flight Tests

Test ID	Test Name	Objective	Configuration	Method	Success Criteria	Status
LV-SFT-01	Flight Test #1 – Baseline Configuration	Identify integration & recovery failure modes	Baseline	Flight test	Data sufficient to identify failure modes	Complete
LV-SFT-02	Flight Test #2 – No Reefing Configuration	Validate recovery without reefing	No Reefing	Flight test	Stable flight & nominal recovery	Planned
LV-SFT-03	Flight Test #3 – Reefed Configuration	Validate full recovery system	Reefed	Flight test	Nominal staged recovery	Planned

D. Subscale Flight-Derived Verifications

Test ID	Verification	Derived From	Method	Success Criteria
LV-SFV-01	Static Stability Verification	LV-SFT-01 through LV-SFT-03	Flight data analysis	≥1.0 cal
LV-SFV-02	Dynamic Stability Observation	LV-SFT-01 through	Flight video and telemetry	No oscillation

		LV-SFT-03		
LV-SFV-03	Structural Deformation Inspection	LV-SFT-01 through LV-SFT-03	Inspection	No permanent deformation
LV-SFV-04	Coupler Compression Verification	LV-SFT-01 through LV-SFT-03	Inspection	No ovalization
LV-SFV-05	Drogue Descent Rate Verification	LV-SFT-01 through LV-SFT-03	Data analysis	Matches prediction within margin
LV-SFV-06	Main Descent Rate Verification	LV-SFT-01 through LV-SFT-03	Data analysis	Within limits
LV-SFV-07	Total Descent Time Verification	LV-SFT-01 through LV-SFT-03	Data analysis	Meets expected decent

E. Full-Scale Flight Tests

Test ID	Test Name	Objective	Configuration	Method	Success Criteria	Status
LV-FFT-01	Flight Test #1 – Baseline Configuration	Identify integration & recovery failure modes	Baseline	Flight test	Partial success acceptable	Planned
LV-FFT-02	Flight Test #2 – Final Configuration	Validate recovery and flight	Payload Onboard	Flight test	Stable flight & nominal recovery	Planned

F. Full-Scale Flight-Derived Verifications

Test ID	Verification	Derived From	Method	Success Criteria
LV-FFV-01	Static Stability Verification	LV-FFT-01 through LV-FFT-03	Flight data analysis	≥1.0 cal
LV-FFV-02	Dynamic Stability Observation	LV-FFT-01 through LV-FFT-03	Flight video and telemetry	No oscillation
LV-FFV-03	Structural Deformation Inspection	LV-FFT-01 through LV-FFT-03	Inspection	No permanent deformation

LV-FFV-04	Coupler Compression Verification	LV-FFT-01 through LV-FFT-03	Inspection	No ovalization
LV-FFV-05	Drogue Descent Rate Verification	LV-FFT-01 through LV-FFT-03	Data analysis	Matches prediction within margins
LV-FFV-06	Main Descent Rate Verification	LV-FFT-01 through LV-FFT-03	Data analysis	Within limits
LV-FFV-07	Total Descent Time Verification	LV-FFT-01 through LV-FFT-03	Data analysis	Meets handbook & expected decent

5.2.2.2 Launch Vehicle Test Descriptions & Justification (Include "how results of a test can cause any necessary changes")

This subsection describes the purpose, methodology, and justification for the launch vehicle tests presented in Section 6.1.2.1. Tests are organized to reflect the progression from material and component qualifications through functional ground testing, and ultimately to integrated system validation via flight testing. Each test is designed to verify a specific aspect of launch vehicle performance or integrity, and collectively they provide confidence that the vehicle will perform as intended under flight conditions.

A. <u>Material & Component Qualification (Pre-Flight)</u>

<u>LV-MAT-01: Inconel 718 Material Testing:</u>

Test Description & Methodology:

Inconel 718 test specimens are evaluated under tension and torsion loading to verify that the material can withstand expected deployment loads with sufficient margin. Stress–strain behavior is recorded to assess yield behavior and failure modes. Testing evaluates the effect of post-processing condition on material strength by comparing annealed and non–post-processed specimens. A total of twelve specimens are tested: three annealed and three non–post-processed specimens in tension, and three annealed and three non–post-processed specimens in torsion. Post-test inspection is performed to identify yielding or permanent deformation.

Justification:

Bulkheads serve as critical load paths during recovery events. This test is necessary to confirm adequate strength margin, validate analytical load assumptions, and assess the impact of post-processing on material performance prior to flight.

Design Impact / Resultant Changes:

Test results inform final bulkhead post-processing selection and confirm structural margin. If insufficient margin is observed, bulkhead thickness, heat treatment, or recovery configuration may be modified to reduce deployment loads.

LV-MAT-02: Cord Material Test:

Test Description & Methodology:

Shock cord materials are subjected to tension testing while varying cord type and applied load. Paramax, Kevlar, and 550 paracord are evaluated to characterize elongation behavior and failure modes. Stress–strain data is recorded to assess suitability for recovery system use.

Justification:

Shock cord failure represents a single-point failure mode for the recovery system. This test is necessary to ensure the selected material can withstand deployment loads without rupture or excessive elongation, which could result in recovery failure or structural damage.

Design Impact / Resultant Changes:

Test results are used to select the final shock cord material and determine appropriate harness configuration. If excessive elongation or damage is observed, cord material, harness length, or recovery configuration may be adjusted.

B. Recovery & Avionics Functional Testing (Pre-Flight)

LV-DEP-01: Full-Scale Drogue Deployment Verification:

Test Description & Methodology:

Full-scale ground deployment tests are conducted to verify reliable drogue parachute deployment using flight-representative hardware. Deployment energy is varied by adjusting CO_2 canister sizing. The recovery system is packed in a flight-like configuration, and separation behavior is observed during deployment.

Justification:

Reliable drogue deployment is required to ensure controlled descent and proper sequencing of recovery events. Ground testing reduces risk prior to flight by validating deployment methodology and energy requirements.

Design Impact / Resultant Changes:

Test results inform final CO_2 canister sizing and packing configuration. If incomplete deployment or excessive shock is observed, canister sizing or recovery configuration may be adjusted.

<u>LV-DEP-02: Full-Scale Main Deployment Verification:</u>

Test Description & Methodology:
Ground deployment tests are conducted to verify reliable main parachute deployment using full-scale, flight-representative hardware. CO_2 canister sizing is varied to ensure sufficient deployment energy without inducing excessive structural loads.

Justification:

Main parachute deployment reliability is critical to vehicle recovery and survivability. This test verifies that the main parachute deploys cleanly and without structural damage prior to flight.

Design Impact / Resultant Changes:

Results inform final canister sizing, parachute packing procedure, and recovery configuration. Observed anomalies may lead to adjustments in deployment energy or hardware configuration.

<u>LV-DEP-03: Reefing Line Cutter Actuation Test:</u>

Test Description & Methodology:
Bench and ground testing are conducted to verify reliable actuation of the reefing line cutter under representative tension loads. Cutter charge mass is varied to determine sufficient energy for consistent line severance.

Justification:

Bench and ground testing are conducted to verify reliable actuation of the reefing line cutter under representative tension loads. Cutter charge mass is varied to determine sufficient energy for consistent line severance.

Design Impact / Resultant Changes:

Test results validate cutter selection and charge sizing. If incomplete cuts or delayed actuation are observed, cutter configuration or charge mass will be adjusted.

<u>LV-DEP-04 & LV-DEP-05: Sub-Scale Drogue and Main Deployment Verification:</u>

Test Description & Methodology:
Sub-scale ground deployment tests are conducted to verify drogue and main parachute deployment prior to flight. Flight-representative packing configurations are used to assess separation behavior and deployment sequence.

Justification:

These tests validate deployment methodology and packing procedures prior to sub-scale flight testing, reducing the likelihood of deployment anomalies during flight.

Design Impact / Resultant Changes:

Results inform recovery packing procedures and confirm deployment logic prior to flight. Observed issues may result in packing or configuration changes.

LV-AV-01: Avionics Arming & Continuity Test:

Test Description & Methodology:
Bench testing and inspection are performed to verify correct arming logic and wiring continuity of recovery avionics. Arming indicators and continuity across deployment channels are verified prior to integration.

Justification:

Proper arming and electrical continuity are required to ensure reliable initiation of recovery events. This test reduces the risk of arming-related failures during flight.

Design Impact / Resultant Changes:

Test results validate wiring configuration and arming procedures. Any continuity or arming anomalies would result in wiring or switch configuration changes.

LV-AV-02: Deployment Channel Output Verification:

Test Description & Methodology:
Bench testing is conducted to verify that altimeter deployment channels fire correctly at programmed events under representative loads.

Justification:

This test verifies recovery sequencing logic and ensures deployment channels function as intended prior to flight.

Design Impact / Resultant Changes:

Results confirm altimeter configuration. Any incorrect firing behavior would require software or wiring adjustments.

LV-AV-03: Battery Endurance Verification:

Test Description & Methodology:
Avionics batteries are tested under representative load profiles to verify sufficient power margin for all mission phases, including pre-launch idle time, ascent, descent, and post-landing operations.

Justification:

Insufficient battery capacity represents a mission-ending failure. This test ensures avionics remain operational throughout the mission with adequate margin.

Design Impact / Resultant Changes:

Results confirm battery selection. If insufficient endurance is observed, battery capacity or power consumption will be adjusted.

C. Subscale Flight Tests

LV-SFT-01: Flight Test #1 – Baseline Configuration:

Test Description & Methodology:
The initial subscale flight test is conducted using the baseline vehicle configuration to evaluate integrated system behavior, recovery sequencing, and structural survivability under flight conditions.

Justification:

This test is necessary to identify integration and recovery failure modes that cannot be fully captured through ground testing or analysis.

Design Impact / Resultant Changes:

This flight identified recovery system anomalies that resulted in partial mission success. Post-flight inspection and data analysis were used to identify root causes, leading to significant recovery configuration and integration changes prior to subsequent flights.

LV-SFT-02: Flight Test #2 – No Reefing Configuration:

Test Description & Methodology:
A second subscale flight is conducted without the reefing mechanism to isolate recovery failure modes and validate baseline recovery performance.

Justification:

Removing the reefing system allows isolation of failure causes observed during the initial flight and provides confidence in the underlying recovery architecture.

Design Impact / Resultant Changes:

Results from this flight are used to validate recovery performance prior to re-introducing the reefing system.

LV-SFT-03: Flight Test #3 – Reefed Configuration:

Test Description & Methodology:
A third subscale flight is conducted with the full recovery system, including the reefing mechanism, to validate staged deployment and controlled descent performance.

Justification:

This test verifies the complete recovery system under flight conditions prior to full-scale implementation.

Design Impact / Resultant Changes:

Results confirm final recovery configuration or identify required refinements prior to full-scale flight.

D. Subscale Flight-Derived Verifications

Subscale flight testing provides system-level verification of multiple performance metrics derived from flight data, inspection, and analysis.

- **Static and Dynamic Stability Verifications (LV-SFV-01, LV-SFV-02)** validate aerodynamic stability assumptions and confirm predictable flight behavior.
- **Structural and Coupler Inspections (LV-SFV-03, LV-SFV-04)** verify that structural components survive flight and deployment loads without permanent deformation.
- **Descent Performance Verifications (LV-SFV-05 through LV-SFV-07)** confirm recovery system performance and validate analytical descent predictions.

E. Full-Scale Flight Tests and Verifications

Full-scale flight tests (LV-FFT-01 and LV-FFT-02) are planned to validate the as-built launch vehicle under representative mission configurations. Flight-derived verifications (LV-FFV-01 through LV-FFV-07) will be performed using flight data and post-flight inspection to confirm stability, structural survivability, and recovery performance at full scale.

5.2.2.3 Launch Vehicle Test Results & Design Impact

This section summarizes results from completed launch vehicle tests and describes how these results have informed design decisions. Tests that are currently in progress or planned are identified, along with the expected impact of their results on the launch vehicle design.

A. Material & Component Qualification (Pre-Flight)

LV-MAT-01: Inconel 718 Material Testing:

Test Status: In-Progress

Results Summary:

Initial testing has been completed for select specimens and has confirmed expected elastic behavior and repeatable stress–strain trends under the loads achievable with available equipment.

Published Inconel 718 properties (ASTM/manufacturer data) indicate material strength substantially above predicted deployment loads using conservative assumptions. Full tensile-to-yield characterization has not yet been completed due to equipment load limits. The team is coordinating access to higher-capacity test equipment to complete testing to yield and/or failure and to finalize post-processing selection based on measured strength and ductility.

Design Impact:

Preliminary results support the current bulkhead material selection and geometry. Final material allowable and post-processing selection will be confirmed after completion of higher-capacity tensile testing. If measured margin is lower than expected, mitigation actions include increasing bulkhead thickness, modifying heat treatment/post-processing, and/or reducing peak deployment loads through recovery configuration adjustments.

LV-MAT-02: Cord Material Test:

Test Status: Complete

Results Summary:

Tension testing was performed on candidate cord materials (Paramax, Kevlar, and 550 paracord) to assess load capability, elongation behavior, and failure mode. Paramax and Kevlar demonstrated acceptable strength and controlled elongation behavior relative to expected deployment loading. 550 paracord exhibited excessive elongation and/or undesirable failure behavior and was eliminated from flight use.

Design Impact:

Test results drove final selection of the recovery harness material and informed harness routing/length decisions to manage peak loads and prevent slack/entanglement. Any future adjustments will focus on harness length and attachment hardware rather than cord selection.

B. Recovery & Avionics Functional Testing (Pre-Flight)

LV-DEP-03: Reefing Line Cutter Actuation Test:

Test Status: Complete

Results Summary:

Bench/ground trials verified reliable cutter actuation and consistent line severance under representative tension. Charge mass testing established a configuration that provides sufficient cutting energy while minimizing unnecessary shock loading.

Design Impact:

Results validate cutter selection and charge sizing for integration. If future testing reveals delayed severance or incomplete cuts under higher tension, charge mass and/or mechanical routing of the reefing line will be adjusted.

<u>LV-DEP-04: Sub-Scale Drogue Deployment Verification</u>
Test Status: Complete

Results Summary:

Subscale drogue ground deployment testing demonstrated clean separation and consistent deployment in a flight-representative packing configuration.

Design Impact:

Validated packing approach and deployment energy baseline for subscale operations; packing procedure is now standardized for flight testing.

<u>LV-DEP-05: Sub-Scale Main Deployment Verification:</u>
Test Status: Complete

Results Summary:

Subscale main ground deployment testing demonstrated clean separation and repeatable deployment in a flight-representative packing configuration.

Design Impact:

Validated packing and retention approach for the main system; results support proceeding to subscale flight testing.

<u>LV-AV-01: Avionics Arming & Continuity Test:</u>
Test Status: Complete

Results Summary:

Bench testing confirmed correct arming indication and continuity across all critical circuits in the recovery avionics harness. No intermittent continuity issues were observed during repeated handling checks.

Design Impact:

Confirms current wiring layout and arming procedure. Any future changes will be limited to connector retention/strain relief if field integration reveals intermittent issues.

<u>LV-AV-02: Deployment Channel Output Verification:</u>
Test Status: Complete

Results Summary:

Deployment output verification were executed using flight-representative loads to confirm correct firing behavior at programmed events.

Design Impact:

This test gates final flight configuration of deployment channel assignments and event timing. Any incorrect outputs will be corrected via altimeter programming updates and/or wiring revisions.

LV-AV-03: Battery Endurance Verification:
Test Status: Complete

Results Summary:

Battery endurance testing will validate ≥150% mission duration under representative load profiles, including pre-launch idle, ascent/descent operation, and post-landing time.

Design Impact:

This test gates final battery selection and power budget closure. Insufficient margin will be mitigated via higher-capacity batteries, reduced duty cycle, or load reduction.

C. Subscale Flight Tests

LV-SFT-01: Flight Test #1 – Baseline Configuration:
Test Status: Complete

Results Summary:

Terra Parvis V1 achieved a successful boost phase and reached an apogee of 4474.5 ft, closely matching simulation predictions (4698 ft). The descent phase experienced consecutive failures including a non-firing drogue deployment event, an oversized main parachute selection, and bulkhead pull-through failure. Despite these failures, the avionics bay was recovered and flight data was extracted, enabling root-cause evaluation and corrective actions. Detailed flight performance comparisons and recovery failure analysis are provided in the Recovery section and associated subscale analysis subsection.

Design Impact:

Corrective actions for Terra Parvis V2 include: replacing PETG bulkheads with ¼” plywood, rebuilding the airframe and coupler as fiberglass, resizing the main parachute from 72” to 48”, and removing PETG structural dependence in the transition by producing the transition fully in fiberglass (using printed parts as molds only). Additional process improvements include implementing more rigorous pre-flight configuration control (e.g., print infill verification, avionics programming verification, and initiator continuity checks) to prevent integration oversights.

D. Subscale Flight-Derived Verifications

This section presents verification results derived from subscale flight testing. Verification is performed using flight telemetry, visual observation, and post-flight inspection. Data from completed subscale flights is used to validate stability, recovery performance, structural survivability, and avionics functionality. Data from LV-SFT-01 is included where applicable; however, anomalous recovery behavior from this flight is excluded from recovery performance verification conclusions.

1) Flight Profile & Stability Verification
   a. Flight Profile Verification
      i. Verification ID(s): LV-SFV-01
      ii. Data Source: Flight computer telemetry, pre-flight simulation outputs, launch conditions inputs, flight video.
      iii. Verification Objective: Verify ascent profile and major event timing (burnout, apogee) match simulation within acceptable variance and identify off-nominal signatures.

   Discussion:
   LV-SFT-01 reached apogee within ~5% of simulation prediction and showed closely matched ascent velocity/acceleration trends, indicating the ascent model and propulsion assumptions were reasonable. Descent profile was off-nominal due to recovery sequence failure; therefore, descent profile is not used to validate recovery modeling for LV-SFT-01.
   Verification Result: Partial Pass (ascent profile validated; descent recovery profile not valid for verification).
   Notes: Updated conclusions pending LV-SFT-02/03.

   b. Static Stability Verification
      i. Verification ID(s): LV-SFV-01
      ii. Data Source: Pre-flight stability analysis + flight behavior observation (video) + ascent telemetry signatures.
      iii. Verification Objective: Verify the vehicle remains statically stable (≥1.0 caliber) as evidenced by predictable, non-divergent ascent behavior.

   Discussion: LV-SFT-01 boost behavior was stable with no observed divergent instability signatures. Stability margin will be reconfirmed for LV-SFT-02/03 using the V2 fiberglass airframe and updated mass properties to ensure model inputs match as-built configuration.
   Verification Result: Provisional Pass based on observed stable ascent; final closure pending LV-SFT-02/03.

Notes: If instability is observed in later flights, mitigations include CG adjustment and fin alignment correction.

c. Dynamic Stability Observation
   i. Verification ID(s): LV-SFV-01
   ii. Data Source: Flight video + telemetry trend review.
   iii. Verification Objective: Verify absence of sustained oscillations during boost/coast (no divergent roll/pitch oscillation).

Discussion: LV-SFT-01 did not exhibit apparent divergent oscillations during ascent. Given high winds and stability of our rocket, the rocket turned into the wind. Flight video and telemetry from LV-SFT-02/03 will be used to confirm dynamic behavior under the updated V2 configuration.

Verification Result: Provisional Pass, pending additional flights.

Notes: Wind and rail departure conditions will be documented for comparison.

2) Recovery System Performance Verification
   a. Deployment Event Timing Verification
      i. Verification ID(s): LV-SFV-05, LV-SFV-06, LV-SFV-07
      ii. Data Source: Flight computer event logs + telemetry + recovery section analysis.
      iii. Verification Objective: Verify correct sequencing (apogee event → drogue; main at programmed altitude; reefing cut timing if applicable).

Discussion: LV-SFT-01 drogue deployment did not occur due to a non-fire at apogee; therefore, recovery event timing from this flight cannot be used to validate recovery performance. LV-SFT-02/03 will be used to verify event timing under flight conditions after completion of avionics output verification and updated configuration control procedures.

Verification Result: Fail for LV-SFT-01 (non-fire); Planned for LV-SFT-02/03.

Notes: Root cause was electrical/procedural in nature; mitigations include channel output checks, programming verification, and improved initiator continuity confirmation.

   b. Drogue Descent Rate Verification
      i. Verification ID(s): LV-SFV-05
      ii. Data Source: Telemetry-derived descent rate during drogue phase (LV-SFT-02/03).
      iii. Verification Objective: Validate drogue descent rate matches prediction within margin and does not exceed structural constraints.

Discussion:

Not verifiable from LV-SFT-01 due to missing drogue deployment. Will be verified using LV-SFT-02 flight data.

Verification Result: Not Verified (LV-SFT-01); Planned (LV-SFT-02).

Notes: Out-of-margin results will drive drogue sizing and/or deployment energy changes.

c. Main Descent Rate Verification
    i. Verification ID(s): LV-SFV-06
    ii. Data Source: Telemetry-derived descent rate during main phase (LV-SFT-02/03).
    iii. Verification Objective: Confirm main descent rate meets landing energy requirements and matches prediction within margin.

Discussion: LV-SFT-01 main deployment occurred at extreme velocity due to drogue non-fire and therefore does not represent nominal main descent performance. V2 main parachute has been resized to 48” specifically to meet landing energy constraints with improved descent profile.

Verification Result: Not Verified (LV-SFT-01); Planned (LV-SFT-02/03).

Notes: Overspeed deployment lessons learned are captured in Recovery section; main sizing corrected.

d. Total Descent Time Verification
    i. Verification ID(s): LV-SFV-07
    ii. Data Source: Apogee-to-landing telemetry duration (LV-SFT-02/03).
    iii. Verification Objective: Verify descent timeline complies with handbook constraints and team safety requirements.

Discussion: LV-SFT-01 descent timing is not representative due to recovery failure. LV-SFT-02/03 will be used for verification closure.

Verification Result: Not Verified (LV-SFT-01); Planned (LV-SFT-02/03).

3) Structural Survivability Verification

a. Structural Deformation Inspection
    i. Verification ID(s): LV-SFV-03
    ii. Data Source: Post-flight inspection + as-landed photos + recovered hardware.
    iii. Verification Objective: Verify airframe and components survive flight without permanent deformation that compromises functionality.

Discussion: Post-flight inspection showed Body Tubes 1 and 3 largely intact (with fin damage), avionics bay recovered, and Body Tube 2 transition failure consistent with high deployment load and PETG-based transition weakness. Bulkheads failed by pull-through at eyebolt washers due to brittle low-infill PETG, compounded by extreme main deployment conditions. These findings directly motivated V2 redesign toward fiberglass structural construction and plywood bulkheads.

Verification Result: Fail (V1 configuration), with corrective redesign implemented for V2.

Notes: Verification closure will occur after LV-SFT-02/03 demonstrate survivability under nominal recovery.

b. Coupler Compression Verification
    i. Verification ID(s): LV-SFV-04
    ii. Data Source: Post-flight inspection and fit checks of couplers/interfaces.
    iii. Verification Objective: Verify couplers and interfaces do not ovalize/crush or bind due to deployment loads.

Discussion: Due to abnormal recovery loads in LV-SFT-01, interface survivability is not representative of nominal operation. Coupler and transition manufacturing has been redesigned to fiberglass to reduce brittle fracture risk and improve interface strength.
Verification Result: Not Closed (LV-SFT-01 off-nominal); Planned closure with LV-SFT-02/03.
Notes: Fit and separation checks will be conducted before and after each SEARS flight.

4) Avionics Data Integrity Verification
    a. Avionics Data Retention Verification
        i. Verification ID(s): LV-INT-02 (or rename to LV-AV-04 if you want consistent naming)
        ii. Data Source: Downloaded flight logs from Telemetrum + file integrity review.
        iii. Verification Objective: Verify the flight computer captures and retains usable data for post-flight analysis.

Discussion: Despite recovery anomalies, the avionics bay was recovered and flight data was successfully extracted, enabling detailed comparison to simulation and diagnosis of failure modes. This confirms the team's ability to capture and use telemetry for iterative development.
Verification Result: Pass (LV-SFT-01).
Notes: Continued verification will be performed for LV-SFT-02/03 to confirm robustness across flights.

5) Avionics Data Integrity Verification

Verification Area	**Flights Used**	**Results**
Stability	LV-SFT-01 (boost phase), LV-SFT-02/03 planned	Provisional pass (V1 boost); closure pending
Recovery Performance	LV-SFT-02/03 planned	Not verified from V1 due to drogue non-fire
Structural Survivability	LV-SFT-01	Fail (V1 configuration); redesign implemented
Avionics Data Integrity	LV-SFT-01	Pass

5.2.3 Payload Testing

Payload testing verifies that the payload can (1) safely deploy from the launch vehicle after landing, (2) autonomously complete the soil collection and pH measurement experiment within the required mission timeline, and (3) log and retain data with correct timestamps. Verification is performed through bench testing, demonstrations, inspection, and (where applicable) flight-derived survivability verification. Payload testing includes interface and separation testing to ensure reliable deployment and compatibility with the launch vehicle.

5.2.3.1 Payload – Required Test Matrix

A. Payload Interface & Separation (Deployment)

Test ID	Test Name	Objective	Variable(s)	Method	Success Criteria	Status
PL-INT-01	Separation Mechanism Functional Test	Verify payload releases reliably	Alignment, surface condition	Bench demo	Release succeeds ≥3/3 trials	Planned
PL-INT-02	Separation Cycle Test	Verify reliability over repeated use	Cycle count	Bench demo	≥ 3/3 cycles with no degradation	Planned
PL-INT-03	Post-Landing Deployment Test	Verify deployment after landing-like conditions	Orientation/tilt, surface condition	Flight Test	Begins deployment within 45 secs and experiences no binding failure	Planned

B. Payload Electrical & Software Functional Testing

Test ID	Test Name	Objective	Variable(s)	Method	Success Criteria	Status
PL-ELEC-01	Power-On / Boot Verification	Verify payload reliably powers on and initializes	Battery voltage	Bench test	Successful startup ≥3/3 trials	Planned
PL-ELEC-02	Battery Endurance Test	Verify power margin for mission	Load profile	Bench test	≥150% mission duration	Planned

PL-ELEC-03	Connector / Continuity Verification	Verify wiring integrity	Handling state	Inspection /bench	No intermittent connections	Planned
PL-SW-01	Data Logging & Timestamp Verification	Verify data retention and timestamp accuracy	Sample rate	Bench test	100% logs retained with valid timestamps	Planned
PL-SW-02	Full Autonomous Sequence Dry Run	Verify complete mission sequence without errors	Config state	Bench demo	End-to-end run completes successfully	Planned
PL-SW-03	Fault Handling / Safe-State Test	Verify safe response to common faults	Sensor disconnect, low voltage	Bench test	Error logged; safe stop or recovery	Planned

C. Payload Mechanisms & Mobility

Test ID	Test Name	Objective	Variable(s)	Method	Success Criteria	Status
PL-MECH-01	Soil Acquisition Mechanism Functional Test	Verify collection mechanism operates reliably	Soil type	Bench demo	Collects soil without jamming	Planned
PL-MECH-02	Mobility / Actuation Test	Verify actuation/mobility performance	Surface condition	Ground demo	Traverses/acts reliably without stalling	Planned
PL-MECH-03	Containment Integrity Test	Verify sample retained	Orientation/tilt	Inspection /demo	No leakage or spillage	Planned

D. Experiment Verification (Soil + pH)

Test ID	Test Name	Objective	Variable(s)	Method	Success Criteria	Status
PL-EXP-01	Soil Volume Collection Verification	Verify minimum sample	Soil moisture/type	Bench measurement	≥50 mL (or team target)	Planned

		volume collected				
PL-EXP-02	pH Calibration Verification	Verify pH accuracy	Buffer solutions	Bench test	Error ≤ .1 pH	Planned
PL-EXP-03	pH Repeatability Test	Verify repeatability of readings	Repeated trials	Bench test	Std dev ≤ .5	Planned
PL-EXP-04	Full Mission Timeline Test	Verify experiment completion within time limit	Terrain/soil	Ground demo	Complete mission ≤15 minutes	Planned
PL-EXP-05	End-to-End Integrated Mission Demo	Verify full payload mission flow post-deploy	Configuration	Ground demo	Deploy → collect → measure → log successfully	Planned

E. Payload Flight Survivability

Test ID	Test Name	Objective	Variable(s)	Method	Success Criteria	Status
PL-SURV-01	Mechanical Survivability Verification	Verify payload mechanisms survive launch & landing	Flight configuration	Flight-derived + inspection	Separation + collection mechanisms functional post-flight	Planned
PL-SURV-02	Electrical/Software Survivability Verification	Verify payload electronics survive launch & landing	Flight configuration	Flight-derived + inspection	Power-on + logging functional post-flight	Planned

5.2.3.2 *Payload Test Descriptions & Justification*

This subsection describes the purpose, methodology, and justification for the payload tests listed in Section 6.1.3.1. Payload testing is structured to verify: (1) reliable deployment/separation after landing, (2) electrical and software reliability for autonomous operation and data capture, (3) mechanical performance for soil acquisition and containment, (4) experimental performance for pH measurement and sample volume requirements, and (5) survivability through launch, descent, and landing loads. Results from these tests are used to refine payload configuration prior to integrated flight testing and competition launch.

A. <u>Payload Interface & Separation (Deployment)</u>

PL-INT-01: Separation Mechanism Functional Test

Test Description & Methodology:
The payload separation mechanism is exercised in a bench environment using flight-representative interfaces to verify reliable release. Trials are conducted with varied alignment and surface conditions to simulate realistic tolerance and friction effects. Release behavior is recorded, including any binding, incomplete separation, or delayed actuation.

Justification:

Successful payload deployment is a prerequisite for completing the mission; failure to separate prevents soil acquisition and pH testing from initiating. This test verifies basic functionality prior to environmental and integrated testing.

Design Impact / Resultant Changes:

If release is inconsistent, modifications may include adjustment of interface tolerances, actuation force/torque margin, mounting alignment features, or mechanism redesign to reduce binding sensitivity.

PL-INT-02: Separation Cycle Test

Test Description & Methodology:
The separation mechanism is repeatedly actuated across a defined number of cycles to evaluate repeatability and identify wear-related degradation. Mechanism performance is assessed by observing release consistency and any change in required actuation force/torque over cycles.

Justification:

The separation system must remain reliable through repeated ground testing, integration cycles, and final flight operations. Cycle testing reduces the risk of wear-induced failure.

Design Impact / Resultant Changes:

If degradation is observed, design changes may include improved bearing surfaces, lubrication strategy, stronger actuation components, or revised geometry to reduce wear.

PL-INT-03: Post-Landing Deployment Test

Test Description & Methodology:
A ground demonstration is performed to simulate post-landing conditions. The payload and separation interface are placed at representative orientations/tilts and on varied surfaces to evaluate deployment performance under realistic landing conditions. The time from trigger/actuation to successful deployment is recorded.

Justification:

The payload must deploy after landing, where orientation and surface conditions are uncertain. This test validates that deployment is robust to these conditions.

Design Impact / Resultant Changes:

If deployment fails at certain orientations or surfaces, changes may include increasing actuation margin, adding guiding features, increasing clearance, or adding fail-safe release behavior.

B. Payload Electrical & Software Functional Testing

PL-ELEC-01: Power-On / Boot Verification

Test Description & Methodology:

The payload is powered on from flight-representative battery conditions and required to boot and enter the correct operational state. Multiple startup trials are performed to verify reliability across expected battery voltage ranges.

Justification:

Reliable initialization is required to execute the autonomous mission sequence. Boot failures represent a mission-ending condition.

Design Impact / Resultant Changes:

If boot reliability is insufficient, changes may include improved power regulation, revised startup sequencing, software watchdogs, or simplified initialization requirements.

PL-ELEC-02: Battery Endurance Test

Test Description & Methodology:

Battery endurance is tested under representative load profiles, including idle, actuation loads, sensing, and logging. Power consumption is measured and compared to battery capacity.

Justification:

The payload must operate reliably for the full mission duration with adequate margin. Insufficient endurance can cause mission failure or data loss.

Design Impact / Resultant Changes:

If endurance margin is insufficient, changes may include increased battery capacity, reduced power draw, revised duty cycles, or software power management.

PL-ELEC-03: Connector / Continuity Verification

Test Description & Methodology:

Continuity and connector integrity are verified through inspection and bench testing. Connectors are checked under handling conditions representative of integration and field operations to identify intermittent connections.

Justification:

Loose or intermittent connections are a common failure mode in launch environments and can lead to resets or incomplete mission execution.

Design Impact / Resultant Changes:

If intermittent behavior is detected, changes may include improved strain relief, connector retention features, cable routing revisions, or connector selection changes.

<u>PL-SW-01: Data Logging & Timestamp Verification</u>

Test Description & Methodology:

The payload logging system is tested to ensure all required variables are recorded and stored with valid timestamps. Logs are downloaded and reviewed to confirm completeness, formatting, and data retention.

Justification:

The mission requires scientifically meaningful data output, including timestamps. Data loss or incorrect timestamps reduce mission credibility and can constitute mission failure.

Design Impact / Resultant Changes:

If data integrity issues occur, changes may include revised logging formats, increased storage robustness, improved timing sources, or error-checking for file write operations.

<u>PL-SW-02: Full Autonomous Sequence Dry Run</u>

Test Description & Methodology:

The full autonomous mission sequence is executed in a bench environment from a simulated "landing" trigger. The sequence includes deployment, soil collection actions, pH measurement, and data logging.

Justification:

End-to-end testing verifies subsystem integration and ensures that individual working components function correctly as a complete system.

Design Impact / Resultant Changes:

If the sequence fails, changes may include revised state machine logic, additional sensor checks, retries/timeouts, or revised task ordering.

<u>PL-SW-03: Fault Handling / Safe-State Test</u>

Test Description & Methodology:

Fault scenarios such as sensor disconnection, low voltage, or actuator stall are introduced to verify that the payload responds safely and records the fault condition.
Justification:

Fault-handling capability increases mission robustness and ensures failures are diagnosable.

Design Impact / Resultant Changes:

If handling is insufficient, changes may include added software fault detection, watchdog timers, safe-state behavior, and more explicit error logging.

C. Payload Mechanisms & Mobility

PL-MECH-01: Soil Acquisition Mechanism Functional Test
Test Description & Methodology:
The soil acquisition mechanism is tested using representative soil types to verify that the mechanism can collect soil without jamming. Observations include collection effectiveness, actuator load behavior, and repeatability across trials.

Justification:

Mechanical soil acquisition is a core mission capability. This test verifies functional performance and identifies mechanical failure modes early.

Design Impact / Resultant Changes:

If jamming or poor collection occurs, changes may include geometry revisions, increased torque margin, improved clearing features, or revised collection strategy.

PL-MECH-02: Mobility / Actuation Test (if applicable)
Test Description & Methodology:
Payload mobility or actuation is tested on varied surface conditions to verify performance and margin against stalling.

Justification:

If mobility is required to reach a sampling location or orient for collection, reliable actuation is necessary for mission success.

Design Impact / Resultant Changes:

If insufficient performance is observed, changes may include improved traction, revised gearing, motor selection changes, or reduced mechanical loading.

PL-MECH-03: Containment Integrity Test

Test Description & Methodology:
Containment is verified through inspection and demonstrations where the payload is tilted and moved through representative orientations to confirm that collected soil remains retained.

Justification:

Sample containment ensures measurement reliability and prevents loss of sample during pH testing or data collection.

Design Impact / Resultant Changes:

If leakage occurs, changes may include improved seals, revised container geometry, or secondary retention features.

D. Experiment Verification (Soil + pH)

PL-EXP-01 Soil Volume Collection Verification

Test Description & Methodology:
Collected soil volume is measured across trials with varying soil moisture/type. Volume measurement is performed using a consistent method to verify compliance with minimum volume requirements.

Justification:

Minimum sample volume is required to satisfy mission constraints and ensure a valid pH measurement.

Design Impact / Resultant Changes:

If volume is insufficient, changes may include increased collection capacity, longer collection duration, improved mechanism efficiency, or revised sample routing.

PL-EXP-02: pH Calibration Verification

Test Description & Methodology:
The pH measurement system is calibrated and verified using known buffer solutions. Measured values are compared against expected values to quantify error.

Justification:

Calibration validation is required to ensure scientific validity of the pH measurement.

Design Impact / Resultant Changes:

If error exceeds tolerance, changes may include revised calibration procedure, sensor replacement, improved signal conditioning, or temperature compensation tuning.

PL-EXP-03: pH Repeatability Test

Test Description & Methodology:
Repeated measurements are performed on the same prepared sample to quantify variability and repeatability.

Justification:

Repeatability ensures that measured results reflect true sample properties rather than measurement noise.

Design Impact / Resultant Changes:

If variability is excessive, changes may include improved mixing, longer stabilization time, filtering/averaging, or revised measurement protocol.

PL-EXP-04: Full Mission Timeline Test

Test Description & Methodology:
A ground demonstration is conducted from simulated landing through completion of soil acquisition and pH measurement to verify the complete experiment is executed within the required time.

Justification:

The mission requires completion within the allowed time window; time overrun constitutes mission failure.

Design Impact / Resultant Changes:

If time exceeds limits, changes may include streamlined sequence steps, shorter stabilization windows, reduced motion paths, or improved actuator speed.

PL-EXP-05: End-to-End Integrated Mission Demo

Test Description & Methodology:
A complete mission demonstration is performed including deployment, collection, pH measurement, and data logging, using flight-representative configuration and procedures.

Justification:

This test verifies integration across mechanical, electrical, and software subsystems, and ensures readiness prior to flight survivability testing.

Design Impact / Resultant Changes:

Observed integration issues may drive changes across hardware interfaces, software sequencing, or experiment procedures.

E. Payload Flight Survivability

PL-SURV-01: Mechanical Survivability Verification

Test Description & Methodology:

Following a flight (subscale or full-scale as applicable), the payload is inspected and functionally checked to verify that separation and collection mechanisms remain operational post-flight.

Justification:

Launch, deployment, and landing loads can damage mechanisms or alter alignments. Survivability verification ensures the payload can operate after flight stresses.

Design Impact / Resultant Changes:

If damage or binding is observed, changes may include reinforcement, revised mounting, damping/isolation, or reduced exposure to shock events.

PL-SURV-02: Electrical/Software Survivability Verification

Test Description & Methodology:

Following flight, the payload is powered on and required to boot and log data to confirm electronics and software robustness.

Justification:

Electronics may reset, disconnect, or corrupt data due to vibration and shock. This test verifies functional survivability.

Design Impact / Resultant Changes:

If issues occur, changes may include connector retention improvements, strain relief, power regulation improvements, or software recovery features.

5.2.3.3 Payload Test Results & Design Impact

At the time of this report, payload testing is in the execution phase and no payload tests have been formally completed. The payload test plan (Section 6.1.3.1) has been defined to verify deployment/separation, electrical/software reliability, mechanism performance, experiment validity (soil volume and pH), and flight survivability. Test execution is scheduled prior to the Critical Design Review, and all results will be incorporated into this section once available.

Although results are not yet available, the following design decisions are explicitly gated on payload test outcomes:

- Separation mechanism design freeze is gated on PL-INT-01/02/03 demonstrating reliable deployment across alignment and orientation conditions.
- Battery selection and power budget finalization is gated on PL-ELEC-02 demonstrating ≥150% mission-duration margin under representative load.
- Autonomy sequence finalization is gated on PL-SW-02/03 end-to-end mission success and robust fault handling.
- Soil collection mechanism configuration is gated on PL-MECH-01 and PL-EXP-01 meeting the minimum sample volume requirement without jamming.
- pH measurement procedure and calibration method are gated on PL-EXP-02/03 meeting accuracy and repeatability targets.

Once testing is completed, this section will be updated to include measured results, pass/fail outcomes, and any resultant payload design changes.

5.2.4 Completed Testing Summary

Testing progress is summarized in Table 6-X. Completed tests provide verification evidence for design integrity and inform configuration decisions. Tests in progress and planned tests will be completed prior to the Critical Design Review or prior to final mission readiness, as applicable.

Category	**Completed**	**In Progress**	**Planned**
Launch Vehicle – Materials	LV-MAT-02	LV-MAT-01	—
Launch Vehicle – Recovery/Avionics	LV-DEP-03, LV-DEP-04, LV-DEP-05, LV-AV-01	—	LV-DEP-01/02 LV-AV-02/03
Launch Vehicle – Subscale Flight	LV-SFT-01	—	LV-SFT-02 LV-SFT-03
Payload – Interface/Separation	—	—	PL-INT-01/02/03
Payload – Electrical/Software	—	—	PL-ELEC-01/02/03 PL-SW-01/02/03
Payload – Mechanisms/Experiment	—	—	PL-MECH-01/02/03 PL-EXP-01–06

Completed testing to date has confirmed key design assumptions and identified improvement areas, particularly in recovery system and integration reliability. Remaining tests are scheduled to close verification gaps for full-scale readiness and to validate end-to-end payload mission performance within the required timeline. Results from all completed tests will be incorporated into the requirements compliance matrix in Section 6.2

5.3 Requirements Compliance

This section documents the verification plan for all requirements defined in Sections 1 through 5 of the NASA Student Launch Handbook. These handbook sections establish the mandatory system-level requirements governing launch vehicle design, recovery system performance, payload mission objectives, and safety constraints. Compliance with these requirements is demonstrated through a combination of test, analysis, demonstration, and inspection.

Each requirement is mapped to one or more verification activities defined in Section 6.1. Verification status is tracked as Planned, In Progress, or Complete. Verification evidence will be documented as testing and analysis are completed prior to Flight Readiness Review (FRR).

5.3.1 Handbook Requirements Verification Plan (Sections 1–5)

5.3.1.1 Launch Vehicle Requirements

Req ID	Requirement (Handbook Derived)	Verification Method	Verification Plan (Section 6.1)	Status	Evidence
HV-LV-01	Launch vehicle is statically stable (≥1.0 caliber)	Analysis + Test	LV-SFT-xx, LV-SFV-01	Complete	Flight data
HV-LV-02	Vehicle survives ascent and recovery loads	Test + Inspection	LV-SFT-xx, LV-SFV-03	Complete	Inspection
HV-LV-03	Full-scale vehicle flown prior to FRR	Test	LV-FFT-01 / 02	Planned	Ghantt Chart
HV-LV-04	Vehicle mass and	Inspection	As-built inspection	Planned	Ghantt Chart

	configuration match final design				

5.3.1.2 Recovery System Requirements

Req ID	Requirement (Handbook Derived)	Verification Method	Verification Plan (Section 6.1)	Status	Evidence
HV-REC-01	Dual-deploy recovery system implemented	Test	LV-DEP-01, 02, 03	Partial	Ground test
HV-REC-02	Drogue deploys at apogee	Test	LV-DEP-01, LV-SFT-xx	Partial	Telemetry
HV-REC-03	Main deploys at programmed altitude	Test	LV-DEP-02, LV-SFT-xx	Partial	Telemetry
HV-REC-04	Descent rates within safe limits	Analysis + Test	LV-SFV-05–07	Complete	Flight data
HV-REC-05	Recovery system causes no structural damage	Inspection	LV-SFV-03, 04	Complete	Inspection

5.3.1.3 Payload Mission Requirements

Req ID	Requirement (Handbook Derived)	Verification Method	Verification Plan (Section 6.1)	Status	Evidence
HV-PAY-01	Payload remains retained during ascent	Inspection	PL-INT-01	Planned	Flight Test

HV-PAY-02	Payload deploys after landing	Demonstration	PL-INT-03	Planned	Flight Test
HV-PAY-03	Payload operates autonomously	Test	PL-SW-02	Planned	Flight Test
HV-PAY-04	Payload completes mission ≤15 minutes	Test	PL-EXP-05	Planned	Ground & Flight Test
HV-PAY-05	Payload collects ≥50 mL soil	Test	PL-EXP-01	Planned	Ground & Flight Test
HV-PAY-06	Payload measures and logs pH	Test	PL-EXP-02, PL-SW-01	Planned	Ground & Flight Test
HV-PAY-07	Payload data retained post-mission	Test	PL-SW-01	Planned	Ground & Flight Test

5.3.1.4 Safety Requirements

Req ID	**Requirement (Handbook Derived)**	**Verification Method**	**Verification Plan**	**Status**	**Evidence**
HV-SAFE-01	Vehicle complies with safety constraints	Analysis	Hazard analysis, FMEA	Complete	Safety docs
HV-SAFE-02	Energetics properly identified and labeled	Inspection	Vehicle drawings	Complete	Drawings
HV-SAFE-03	Recovery system minimizes landing energy	Analysis + Test	Descent analysis, flight data	Complete	Calculations

5.3.2 Team-Derived Requirements Verification Plan

5.3.2.1 Launch Vehicle – Team-Derived Requirements

Team Req ID	Requirement	Verification Method	Verification Plan	Status
TDR-LV-01	Bulkheads maintain SF ≥ 5 vs deployment loads	Analysis + Test	LV-MAT-01	In-Progress
TDR-LV-02	Av-bay separation is reliable and repeatable	Demo/Test	LV-DEP-01/02 + LV-SFT-02/03	Planned
TDR-LV-03	Vehicle mass properties match simulation inputs	Inspection	LV-ACC-03 + analysis update	Planned

5.3.2.2 Recovery System – Team-Derived Requirements

Team Req ID	Requirement	Verification Method	Verification Plan	Status
TDR-REC-01	Reefing cut occurs reliably under load	Test	LV-DEP-03	Complete
TDR-REC-02	Packing process is repeatable with clean separation	Demo/Test	LV-DEP-01/02 + LV-DEP-04/05	Partial
TDR-REC-03	Harness configuration prevents entanglement	Demo/Test	LV-SFT-02/03	Planned

5.3.2.3 Payload Mission – Team-Derived Requirements

Team Req ID	Requirement	Verification Method	Verification Plan	Status
TDR-PAY-01	Payload boots successfully ≥3/3 attempts in flight config	Test	PL-ELEC-01	Planned

TDR-PAY-02	End-to-end mission completes without operator input	Demo/Test	PL-EXP-06	Planned
TDR-PAY-03	Data file is complete and readable post-mission	Test	PL-SW-01	Planned

5.3.3 Requirements Verification Status Summary

At the time of the Critical Design Review, launch vehicle structural, stability, and subscale recovery requirements have been partially or fully verified through analysis, ground testing, and subscale flight testing. Payload verification activities are planned and scheduled prior to FRR.

5.4 Budget

The budget shown below provides a comprehensive, line-item breakdown of all components required for the project. Items not listed are materials, tools, or components already available to the team through existing university or lab resources. Some costs are estimated or flexible due to available inventory and shared resources, but all major procurements and critical hardware are fully accounted for.

Terra Nova Project Budget

Item	Description	Quantity	Estimate Shipping & Handling	Price ea.	Total	Link	Status
Full Scale Airframe and Motor							
Nose Cone	Fiberglass Nosecone FNC-6.0	1	$40.00	$190.98	$230.98	(Click Here)	Not Ordered
6in ID Body Tube	6" fiberglass airframe 60in	1	$55.00	$279.81	$0.00	(Click Here)	Inventory
5in ID Body Tube	5" fiberglass air-frame 30in	2	$40.00	$129.67	$299.34	(Click Here)	Not Ordered
6" coupler	G12CT-6.0 12" lengths	1	$15.00	$71.00	$0.00	(Click Here)	Inventory
5" coupler	G12CT-5.0 12" lengths	3	$20.00	$62.57	$207.71	(Click Here)	Not Ordered
Motor Tube	BT-3.00	1	$10.00	$14.16	$24.16	(Click Here)	Not Ordered
Motor Retainer	RA75L - 75MM MOTOR RETAINER ASSEMBLY FOR LOC TUBES	1	$10.00	$57.00	$67.00	(Click Here)	Purchased
Fiberglass Sheet	1/4 INCH G10 FIBERGLASS SHEET 1 SQUARE FOOT for fins	4	$25.00	$46.00	$209.00	(Click Here)	Not Ordered
Rail guides	Large Airfoiled Rail Buttons (fits 1.5" Rail - 1515)	1	$10.00	$11.73	$21.73	(Click Here)	Purchased
Standoffs for the rail buttons	5/16" Rail Button Standoff	2	$10.00	$3.74	$17.48	(Click Here)	Not Ordered
Rocket Motor	RMS-75/5120 Reload Kit (1 Pack) - 12142P - Hazmat FedEx surcharge	3	$85.00	$376.99	$1,215.97	(Click Here)	Not Ordered
Adhesives	Epoxy fillers	1	$0.00	$100.00	$100.00	(Estimates)	Purchased
Paint / finishing supplies	Used for both rocket airframes	1	$0.00	$150.00	$150.00	(Estimates)	Not Ordered
Fasteners & hardware(bolts, screws, ect.)	Used for both rocket airframes	1	$0.00	$75.00	$75.00	(Estimates)	Purchased
3D printer Filiment	nter Filament, Excellent Toughness High Precision, Odorless Non-Tox	1	$0.00	$79.99	$0.00	(Click Here)	Inventory
Fiberglass Cloth	**#120-38 Standard E-Glass Fiberglass Cloth 38**	6	$15.00	$9.90	$74.40	(Click Here)	Purchased
						Total:	$2,692.77
Sub-Scale Airframe and Motor							
Large Diamter Body Tube	Phenolic Airframe Tubing - 3.900" ID	1	$5.00	$28.04	$33.04	(Click here)	Purchased
Small Diamter Body Tube	Phenolic Airframe Tubing - 3.002" ID	2	$5.00	$25.79	$56.58	(Click here)	Purchased
Large Coupler Tube	PML Phenolic Coupler Tube - 3.78" x 36"	1	$5.00	$33.65	$38.65	(Click here)	Purchased
Small Coupler Tube	PML Phenolic Coupler Tube - 2.88" x 36"	1	$5.00	$29.16	$34.16	(Click here)	Purchased
Fiberglass Sheet	1/8 INCH G10 FIBERGLASS SHEET 1 SQUARE FOOT	2	$15.00	$19.80	$0.00	(Click Here)	Design Change
Motor Retainer	Screw On Retainers for 54mm Tubes	1	$10.00	$44.08	$54.08	(Click Here)	Purchased
RMS	54/1280 Complete Motor Hardware Set (w/ Forward Seal Disc) - 5412M	1	$10.00	$232.99	$242.99	(Click here)	Purchased
MOTOR	RMS-54/1280 Reload Kit (1 Pack) - 108014 - - Hazmat FedEx surcharge	1	$65.00	$135.99	$200.99	(Click here)	Purchased
Rail Buttons	Standard Rail Button (fits 1" Rail - 1010)	1	$5.00	$5.58	$10.58	(Click here)	Purchased
Nose Cone	Plastic Nosecone PNC-3.9	1	$15.00	$24.95	$39.95	(Click here)	Purchased
Main Chute	48" Elliptical Parachute - 8.6lb at 20fps	1	$20.00	$161.21	$181.21	(Click here)	Purchased
Droge Chute	12" Elliptical Parachute - 0.5lb at 20fps	1	$20.00	$77.51	$97.51	(Click here)	Purchased
						Total:	$989.74
Recovery Systems							
Shock Cord	Paramax Shock Cord - 100 ft	1	$10.00	$35.00	$45.00	(Click Here)	Purchased
All-thread rods + washers + nuts	Connect bulkheads internally for payload & motor retention structure.	1	$0.00	$40.00	$40.00	(Estimates)	Purchased
Black Powder / E-matches	E-matches (80 pieces and 3ft)	1	$75.00	$66.40	$141.40	(Click Here)	Purchased
C02 Canisters		2	$25.00	$25.00	$75.00	(Estimates)	Purchased
Flight Compuer	Eggtimer Proton	2	$20.00	$80.00	$180.00	(Click Here)	Purchased
Sub Scale Apogee Drouge	12" Elliptical Parachute - 0.5lb at 20fps	1	$20.00	$77.51	$97.51	(Click Here)	Purchased
Sub Scale Main Chute	48" Elliptical Parachute - 8.6lb at 20fps	1	$20.00	$161.21	$181.21	(Click Here)	Purchased
Full Scale Apogee Drouge	15" Compact Elliptical Parachute - 0.8lb at 20fps	1	$25.00	$88.20	$113.20	(Click Here)	Not Ordered
Full Scale Main Chute	Iris Ultra 120" Compact Parachute - 79lb @ 20fps; 44lb @ 15fps	1	$25.00	$685.59	$710.59	(Click Here)	Not Ordered
Shear Pins	Small Nylon Shear Pins - 20 pack	2	$5.00	$4.63	$14.26	(Click Here)	Purchased
Full Scale CO2 Ejection System	The "Double Eagle" Kit	1	$35.00	$279.00	$314.00	(Click Here)	Not Ordered
Subscale CO2 Ejection System	Peregrine CO2 Ejection Device (8g, 12g)	1	$35.00	$239.99	$274.99	(Click Here)	Purchased
Battery packs		3	$5.00	$25.00	$80.00	(Estimates)	Purchased
Wiring, connectors, switches & small electronic	Electronic Sytems for Both Subscale and Full	1	$15.00	$75.00	$90.00	(Estimates)	Purchased
						Total:	$2,357.16
Payload System							
Micro SD	Micro SD Storage Device	2	$0.00	$10.98	$0.00	(Click Here)	Inventory
Ball Bearing	6 mm ID Flanged Ball Bearing (12mm OD, 4mm Thickness) - 2 Pack	4	$20.00	$3.39	$33.56	(Click Here)	Not Ordered
Servo	2000 Series Dual Mode Servo (25-2, Torque)	1	$0.00	36.99	$36.99	(Click Here)	Not Ordered
D-Shaft	6mm D-Shaft (Stainless Steel, 30mm Length)	2	$0.00	$2.79	$5.58	(Click Here)	Not Ordered
D-Shaft	6mm D-Shaft (Stainless Steel, 60mm Length)	2	$0.00	$3.39	$6.78	(Click Here)	Not Ordered
Timing Pulley	2mm Pitch GT2 Pinion Timing Pulley (8mm REX™ Bore, 20 Tooth)	1	$0.00	$7.99	$7.99	(Click Here)	Not Ordered
Gearmotor	19:1 Metal Gearmotor 37Dx52L mm 12V (Helical Pinion)	3	$5.00	$44.95	$139.85	(Click Here)	Not Ordered
Clamping Collar	0 Series Aluminum Clamping Collar (6mm ID x 19mm OD, 9mm Len	2	$0.00	$4.49	$8.98	(Click Here)	Not Ordered
Tank Tracks	n Wheel Part, RC Tank Tracks Full Scale Easy Installation for Replacer	1	$0.00	$36.79	$36.79	(Click Here)	Purchased
Arduino Nano	Arduino Nano [A000005] - Compact ATmega328P	3	$0.00	$0.00	$0.00	https://nam04.safelinks	Inventory
Lipo Battery	OVONIC 3S Lipo Battery 5200mAh 11.1V 80C RC	1	$0.00	$26.99	$26.99	https://nam04.safelinks	Not Ordered
VERTICAL PUMP	SUBMERSIBLE 3VDC VERTICAL PUMP	1		$2.95	$2.95	https://nam04.safelinks	Not Ordered
High-Power Metal Gear	Crossed, 1045 Carbon Steel, 1 Module, 13 Teeth, Right-Hand	4	$5.00	$17.31	$74.24	(Click Here)	Not Ordered
PCB Building		1	$15.00	$175.00	$190.00	(Estimates)	Not Ordered
Microcontrollers & electronic components	Anything extra we don't have in lab	1	$0.00	$100.00	$100.00	(Estimates)	Partially Purchased
Prototyping parts	Anything extra we don't have in lab	1	$0.00	$40.00	$40.00	(Estimates)	Partially Purchased
Soil Sampling Kit	PH Probe	1	$35.00	$170.00	$205.00	(Click Here)	Purchased
						Total:	$915.70

Transportation, Lodging, Community Outreach, & Misc.							
Campus Veihcle	Provided by Florida State University - Panama City	1	Not aplicable	$0.00	$0.00	Not Applicable	-----
Gas	Estamite For all Travel both community events and launches	1	Not aplicable	$400.00	$400.00	(Estimate)	-----
Lodging	Air B&B	1	Not aplicable	$2,000.00	$2,000.00	(Click Here)	-----
Shirts	Custom designs incorporationg University branding, team logo, and team sponsers	8	30	$38.00	$304.00	(Click Here)	Purchased
Food	Food for students for occasians like launch days and the Hudson Trip	1	Not aplicable	$600.00	$600.00	(Estimates)	-----
Miscellaneous Supplies	Anything outside of the currently stated budget	1	Not aplicable	$200.00	$200.00	(Estimates)	-----
3D printer Filiment	Creality PETG Filament 1.75 Black 4 kg	1	$0.00	$79.99	$79.99	(Click Here)	Partially Purchased
						Total:	$3,583.99

Fundraised			
Company	**Description**	**Given Funds**	**Contact**
Floirda State Universtiy Panama City	Seed funding through the Deans Office to participate in the NASA USLI Competition	$2,500.00	
Donations	Community Funding	$1,400.00	
Stem Funding	Actively coordinating with multiple companies to raise additional funds and increase the overall budget.	$0.00	
		$0.00	
		$0.00	
		$0.00	
NOTE: The variance is set to add 10% to help mitigate any unexpected price changes throughout the year.		Subtotal:	$9,713.90
		Variance:	$971.39
		Total:	$10,685.29
		Amount Spent:	$2,699.56
		Fundraised:	$3,900.00
		Remaining Funds:	$1,200.44

Figure 5-1 Line-Item Budget

5.5 Fundraising

5.5.1 Funding Plan

The Terra Nova Team has a comprehensive finance plan to support participation in the 2026 NASA University Student Launch Initiative.

The strategy utilizes institutional support, industry partnerships, and community engagement to secure funding for all project phases: design, fabrication, testing, competition participation, and STEM outreach.

The projected total budget is $10,685.29, as outlined in Table 6-1.

5.5.2 Funding Sources

5.5.2.1 University Support

- Florida State University – Panama City (FSU-PC):
  The Dean's Office has committed $2,000–$3,000 in seed funding to establish the project foundation.

- INSPIRE Program and their Industrial Partners:

The FSU Inspire Program, in collaboration with Astro America and Maritech Machining Inc., has offered in-kind manufacturing support including 3D-printed metal parts and heat treating at no cost to the team.

5.5.2.2 Industry Partnerships

The team is establishing sponsorships with local and regional STEM-oriented organizations through FSU-PC's Office of Advancement.
A tiered recognition system encourages partner engagement.

Sponsorship Tiers:

Tier	Contribution	Recognition
Garnet Sponsor	$500	Logo on team apparel; social media & event recognition
Gold Sponsor	$1,500+	Prominent logo placement; full recognition at events; company name on rocket internals

Target Partners:

- Tech Farms (STEM incubation hub)
- Booz Allen Hamilton (aerospace & defense)
- Maritech Machining Inc. (precision manufacturing)
- VSC Fire & Security (systems integration)
- Bay County Chamber of Commerce (community and business outreach)

5.5.3 Outreach and Fundraising Strategy

Phase Approach:

Phase	Status	Description
Phase 1 University Funding	Complete	Secured seed funding from Dean's Office.
Phase 2 Industry Engagement	In Progress	Contacting STEM partners via FSU-PC connections.

Phase 3 Community Outreach	Not Begun	Broaden outreach to local businesses emphasizing STEM education impact.

Community & STEM Engagement

- **Chamber of Commerce Networking:** Attend First Friday events and Business-After-Hours sessions for sponsor outreach.

- **Public Engagement:** Conduct community STEM launches, school presentations, and local science fairs to increase visibility and attract sponsors.

5.5.4 Contingency Planning and Risk Mitigation

5.5.4.1 Contingency Reserve

A 10% contingency reserve of $971.39 is allocated for unexpected expenses, including design modifications, supply delays, or cost fluctuations.

5.5.4.2 Risk Mitigation Tiers

Tier	Strategy	Purpose
Tier 1	Request additional university resources (departments, Student Government Council).	Immediate backup funding
Tier 2	Expand regional/industry sponsorship outreach.	Sustain project growth.
Tier 3	Conduct emergency fundraising or community events.	Cover shortfalls.
Tier 4	Leverage in-kind support (Maritech, Astro America).	Reduce direct costs.

5.5.5 In-Kind Contributions

The team receives substantial non-monetary support through:

- Astro America – 3D metal printing and prototype support.
- Maritech Machining – Fabrication and heat-treating services.

These contributions significantly reduce the team's direct expenditure and fabrication lead times.

5.5.6 Financial Oversight and Accountability

- Project Lead (Krieg Conrad) maintains primary financial oversight.
- Weekly budget reviews occur at team meetings, with monthly financial summaries documented in team minutes.
- Procurement Process:
    - All purchases follow FSU-PC purchasing policies.
    - Each expenditure requires approval by Project Lead and relevant subsystem lead.
    - Receipts and transaction records are maintained in the team's shared financial log.

5.5.7 Summary

The Terra Nova Team has developed a diversified and sustainable financial strategy that combines university support, industry partnerships, and community outreach.
With multiple funding tiers, in-kind contributions, and a 10% contingency reserve, the team is financially positioned for full mission execution — from subscale testing to launch week in Huntsville.

The Terra Nova Team is confident that this plan will meet the $10,753.57 project requirement and ensure successful participation in the NASA USLI 2026 competition.

5.6 Timeline

Our timeline is shown here, along with a Gantt chart that breaks it down into weekly objectives. In addition to working on the documentation deliverables, several system officers and their teams will focus on certain tasks. Depending on the availability of the local schools, we will continue our community participation on Fridays and weekends with the goal of visiting K–12 institutions at least once a month. We'll also go to university events like Tech Days and Discovery Days. To guarantee timeline compliance, the leadership team will convene once a week. We are currently meeting every Wednesday at 1pm. The Terra Nova Project's timetable and Gantt chart are shown below, starting after the submission of the PDR. By using both, we are able to get a visual depiction of our project status as well as a work list.

Week	Dates	Documentation & Analysis	Build, Fabrication & Testing	Outreach & Administrative
Week 16 **CDR Due** **Re-Flights**	Jan 5 – 11	**Submit CDR (Jan 7)**; prepare subscale data templates.	Subscale relaunch preparation; avionics integration. **Subscale Flight Tests (2 flights)** – Samson, AL.	Sponsor updates. Media capture and Outreach for launch.
Week 17	Jan 12 – 18	Analyze subscale data; document anomalies; update recovery configuration.	Begin **payload fabrication**; minor vehicle rework if required.	Logistics prep for test launches.
Decision Gate #1	**Evaluate subscale performance and recovery behavior to authorize full-scale build.**			
Week 18	Jan 19 – 25	Update FRR skeleton; update CAD to as-built.	Continue payload build; **full-scale recovery ground tests**.	Confirm spring outreach calendar.
Week 19	Jan 26 – Feb 1	Hazard analysis update; test procedure documentation.	Full-scale vehicle integration.	Sponsor engagement; finalize Huntsville travel.
Week 20	Feb 2 – 8	Draft **FRR verification sections**; launch procedures.	Final integration; parachute packing; continuity checks	Mentor review.
Decision Gate #2	**Verify integration, dual ejection tests, and safety sign-offs before proceeding to full-scale launch and FRR submission.**			
Week 21 **FRR Q&A**	Feb 9 – 15	Team photos (Feb 9); attend **FRR Q&A (Feb 11)**; integrate feedback.	Final readiness checks; buffer.	Plan community flight-test event.
Week 22 **Full-Scale Flight Test**	Feb 16 – 22	Draft Full-Scale Results templates ahead of test.	**Full-Scale Flight Attempt #1 (SEARS or Propel Science Foundation)** flight execution & inspection.	Host outreach at launch; capture media.

Week 23 Backup Flight	Feb 23 – Mar 1	Insert Full-Scale Results (if flown Feb 21); finalize FRR draft.	**Backup Launch Feb 28** (if needed); disassembly & inspection.	Outreach/media for backup launch.
Week 24	Mar 2 – 8	Proofread FRR; mentor safety review; compliance check. **Post 2nd Flight Evaluation**	**Full-Scale Flight Attempt #2 (Final Configuration).**	Prepare FRR outreach materials.
Week 25 FRR Due	Mar 9 – 15	**Submit FRR (Mar 9)**; debrief; prep Addendum placeholders.	Rebuild/reintegrate per NASA feedback; continue payload testing.	Finalize travel accommodations.
Week 26 Spring Break	Mar 16 – 22	Prepare FRR Addendum templates; update STEM documentation.	Touch-ups, paint, pack spares.	Light outreach; travel logistics.
Week 27	Mar 23 – 29	Draft Launch-Week logistics & outreach report updates.	Final rocket prep; test ejection/payload.	Confirm travel roles; safety docs.
Week 28	Mar 30 – Apr 5	Finalize Launch-Week docs; print checklists; plan PLAR structure.	Final bench & power-cycle tests.	Pre-travel outreach posts; community reminders.
Week 29 Addendum (if req.)	Apr 6 – 12	Submit FRR Addendum (if requested); final proofing.	Final maintenance & packing for transport.	Confirm lodging & transport.
Week 30 Launch Week Q&A	Apr 13 – 19	Attend **Launch Week Q&A (Apr 15)**; finalize checklists; compliance verification.	Final readiness & continuity tests; crate packing.	Final travel docs; emergency contacts.
Week 31 Competition Launch Week (Huntsville)	Apr 20 – 26	**Apr 20–26 Detailed Ops:** **Mon:** Checklist print & travel prep. Tue: Transport arrival & inventory. **Wed:** RSO inspection & manifest upload. **Thu:** Full assembly & RSO approval. **Fri:** Internal rehearsal & final checks.	Field operations, launch execution, and post-flight inspection.	On-site outreach & sponsor engagement; attend closing ceremonies.

		Sat Apr 25: Competition Launch Day (Flight & retrieval). Sun: Post-flight data backup & teardown.		
Week 32	Apr 27 – May 3	Draft PLAR; insert flight & payload analysis; lessons learned.	Post-flight teardown; anomaly analysis.	Publish outreach report; send sponsor thanks.
Week 33 — PLAR Due	May 4 – 10	**Submit PLAR (May 10)**; archive documentation & raw data.	—	Final outreach report; budget closeout; mentor debrief.

Table 5-2 Project Timeline

Full-scale flight attempts are planned in coordination with either the Southeast Association of Rocketry (SEARS) under the SEARS Treasurer John Hansel or the Propel Science Foundation, both of which have confirmed launch availability within the project timeline.

The FRR schedule is structured to allow incorporation of subscale relaunch data and at least one full-scale flight prior to FRR submission, with a second full-scale opportunity available for final configuration validation.

The Terra Nova Project - Gantt Chart

Week Highligh 16

Category	ACTIVITY	PLAN START	PLAN DURATION	ACTUAL START	ACTUAL DURATION	PERCENT COMPLETE
PDR Development						
	Draft PDR Outline & Assign Content	1	1	1	1	100%
	Write Subsystem Sections:	2	2	2	2	100%
	Draft Hazard Analysis & FMEA:	2	2	2	2	100%
	Mission Performance Sims	3	1	3	1	100%
	Format & Peer Review PDR:	4	2	4	2	100%
	PDR Mock Presentation	6	1	6	1	100%
	Submit PDR Deliverables:	7	1	7	1	100%
Material Testing						
	Prepare Inconel Specimens:	9	1	9	1	50%
	Conduct Material Tests:	10	4	10	4	50%
Subscale Build & Test						
	Subscale Design Finalization:	1	2	1	2	100%
	Subscale Fabrication:	8	5	8	6	100%
	Subscale Launch:	12	0	12	5	33%
	Analyze Subscale Flight Data:	13	1	13	4	33%
CDR Development						
	CDR Draft:	8	2	8	2	100%
	Integrate Subscale Lessons:	13	2	13	2	100%
	Finalize CDR Report & Slides:	15	2	15	2	100%
	Submit CDR Deliverables:	16	1	16	1	100%
Full-Scale Build & Test						
	Begin Full-Scale Fabrication:	16	3	16	3	0%
	Ground Ejection Tests:	18	2	18	2	0%
	Payload Vibration Tests:	19	2	19	2	0%
	Final Integration & Assembly:	20	2	20	2	0%
	Full-Scale Launch (VDF):	22	3	22	3	0%
FRR Development						
	Draft FRR Report & Slides:	22	3	18	7	0%
	Submit FRR Deliverables:	25	0	25	0	0%
Launch Week						
	Prepare Logistics & Travel:	26	2	26	2	0%
	Competition Launch:	31	0	31	0	0%
Post Launch						
	Draft PLAR & Lessons Learned:	32	2	32	2	0%
	Submit PLAR:	33	0	33	0	0%
Ongoing Activities						
	STEM Outreach Visits:	1	30	1	30	55%
	Sponsor & Budget Mgmt:	1	33	1	33	50%
	Safety & Risk Updates:	2	30	2	30	50%

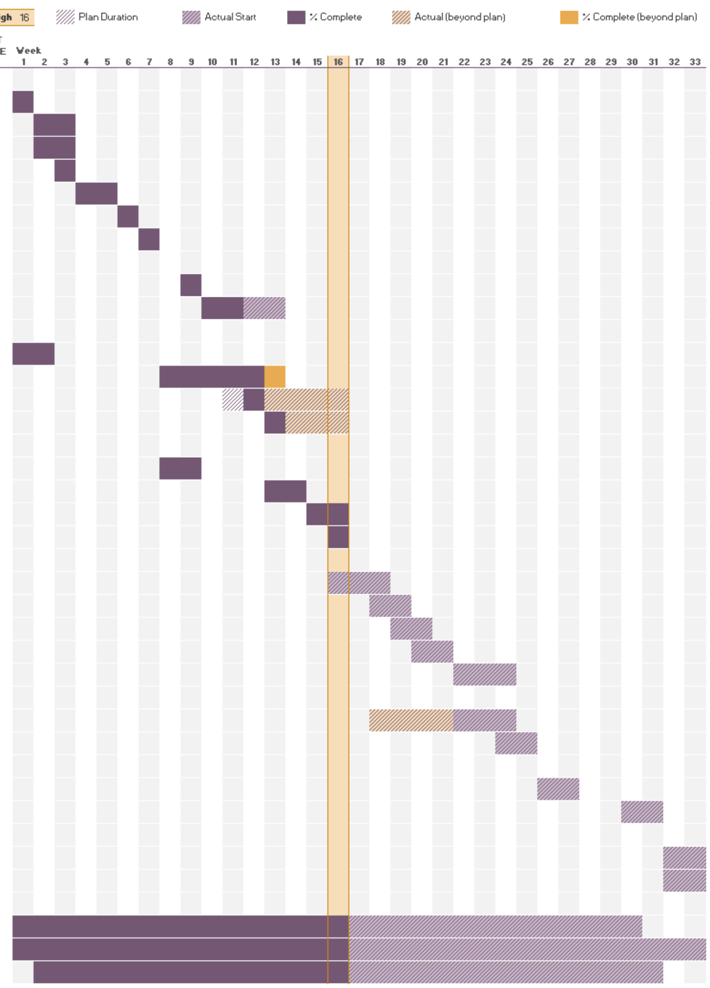


Table 5-3 Gantt Chart

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