Tacho Lycos

2026 NASA Student Launch

Flight Readiness Review Addendum

High-Powered Rocketry Club at NC State University
1840 Entrepreneur Drive
Raleigh, NC 27606

April 6, 2026

Common Abbreviations and Nomenclature

AIAA = American Institute of Aeronautics and Astronautics
APCP = Ammonium Perchlorate Composite Propellant
AV = Avionics
AVAB = Avionics and Air Brakes Bay
CAD = Computer Aided Design
CATO = Catastrophe at Take Off
CDR = Critical Design Review
CFD = Computational Fluid Dynamics
CG = Center of Gravity
CNC = Computer Numerical Control
CP = Center of Pressure
ECE = Electrical and Computer Engineering
ETF = Educational and Technology Fee
EYE = Engineer Your Experience
FAA = Federal Aviation Administration
FMEA = Failure Modes and Effects Analysis
FRR = Flight Readiness Review
GrAVE = Ground Activated Vehicle Ejector
GPS = Global Positioning System
GUI = Graphical User Interface
HAUS = Habitat for Agricultural Utilization Study
HPRC = High-Powered Rocketry Club
ID = Inner Diameter
IMU = Inertial Measurement Unit
INS = Inertial Navigation system
LED = Light Emitting Diode
LiPo = Lithium Polymer
LRR = Launch Readiness Review
LS = Likelihood Severity
MAE = Mechanical & Aerospace Engineering
NAR = National Association of Rocketry
NASA = National Aeronautics and Space Administration
NCSG = North Carolina Space Grant
NCSU = North Carolina State University
NFPA = National Fire Protection Association
NPK = Nitrogen, Phosphorus, and Potassium
OD = Outer Diameter
PCB = Printed Circuit Board
PDR = Preliminary Design Review
PDF = Payload Demonstration Flight
PEM = Penn and Manufacturing Corp.
PETG = Polyethylene Terephthalate Glycol
PLA = Polylactic Acid
PLAR = Post-Launch Assessment Review
PPE = Personal Protective Equipment
RSO = Range Safety Officer
RVM = Requirement Verification Matrices
SDS = Safety Data Sheets
SGA = Student Government Association
SL = Student Launch
STEM = Science, Technology, Engineering, and Mathematics
TRA = Tripoli Rocketry Association
VDF = Vehicle Demonstration Flight
ZOMBIE = Z-axis Orienting Mechatronic Botanical Investigative Excavator

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1 Summary of FRR Addendum Report

1.1 Team Summary

Table 1.1: Team Information

Information Required	Details
Team Name	Tacho Lycos
Mailing Address	1840 Entrepreneur Drive, Raleigh, NC 27606
Team Lead	Elizabeth Bruner
Team Lead Email Address	eabruner@ncsu.edu
Team Email Address	rocketry-org@ncsu.edu

1.2 Purpose of Flight

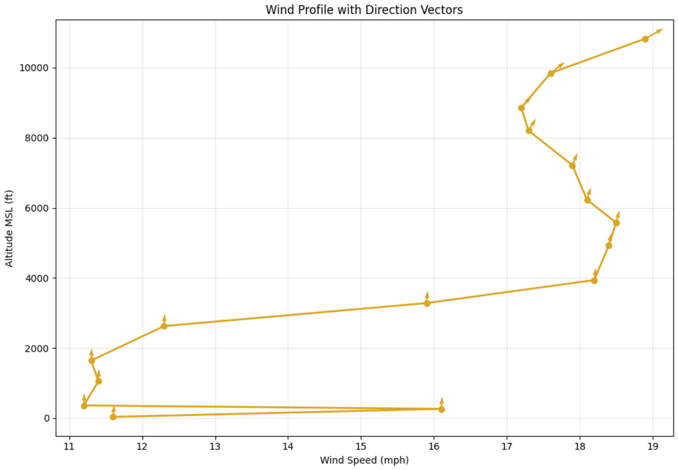
The purpose of the flight conducted was to fufill the requirements for both the Payload Demonstration Flight and the Vehicle Demonstration Re-flight.

1.3 Flight Summary Information

1.3.1 Launch conditions

The Vehicle Demonstration Flight and Payload Demonstration Flight, VDF and PDF, respectively, were completed on April 4th, 2026, in Monterey, VA. The range has a plethora of trees within a 500 ft radius of the launch location. Apogee during launch was kept low to avoid damage to the parachutes and to ensure nominal operation of ZOMBIE and GrAVE.

The vehicle was launched from a 15 ft. 1515 rail, which was verified to be at a 1-degree cant angle away from the assembly area. The predicted rail exit velocity was simulated to be 84 ft/s for Launch Vehicle stability upon exit from the 15 ft. rail. An AeroTech FirstFire high-power igniter was utilized for motor ignition and initiated via a Wilson F/X wireless electronic launch pad controller with a 12V power supply.



(a) Wind Profile for launch day

Parameter	Value
Average Wind Speed	12.3 (mph)
Ground Temperature	75.20 °
Ground Pressure	29.91 (inHg)
Heading	90°
Launch Rail Cant.	1°

(b) VDF Launch Day Conditions

1.3.2 Motor flown

The motor flown in the vehicle was the AeroTech L1390G.

1.3.3 Ballast and Stability

The Launch Vehicle was flown with no ballast. The stability of the Launch Vehicle in its flight-ready configuration was found to be 3.30 calipers.

1.3.4 Final Payload Flown

The payload flown consisted of the ZOMBIE lander and the GrAVE deployment system. Both parts were flown in the final flight configuration, and the code used for retention is the code that will be used at the competition. ZOMBIE and GrAVE used the same Finite State Machine to detect launch and landing, taking in data from an Inertial Navigation System. When landing was detected, GrAVE released its electronic latch and activated a leadscrew pusher plate. This ejected ZOMBIE from the nosecone.

1.3.5 Air Brakes

The Air Brake system performed as designed, reducing apogee by 922 (ft). The fins deployed and safely retracted during flight. No damage, warping, or shearing was observed on the fins upon recovery of the vehicle. The Air Brakes apogee prediction algorithm undershot the target apogee set in code by 355 (ft). This was primarily due to bias in the Magnetometer and IMU data accumulating during the integration strategy for apogee prediction. The bias accumulated into a large sensor drift, which kept on feeding the apogee prediction algorithm slightly higher pressure, orientation, and acceleration values than expected. This resulted in a 65 (ft) apogee discrepancy between the Silicdyne Fluctus apogee and the Air Brakes system apogee.

1.3.6 Altitude Summary

The target apogee for this launch was 4000 (ft). The Launch vehicle reached an apogee of 3645 (ft) with a time to apogee of 15.3 (s) and total flight time of 69 (s). The simulated apogee without Air Brakes deployment was 4567 (ft). Simulations during VDF garnered an apogee of 3627 (ft), time to apogee of 13.8 (s), and total flight time of 72 seconds.

1.3.7 Off-nominal Events

Due to off-nominal deployment of the main parachute, the parachute was not able to fully inflate, causing for an increase in the expected kinetic energy upon landing.

Due to off-nominal sensor drift, a 65 ft apogee discrepancy between the Silicdyne Fluctus and Air Brakes systems was caused by accumulated magnetometer and IMU bias, and fed the apogee prediction algorithm inflated pressure, orientation, and acceleration values.

2 Changes Made Since FRR

2.1 Changes Made To Vehicle Criteria

Table 2.1: Changes Made to Vehicle

Change Description	Justification	Affected Subsystem(s)
Main parachute switched from Fruity Chutes 96 (in) Iris Ultra to SkyAngle CERT-3 XL	Damage to original parachute	Recovery
Shear pin size changed from 4-40 to 6-32	Calculations revealed potential of vehicle separation upon ascent	Recovery
Drogue primary and secondary ejection charges changed from 3.5 (g) and 4.5 (g) to 4.5 (g) and 5.5 (g), respectively.	Increased shear pin size	Recovery
Main primary and secondary ejection charges changed from 3.8 (g) and 4.8 (g) to 7 (g) and 8 (g), respectively.	Increased shear pin size	Recovery

2.2 Changes Made to Payload Criteria

Table 2.2: Changes Made to Payload

Change Description	Justification	Affected Subsystem(s)
Reprint 3D printed payload components from PETG instead of PLA	Increased component heat tolerance needed	Payload
ZOMBIE leg thickness increased from 0.25 (in) to 0.375 (in)	Increases the stiffness of the legs for deployment	Payload
ZOMBIE leg deployment motor switched from lead screw stepper motor to servo with gears	Increased the force on the leg deployment collar to enable leg deployment from horizontal	Payload
Payload total mass changed from 7.859 (lbm) to 8.201 (lbm)	Mass increased due to the changes above	Payload

3 Payload Demonstration Flight

3.1 Mission Overview

The objective of the Habitat for Agricultural Utilization Study, or HAUS, is to secure the STEMnauts and to collect soil measurements. The HAUS must have an atmosphere-isolated compartment that retains the STEMnauts for the entirety of the flight. Additionally, the HAUS will collect and retain a 50 (mL) sample of soil within 15 minutes of landing. This soil will be tested for its pH level, electrical conductivity, and Nitrate-Nitrogen content.

The HAUS consists of two parts: ZOMBIE and GrAVE. ZOMBIE, or the Z-axis Orienting Mechatronic Botanical Investigative Extractor, is a self-righting lander that deploys after landing. GrAVE, or the Ground Activated Vehicle Ejector, is the mechanism that expels ZOMBIE from the launch vehicle. The soil sample will be collected using an auger drill located at the bottom of ZOMBIE. To ensure this drill is pointed at the ground, ZOMBIE will deploy from the launch vehicle after landing using GrAVE. ZOMBIE will then self-right by deploying its legs. The soil collected by the auger will be deposited into a container within the payload that contains the soil sampler. The soil sensor will begin taking readings after a set time period and will record these readings using a Raspberry Pi.

3.1.1 Retention System

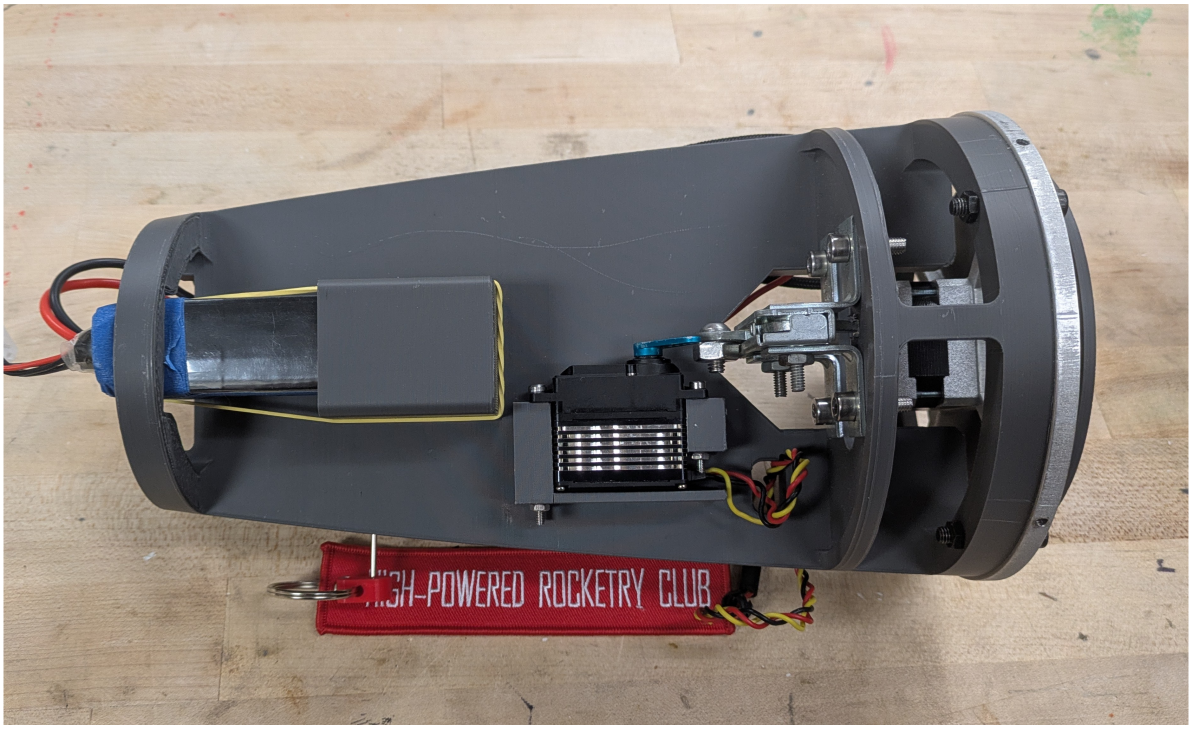


Figure 3.1: GrAVE with the aluminum mount and latch mechanism visible.

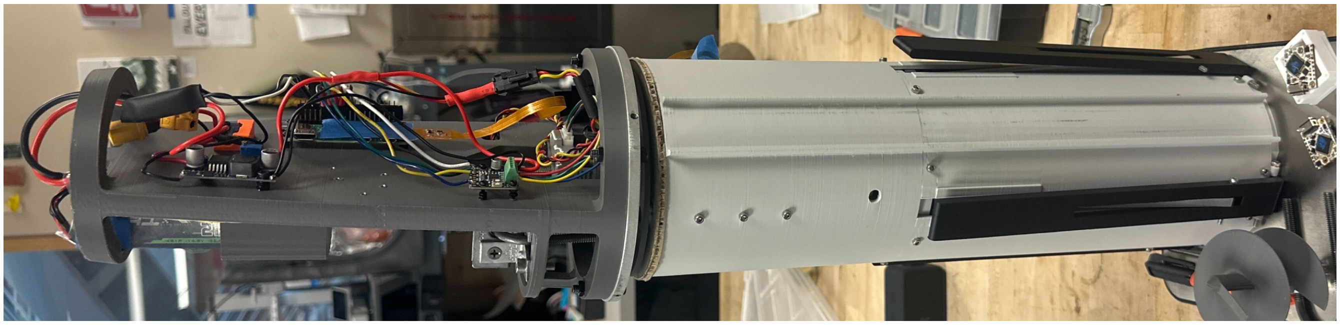


Figure 3.2: ZOMBIE retained by GrAVE's latch.

GrAVE is the payload's retention system. It is mounted inside the nosecone by four #6-32 screws set through the nosecone airframe into GrAVE's aluminum mount. This mount is attached to a 3D printed sled by four M5 screws. The sled has mounting points for a servo and mechanical latch, which holds ZOMBIE's upper U-bolt during flight to retain ZOMBIE. The latch can only be triggered by the rotation of the servo arm, which means that ZOMBIE will be retained by the latch until landing is detected and the checks are passed to initiate the servo. ZOMBIE is held in place laterally and rotationally by the two carbon fiber guide rails epoxied into the nosecone coupler.

3.1.2 Landing Detection

Landing detection is done by the Finite State Machine program with data from the Inertial Navigation System (INS). The state machine code is shared by both ZOMBIE and GrAVE, and the systems diverge after landing. When the program initiates, it defaults into Standby state. Launch is detected when the total measured acceleration exceeds 50 (m/s^2). There are two systems that are used to detect landing during launch. The first is an active system that checks if the detected pressure altitude is under 5 (m) and that acceleration variation is measured at 1 ± 0.03 (G) for the past 5 (s). This is only designed to check after 90 (s), the rough expected time of flight, have passed since launch. The timer was included to reduce the chance of activation before landing by checking only around when the vehicle is expected to land. Factors such as the main parachute deploying at apogee or windy conditions dragging the vehicle after landing could prevent the system from detecting landing. If the first system is not triggered within 345 (s) of launch, the state machine is moved to the landed state anyways. This option allows for a fallback if landing is not detected, waiting the worst-case descent time of main at apogee. After this, GrAVE begins the ZOMBIE deployment process. ZOMBIE waits for this process to complete using a timer before beginning its own deployment.

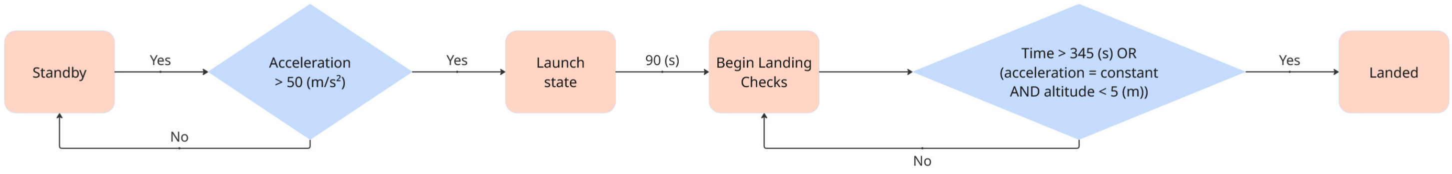


Figure 3.3: Landing detection diagram.

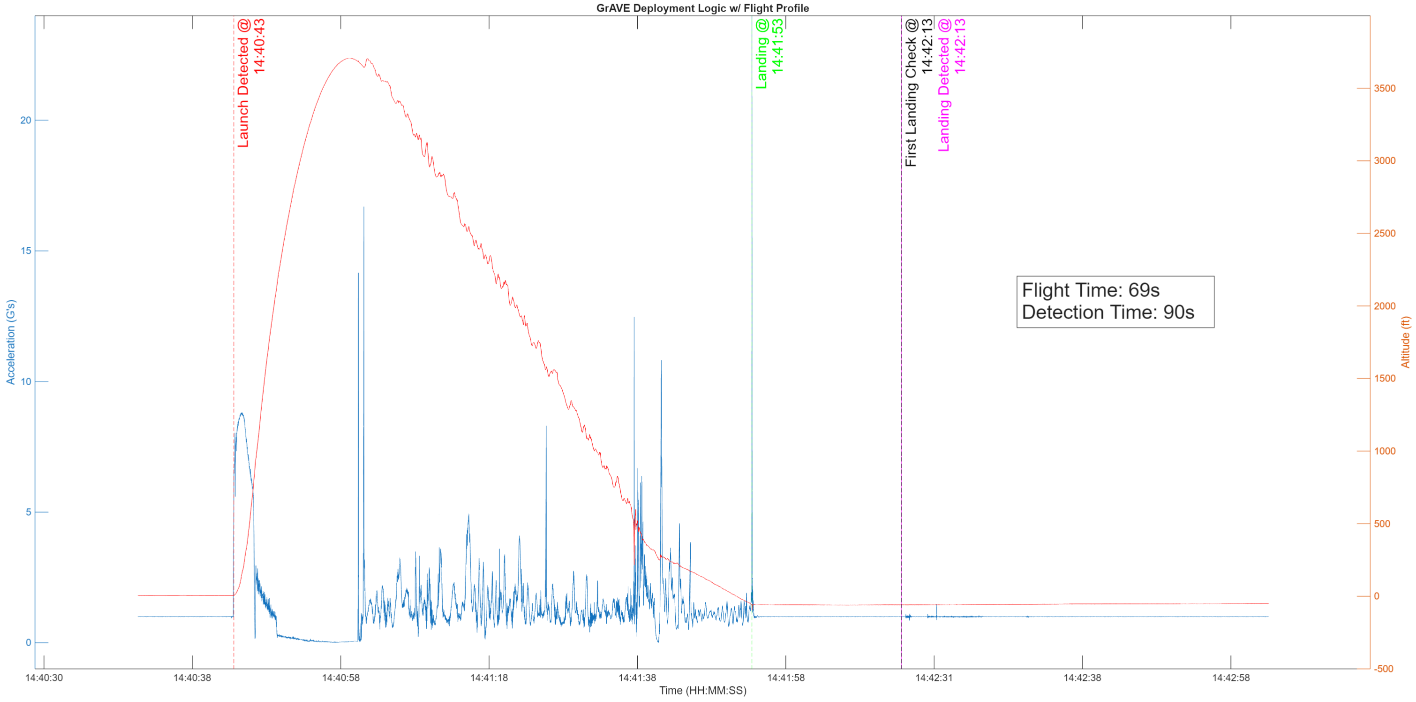
3.2 Post Flight System Analysis

The payload demonstration flight was completed on April 4, 2026 in Monterey, Virginia. The flight was a complete success. GrAVE successfully retained ZOMBIE during flight, and upon landing, ZOMBIE was released and deployed. Altimeter flight profiles are provided in Section 4.4, as the purpose of this flight was also a Vehicle Demonstration Re-Flight, described in Section 4.

3.3 Successful Systems

Flight Logic

During flight, both ZOMBIE and GrAVE used an INS to record flight data. Both systems were completely independent and used separate INSs. Dispite this, ZOMBIE and GrAVE recorded extremely similar flight profiles and the logic was consistent across both systems.



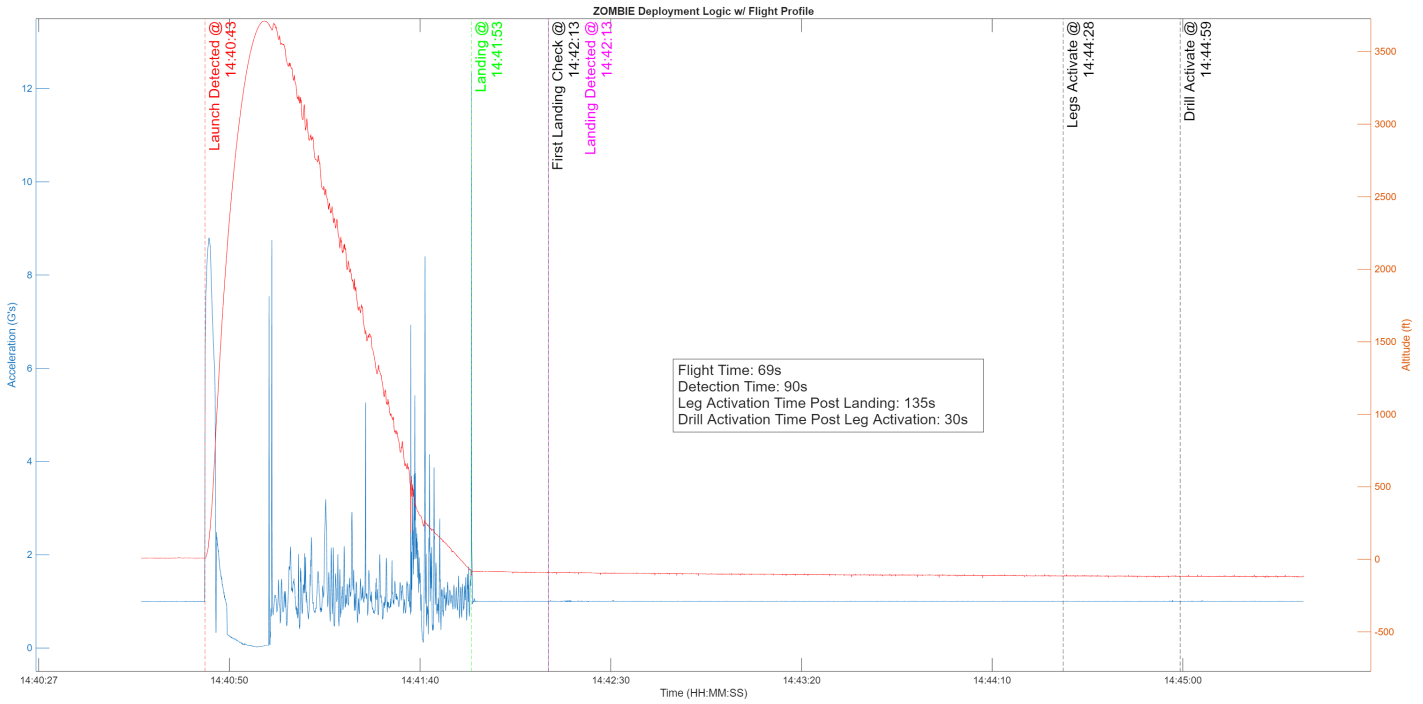


Figure 3.4: GrAVE (top) and ZOMBIE (bottom) deployment logic over recorded flight profiles.

Seen above in Figure 3.4 are the flight profiles of both ZOMBIE and GrAVE and the logic decisions overlayed on top. All state changes were triggered simultaneously on both systems. Launch was detected at 14:40:43 and landing was detected at 14:42:13. Post-launch analysis determined the actual flight time of the vehicle to be 69 (s). The landing detection system is only written to check 90 (s) post-launch as described in section 3.1.2, and as seen in Figure 3.4 landing is detected on the first check cycle, 90 (s) after launch. This is consistent across both systems, proving that the landing detection worked as designed.

After landing is detected, there is a branch in logic between the two systems. GrAVE is designed to immediately deploy ZOMBIE after landing is detected. This was not graphed as it is marked at the exact same time as the first landing check and the actual landing detection. ZOMBIE has a more complex system of logic that needed to be tested. The ZOMBIE flight profile in Figure 3.4 shows the timing for leg activation and the drill activation. The legs activated 135 (s) after landing was detected, this is what is set in the code to allow GrAVE to finish deploying ZOMBIE. After the legs are deployed, the code activated the drilling sequence. The drilling state starts 30 (s) after the legs are activated. This is 40 (s) before the legs are fully deployed. The 40 seconds of time is used to activate the soil sensor and ready the drilling motors for when the legs are finished righting ZOMBIE. The demonstrated logic in PDF is consistent with the designed CONOPS of the system, making it a complete success.

GrAVE Deployment

All systems regarding GrAVE deployment were successful. All systems were initialized successfully on the launch pad. During launch, the electronic latch remained closed around ZOMBIE's U-bolt. Landing detection is outlined in section 3.1.2. During post-launch analysis, it was determined that landing was detected from the altitude and acceleration measurements, verifying that these checks work successfully.

After landing detection triggered, GrAVE released ZOMBIE's latch via the activation of a stepper servo. The servo applied a force to the release lever of the latch, letting the latch spring into the open position. ZOMBIE's U-bolt was freed, and the lead screw extending pushed it out of the nosecone. The lead screw is attached to a pusher plate that slots over ZOMBIE's U-bolt. Actuating this lead screw pushes ZOMBIE out of the nosecone. Both the latch and lead-screw pusher plate mechanisms worked as designed. While this was not directly observed as the events occurred directly after landing, it was observed that ZOMBIE was extruded past the edge of the nosecone. That position is not possible unless the latch was released before ejecting. The latch also could not have released early because that would have caused the ZOMBIE to fall during descent. Furthermore, when recovering the payload, ZOMBIE was able to be removed from the nosecone without having to unlatch the latch, which supports the fact that the latch was released after landing automatically.

Figure 3.5: Payload landing configuration.

GrAVE only deployed ZOMBIE 25 (mm) instead of the full 470 (mm), but this was by design. There were many trees on the launch field used for PDF, and simulations ran by the aerodynamics subteam before launch suggested that there was a likelihood we would land in them. Efforts were taken to mitigate the risks of landing in a tree. If GrAVE were to deploy ZOMBIE fully while stuck in the branches of a tree, ZOMBIE would fall and sustain structural damage. It is also a safety concern, as ZOMBIE could fall from the vehicle, or the tree, while members approach the Launch Vehicle, risking hitting them and causing harm. For this reason, the deployment distance was reduced.

Figure 3.6: PDF launch site with surrounding tree cover.

While hanging in a branch, ZOMBIE is much less likely to fall out if still mainly inside the nosecone. This was verified with quick tests where ZOMBIE was partially deployed and then swung from tree branches at NC State. ZOMBIE did not fall, supporting that this change mitigated the described risk. While the deployment distance used was different than the final deployment distance, the code to detect landing and initiate the motor was the same. If the deployment distance is changed, and no other part of the code, ZOMBIE will deploy fully as intended. This was verified in the GrAVE Deployment Test. For this reason, this is the final, active code for ZOMBIE ejection.

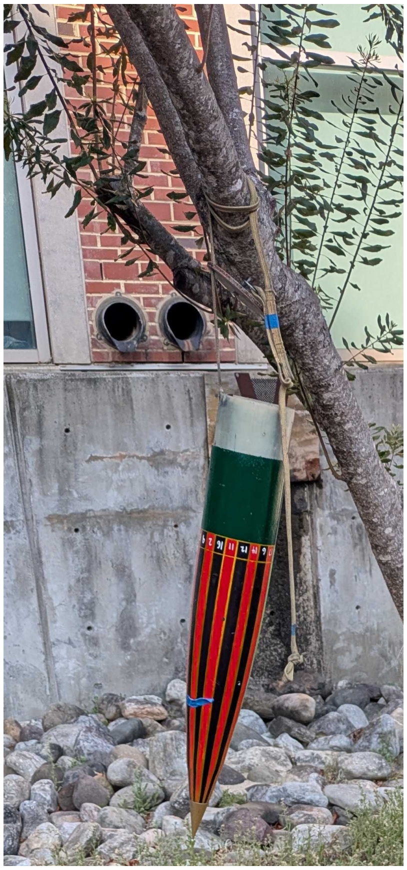


Figure 3.7: The payload hung from a tree to test ejection safety.

GrAVE did not receive any damage during flight. When the Team returned to the lab, the nosecone was disassembled and inspected for any possible damage that might have manifested structurally or electronically. All systems were still operational and the structure was ready to fly again.

Figure 3.8: No damage was observed on GrAVE after launch.

ZOMBIE

Figure 3.9: No damage was observed on ZOMBIE after launch.

All systems on ZOMBIE worked as designed, and it was undamaged upon post-flight inspection. ZOMBIE functioned in a reduced role from the final design, but because it was retained, still completed the PDF requirements. ZOMBIE's reduced functionality was a result of the safety risks posed by the launch field's surrounding tree cover. If the legs and drilling sequences were to physically operate, the legs would break inside the nosecone because it was only deployed 25 (mm). Despite this limitation, ZOMBIE gathered data from the INS and ran the same state-machine program that will be used for the competition, verifying launch and landing detection. It is important to note that this change only affected whether the motors spun or not, and did not change any system logic. During the competition flight, this exact same code will be run with the motors activated, allowing for full functionality. Since this functionality was demonstrated by the code without risking the motors, this was considered a success.

Figure 3.10: The HAUS experienced no damage during flight and all STEMnauts remained safely secured inside.

3.4 Unsuccessful Systems

Since all systems functioned as intended for the PDF, no systems were deemed unsuccessful. Additionally, all systems were inspected for damage after the flight, and no components were found to have any damage sustained.

3.5 Lessons Learned

There was one minor error encountered during vehicle assembly which did not impact the flight. There was a small bug that would occur if the computer were to slow down while initializing the servo. This glitch caused the latch servo to center itself, prematurely releasing ZOMBIE. Since this issue happened on the ground, no damage was sustained to the payload. The bug was identified through thorough testing and technical troubleshooting. This was not found before even though the GrAVE system had been tested dozens of times successfully before on the ground. Most likely the conditions of the launch field and the poor cellular reception led to conditions that would slow the Raspberry Pi down enough to experience the bug. The code was updated to prevent premature centering by preventing initialization from changing the servo position before it is explicitly set. The payload team learned that launch day conditions are more chaotic than lab testing conditions, and that sensitive hardware should be hardened or planned for.

Otherwise, the success of the payload meant the main lesson learned was to continue work as normal. The preparation done for launch, thorough testing, and integration of mock scripts all allowed for a smooth and successful flight.

4 Vehicle Demonstration Re-flight Results

4.1 Successful Systems

The Launch Vehicle structural components performed as expected. No damage was sustained to any part of the Launch Vehicle during flight or landing. All shear pins cleanly sheared between the airframe and coupler sections during drogue and main parachute deployment. All countersunk screws remained firmly seated, connecting the Main Parachute Bay to the AV Bay and Drogue Parachute Bay to the Fin Can.

The recovery avionics performed as designed with the primary altimeter separating the Launch Vehicle to deploy the drogue parachute at apogee and main parachute at 550 ft. The drogue parachute also performed as designed, slowing the Launch Vehicle's descent to 92.7 (fps).

The Air Brake system performed as designed, reducing apogee by 922 (ft). The fins deployed and safely retracted during flight. No damage, warping, or shearing was observed on the fins upon recovery of the vehicle.

4.2 System Failures

The Air Brakes apogee prediction algorithm undershot the target apogee set in code by 355 (ft). This was primarily due to bias in the Magnetometer and IMU data accumulating during the integration strategy for apogee prediction. The bias accumulated into a large sensor drift, which kept on feeding the apogee prediction algorithm slightly higher pressure, orientation, and acceleration values than expected. This resulted in a 65 (ft) apogee discrepancy between the Silicdyne Fluctus apogee and the Air Brakes system apogee.

During deployment, the main parachute shroud lines twisted around themselves, preventing the parachute from fully inflating. This resulted in a faster descent rate under main, thereby increasing the kinetic energy at landing.

4.3 Payload Status

4.3.1 Competition Payload Flown

We tested the structural integrity of the GrAVE retention system by flying the fully weighted configuration inside the Nosecone. The purpose of the flight was also to demonstrate the Payload, described in Section 3.

4.3.2 Flight and Air Brakes Status

The Launch vehicle reached an apogee of 3645 (ft) with a time to apogee of 15.3 (s) and total flight time of 69 (s). The simulated apogee without air brakes deployment was 4567 (ft). Simulations during VDF garnered an apogee of 3627 (ft), time to apogee of 13.8 (s), and total flight time of 72 seconds. The target apogee for this launch was 4000 (ft). Figure 4.1 below highlights the Openrocket vs Fluctus flight trajectory.

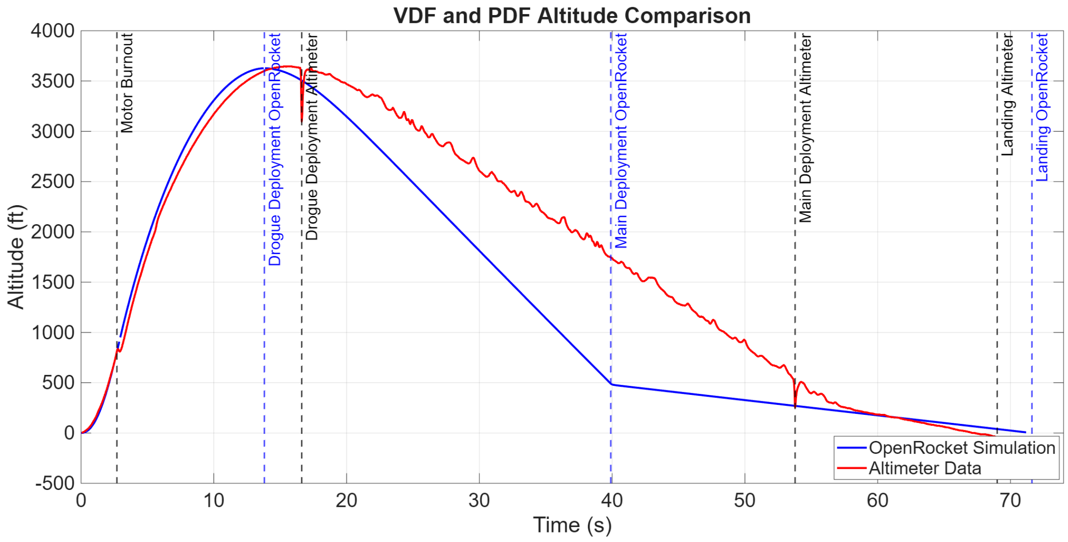


Figure 4.1: OpenRocket and Fluctus altitude over time.

Moreover, the landing location of the vehicle was able to be predicted within a 3 sigma confidence as shown in Figure 4.2

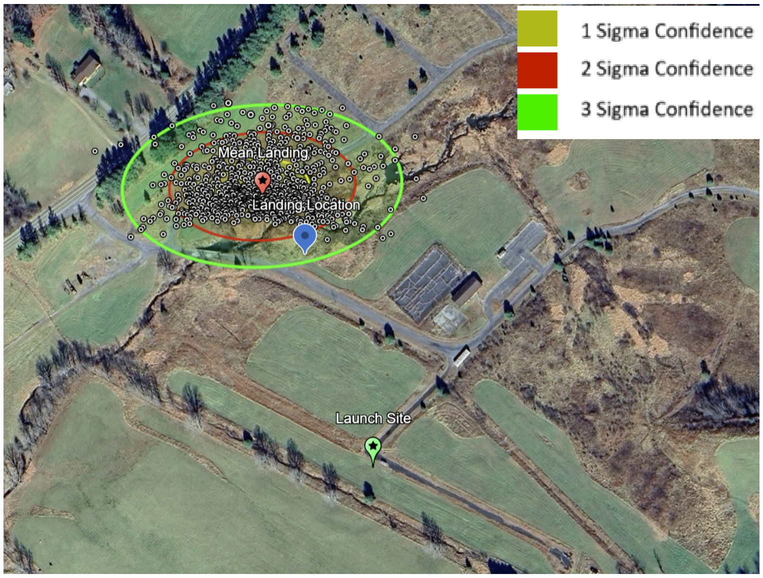


Figure 4.2: Monte Carlo Simulation of landing locations with confidence ellipses

As shown in Figure 4.3 shows the Air Brake fins fully deployed, as observed from the onboard camera in the Nosecone. The new SPV 0508 servo successfully deployed and retracted the fins at max q and 7 seconds into flight respectively. This further satisfies Team Derived Requirement AF 2 and AF 1, as the onboard camera confirms that the Air Brake system was deployed and that it controlled only the vehicle's vertical ascent.

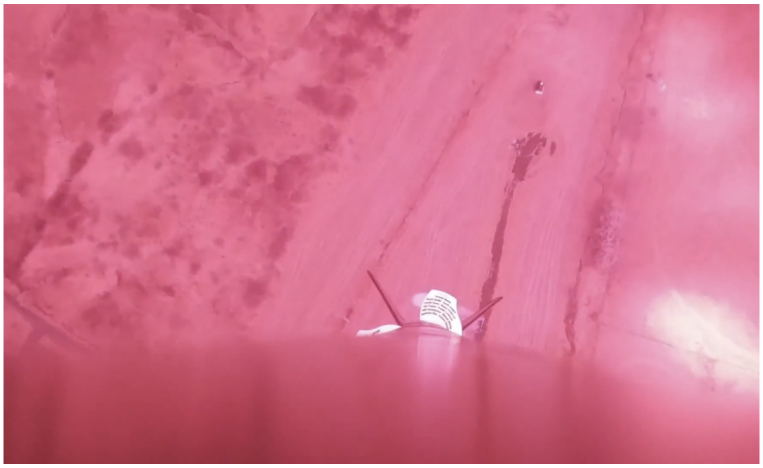


Figure 4.3: Air Brake Fins fully deployed, captured from on board camera.

Lastly, the Air Brakes prediction algorithm was 65 (ft) under what the Fluctus reported at apogee, and it converged to this value within 10 seconds of flight. Figure 4.4 shows the altitude over time for both the Fluctus data, INS estimated pressure altitude, and predicted apogee. The pressure altitude was able to be accounted for due to the INS' custom unscented Kalman filter but due to the sensor bias and drift present in the system the apogee prediction algorithm failed to bring the vehicle to the target apogee of 4000 (ft).

Fine tuning of the apogee prediction algorithm will be employed at launch day. This will be accomplished by estimating the inherent bias of each sensor in the system, and when the air brakes system detects liftoff, sensor values will be corrected by the estimated drift. Furthermore, Air Brakes will be deployed later into flight for less noisy data due to the lower velocity.

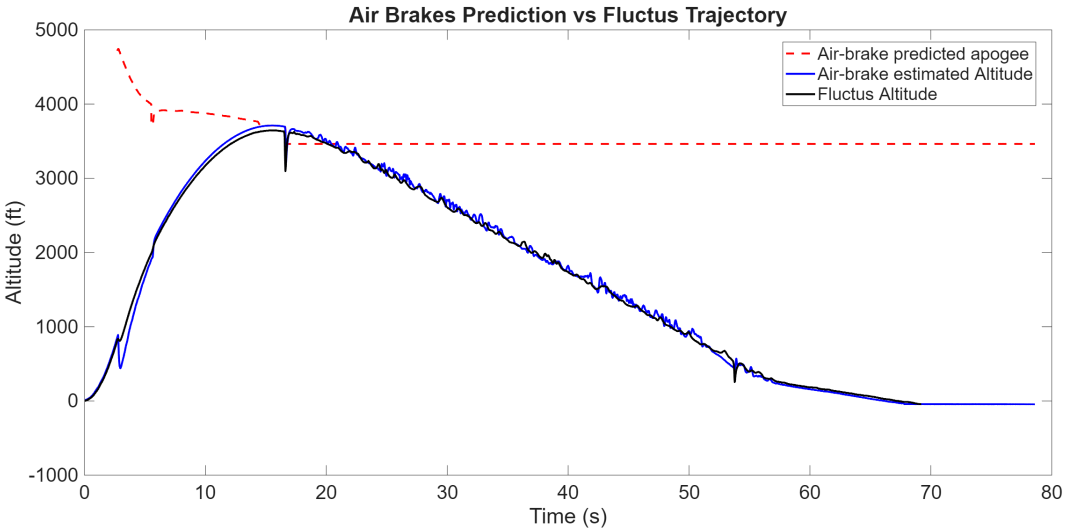


Figure 4.4: Air Brake Fins Apogee prediction during flight.

4.4 Flight Profile Data

For the Vehicle Demonstration Reflight, the Fluctus reported a max altitude of 3645 (ft) and the EasyMini reported 3639 (ft). The onboard altimeters demonstrated a 0.16% difference in apogee reports. The raw data from each altimeter, altitude and velocity, is are shown in Figures 4.5 and 4.6.

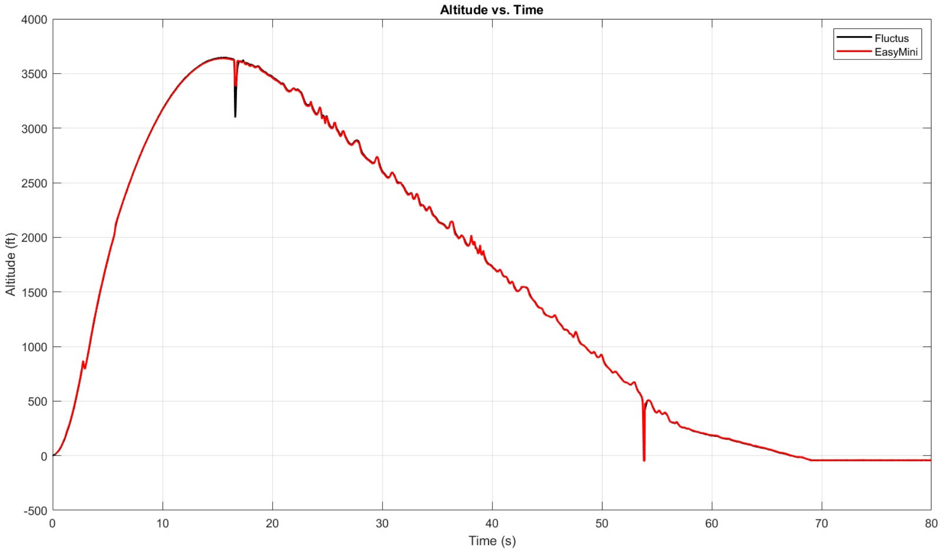


Figure 4.5: Fluctus and EasyMini altitude data during VDF.

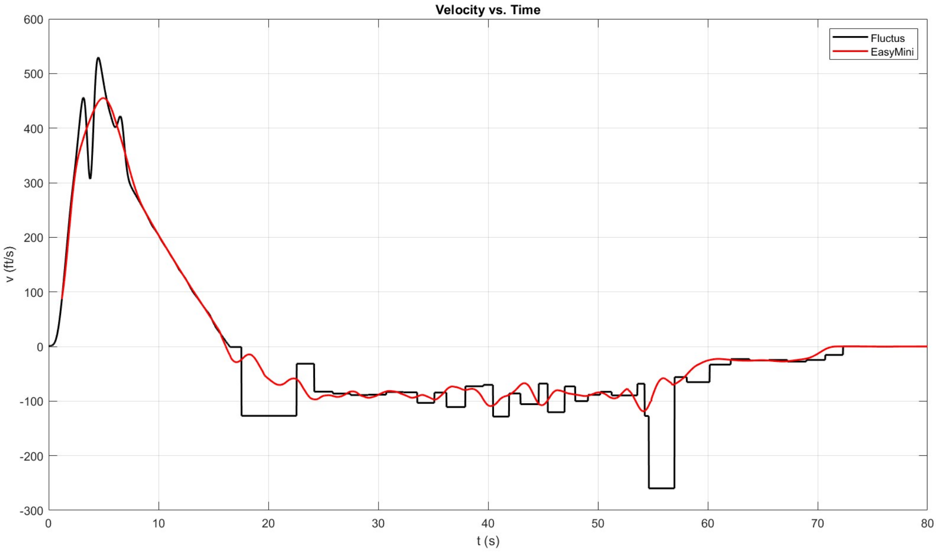


Figure 4.6: Fluctus and EasyMini velocity data during VDF.

In analyzing the altitude data from each altimeter, a small drop in altitude during parachute deployment was observed from both the Fluctus and EasyMini. The Fluctus saw a 548 (ft) drop at drogue parachute deployment and a 294 (ft) drop at main deployment. The EasyMini data shows a 254 (ft) drop at drogue deployment and a 500 (ft) drop at main deployment. These pressure spikes, resulting from ejection charge gases leaking into the AV Bay, are much improved compared to previous flights.

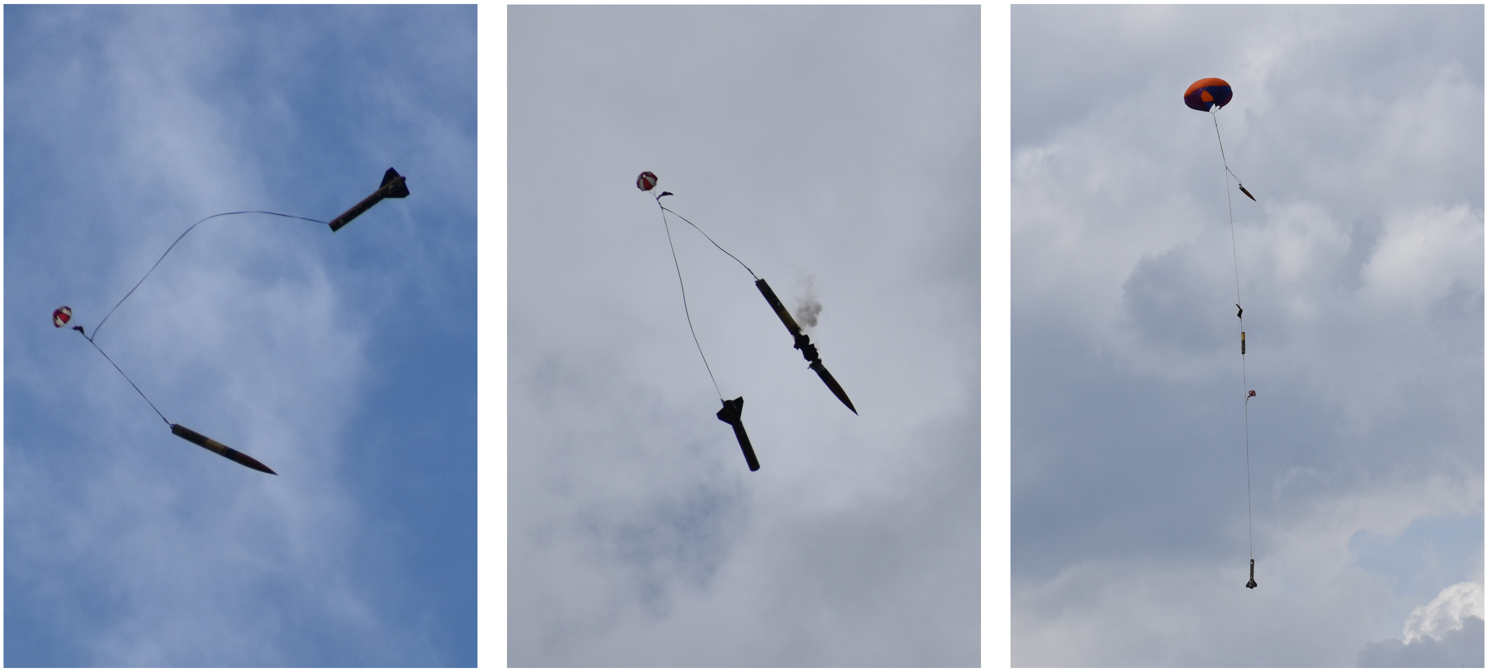
Drogue and main descent velocities are calculated by averaging the velocity during the respective descent phases. The average drogue descent velocity, taken between 19.16 (s) and 52.02 (s), was determined to equal 92.7 (fps). The average main descent velocity, taken between 62.42 (s) and 70.61 (s), was determined to equal 25.2 (fps). A summary of the launch data from each altimeter is shown in Table 4.1.

Table 4.1: Launch data summary.

	Apogee (ft)	Drogue Descent Velocity (fps)	Main Descent Velocity (fps)
Primary Altimeter (Fluctus)	3645	92.7	25.2
Secondary Altimeter (EasyMini)	3639	87.6	25.4

4.5 Recovery System

The Launch Vehicle's recovery system functioned as intended during the Vehicle Demonstration Flight. The drogue parachute was deployed at apogee and the main parachute was deployed at 550 (ft). Photos of the launch vehicle during drogue descent, deployment of the main parachute, and main descent are shown in Figure 4.7.



(a) Descent under drogue parachute.

(b) Main parachute deployment.

(c) Descent under main parachute.

Figure 4.7: VDF descent phases.

4.5.1 Landing Configuration

A photo of the full vehicle landing configuration is shown in Figure 4.8. The vehicle sustained no damage during landing.

Figure 4.8: VDF full landing configuration.

Photos of each individual section are shown in Figures 4.9a, 4.9b, and 4.9c. Upon recovery of the vehicle, the vehicle was inspected for unblown ejection charges and damage. All charges were initiated during flight and no damage to any component was observed.

(a) Fin Can and Drogue Bay.

(b) AV Bay and Main Bay

(c) Nose Cone

Figure 4.9: VDF landing configuration

The vehicle landed as expected, with the Fin Can and AV Bay connected to the drogue parachute and the Nosecone and AV Bay connected to the main parachute. Neither the shock cord or drogue parachute were tangled upon landing. The main parachute shroud lines appeared to be twisted together, as also seen in the photo of the vehicle during main descent in Figure 4.7c. The lines being twisted around each other resulted in the canopy not being able to fully inflate, impacting the vehicle's descent velocity.

4.5.2 Kinetic Energy Analysis

The mass of the Launch Vehicle on the launchpad and the mass of the vehicle after motor burnout are displayed in Table 4.2. The dry mass was used for recovery calculations, including descent velocity and time.

Table 4.2: Launch day vehicle mass.

Vehicle Mass	
Gross Launch Mass	42.0 (lbm)
Dry Mass	37.65 (lbm)

The kinetic energy of each vehicle section under drogue and main parachute is shown in Table 4.3. For the purpose of calculating kinetic energy, the mass of the drogue parachute, Nomex, and respective shock cord were excluded from the Drogue Bay mass and the main parachute, deployment bag, and respective shock cord were excluded from the mass of the Main Bay. The maximum kinetic energy for any one section of the vehicle during Vehicle Demonstration Flight was calculated to be 146.4 (ft-lbf). This value is significantly larger than what NASA requirement 3.2 allows. This high kinetic energy is largely due to the main parachute lines twisting around themselves during deployment, inhibiting the parachute from fully inflating. Had the parachute been able to fully inflate, the descent velocity, and thus kinetic energy upon landing, would have been signigicantly lower.

Table 4.3: Launch day kinetic energy for each section of the vehicle.

Section	Mass (lbm)	Descent Velocity under Drogue (fps)	Kinetic Energy under Drogue (ft-lbf)	Descent Velocity under Main (fps)	Kinetic Energy under Main (ft-lbf)
Fin Can, Drogue Bay	14.85	92.7	1982	25.2	146.4
AV Bay, Main Bay	5.62		2458		55.4
Nosecone	12.8				126.2

Table 4.4 shows a comparison of the predicted recovery system performance and the actual performance during Vehicle Demonstration Flight. A percent difference is included to better quantify the accuracy of the predictions. The largest difference the Team saw was a

96.4% difference between predicted and actual kinetic energy. Because the main parachute did not fully inflate, the vehicle descent rate under main was 51% faster than anticipated, resulting in an exponential increase in kinetic energy. This increased main descent velocity also resulted in a shorter overall descent time. The vehicle's drogue descent velocity, however, was consistent with predicted values.

Table 4.4: VDF descent comparison.

Parameter	Predicted Value	Actual Value	Percent Difference
Drogue Descent Velocity (fps)	98.5	92.7	-6.1%
Main Descent Velocity (fps)	14.9	25.2	+51.4%
Descent Time (s)	71.9	53.1	-30.0%
Fin Can + Drogue Bay KE ft-lbf)	51.2	146.4	+96.4%
AV Bay + Main Bay KE (ft-lbf)	19.4	55.4	+96.4%
Nosecone KE (ft-lbf)	44.1	126.2	+96.4%

4.6 Post-Flight Simulations

Using data collected from the Fullscale launch flight, the vehicle's Cd can be measured and compared with that derived from CFD analysis in ANSYS Fluent. The derivation applies only during the coast phase of flight, as the primary forces acting on the vehicle are limited to weight and drag. Equation 1 shows that the relationship between weight, acceleration, and drag force can be approximated via Newton's Second Law of motion for the Launch Vehicle. Equation 2 is the total drag force when Air Brakes is deployed. M is burnout mass, acceleration, a, is given from the Fluctus data, gravitation acceleration is constant and represented by g, $F_{d,roc}$ is the drag force for the rocket, and $F_{d,AB}$ is the Air Brakes drag force.

$$F_{d,roc} = -m(g + a) \tag{1}$$

$$F_{total} = F_{d,roc} + F_{d,AB} \tag{2}$$

Once F_d can be calculated, Equation 3 shows C_d can be calculated.

$$C_d = \frac{2\,F_{\text{total}}}{\rho\,v^2\left(S_{\text{ref,rocket}} + S_{\text{ref,airbrakes}}\right)}. \tag{3}$$

Table 4.5 tabulates the mean C_d garnered from the equations above for the data during the coast phase of flight. The frontal area with Air Brakes deployed was found to be 29.5 (in^2). Table 4.5 highlights the estimated and predicted values. As shown, the predicted C_d was 16% higher than was estimated from the Fluctus data. This shows that Simulations are slightly overestimating the total drag around the vehicle but still gives overall confidence in simulation trajectories since as shown in figure 4.1 the apogee differs only by 0.5%.

Table 4.5: Fullscale C_d Error Analysis

C_d Estimated	C_d Predicted	Percent Difference
0.81	0.97	16 %

4.7 Lessons Learned

The Launch Vehicle experienced higher than expected kinetic energy during descent due to the twisting of the main parachute. We folded the parachute according to the SkyAngle Cert -3 XL deployment bag recommendations. For future flights we will ensure that these recommendations are cohesive with our recovery setup. We will attempt different folding techniques, and test them by performing drop tests. We have never experienced an issue with manufacturer folding recommendations, as they have always allowed our parachutes to deploy nominally. For future flights, more rigorous testing will be completed.

5 Payload Testing

Since the submission of the FRR document, the payload team has completed all planned tests. These tests verified that the payload was ready to be flown on PDF. The tests are outlined below in Table 5.1, being the ZOMBIE self-righting test, the GrAVE deployment test, and the Ground Simulation of Payload Hardware. All tests were successful.

Table 5.1: Payload Tests Completed Since FRR

Test Name	Date Completed	Requirement/Hazard	Verification
ZOMBIE Self-Righting Test	Mar 11th, 2026	PF 3, PE 2, DHZ 40	Verified
GrAVE Deployment Test	Mar 8th, 2026	PF 2, PD 3, DHZ 45,46,48,56,59	Verified
Ground Simulation of Payload Hardware	Apr 2nd, 2026	PF 2, PF 3, PD 3, PD 4, PD 5	Verified

5.1 GrAVE Deployment Test

The GrAVE deployment test verified the retention and release mechanisms GrAVE uses on ZOMBIE. While in flight, a latch holds ZOMBIE in place. On landing, this latch is released, and a pusher plate is extended. This test is a ground simulation of this procedure to ensure that it works properly in flight. Since FRR, it was verified that the lead screw motor has enough force to deploy ZOMBIE at any orientation.

Table 5.2: GrAVE deployment test success criteria.

Success Criteria	**Status**
Latch retains ZOMBIE during flight. Once commanded, the latch releases and ZOMBIE is free to deploy	Verified
Pusher plate is actuated by a lead screw motor and extends when commanded	Verified
Pusher plate can move the full weight of ZOMBIE	Verified

Control Variables

- Latch assembly design
- Pusher plate mechanism design
- Electrical layout and battery power
- Motor programming
- Configuration within nosecone
- ZOMBIE installation method

Required Facilities, Equipment, Tools, and Software

- GrAVE electronics sled, assembled with batteries and latch
- Pusher plate with threaded rod
- Launch vehicle nosecone
- GrAVE mounting hardware
- ZOMBIE Lander
- Laptop with command line interface
- WiFi Hotspot connected to both the laptop and GrAVE Raspberry Pi
- High-fidelity testing code

Methodology

1. Assemble the GrAVE sled, including Raspberry Pi Zero, motor drivers, and latch servo
2. Insert the batteries and check voltage.
3. Start a WiFi hotspot from the laptop.
4. Ensure Raspberry Pi is receiving power and connects to the hotspot.
5. Install GrAVE in launch vehicle nosecone.
6. Thread in the lead screw pusher plate, aligning it with the latch on GrAVE.
7. Insert ZOMBIE into the launch vehicle nosecone, sliding it over the rails until it connects to GrAVE's latch. Tug to ensure secure connection.
8. Connect to the GrAVE Raspberry Pi using SSH protocol from the laptop's command line interface.
9. Run commands to change directory to the correct location, then run any commands necessary to prime the code.
10. Run the code using the same process that the state machine will call
11. Listen for the latch to release, then watch ZOMBIE be ejected from the nosecone.

Results

The GrAVE deployment system was tested in multiple scenarios to ensure ZOMBIE ejection, and all tests were successful. Tests were conducted with the nosecone on its side, deploying ZOMBIE horizontally, with the nosecone and deployment happening up a hill, and with the nosecone speared into the ground, leading to vertical deployment.

Figure 5.1: GrAVE Deployment Horizontal

Figure 5.2: GrAVE Deployment Upright

5.2 ZOMBIE Self-Righting Test

The ZOMBIE self-righting test demonstrated the ability for ZOMBIE to self-right using its deployable legs. After being deployed from GrAVE, ZOMBIE actuates a collar that forces the legs to rotate outwards. The reaction from this moment against the ground rights ZOMBIE from the horizontal to vertical position. To achieve the highest level of reliability, this test was conducted on multiple outdoor soil types.

Table 5.3: Self-righting test criteria.

Success Criteria	Status
Lead screw motor can actuate ZOMBIE's legs unloaded	Verified
Lead screw motor can actuate ZOMBIE's legs with full load	Verified

Control Variables

- Leg deployment mechanism
- ZOMBIE mass
- Starting leg deployment angle
- Testing surface
- Leg deployment speed

Required Facilities, Equipment, Tools, and Software

- ZOMBIE Lander
- Leg deployment system within ZOMBIE
- Laptop with command line interface
- WiFi Hotspot connected to both the laptop and GrAVE Raspberry Pi
- High-fidelity testing code
- Soil-like surface

Methodology

1. Assemble ZOMBIE in its full flight configuration, ensuring accurate mass
2. Retract ZOMBIE's legs
3. Plug in batteries and activate ZOMBIE electronics
4. Connect Raspberry Pi and laptop to the same WiFi hotspot, SSH in, and prime code
5. Place ZOMBIE on a soil surface outside similar to the conditions of the launch field in Huntsville, Alabama
6. Run the code that activates the lead screw, translates the collar, and forces the legs to deploy outwards
7. Cut the code when ZOMBIE reaches the fully upright state

Results

All recent self-righting tests were successful. After switching from a lead screw stepper motor to a servo and gear ratio system, the legs achieved enough torque to deploy correctly. The first series of tests were conducted inside on a flat table. When these tests proved successful, the environment was matched to that experienced after launch. ZOMBIE successfully self-righted on grass, exposed soil, mulch, and on a slight hill. This range of testing ensures success in any condition.

Figure 5.3: Results of the ZOMBIE Self-Righting test.

5.3 Ground Simulation of Payload Hardware

The final payload test will be the ground simulation of payload hardware. This test re-created the entire payload operation after landing. GrAVE deployed ZOMBIE, ZOMBIE self-righted, and finally ZOMBIE drilled to collect soil. This is essentially a combination of all previous tests. Its purpose is to validate integration between all subsystems.

Table 5.4: Payload Ground Test Criteria.

Success Criteria	Status
GrAVE deployment	Verified
ZOMBIE self-righting	Verified
ZOMBIE drilling sequence collects 75 (mL) and sensor data	Verified

Control Variables

- GrAVE Configuration
- ZOMBIE Configuration
- Deployment location
- Deployment sequencing
- State machine starting condition

Required Facilities, Equipment, Tools, and Software

- GrAVE in the fully assembled configuration
- ZOMBIE in the fully assembled configuration
- Launch vehicle nosecone
- An outdoor location on soil similar to that of Huntsville, Alabama
- WiFi Hotspot connected to both the laptop and payload computers
- High-fidelity testing code

Methodology

1. Assemble GrAVE and ZOMBIE into the nosecone in the launch configuration
2. Activate electronics, connecting ZOMBIE, GrAVE, and the laptop to the same hotspot
3. Connect via SSH and ”trick” ZOMBIE and GrAVE into thinking they have just landed
4. Observe as the GrAVE code runs that ejects ZOMBIE
5. Ensure ZOMBIE self-rights
6. Validate that ZOMBIE collects 75 (mL) of soil and logs sensor data

Results

This test proved successful and served as good preparation for the Payload Demonstration Flight. With this test, all payload systems were tested together and demonstrated successful integration. A key result from this test was the tuning of ZOMBIE’s deployment timer. If ZOMBIE were to deploy prematurely, it would remain within GrAVE and damage itself to late, and soil collection time is lost. In addition, it was demonstrated that ZOMBIE can self-right without interfering with the nosecone or other deployment hardware, ensuring success on PDF.

Figure 5.4: GrAVE and ZOMBIE Deployment test.

6 Requirements Compliance Status

6.0.1 Verification Plan

Below are requirements that were not verified by the FRR deadline due to them needing a successful PDF or VDF re-flight. Remaining requirements will be verified in the PLAR document.

6.0.2 Competition Requirements

Table 6.1: 2025-2026 Vehicle Requirements

ID	SHALL Statement	Planned Action	Verification Method & Success Criteria	Status	Performing Subsystem	Results
2.1	The vehicle shall deliver the payload to an apogee altitude between 4,000 and 6,000 feet above ground level (AGL). Teams flying below 3,500 feet or above 6,500 feet on their competition launch will receive zero altitude points towards their overall project score and will not be eligible for the Altitude Award.	The Structures and Aerodynamics Leads will design the full Launch vehicle to be capable of delivering the Payload to an apogee between 4,000 and 6,000 ft. AGL. The Structures Lead will facilitate the manufacturing of the launch vehicle with the team.	**(1) Analysis:** Simulations run by the Aerodynamics Lead show the launch vehicle reaching an apogee between 4,000 and 6,000 ft. AGL **(2) Demonstration:** The launch vehicle recovery altimeter data shows between 4,000 and 6,000 ft. for the VDF and PDF flights	PV	Aerodynamics & Structures	**(1)** Section 5.1 of FRR [1] shows the launch vehicle reaching a minimum apogee of 4500 (ft) and a max apogee of 5000 (ft), putting the Launch Vehicle inside the NASA Required Range. **(2)** VDF and PDF results are detailed in Section 4.
2.2	The launch vehicle and payload shall be capable of remaining in launch-ready configuration on the pad for a minimum of 3 hours without losing the functionality of any critical on-board components, although the capability to withstand longer delays is highly encouraged	The Recovery Lead and Payload Team will use batteries that have a large enough capacity that they can power all avionic and payload electronics for a minimum of 3 hours without losing the capability of any critical components. The Integration will verify the batteries will function via a ground test before launch	**(1) Analysis:** Electronic power draw combined with battery capacity calculations confirm functionality for >3 hours. **(2) Demonstration:** All avionics and payload electronic systems maintain full operational functionality for > 3 hours.	V	Payload & Recovery	**(1)** Recovery battery analysis is located in Section 4.4.1 of FRR. Payload battery analysis is located in Section 6.4 of FRR. **(2)** VDF and PDF shows full functionality of the recovery and air brakes batteries during launch prep, pad idle time, launch, recovery, and payload operations, detailed in Section 4.
2.11	The launch vehicle shall have a minimum static stability margin of 2.0 while sitting on the pad.	The Aerodynamics Lead shall design the launch vehicle such that it will have a minimum static stability margin of 2.0 while on the pad.	**(1) Analysis:** Analysis shows the projected launch vehicle has a stability a minimum of 2.0 in its launch ready configuration. **(2) Demonstration:** The Launch Vehicle design has a static stability of greater than 2 in its launch ready configuration.	V	Aerodynamics	**(1)** Section 5.4 of FRR [1]shows the projected stability margin of the launch vehicle. **(2)** Stability for the VDF and PDF is shown to be 3.25, shown in Section 1.3.3.
2.14	The launch vehicle shall accelerate to a minimum velocity of 52 fps at rail exit	The Aerodynamics Lead will select a commercially available motor that provides enough thrust such that the velocity of the launch vehicle at the exit of the rail is at minimum 52 fps.	Analysis: The selected motor provides the launch vehicle with a velocity off the rod of a minimum of 52 fps.	V	Aerodynamics	Velocity off the rod is shown to be over 52 fps in Section 1.3.1.

Table 6.1: 2025-2026 Vehicle Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method & Success Criteria	Status	Performing Subsystem	Results
2.18	Vehicle Demonstration Flight—The purpose of the Vehicle Demonstration Flight is to validate the launch vehicle's stability, structural integrity, recovery systems, and the team's ability to prepare the launch vehicle for flight. A successful flight is defined as a launch in which all hardware is functioning properly (drogue chute at apogee, main chute at the intended lower altitude, functioning tracking devices, etc.).	Project Management will ensure that the launch Vehicle Performs a Vehicle Demonstration flight, wherein all associated subsystems (Recovery, Structures, Aerodynamics, etc) perform as intended and in the same configuration as the competition prior to the FRR [1] Deadline.	**Demonstration:** The VDF flight confirms the full functionality of the launch vehicle including the recovery system and structural components.	V	Project Management	VDF results are located in Section 4.
2.20.6	The launch vehicle shall not exceed Mach 1 at any point during flight	The Aerodynamics Lead will select a motor such that the designed Launch Vehicle does not exceed Mach 1 at any point during its flight.	**(1) Analysis:** The launch vehicle is simulated to reach velocities below mach 1. **(2) Demonstration:** During the VDF flight, altimeter data shows the launch vehicle does not reach velocities above Mach 1.	V	Aerodynamics	**(1)** Flight profiles depicting velocities during flight are identified in Section 5.2 of FRR. **(2)** VDF flight velocity profiles are located in Section 4.4.

Table 6.2: 2025-2026 Recovery Requirements

ID	SHALL Statement	Planned Action	Verification Method & Success Criteria	Status	Performing Subsystem	Results
3.1	The full-scale launch vehicle shall stage the deployment of its recovery devices, where a drogue parachute is deployed at apogee, and a main parachute is deployed at a lower altitude. Tumble or streamer recovery from apogee to main parachute deployment is also permissible, provided that kinetic energy during drogue stage descent is reasonable, as deemed by the RSO.	The Recovery Lead will design a recovery system such that the drogue parachute is deployed at apogee and the main parachute is deployed at a lower altitude.	**Demonstration:** The launch vehicle's altimeters are pre-programmed to deploy its drogue parachute at apogee and main parachute at a specified altitude during descent. This is verified during the VDF flight	V	Recovery	VDF flight confirms the functionality of the recovery altimeters, described in Section 4.5.
3.1.1	The main parachute shall be deployed no lower than 500 feet.	The Recovery Lead will design the recovery system such that the main parachute is deployed at no lower than 500 feet.	**Demonstration:** The Launch vehicle's altimeters are pre-programmed to deploy its main parachute at an altitude greater than 500 ft during descent. This is verified during the VDF flight.	V	Recovery	Main parachute deployment is described in Section 4.5.

Table 6.2: 2025-2026 Recovery Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method & Success Criteria	Status	Performing Subsystem	Results
3.1.2	The apogee event shall contain a delay of no more than 2 seconds.	The Recovery Lead will design the recovery system such that the drogue event will contain a delay of no more than 2 seconds.	**Demonstration:** The launch vehicle's altimeters are pre-programmed to deploy its drogue parachute no more than 2 seconds after apogee. This is verified during the VDF flight.	V	Recovery	Drogue parachute deployment is described in Section 4.5.
3.2	Each independent section of the launch vehicle shall have a maximum kinetic energy of 75 ft-lbf at landing. Teams whose heaviest section of their launch vehicle, as verified by vehicle demonstration flight data, stays under 65 ft-lbf will be awarded bonus points.	The Recovery Lead will select or manufacture parachutes such that each independent section will have a maximum kinetic energy of 75 ft-lbf at landing.	**(1) Analysis:** Kinetic energy calculated for each section has a maximum kinetic energy of 75 ft-lbf. **(2) Demonstration:** During VDF the drogue and main parachute delivers each individual launch vehicle section to the ground with a maximum kinetic energy of 75 ft-lbf.	PV	Recovery	**(1)** Kinetic energy calculations are described in Section 5.6 of FRR. **(2)** VDF described in Section 4.5.2 shows the kinetic energy of each individual section.
3.8	Removable shear pins shall be used for both the main parachute compartment and the drogue parachute compartment.	The Recovery Lead will design the Recovery system such that shear pins will be used for both main and drogue parachute compartments. The Structures Lead will ensure the launch vehicle is designed such that shear pins will be to retain the main and drogue compartments.	**Inspection:** The recovery system is designed such that separating sections utilize shear pins.	V	Recovery & Structures	The shear pin sizes are described in Section 2.
3.11	Descent time of the launch vehicle shall be limited to 90 seconds (apogee to touch down). Teams whose launch vehicle descent, as verified by vehicle demonstration flight data, stays under 80 seconds will be awarded bonus points.	The Recovery Lead will select appropriately sized parachutes to be used for the recovery system such that the launch vehicle's descent time is under 90 seconds.	**(1) Analysis:** The recovery system is designed such that the parachutes will deliver the launch vehicle to the ground in under 90 seconds. **(2) Demonstration:** VDF flight confirms descent time is under 90 seconds.	V	Recovery	**(1)** Descent time calculations are described in Section 5.2 of FRR. **(2)** Descent time for VDF is described in Section 4.5.2.
3.12	Electronic GPS Tracking device(s) shall be installed in the launch vehicle and will transmit the position of the tethered vehicle and any independent section(s) to a ground receiver.	The Recovery Lead will utilize electronic GPS tracking devices in every untethered independent section of the launch vehicle that are capable of transmitting position of the section to a ground receiver.	**(1) Inspection:** The recovery system is designed with GPS tracking devices in each untethered independent section. **(2) Test:** GPS transmitters are ground tested to confirm functionality. **(3) Demonstration:** VDF confirms functionality of GPS tracking devices.	V	Recovery	**(1)** GPS tracking devices are specified in Section 4.4.2 of FRR. **(2)** Ground testing results are described in Section 10.1.3 of FRR. **(3)** GPS tracker functionality was verified during VDF, described in Section 4.

Table 6.3: 2025-2026 Payload Requirements

ID	SHALL Statement	Planned Action	Verification Method & Success Criteria	Status	Performing Subsystem	Results

Table 6.3: 2025-2026 Payload Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method & Success Criteria	Sta-tus	Performing Subsystem	Results
4.1	After landing, teams shall autonomously collect and retain a soil sample of at least 50 milliliters.	The Payload Team will design a payload that can autonomously collect 50 ml of soil from the landing site.	**Demonstration:** Upon landing the payload collects a minimum of 50 ml of soil from the landing site.	V	Payload Team	Payload soil collection is described in Section 3.6.1 of FRR. PDF is described in Section 3. Soil collection is verified in Section 5.3.
4.1.1	All soil collection and analysis must be completed within 15 minutes of landing.	The Payload Team will design the payload such that it will collect the soil sample within 15 minutes of the launch vehicle landing.	**Demonstration:** Upon landing the payload collects a minimum of 50 ml of soil from the landing site within 15 minutes.	V	Payload Team	Payload logging is described in Section 6.4.1 of FRR. Payload soil collection is described in Section 5.3. PDF is described in Section 3.
4.2	Teams shall autonomously test the collected sample for at least one of the following: Nitrate-Nitrogen content, pH level, or electrical conductivity	The Payload Team will design the payload such that it is able autonomously test soil samples for its Nitrate-Nitrogen content, pH level, and Electrical Conductivity.	**Demonstration:** The payload tests collected soil sample for Nitrate-Nitrogen content, pH level, and Electrical Conductivity.	V	Payload Team	Payload Sensor is described in Section 6.4 of FRR. PDF is described in Section 3. Soil collection testing is described in Section 5.3.
4.2.1	Analysis results shall include time stamps for verification.	The Payload will be programmed such that it includes timestamps for every important Analysis result.	**Demonstration:** The payload logs and collects timestamps for each state change.	V	Payload Systems Lead	Payload logging operation is described in Section 6.4.1 of FRR. PDF is described in Section 3. Soil collection testing is described in Section 5.3.

Table 6.3: 2025-2026 Payload Requirements (continued)

ID	SHALL Statement	Planned Action	Verification Method & Success Criteria	Status	Performing Subsystem	Results
4.4	The HAUS's structure shall include an atmosphere isolated compartment to serve as living quarters for 4 STEMnauts. The compartment shall be enclosed and separated from the external atmosphere;No additional requirements for "living conditions" are included, but teams are encouraged to consider appropriate accommodations the STEMnauts may need for an extended excursion on a lunar or planetary body. STEMnauts are assumed to have all qualities typical of astronauts. It is up to teams to be creative in how to depict their four STEMnauts in the HAUS design. "Atmosphere isolated compartment" means the living quarters must be enclosed and separated from the external atmosphere. Pressure equalization holes are exempt from this isolation requirement.	The Payload Structures Lead will design a HAUS enclosure to serve as living quarters for 4 STEMnauts. The HAUS enclosure will be separate from the external atmosphere, with a hole to equalize pressure if deemed necessary.	**Inspection:** The Payload contains the HAUS enclosure where STEMnauts live separate from the external atmosphere.	V	Payload Structures	Finalized HAUS design is located in Section 6.3 of FRR.

6.0.3 Vehicle Team Derived Requirements

Table 6.4: 2025-2026 Team Derived Vehicle Requirements

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
Functional Requirements							
LVF 6	The Launch Vehicle Shall not be overstable, defined as having a static stability >4.	NASA requirement 3.10 requires the launch vehicle drifts less than 2,500 (ft) from the launch pad. Vehicles with excessive stability are more likely to weathercock, increasing horizontal travel and creating greater risk of violating this requirement. Setting the bound at 4 calipers maintains compliance with NASA requirement 2.11 as well.	Final center of gravity and center of pressure will be determined using mass properties and aerodynamic simulation. Stability will be calculated prior to flight.	**(1) Demonstration:** The launch vehicle has a stability less than 4 in its final launch configuration. **(2) Analysis:** Simulated static margins are < 4 for all configurations.	V	Aerodynamics	**(1)** VDF flight confirms stability of the Launch Vehicle, described in Section 1.3.3. **(2)** Stability calculations are described in Section 5.4 of FRR.
Environmental Requirements							

Table 6.4: Team Derived Vehicle Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
LVE 2	All structural components SHALL be designed to operate nominally with an ambient temperature range of 25°F to 100°F	Launch day conditions can vary in temperature, historically between 25°F to 100°F. Designing structural components to remain functional across 25°F to 100°F ensures consistent structural integrity and safe vehicle operation across expected environmental conditions. Composite property materials can be affected by differences in temperature, which is why materials chosen must be operational during all potential launch day conditions.	Materials such as fiberglass, carbon fiber, and compatible adhesives with demonstrated performance across 25°F to 100°F will be selected for structural components.	**(1) Analysis:** Structural materials and adhesives are verified through documentation or engineering analysis to maintain sufficient strength and performance from 25°F to 100°F. **(2) Demonstration:** Structural components experience no failure or degradation during ground and flight operations conducted within the specified temperature range.	V	Structures & Safety	**(1)** Material properties are described in Section 3.6 of FRR. **(2)** VDF is described in Section 4.
LVE 3	All structural components SHALL be resistant to ambient humidity up to 90% relative humidity without degradation in adhesive or performance.	Launch day humidity conditions can vary and may affect composite materials . High humidity can reduce adhesive strength and degrade structural margin if not accounted for. Selecting materials and adhesives proven to perform at up to 90% relative humidity when cured mitigates these risks. A 90% relative humidity threshold was selected to represent near-saturation ambient conditions while avoiding unrealistic condensation scenarios not expected during launch day operations.	Materials and adhesives will be selected based on documented manufacturer data and prior aerospace use demonstrating structural stability and adhesive performance at up to 90% relative humidity.	**(1) Analysis:** Documentation or engineering analysis confirms that selected structural materials and adhesives maintain required performance at humidity levels up to 90%. **(2) Demonstration:** Structural components exhibit no structural degradation, adhesive weakening, or failure attributed to ambient humidity during ground and flight operations.	V	Structures	**(1)** Material properties are described in Section 3.6 of FRR. **(2)** VDF is described in Section 4.
LVE 4	Structural components SHALL be capable of withstanding ground impact from soil, gravel or sparse vegetation without compromising future reusability.	Launch vehicle landings can occur on variable terrain including soil, gravel, or light vegetation, and structural components must survive impact without critical damage to ensure reusability and kinetic energy requirements (NASA Req 2.4, 3.2). Maintaining structural integrity upon landing supports recovery requirements and mitigates structural damage.	Structural components will be designed using impact-resistant composite materials with adequate toughness. Drop will be performed on representative components to verify survivability.	**(1)** Testing: Drop testing demonstrates structural components withstand expected landing impact loads without structural failure or permanent damage affecting reuse. **(2) Demonstration:** Vehicle Demonstration Flight confirms that structural components remain intact and reusable following landing.	V	Structures	**(1)** Completed Fin can drop test is described in Section 10.1.7 of FRR. **(2)** VDF is described in Section 4.

6.0.4 Recovery Team Derived Requirements

Table 6.5: 2025-2026 Team Recovery Vehicle Requirements

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
Functional Requirements							

Table 6.5: Team Derived Recovery Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
RF 4	Post-flight inspections SHALL be done for any and all recovery components including parachutes, harnesses and altimeter housings.	NASA requirement 2.4 requires that the launch vehicle is completely reusable, thus all recovery components must also be able to be reused. Recovery components experience significant loading during deployment, inflation, descent, and landing. Inspection allows confirmation that the Recovery system survived all loads and is reusable.	After each flight, the Recovery Lead will inspect all recovery components for fraying, tearing, deformation, thermal damage, hardware deformation, or electronic enclosure damage. Any damaged components will be repaired or replaced prior to future flight use.	**Inspection:** Post-flight inspection is documented for all recovery components. Visual inspection confirms no unacceptable wear or structural degradation; any damaged components are removed from service and repaired or replaced prior to reuse.	V	Recovery	VDF Results are described in Section 4.
RF 8	Descent velocity SHALL be kept below 135 fps at all times during descent.	If descent velocity is too high, the main parachute and harness can see excessive opening shock, increasing the risk of canopy damage, shroud-line failure, or zippering damage to the airframe; if descent velocity under drogue is too low, the vehicle spends more time aloft, increasing drift and the chance of landing outside the safe recovery area. NASA limits kinetic energy for each independent section to 75 ft-lbf or less (Requirement 3.2) and caps drift distance at 2,500 ft (Requirement 3.10), while also requiring that total descent time be kept within competition limits (Requirement 3.11). Subscale flight data showed that a larger drogue and in gusty winds led to an extended drogue descent, showing that a higher drogue descent rate is needed to better meet these NASA constraints. Setting a 135 fps upper limit allows the Recovery Lead to size a smaller drogue to reduce descent time while still bounding the maximum descent speed to a value that combined with proper main sizing maintains landing kinetic energy and opening loads within acceptable limits.	The Recovery Lead will select appropriately sized drogue and main parachutes to ensure the total descent profile remains under 135 fps. Analysis will be done to verify that descent velocities remain within the limit.	**(1) Analysis:** Predicted descent rates under the selected drogue and main parachutes show that the vehicle remains below 135 fps during all descent phases. **(2) Demonstration:** Flight data from the Vehicle Demonstration Flight confirms that measured descent velocities remain below 135 fps throughout descent.	V	Recovery	**(1)** Descent velocity is calculated and described in Section 5.6 of FRR. **(2)** Descent velocities are described in Table 4.3.

6.0.5 Air Brakes Team Derived Requirements

Table 6.6: Team Derived Air Brakes Requirements

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
Functional Requirements							

Table 6.6: Team Derived Air Brakes Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
AF 2	The Launch Vehicle SHALL include a camera to verify deployment of the Air Brakes system.	If Air Brakes deployment is verified only through acceleration data or software, a false-positive confirmation may occur in scenarios such as partial deployment or code reporting success without physical actuation. If the system is programmed base off of inccorrect deployment, it can lead to the launch vehicle failing to reach the intended apogee (DHZ 20). Providing visual verification ensures that deployment is physically confirmed, and distinguishes between successful and failed deployment states, mitigating these hazards.	An on board camera will be integrated into the launch vehicle with a clear view of the Air Brakes deployment.	**(1) Inspection:** Camera mount and placement are confirmed to provide a clear field of view of the Air Brakes system. **(2) Demonstration:** VDF flight footage confirms capture of Air Brakes deployment.	V	Structures & Aerodynamics	**(1)** Camera location is described in Section 3.6.2 of FRR. **(2)** VDF results including images taken with on board camera is shown in Figure 4.3.
AF 3	The Air Brakes system SHALL remain retracted until the launch vehicle's boost phase has ended.	If the Air Brakes deploy during the boost phase, they introduce significant drag while the rocket is still under high thrust loading, which can create unexpected aerodynamic forces, high stress on the Air Brake components, potentially causing them to break. Air Brake fins breaking could result in unexpected moments and failure to deploy at all, causing the launch vehilce to fail to reach its intended apogee (DHZ 19, 20). By Ensuring that Air Brakes cannot deploy during the boost pahse of flight prevents Air Brakes from deploying under extreme loading, mitigating these risks and ensuring the launch vehicle reaches its intended apogee.	Air Brakes control logic will incorporate a burnout detection condition, preventing deployment commands until burnout has been confirmed.	**Inspection:** VDF onboard camera footage and flight data confirm that Air Brakes remain fully retracted during boost and deploy only after burnout.	V	Aerodynamics	VDF results including images taken with on board camera is described in Figure 4.3.
			Design Requirements				
AD 1	Air Brakes gear mechanisms SHALL be designed such that all fins retract and deploy simultaneously.	If the Air Brakes deploy or retract asymmetrically, one side of the rocket may experience greater drag than the other, which can introduce unintended roll, yaw, pitch disturbances, or loss of aerodynamic stability. This would result in instability and unpredictable flight path due to moments acting on the launch vehicle (DHZ 19 20). By designing the Air Brakes to only deploy simetaneusly, they will not induce moments on the launch vehicle, allowing for a successfull and predictable flight.	The Air Brakes mechanism will utilize a mechanical gear system that physically couples all fins, ensuring simultaneous actuation.	**(1) Test:** Ground testing confirms that all Air Brakes fins deploy and retract simultaneously. **(2) Demonstration:** VDF flight testing confirms synchronized fin deployment and retraction under flight conditions.	V	Aerodynamics	**(1)** Ground testing for Air Brakes deployment is located in Section 10.2.5 of FRR. **(2)** VDF results including Air Brakes deployment is described in Section 4.3.2.

Table 6.6: Team Derived Air Brakes Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
AD 2	The Air Brakes system SHALL utilize state-based software.	If the Air Brakes software responds directly to raw sensor inputs without defined operating states, noise or incorrect measurements can trigger unintended or incorrect deployment behavior, leading to the Air Brakes failing to bring the Launch Vehicle to the intended apogee (DHZ 21). Employing a state based architecture ensures deterministic behavior by restricting Air Brakes actions to well-defined flight phases (e.g., standby, boost, coast, descent), preventing unintended actuation due to transient data and directly mitigating these hazards.	The Air Brakes software will be implemented as a state machine with clearly defined states such as standby, motor burn, coast, free fall, and landing. Transitions will occur only under verified conditions, and each state will explicitly define allowed Air Brakes actions.	**(1) Inspection:** Software documentation and code review confirm use of a state-based architecture with clearly defined states and transitions. **(2) Demonstration:** During VDF, Air Brakes behavior is observed to transition deterministically according to flight states with no unintended deployment responses.	V	Aerodynamics	**(1)** Air Brakes software design is described in Section 7.2 of FRR. **(2)** Air Brakes results are described in Section 4.3.2.
AD 4	The Air Brakes System SHALL be sealed such that Air Brakes barometric pressure data will not be adversely affected from deployment.	If the Air Brakes are not properly sealed, Air Brakes Deployment can cause pressure spikes inside the Air Brakes sensor housing. This can cause incorrect readings to influence the software, causing Air Brakes to perform inccorectly, meaning the launch vehicle fails to reach its intended apogee (DHZ 22). By properly sealing the Air Brakes sensors, the risk of inaccurate barmoetric pressure data is reduced.	Air Brakes are designed using an o-ring to isolate the barometric pressure sensor used in Air Brakes. Sealing methods will be verified during ground testing.	**(1) Test:** Ground testing verifies that Air Brakes deployment does not introduce pressure spikes or instability in sensor readings. **(2) Inspection:** Post-flight barometric data review confirms stable and reliable sensor performance throughout deployment events during VDF.	V	Aerodynamics	**(1)** Air Brakes testing is described in Section 10.2.6 of FRR. **(2)** Air Brakes results are described in Section 4.3.2.

6.0.6 Payload Team Derived Requirements

Table 6.7: Team Derived Payload Requirements

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
Functional Requirements							
PF 2	The payload lander system SHALL be capable of fully removing itself from the nosecone section autonomously.	If the payload lander cannot autonomously separate from the nose cone, separation would rely on external radio commands, introducing additional points of failure such as communication losses or timing errors. If the Lander is unable to eject, the Payload mission would be unable to collect and test soil (NASA Req 4.1 -4.3). By designing the Payload to eject autonomously, the failure point of loss of communication is removed.	The payload lander will be equipped with a mechanical release mechanism triggered by a state-based command. Ground testing will verify successful autonomous separation under various conditions.	**(1) Test:** Ground testing shows autonomous ejection of the payload lander. **(2) Demonstration:** PDF verifies the autonomous ejection of the payload lander.	V	Payload Team	**(1)** Payload deployment test is described in Section 10.2.1 of FRR. **(2)** PDF is described in Section 3.

Table 6.7: Team Derived Payload Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
PF 3	The payload lander SHALL be capable of recognizing and orienting itself upright upon landing and deploying from the nosecone section.	If the payload lander cannot determine its orientation or self-right after landing, it may remain tipped, preventing deployment of the auger and resulting in inability to collect soil. Balance could also be lost during drilling operation by environmental or drag from the main parachute (DHZ 40 & 59). Providing autonomous orientation recognizing and self-righting capability ensures the payload can recover from off-nominal landing attitudes and potential losses of balance ,successfully completing its mission.	The payload lander will utilize sensors and a self-righting mechanism capable of detecting landing orientation and correcting orientation prior to payload operation. Ground testing will verify functionality.	**Test:** Testing shows autonomous ejection of the payload lander.	V	Payload Team	Payload self-righting testing is described in Section 10.2.3 of FRR. Payload lander self-righting testing is described in Section 5.2.
PF 5	The Payload SHALL have a combined weight of no more than 8.5 (lbs).	If the payload mass increases too much, the launch vehicle mass properties, stability margin, and predicted apogee will deviate from analysis, potentially leading to inaccurate apogee control, off-nominal flight behavior, or recovery performance issues (DHZ 58). Not allowing for uncontrolled mass growth ensures that simulations can be valid up until the cap set. 8.5 lbs is the maximum allowable weight for the Payload before the launch vehicle becomes over-stable, failing to comply with LVF 6	The payload will be designed using lightweight materials and components, and mass will be tracked throughout design and fabrication to ensure the total payload mass remains below 8.5 lb.	**Inspection:** The fully assembled payload mass is measured using a scale and verified to be ≤ 8.5 lb .	V	Payload Team	Payload mass is described in Section 2.2, and is shown to be under 8.5 (lbs).
PF 6	The Payload SHALL retain the soil in a contaminant free chamber for testing.	Contamination from flight residue, airborne particulates and other launch vehicle specific chemicals could alter the pH, nitrate content, or electrical conductivity of collected soil samples. This would lead to the sensors picking up false data, compromising the mission (DHZ 53, 55). By keeping the chamber free of contaminants, the risk of contamination is mitigated.	The payload soil chamber will be designed such that it minimizes contaminants.	**Inspection:** Collected soil samples are examined after retrieval and show no visible contamination or only negligible contaminant is found.	V	Payload Structures	Payload soil chamber design is described in Section 6.4.1 of FRR. Soil collection testing is described in Section 5.3.
Design Requirements							

Table 6.7: Team Derived Payload Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
PD 2	The Soil containment chamber SHALL hold a minimum of 75 (mL) of soil.	If the soil containment chamber holds at most 50 ml of soil, then any breaking or damage to the chamber or sample spillage would result in a failure to meet NASA Requirements for the amount of soil needed (NASA Req 4.1, DHZ 47). Designing the chamber to hold 150% of the required sample volume provides margin against spillage, incomplete transfer, or chamber damage.	The soil collection chamber will be designed to hold a minimum of 75 (mL) of soil.	**Inspection:** Measured internal volume of the soil containment chamber meets or exceeds 75 mL prior to integration.	V	Payload Structures	Soil collection chamber is described in Section 6.7.1. Soil collection testing is described in Section 5.3.
PD 3	The interface between the Payload and the nosecone SHALL include features to ensure smooth ejection.	If the Payload-nose cone interface does not provide alignment during separation, the payload may experience binding, jamming, or payload components shearing off, preventing the lander from separating completely and resulting in a failure to complete the mission (DHZ 46,48). Incorporating alignment features such as guides or rails ensures repeatable ejection from the nose cone and directly mitigates hazards.	The payload-nose cone interface will incorporate alignment features such as guides or rails to constrain motion during ejection. Ground testing will be conducted to verify consistent, smooth separation without binding or misalignment.	**(1) Test:** Ground testing demonstrates that the payload ejects smoothly and consistently with no binding, misalignment, or lateral interference across all tested scenarios. **(2) Demonstration:** Payload Demonstration Flight (PDF) confirms smooth payload ejection upon landing under flight conditions.	V	Payload Structures	**(1)** Payload lander ejection test is described in Section 10.2.1 of FRR. **(2)** PDF is described in Section 3, detailing Payload ejection.
PD 4	The payload SHALL log timestamps of all operations for NASA verification as well as post launch analysis.	If payload operations are not time-stamped, it becomes difficult to correlate behavior with launch vehicle flight states, making it challenging to verify correct performance from the system. Additionally, NASA requires time stamps during solid collection (NASA Req 4.2.1). Logging timestamps for all major payload operations ensures traceability, verification and troubleshooting, and mitigates these hazards by providing evidence of payload behavior.	Payload software will integrate sensors and internal timing functions to record time-stamped logs for all major payload events, including state transitions, deployment actions, and sample collection. Logged data will be stored onboard and retrieved post-flight for analysis. Ground testing will validate timestamp accuracy and reliability.	**(1) Test:** Ground testing confirms the payload logs accurate timestamps for all commanded operations. **(2) Demonstration:** Payload Demonstration Flight (PDF) data confirms that all critical payload operations are logged with timestamps during flight.	V	Payload Team	**(1)** Payload software testing is outlined in Section 6.5 of FRR. **(2)** PDF is described in Section 3, detailing all timestamps during flight.

Table 6.7: Team Derived Payload Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
PD 5	Auger operation SHALL be controlled such that jamming and over-torquing is negligible.	If the auger experiences jamming or excessive torque during operation, the auger bit could break, gear components may deform or fail, all preventing soil collection and resulting in mission failure (DHZ 41, 44). By Preventing such jamming and over-torquing, these risks are mitigated.	The auger system will be designed such that jamming and over-torquing is negligible.	**Test:** Ground testing demonstrates the auger operates without jamming or over-torquing across representative soil conditions and load cases.	V	Payload Team	Payload operation testing schedule is outlined in Section 10.2.2 of FRR. Soil collection testing is described in Section 5.3.
Environmental Requirements							
PE 1	No components of the payload SHALL be released into the environment.	If Payload components are released into the environment, they pose a risk as pollution and can be harmful to vegetation and wildlife (EHZ 2). By ensuring that all payload components are secured inside the launch vehicle, pollution is mitigated.	The payload will be designed such that components are securely attached to the launch vehicle.	**(1) Test:** Ground testing confirms that all payload components remain securely attached under representative handling, deployment, and operational conditions. **(2) Demonstration:** Payload Demonstration Flight (PDF) confirms that no payload components are released into the environment during flight or payload operation.	V	Payload Team	**(1)** Payload operation testing is outlined in Section 10.2.3 of FRR. **(2)** PDF is described in Section 3, confirming no payload components are released into the environment.
PE 2	The payload lander SHALL be capable of up-righting on soil conditions ranging from dry loose sand to damp compacted dirt.	If the payload lander cannot self-right across the range of soil conditions present at the launch site, it may remain tipped or unstable after landing depending on the location of landing site. This would result in mission failure, as the inability to self-right prohibits the ability of the lander to collect soil (DHZ 40). Designing the self-righting mechanism to function across a range of soil types mitigates these hazards.	The self-righting mechanism will be designed and tested to operate on multiple soil types, including dry sand, loose dirt, and damp compacted soil. Ground testing will simulate a variety of conditions to verify functionality.	**Test:** Ground testing demonstrates that the payload lander consistently self-rights from different initial orientations across different soil conditions without manual assistance.	V	Payload Team	Payload self righting tests are described in Section 5.2.
Safety Requirements							

Table 6.7: Team Derived Payload Requirements (continued)

ID	SHALL Statement	Justification	Planned Action	Verification Method	Status	Performing Subsystem	Results
PS 1	All Payload systems SHALL be designed and fabricated more than 24 hours before planned Launch.	If payload fabrication or assembly is completed too close to launch, there may be insufficient time to verify fit, identify integration issues, correct defects, or address safety concerns, increasing the likelihood of rushed assembly, human error, or unverified payload behavior (PHZ 56, DHZ 48, 56, 58). By Ensuring Payload systems and designed and fabricated a minimum 24 hours before launch, time is left to access and fix any potential integration problems that may arise.	The payload will be completed, fully assembled and dry fitted at least 24 hours before launch.	**Inspection:** Payload is fully designed, fabricated, and assembled > 24 hours before launch.	V	Payload Team	Payload was fully manufactured before PDF, described in Section 3.
PS 2	The payload lander SHALL not create pinch points that could injure personnel during handling or assembly.	If exposed pinch points exist on the payload lander, personnel may suffer hand or finger injuries during assembly, handling, transport, or field operations, particularly when interacting with moving mechanisms such as the self-righting system or auger (PHZ 7, 9, 34). By ensuring the Payload does not include pinch points, risk of damage to personnel is mitigated.	The payload lander and assembly checklist will be designed to eliminate exposed pinch points.	**Inspection:** No pinch points are present in the self-righting mechanism during assembly or handling.	V	Payload Structures	Payload was fully manufactured before PDF, described in Section 3.
PS 3	The Auger bit SHALL be retracted until needed after launch.	If the auger bit is exposed prior to intended deployment, it may present a laceration or puncture hazard to personnel during handling, transport, and assembly (PHZ 30). By Ensuring that the Auger bit is retracted until needed, risk to personnel is mitigated.	The auger system will be designed such that the auger bit is fully stowed until commanded for deployment.	**Demonstration:** Observation confirms the auger bit remains fully retracted during handling, integration, flight, landing, and recovery, and only deploys when commanded to.	V	Payload Team	Soil collection testing is described in Section 5.3.

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