

Systems Integration and Test of the Lunar Flashlight Spacecraft

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ABSTRACT

Lunar Flashlight is a 6U CubeSat launching in late 2022 or early 2023 that will search for surface water ice content in permanently shadowed regions at the south pole of the Moon using infrared relative reflectance spectroscopy. The mission will act as a technology demonstration of an Advanced Spacecraft Energetic Non-Toxic (ASCENT) green propulsion system and active laser spectroscopy within the CubeSat form-factor. This paper provides an overview of the entire Systems Integration and Test campaign which took place at the Jet Propulsion Laboratory and the Georgia Institute of Technology. From initial testing of the isolated avionics and payload subsystems to the final tests with a fully integrated spacecraft, the project's integration and test campaign is reviewed, with a focus on lessons learned.

PROJECT OVERVIEW

Lunar Flashlight is a 6U CubeSat developed by the NASA Jet Propulsion Laboratory (JPL) in conjunction with the NASA Marshall Space Flight Center (MSFC), the University of California, Los Angeles (UCLA), the Georgia Institute of Technology (GT), and the Georgia Tech Research Institute (GTRI). This mission will demonstrate the capabilities of green monopropellant propulsion systems and near-infrared laser spectroscopy within the CubeSat form-factor. The propulsion system, developed by GT and MSFC,¹ will complete a lunar orbit insertion (LOI) and multiple trajectory correction maneuvers (TCMs), placing the 14 kg spacecraft in a near-rectilinear halo orbit around the Moon. Once in lunar orbit, four near-infrared lasers will detect and map, via relative reflectometry, water ice in permanently-shadowed regions near the lunar south pole.² This project addresses one of NASA's Strategic Knowledge Gaps: the composition, quantity, distribution, and form of water/H species and other volatiles associated with lunar cold traps.³

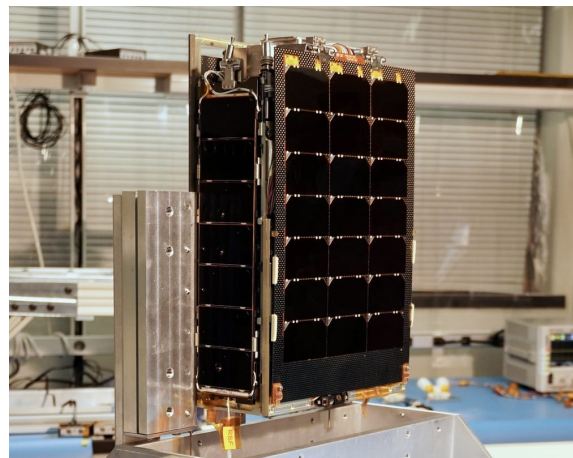


Figure 1: Fully Integrated Spacecraft

Spacecraft Overview

The 6U CubeSat has a mass of approximately 14 kg after fueling, and includes power, command and data handling, communications, attitude control, propulsion, and payload subsystems (Figure 2).

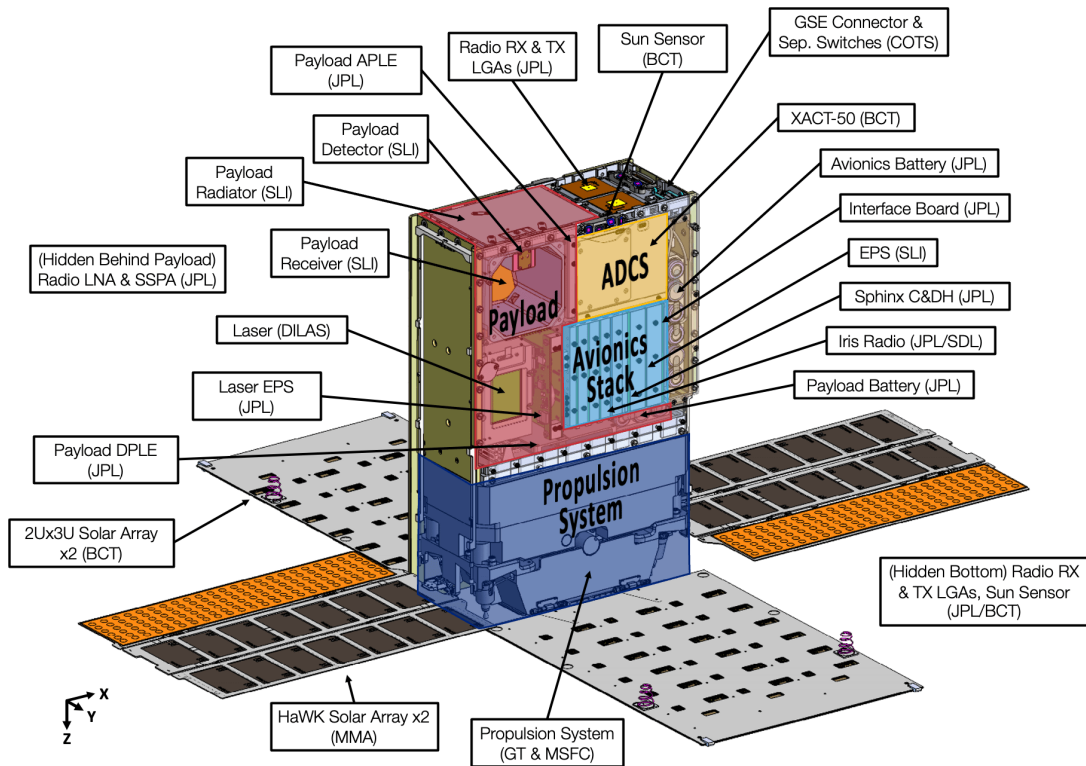


Figure 2: Overview of the Lunar Flashlight 6U CubeSat

The power subsystem includes four solar arrays developed by Blue Canyon Technologies (BCT) and MMA, an Electrical Power Systems (EPS) management board, and a battery built with Panasonic NCR18650B cells.⁴ The arrays are capable of providing over 55 W at end of life.

Command & Data Handling (C&DH) is provided by a JPL-developed Sphinx single-board computer, which includes a GR712RC radiation-hardened microprocessor and a ProASIC3 FPGA. The flight software utilizes JPL's F Prime framework.⁵

The spacecraft uses an Iris Radio, a small satellite transponder developed by JPL and built by Utah State University's Space Dynamics Laboratory.⁶ A pair of low-gain antennas sits at each end of the spacecraft on the Z-axis, providing transmit and receive capability independent of the spacecraft's orientation.

The attitude determination and control system (ADCS) for the spacecraft is provided by a BCT XACT-50. It utilizes sun sensors mounted around the spacecraft as well as an internal star tracker and three internal reaction wheels.⁷

The Lunar Flashlight Propulsion System (LFPS) was designed and built by Georgia Tech's Glenn Lightsey Research Group in collaboration with the NASA Marshall Space Flight Center.⁸ It can pro-

vide over 3300 N·s of impulse, allowing for lunar orbit insertion, correctional maneuvers, and reaction wheel desaturation during the mission.⁹

Lunar Flashlight's science instrument payload is a compact Short Wave Infrared (SWIR) Laser reflectometer designed to find signatures of water ice in the lunar regolith. Diode lasers emit infrared energy at four different wavelengths sequentially. This light reflects off the lunar surface and is picked up by the onboard paraboloidal receiver mirror which collects the incoming light onto an InGaAs detector.¹⁰ Differences in the detector's amplitude across the sequence can be used to identify the water ice signature.² Due to high power requirements, the lasers are powered from a separate battery built with Sony NCR18650B cells.⁴

SI&T Overview

The goal of the Systems Integration and Test (SI&T) process was to integrate each subsystem into a single cohesive system, build the spacecraft for flight, and confirm its capabilities through an extensive suite of tests. Some subsystems were delivered ready for integration. Other subsystems, such as the payload, required further development by the SI&T team to meet all functional requirements and inter-

face properly with other subsystems. While initial activities took place at JPL, Georgia Tech was selected to finish the SI&T process, as well as perform Mission Operations. This gave an opportunity for the Operations team to get familiar with the spacecraft prior to launch.

Mission Assurance

Lunar Flashlight is classified by NASA as a “technology demonstration” payload, governed by NASA 7120.8, and classified by JPL as a “Type II” project. These classifications helped to shape the project’s Safety and Mission Assurance Plan (SMAP) as well as the Quality Assurance (QA) Requirements Tailoring Agreement (QARTA).

Various internal tools were utilized at JPL to track the state of hardware during SI&T, including the Instructions/Procedures for Build, Assemble, and Test (IBAT/PBAT) systems, the Quality Assurance Reporting System (QARS), and the Problem Reporting System (PRS). DOORS NG software was used to track requirements.

During final SI&T activities at JPL, QA often acted as a witness to final mates and torques as well as staking of flight hardware. However, the QARTA allowed for peer witness verification of many activities. The JPL Quality Critical Items (QCI) classification enforces additional QA oversight, but as only a few flight hardware items were designated QCI, oversight requirements were limited.

JPL Mission Assurance personnel surveyed GT’s facilities for flight SI&T readiness prior to delivery of the upper spacecraft. Once integration began at GT, JPL Mission Assurance and QA continued to support the project in an oversight role, participating in daily tag-ups, activity planning, and reviews. QA monitored critical activities with a mix of in-person and remote video support.

Integration Approach

Integration of the upper spacecraft took place in the JPL CubeSat Lab cleanroom, while other integration activities were primarily done in GTRI’s C-SHAFT cleanroom. Pre-approved work procedures and a “buddy system” were required for all activity involving flight hardware. Integration work was logged in the IBAT system or in procedure archives, and was captured through extensive photography.

Throughout the SI&T process, the team drew on JPL’s in-house Electronic Manufacturing, Packaging & Tech Services section to manage and execute complex board reworks. For other reworks, the project worked directly with JPL technicians, providing

Cognizant Engineer (CogE) and QA-approved build instructions to the technicians. For mechanical manufacturing and rework activities, the team utilized JPL’s Mechanical Fabrication & Test section. Premixed epoxies and other polymerics were provided by JPL’s polymerics team.

The team relied on a JPL flight mechanical assembly technician to help build the mechanical assemblies for flight. This technician brought invaluable knowledge, skills, and tools that allowed the team to resolve a number of setbacks.

Once at Georgia Tech, GTRI technicians, students, and engineers performed most integration activities. JPL mechanical engineers were present to support installation of the propulsion system and solar panels, and JPL maintained an on-site SI&T lead as final integration and test activities were performed. Additionally, the JPL mechanical assembly technician flew to Atlanta to support certain intricate integration tasks.

Test Approach

As with integration procedures, all tests involving flight hardware required pre-approved procedures and use of the buddy system. Procedures were approved by the deputy project manager, QA, and, when applicable, component CogEs. The team was particularly diligent in non-cleanroom environments to mitigate contamination risks and keep the hardware covered as much as possible.

JPL’s SI&T team utilized L^AT_EX to write procedures. This allowed for easy version-control, formatting consistency, and central updating of subprocedures. Multiple test procedures were developed over time into a single “choose your own adventure” full functional test procedure. As configurations evolved, operators could indicate which components were in a test setup, then navigate to the relevant sections to set up and test the equipment.

Georgia Tech was responsible for authoring and executing most procedures at GT. Joint Test Readiness Reviews (TRRs) were hosted for each activity, allowing for feedback from JPL project management, mission assurance, CogEs, and flight engineers. These procedures referenced the JPL SI&T-authored full functional test procedure when completing standard tasks, such as power-on, charging, and software updates.

iPads were used extensively for running procedures at JPL and GT. This approach eliminated paper clutter, added the ability to seamlessly drop in photographs, made edits and color-coded notes easy to add, and sped up the archival and sharing process.

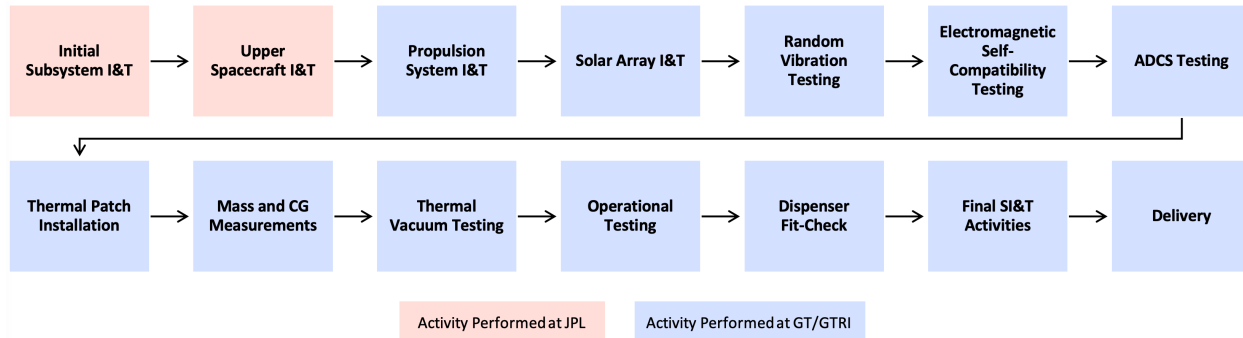


Figure 3: Lunar Flashlight SI&T Activity Flow

INTEGRATION AND TEST ACTIVITIES

The Lunar Flashlight Systems Integration and Test campaign began at JPL in 2018 with initial work at the subsystem level. The team followed the flow seen in Figure 3, finishing in 2022 at GT with the delivery of the fully-built and tested spacecraft.

Initial Subsystem I&T

Prior to assembling the spacecraft for flight, an array of integration and test activities were performed with each subsystem. Early in the project, standalone electrical tests took place with delivered components. Later on, as the system became more stable and integrated, more complex system-level functional tests were performed.

Initial Avionics Power-On Checkout

Testing of all components began with simple electrical integration, power-on, and telemetry checkout tests. A “safe-to-mate” was performed on all electrical interfaces, where characteristics of each pin were measured with a digital multimeter. Once SI&T was satisfied with the results of these tests, the component was integrated into the system. In the primary avionics workflow, SI&T built up the avionics assembly with the C&DH board, EPS board, radio, and interface board. Figure 4 shows one step of this “avionics stack” assembly process.

During the initial avionics integration process, the project was able to leverage the commonality of the C&DH, EPS, and Iris Radio with the Near-Earth Asteroid Scout (NEA Scout) mission, which also shared some personnel at JPL. For example, when NEA Scout found a floating signal between C&DH and EPS boards causing unexpected system resets, Lunar Flashlight was able to implement the hardware fix before beginning testing.

Power-on and setup steps developed in support of this early testing were passed into the full functional test procedure. Additionally, as test data was accumulated, this procedure was updated to inform testers of more accurate expected values and power consumption profiles.

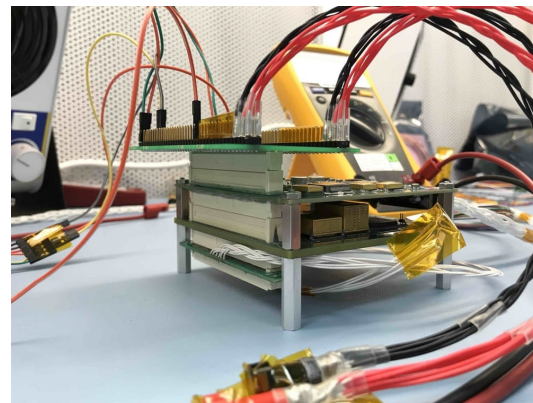


Figure 4: Avionics Stackup Test with EPS and Sphinx C&DH Boards

Deep Space Network RF Compatibility Test

A DSN RF compatibility test was conducted at the Peraton DTF-21 facility near Los Angeles and was the only opportunity to test the spacecraft’s FSW → C&DH → Iris transceiver → Iris TX/RX amplifier signal chain connected directly to an actual DSN receiver. It also served as an opportunity for the ground data systems team to test their interface to the DSN. As the test occurred prior to much of the avionics stack being ready for integration, the test was conducted in a flat-sat setup consisting of only the C&DH board and Iris Radio system. Testing took place with coaxial wired connections. All flight uplink and downlink configurations were tested, as well as signal lock time and ranging modes.

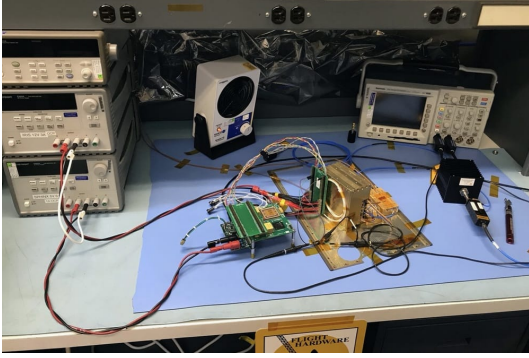


Figure 5: C&DH and Radio Setup for DSN Compatibility Test

Payload End-To-End Tests

As part of the payload’s subsystem-level testing, four payload end-to-end (ETE) tests of increasing complexity were conducted. The system configuration evolved from test to test, moving from a “flat-sat” payload configuration in the first test to a fully assembled payload/avionics setup in the final test. These tests verified a variety of payload performance details, such as power draw characteristics of the lasers and performance of the payload battery. Thermal performance was also tested, verifying the thermal control of the Phase Change Material (PCM) ($+18\pm 1^\circ\text{C}$) and radiator ($-61.0\pm 0.5^\circ\text{C}$) with on-board heaters.

Response time and noise performance of the detector’s analog front-end measurement chain were iteratively tested and improved.¹¹ Data was collected to quantify the stability of the laser’s power supply, output power, and output spectrum with respect to laser temperature over multiple 90-second laser-firing sequences. The ETE tests also provided an opportunity to control the payload through the flight avionics interface and uncover any operational idiosyncrasies. “Experiment data” is captured at 25 kHz during laser firing. The procedure for capturing, downlinking, and processing this data was tested extensively.

For these tests, hardware was placed in a thermal vacuum chamber. A two-stage cryochiller was thermally connected to the receiver housing and radiator using copper sheets, and a recirculating chiller was connected to the PCM using a plumbed copper block. Heaters were attached to the copper sheets to keep temperatures near the target zones.

One challenge the team faced was avoiding detector measurement saturation while still firing the 75 W lasers at full power. To accomplish this, a diffuser was placed in the beampath of the laser, and

a temporary cover outfitted with a pinhole slot was installed over the receiver window.

These tests verified the reliability and performance of the payload, demonstrating a Laser Electronic Power Supply (LEPS) capable of outputting 45 A continuous current for 90 seconds and Analog Signal Front-End Electronics (ASFE) capable of measuring a photodetector current input range of 0 nA – 10 nA with RMS noise of 6.9 pA.

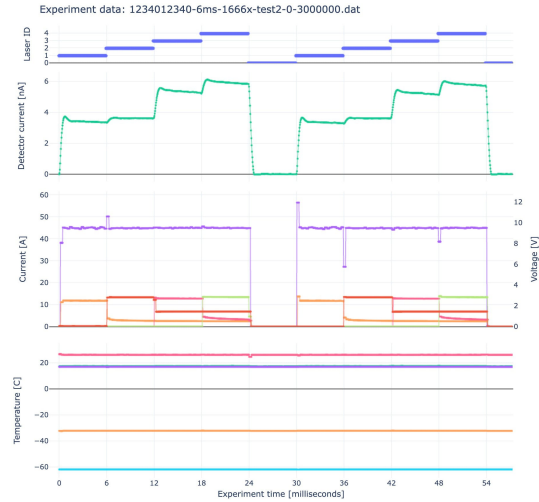


Figure 6: Example ETE Experiment Data

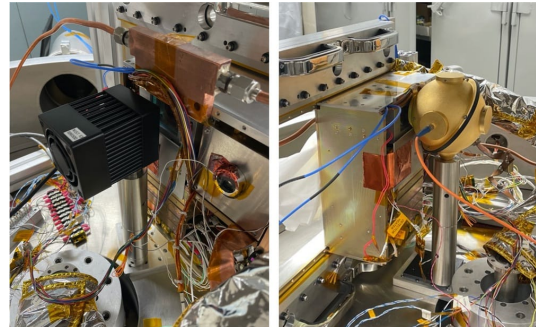


Figure 7: Laser Output Power (Left) and Laser Output Spectrum (Right) Measurements

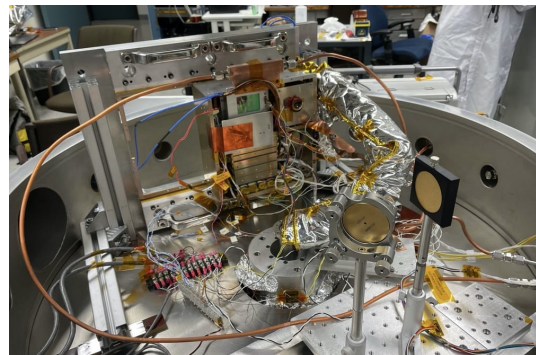


Figure 8: Payload End-To-End Test 4 Setup

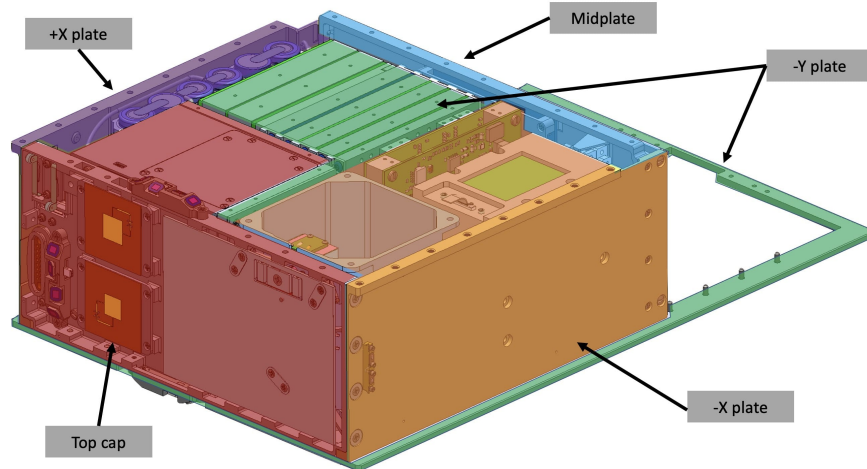


Figure 9: Upper Spacecraft Assembly with the Five Primary Sub-Assemblies Highlighted

Upper Spacecraft I&T

The upper spacecraft consists of the C&DH, avionics battery, payload, and most of the ADCS and radio components. The only components not included in the upper spacecraft are the propulsion system, solar array assemblies, and some radio, power, and ADCS components that mount to the propulsion system. The upper spacecraft is comprised of five primary mechanical subassemblies: the $-Y$ plate, the $+X$ plate, the $-Z$ plate (aka “top cap”), the $-X$ plate, and the “midplate”. Figure 9 shows the locations of each sub-assembly. Each sub-assembly was built separately, then the four “walls” were assembled onto the $-Y$ plate. Positioning pins ensured that the cornerstone of the spacecraft was always established in the $+X / -Y / -Z$ (top cap) corner. After this reference was established, the rest of the subassemblies could be torqued down securely. Figure 10 through Figure 12 show how these sub-assemblies came together for the final installation.

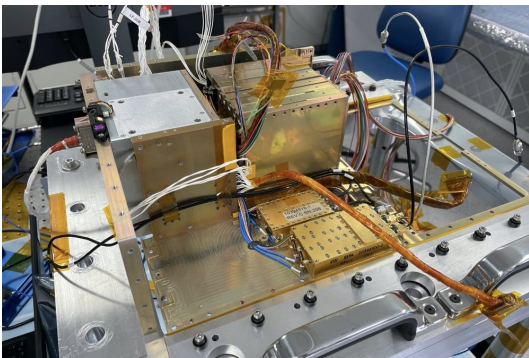


Figure 10: Top Cap and $-Y$ Plate Sub-Assemblies During Integration

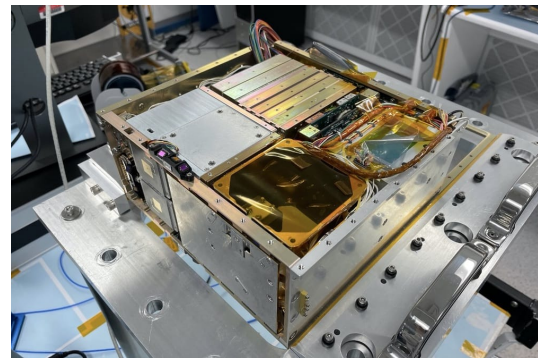


Figure 11: Integrated Upper Spacecraft Prior to $+Y$ Plate Installation

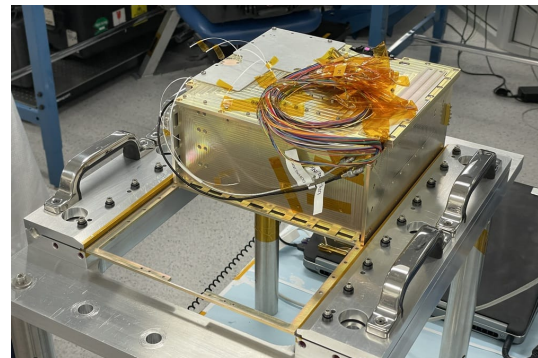


Figure 12: Integrated Upper Spacecraft with the $+Y$ Plate Installed for Transport to GT

Propulsion System I&T

The Lunar Flashlight Propulsion System was designed and assembled at Georgia Tech,⁹ and it was the first subsystem to be integrated into the spacecraft following the arrival of the upper spacecraft to

GTRI's C-SHAFT facility. Integration was followed by functional and thermal performance testing of the propulsion system.

Propulsion System Integration

The LFPS integration process began with initial electrical compatibility tests prior to shimming and mating with the upper spacecraft. After performing a standalone functional test of the propulsion system along with a safe-to-mate procedure, the LFPS was electrically mated to the spacecraft for an end-to-end electrical test (Figure 13). Once communication was established, the spacecraft C&DH system could successfully command the LFPS and receive telemetry using the XACT as a relay.

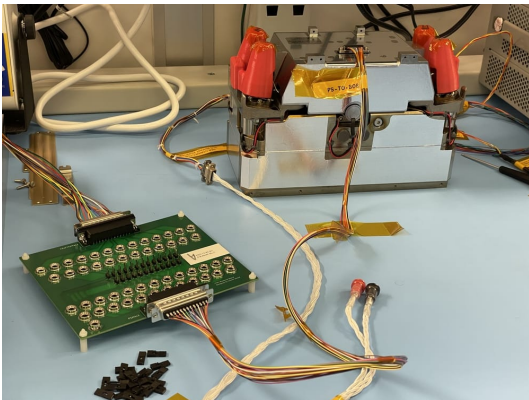


Figure 13: Initial End-To-End Electrical Test of Prop System with Spacecraft

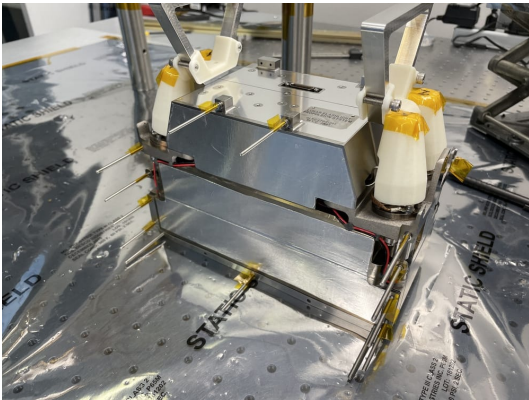


Figure 14: Handles Installed on the LFPS with Shims Loaded and Ready for Integration

Installation of the LFPS into the spacecraft required the selection and installation of shims, which provided a precise fit and alignment. Different shims were tested until an optimal fit was found. Threaded rods were used to hold the shims and align the propulsion system with the upper spacecraft while it

was lowered onto the upper spacecraft chassis (Figure 14). After fully lowering the LFPS onto the chassis (Figure 15), the rods were replaced with screws secured with Loctite 242. Additional shims were added at the other interfaces between the LFPS and the upper spacecraft, then screws were installed at those interfaces.

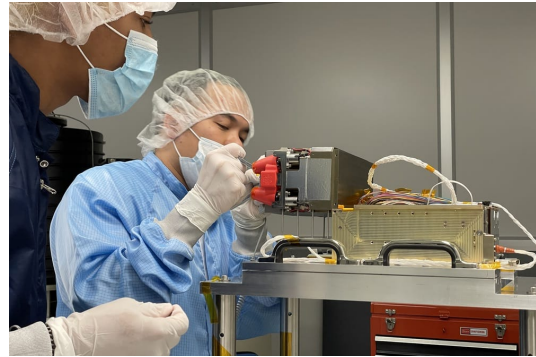


Figure 15: Lowering the Propulsion System Onto the Spacecraft Using Threaded Rods

Following propulsion system integration with the upper spacecraft, two X-band patch antennas, a sun sensor, and an RBF power inhibit assembly were installed on the +Z face of the LFPS (Figure 16).

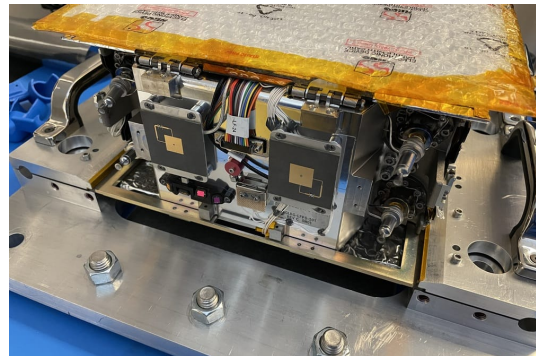


Figure 16: Peripherals Installed on LFPS

The non-flight handles (Figure 14) used for maneuvering the LFPS became the primary spacecraft handling feature following integration. These were used to help move the spacecraft inside of the "Dreadnaught", a multi-use fixture provided by the NEA Scout project. They were also used when moving the spacecraft between various fixtures.

Propulsion System Tests

The propulsion system underwent a variety of functional testing following integration. First, a standardized functional test was developed as part of the larger spacecraft full functional test procedure. This test verified two-way communication between the C&DH system and the LFPS through

the XACT, verified temperature and pressure sensor feedback, exercised all heater channels at safe temperatures, and optionally exercised the valves.

Second, a thruster preheat test was developed to test the performance of the heaters in the integrated system. The LFPS thrusters must preheat to $> 380^\circ\text{C}$ before they can begin maneuvers. This preheat operation is one of the most power-intensive operations performed by the spacecraft and the warm-up curve depends on the voltage drop of the integrated system, which was unknown prior to integration. Identifying this curve was critical for understanding the operational power budget and for planning sequences.

The test was performed while the spacecraft was under vacuum, at both a hot and a cold plateau of the TVAC test (Figure 17), as well as after a thruster was replaced. This test mirrored thruster preheat testing performed during acceptance testing of the propulsion system with commanding and power now being provided through the spacecraft. The preheat was successfully completed within the power requirements at both the hot and cold TVAC plateaus. The cold test struggled to reach the desired initial firing temperature, which was due to two non-flight-like conditions. First, the propulsion system will be sun-pointing during preheat, so the system is expected to be much warmer. Second, the externally supplied current was only 2.5 A rather than the 4.7 A expected from the solar arrays when preheating during flight. The higher current will translate to a higher input voltage and higher preheat power during flight.

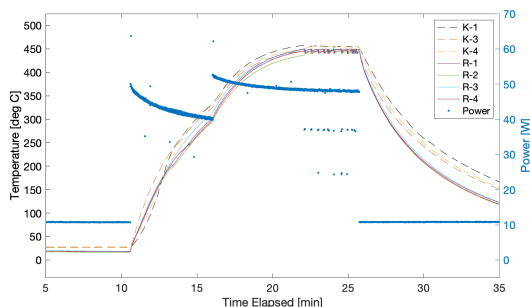


Figure 17: Thruster Preheat Results During TVAC Hot Dwell

Finally, a thruster valve commandability test was performed. This showed that, for each valve commanded by the XACT, the appropriate valve on the propulsion system was actuated. First, each thruster nozzle was sealed with Kapton tape. Then, with the propellant tank isolation valve open, each thruster valve was individually actuated via an XACT ManualBurn command. Helium, which was left in the propellant tank to ensure a positive pressure, would es-

cape through the nozzle under test, pressing against the tape to create a visual indicator of the active thruster valve. The location of each indicating valve was compared with the XACT ICD, which confirmed the proper valve was actuating.

Thruster Replacement

After the propulsion system had been integrated into the spacecraft, while System I&T continued at GT, a parallel qualification program on spare thrusters was underway at MSFC. One of the spare thrusters failed a hot fire test after completing random vibration testing, which revealed that a feed tube configuration in one of the flight thrusters was likely to develop a leak during flight. After several months of additional analysis, testing, risk assessments, and discussions, the project decided to replace the affected flight thruster.

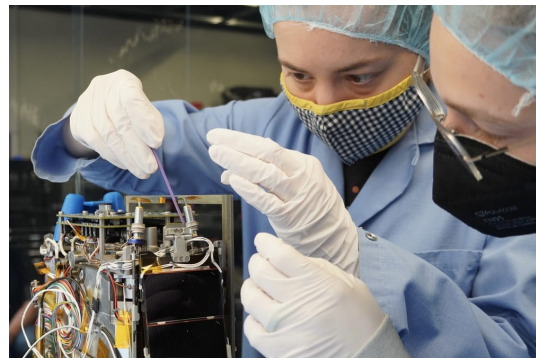


Figure 18: Removal of Thruster

Replacement of the thruster was a collaborative effort between MSFC, the thruster vendor, GT, and JPL. All peripheral hardware attached to the “muffin tin” (which covers the LFPS electronics) was carefully removed. This included the +Z Iris Radio antennas, +Z sun sensor, BCT solar panels, and RBF power inhibit assembly. Then, the $-Y$ backplate was gently pulled back to allow the muffin tin to be extracted, allowing access to the thruster and its electrical connections. Staking had been used on the thruster heater and thermocouple connectors, which had to be carefully removed. After this, the thruster fasteners were removed and the thruster was taken off of the LFPS (Figure 18).

The new thruster was installed and a flow test was performed with helium to verify the expected flow through the thruster (Figure 19). The system was then reassembled, first with installation of the muffin tin, followed by installation of peripheral hardware and solar panels. To mitigate workmanship risks as the spacecraft had already completed random vibration testing, staking was applied to all

relevant fastener heads as an additional locking measure. Finally, the fully reassembled spacecraft was placed in a vacuum chamber where a full thruster preheat cycle was successfully performed.

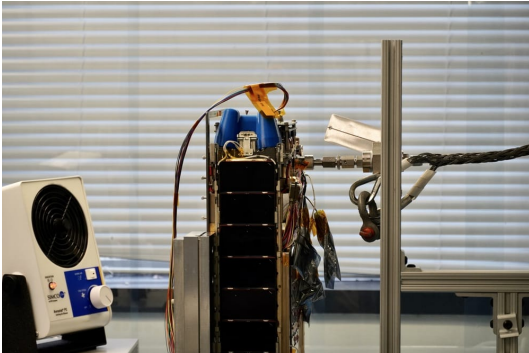


Figure 19: Flow Test with New Thruster

Solar Array I&T

Following integration of the propulsion system into the spacecraft, the four solar arrays were integrated and then tested for functionality.

Solar Array Integration

The two BCT arrays were installed on the +Y and -Y faces of the spacecraft. Non-flight screws were installed opposite the spring-loaded brackets to keep each BCT panel stowed against the side of the spacecraft while it was outside of the dispenser.

The two MMA tri-fold arrays were installed on the +X and -X spacecraft faces. This process included the installation of a spring-loaded burn-wire-triggered launch cage for stowing the panels and the application of Kapton-covered padding for protecting the stowed panels. As part of the solar array installation process, the pigtail harnesses on all four arrays were cut to size and soldered into Cristek Micro Strip connectors (Figure 20).

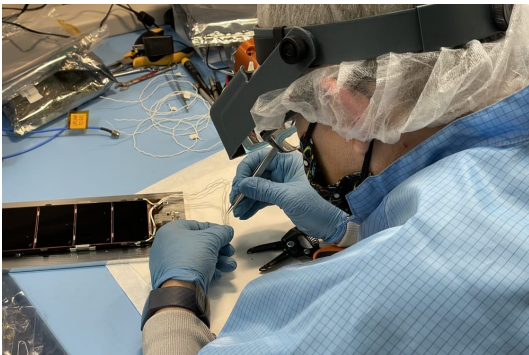


Figure 20: Solar Array Harnessing

To protect the BCT solar arrays during dispenser installation, launch, and deployment, snubbers and bumpers were installed. Snubbers are silicone pads mounted to the spacecraft chassis under the stowed arrays to dampen vibrations and prevent contact between the panels and the spacecraft chassis.¹² Arrays of snubbers were installed on the +Y and -Y faces of the spacecraft using Momentive RTV566 silicone sealant (Figure 21).

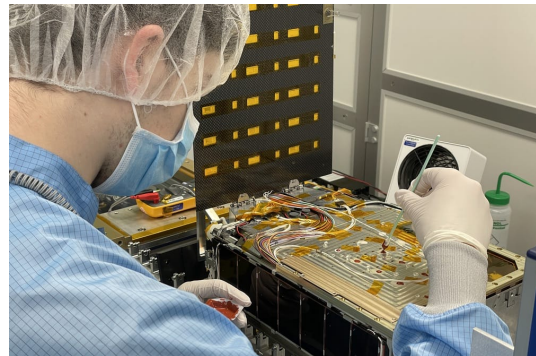


Figure 21: Installation of Solar Array Snubbers

Bumpers are 3D-printed ULTEM spacers that keep the solar arrays from contacting the dispenser (Figure 22). Optimal locations for the bumpers were determined by sliding the spacecraft in and out of the dispenser-emulating segment of the Dreadnaught fixture. They were adhered to the sides of the BCT arrays using Henkel 9394.

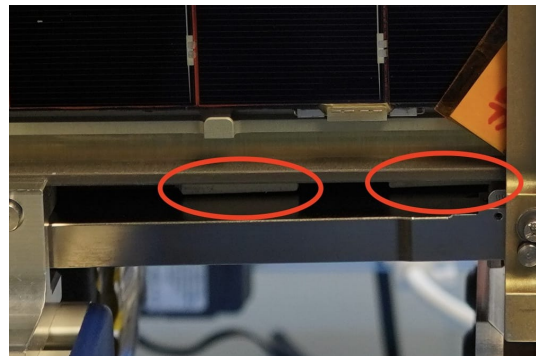


Figure 22: Bumpers Keep the Solar Cells From Touching the Dispenser Door

Solar Array Tests

The Gravity Offload Fixture (GOLF) provided by the NEA Scout project was used to perform solar array deployment tests (Figure 23). This fixture uses constant-force springs to counteract gravity, allowing the solar array springs to deploy the panels vertically in a simulated zero-g setup. For the MMA

tri-fold panels, burn wires were activated by an operator, and each mechanism successfully demonstrated an appropriate melting of the melt rod and subsequent deployment. For the BCT panels, the GOLF included two motor-actuated brackets that held the panels in a stowed position. Upon manual activation of the motors, the brackets retracted, simulating deployment from the flight dispenser. This test also demonstrated a successful deployment.

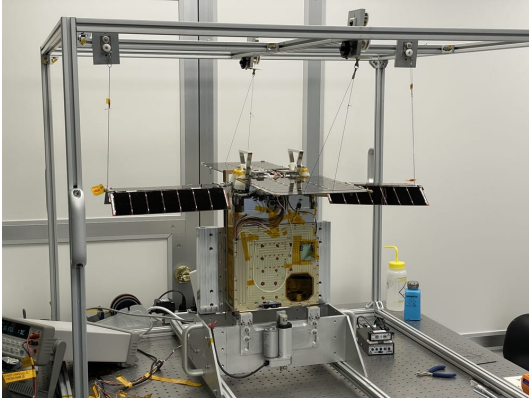


Figure 23: Solar Array Deployment Test Performed with Gravity Offload Fixture

Following the final integration of solar arrays, a charging test was performed with a 500 W lamp to ensure that the spacecraft EPS could receive power from each array (Figure 24). This test successfully demonstrated functionality of the solar arrays, capturing approximately 100 mA from each MMA array and 250 mA from each BCT array.

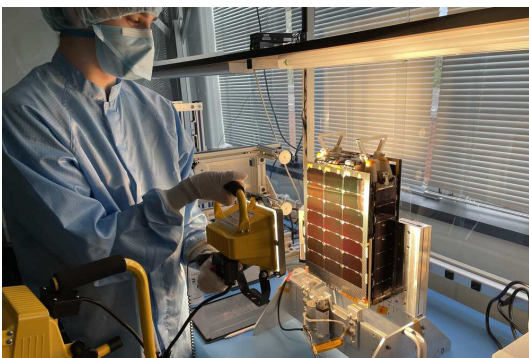


Figure 24: Illuminating the +Y BCT Panel

Random Vibration Testing

A random vibration environmental test was performed to show that the spacecraft, as designed and built, can survive the vibration loads expected during launch and function as required after launch to complete the mission. The test profile for each axis was determined by a JPL dynamics engineer

based on launch loads and other requirements from the launch provider. For Lunar Flashlight, a vibration analysis was initially performed assuming the NASA Space Launch System (SLS) vehicle. When the project switched to the SpaceX Falcon 9 launch vehicle, the new vibration profiles were enveloped by the previous SLS profiles. This simplified test preparations, as the previous analysis had already shown structural compatibility with higher vibration levels.

The random vibration test was performed at GTRI's Cobb County Research Facility (CCRF), following the test plan in Table 1 (Figure 25). The fixture used for holding the spacecraft during random vibration was shared by the NEA Scout project, as both projects used the same launch dispenser.

Table 1: Random Vibration Test Plan

Test	Description
1	Hardware Functional Test
2	X-axis Resonance Search
3	X-axis Random Vibration Test Profile
4	Rotate Spacecraft
5	Z-axis Resonance Search
6	Z-axis Random Vibration Test Profile
7	Reconfigure Shaker for Vertical Test
8	Y-axis Resonance Search
9	Y-axis Random Vibration Test Profile
10	Hardware Functional Test

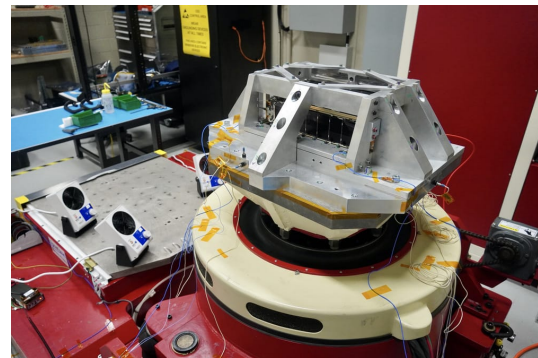


Figure 25: Random Vibration Z-Axis Setup

Before and after the random vibration test, laser alignment tests were performed to verify performance of the laser and confirm no major alignment changes occurred during the vibration test. For the first test, the laser was set up to face an off-axis parabolic mirror. This mirror was then adjusted to face a gold-coated diffuse reflectance target. Finally, short laser firing sequences were run, and neutral-density filters were iteratively added in front of the receiver to attenuate the detector signal until non-saturated values were recorded.

The post-vibration laser alignment test setup mirrored the pre-vibration setup including the final neutral-density filter configuration. New measurements were compared with the final pre-vibration measurements, confirming minimal changes in laser performance and alignment. As part of the second test, the laser output was also captured with a FLIR SC6100 thermal camera (Figure 26).

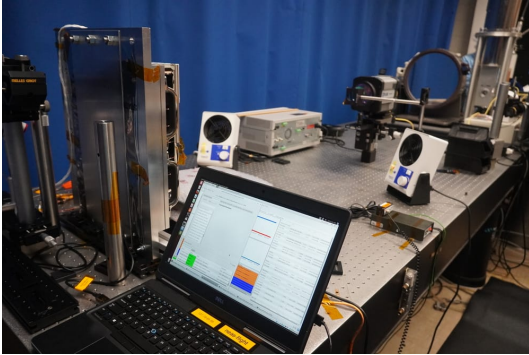


Figure 26: Laser Thermal Image Capture

Electromagnetic Self-Compatibility Testing

An electromagnetic self-compatibility test was performed in an anechoic chamber at GTRI’s CCRF location (Figure 27). Lunar Flashlight has two pairs of planar patch antennas, one on the propulsion system (+Z) side and one on the payload radiator (−Z) side. This test was designed to confirm the capability of the spacecraft to communicate using each pair of antennas. For each antenna pair, a “plugs-in” test was performed, where operators adjusted the spacecraft and ground support equipment (GSE) settings to achieve a reliable RF lock on uplink and downlink channels. This was followed by a “plugs-out” test, where operators disconnected the spacecraft from external power and wired communications, and used the settings determined previously to communicate wirelessly with the spacecraft.

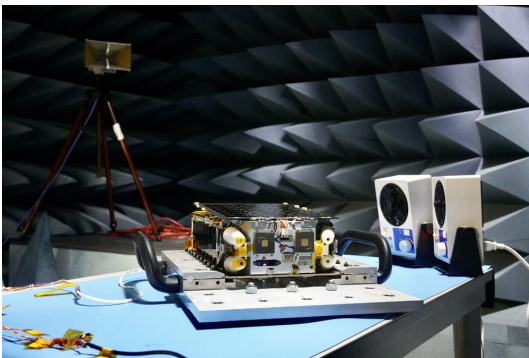


Figure 27: Electromagnetic Self-Compatibility Test Setup

By following the test plan in Table 2, this test successfully showed that the spacecraft could communicate wirelessly in full-duplex using each pair of antennas while fully disconnected from ground support equipment (GSE).

Table 2: Electromagnetic Self-Compatibility Test Plan

Test	Description
1	“Plugs-In” +Z antenna test to verify setup.
2	“Plugs-Out” +Z antenna test, with checkout of payload and propulsion system. (No external power or wired UART)
3	Rotate spacecraft 180° for -Z test.
4	“Plugs-In” -Z antenna test to verify setup.
5	“Plugs-Out” -Z antenna test, with checkout of payload and propulsion system. (No external power or wired UART)

ADCS Testing

After final integration of the XACT and its external sun sensors, tests were performed to ensure each XACT component was integrated and operating correctly.

The XACT utilizes sun sensors to keep the spacecraft’s solar arrays sun-pointing by default, ensuring the spacecraft batteries remain charged. The functionality of these sensors was tested after installation to ensure proper connection, orientation, and operation (Figure 28). This test utilized a 500 W work lamp as a basic solar simulator.

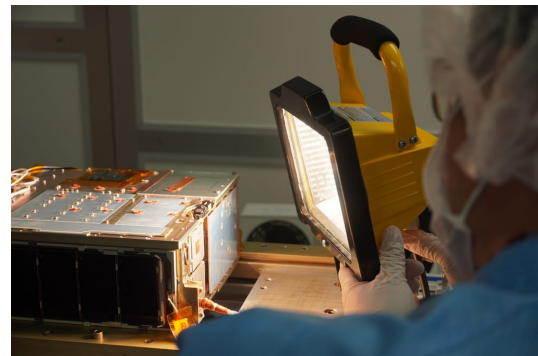


Figure 28: Test of the +Y XACT Sun Sensor

On Lunar Flashlight, each sun sensor assembly includes four photodiodes with unique pointing vectors. For each assembly, the lamp was first positioned directly parallel with the boresight of the assembly. Then, the sun vector telemetry was checked to confirm that the XACT was detecting the sun along the axis of the tested sensor. Next, the light was moved approximately 30° from the boresight and the telemetry was examined to confirm that the

XACT's sun vector had moved accordingly. Finally, the light was moved approximately 30° from the boresight in the opposite direction, and the telemetry was again checked to confirm the updated sun vector matched expectations. This test was successfully performed on all four sun sensor assemblies.

The XACT includes a star tracker or stellar reference unit (SRU) which is used to determine the spacecraft's attitude during flight. The SRU uses a camera to measure the brightness of stars in its field of view, and then compares these measurements to an internal star field catalog using a pattern-matching star identification algorithm.¹³

Following integration of the XACT into the spacecraft, the SRU was tested using a JPL-developed star field simulator (SFS).¹⁴ This simulator works by generating a star field image onto a display that is positioned in front of the XACT's star tracker camera (Figure 29). First, a functional test was performed to calibrate the SFS, ensuring that the XACT was able to detect the simulated stars and determine where it was pointing in the simulated environment. After confirming the setup and basic functionality of the SRU, phasing tests were performed for each axis.

For each phasing test, the SFS was configured to display a star field and then rotate the field around a specific axis at $0.5^\circ/\text{s}$ for approximately 180 seconds. As the star field rotated, the XACT's SRU-based attitude vector was reviewed to confirm that it was rotating along the expected axis and in the appropriate direction.

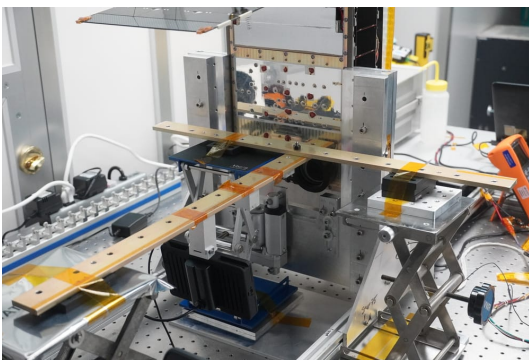


Figure 29: Test of the XACT Star Tracker Using Starfield Simulator Display

Lastly, while the XACT camera is primarily for use as a star tracker, it can be used to capture images during operations. This functionality was tested using a few team members as subjects (Figure 30).



Figure 30: Test Image Captured with the XACT Onboard Camera

Thermal Patch Installation

Thermal analysis was performed on the final spacecraft design, which resulted in recommendations for the installation of thermal control surfaces. Silver-coated Teflon was installed on five faces of the spacecraft, including the payload's optical receiver cryogenic radiator, to increase the passive cooling capability of the system (Figure 31). After installing the material, Scotch-Weld 2216 staking was added to provide additional bonding. Finally, Eccobond 57C was applied to ground each Teflon sheet to the spacecraft chassis, and each Eccobond stripe was covered with an overcoat of Arathane 5750.

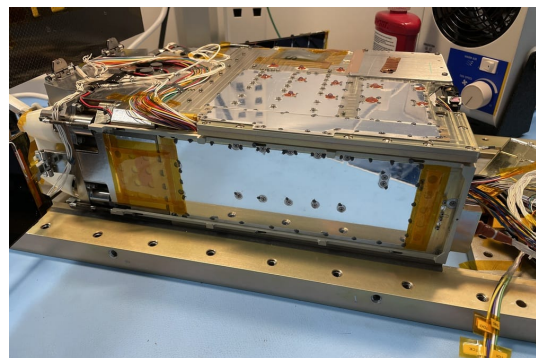


Figure 31: Silver Teflon Applied to Spacecraft

Mass and CG Measurements

A measurement jig was developed by GTRI to aid in the determination of the spacecraft's center of gravity (Figure 32). This fixture was made using 80/20 T-slot rails combined with 3D-printed rails, and could be reconfigured to hold the spacecraft on each axis. In each configuration, the spacecraft sat on two balance points with the ability to rock along

the axis created by those points. Careful adjustments were made until the spacecraft was balanced on these two points. Once fully balanced, the position of the balance points, which indicated the center of gravity of the spacecraft along this specific axis, was measured with respect to a spacecraft reference point. This test was repeated for all three axes to calculate the three-dimensional center of gravity. After this, a mass measurement was performed, confirming the unfueled spacecraft met the project’s mass requirements at 11.5 kg.

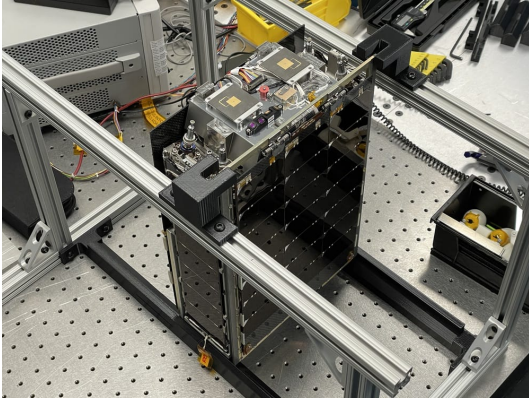


Figure 32: Y-Axis CG Measurement Setup

The team had begun to identify a need for additional handles during the SI&T process, and at this point a sheet metal handle was manufactured which utilized existing screw interfaces on the chassis (Figure 33). Once placed into service, this handle continued to be used as an aid along with the propulsion system handles for two-person transfers of the spacecraft between fixtures.

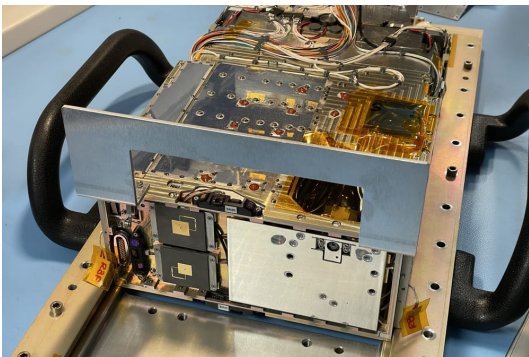


Figure 33: Aluminum Handle Used for Carrying Spacecraft Between Fixtures

Thermal Vacuum Testing

Spacecraft-level thermal vacuum (TVAC) testing was performed to demonstrate the functionality of the spacecraft in the expected space environment.

A 120-hour bakeout was also combined into the hot dwell portions of the test. The TVAC test required a vacuum of $< 1 \times 10^{-5}$ torr and two thermal cycles with hot plateaus at $+40^\circ\text{C}$ and cold plateaus at -10°C . The avionics flight spare battery was used during this test to allow for higher temperatures without stressing the flight battery. The test outline is shown in Figure 34.

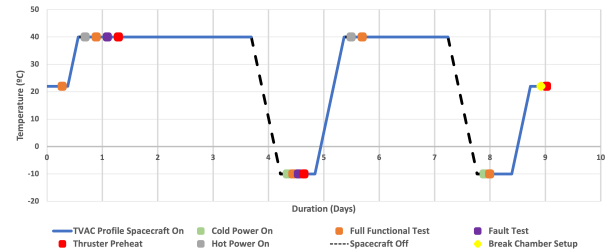


Figure 34: LF TVAC/Bakeout Test Plan

A LACO 1P23232 single-walled thermal vacuum chamber was utilized for this test. Initial tests with the chamber’s single JULABO chiller and a mass model showed that a more powerful system would be needed to heat and chill the spacecraft to the necessary temperatures. This was accomplished by augmenting the system with a copper shroud and an additional JULABO chiller (Figure 35). The shroud was constructed out of 1/8” 110 copper sheets and surrounded by seven loops of 1/2” OD 122 copper tubing. The second chiller pumped Thermal C5 fluid through this tubing to heat or cool the shroud.

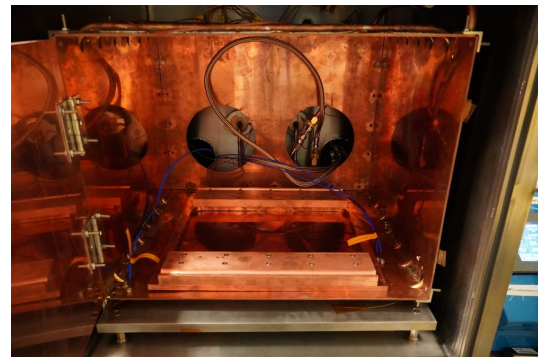


Figure 35: Copper Handling Fixture and Liquid-Cooled Copper Shroud

Aluminum rods wrapped in copper gauze were placed between the copper box and the LACO chamber “platen” to improve the thermal pathway to the primary chiller. With both chillers set to -40°C , this new setup achieved mass model temperatures below the -10°C requirement.

For the flight system test, the standard aluminum spacecraft handling fixture was replaced with a copper fixture to provide higher thermal conduc-

tivity between the chillers and the spacecraft (Figure 35). Copper gauze was also added between the spacecraft chassis and the shroud to provide additional thermal pathways (Figure 36).

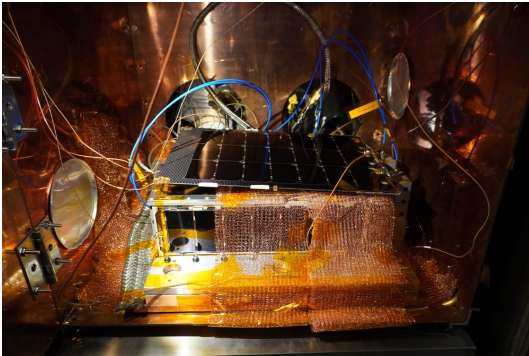


Figure 36: Spacecraft in TVAC Chamber

Thermocouples were applied to strategic locations on the CubeSat for this test (Figure 37). Data from these thermocouples augmented the spacecraft thermal telemetry, allowing operators to effectively monitor and set the temperature of the system even when the spacecraft was powered off. Additionally, this data was used to validate the spacecraft’s thermal model.

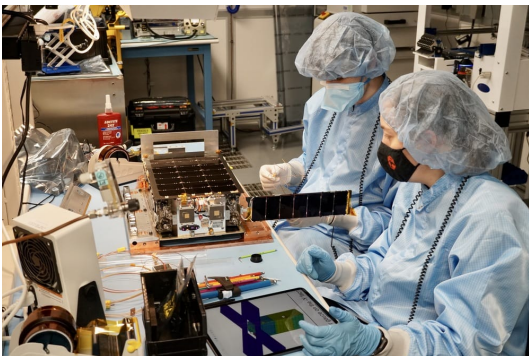


Figure 37: Installing Thermocouples for TVAC

The thermal vacuum test was completed over 10 days with continuous in-person monitoring. A team of three to four core members was responsible for running functional and other prescribed tests, while approximately 10 others were responsible for the continuous monitoring in shifts. Two hot and cold cycles were performed, with the hot plateaus being extended to cover the bakeout requirement. Witness plates were also used to analyze contamination released from the system during bakeout. Each thermal cycle began with the spacecraft powered off to meet non-operational requirements. The spacecraft was also powered off during hot-to-cold ramps to reduce the ramp durations. This TVAC test was suc-

cessful, satisfying the temperature cycling requirements and confirming that the spacecraft can operate effectively across its expected temperature range.

Helium Purge Test

As part of the launch process, the Falcon 9 stage housing the Lunar Flashlight spacecraft will be exposed to a helium environment. Certain electrical devices can be damaged by helium exposure, so SpaceX required a helium purge test of the spacecraft. This test consisted of placing the spacecraft in the TVAC chamber and purging the sealed chamber with helium at a pressure of approximately 15 psig for 24 hours. To simulate launch, a vacuum was then pulled, followed by a powered spacecraft functional test to verify continued nominal operation.

To purge the chamber with helium, a helium K-bottle was connected to a single stage regulator which regulated the bottle pressure of 2000-3000 psig down to approximately 1 psig. The output of the regulator was then connected to a flow meter which fed into the chamber. The flow meter allowed operators to monitor the flow rate into the chamber and measure the leak rate of the system. To minimize the leak rate, multiple c-clamps were placed around the door to further compress the o-ring seal.

To start the test, the chamber was pumped down to rough vacuum. The chamber was then purged with helium at a pressure of 1 psig. As the chamber reached equilibrium, the flow rate decreased to the leak rate. This setup was held for 24 hours, and then the chamber was again pumped down to rough vacuum. This was followed by a full functional test which verified that the spacecraft continued to function nominally after 24 hours of helium exposure.

Operational Testing

Two mission operation simulation tests were performed as a collaboration between the Operations and SI&T teams: “Day in the Life” and “Fault Protection”. These tests served as an opportunity for the Operations team to gain experience working directly with the spacecraft, acquiring higher fidelity data than the testbed could provide, while also gathering data needed to verify requirements were met.

The Day in the Life test exercised various mission scenarios, from the initial safe mode sequence at post-deployment power-on, to autonomous propulsion maneuver command sequences. The Fault Protection test explored flight software responses to various fault scenarios, ensuring that the system responded to faults as expected and successfully enforced autonomous flight rules.¹⁵

Dispenser Fit-Check

A dispenser fit-check was performed with support from the launch integrator, Maverick. Using the Dreadnaught fixture, the spacecraft was inserted into the spring-loaded dispenser (Figure 38). The team demonstrated that the spacecraft could move smoothly along the dispenser rails, the dispenser door could properly latch closed, and the spacecraft avionics battery could be charged through the dispenser maintenance port while the spacecraft was stowed.

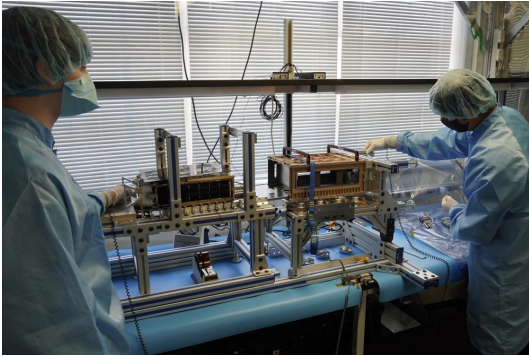


Figure 38: Installation of Spacecraft Into Dispenser Using Dreadnaught Fixture

Final SI&T Activities

A collection of closeout activities was performed to wrap up the SI&T campaign. This included installation of the flight avionics battery, cleaning of solar arrays, additional staking of external wires, and the installation and testing of new flight software releases. Once these final SI&T activities were completed, the spacecraft was nitrogen-purged and placed in storage awaiting fueling.

LESSONS LEARNED

Numerous lessons were learned during this mission's SI&T process, from the value of harness engineering support to the benefits of a flight-like testbed and appropriate tools for accessing and manipulating data.

Harness Engineering

Effective harness engineering is extremely valuable when working on a project of this complexity and compactness. With no dedicated harnessing engineer, the project encountered significant wiring issues during integration. For example, the team frequently spent a lot of effort determining how to

fit harnesses into allocated spaces. Unwieldy laser power cables had to be replaced with thinner and more precisely manufactured busbars. Shielding and strain relief tape were often reworked multiple times until they met bend radius requirements. Interference with avionics battery harness staking required removal of material from the spacecraft chassis.

SpaceWire data corruption was another issue that could have been avoided with more in-depth harness engineering. It was discovered that PWM-controlled heater wires running near a SpaceWire harness caused intermittent data integrity issues when active. This was resolved with a flight rule limiting when the heaters are used, but this could have been avoided with better routing or shielding.

Imprecise lengths of cables also caused trouble during SI&T. For example, unexpected differences in the length of coaxial patch cables necessitated an unplanned flip of antenna positions, which caused confusion during later RF testing when equipment was set up incorrectly. Also, excess length in the propulsion system harness had to be handled on the external surface of the spacecraft, which necessitated a complicated routing and staking strategy.

Limited harness modeling also made it difficult to estimate the voltage drop and related power limitations that each subsystem would experience prior to integration. This was especially concerning for the propulsion system since it had high power requirements for preheating the thruster catalyst beds and yet was one of the last subsystems to be integrated.

Dimensional Checks

The value of dimensional checks and “fit checks” was underscored throughout the SI&T process. The team used the rapid prototype approach for various fit checks, successfully testing out new designs using 3D-printed models. For example, prior to manufacturing metal busbars, full-size 3D-printed models were created. These were used to quickly iterate the design prior to starting the more complicated metal fabrication process. When plans were being developed to modify the payload radiator, a 3D-printed model was created to test the fit of the new design. This approach was critical, as an improper modification to the radiator could have necessitated the manufacturing of an entirely new part, which would have had significant schedule impacts.

Mishap Prevention and Detection

A few mishaps during the SI&T process emphasized how appropriate documentation, procedures, and hardware protections can prevent accidents or

reduce their effects. One mishap was the accidental application of 14 V to a 5 V C&DH power rail. Fortunately, internal protection diodes clamped the voltage and no hardware needed to be replaced, but the follow-up testing and analysis caused non-trivial delays in the schedule. A better-designed GSE setup could have avoided this. For example, an external interface board could have monitored for overvoltage before passing power on to the spacecraft. The spacecraft power-on procedure was modified after this event to add additional checks and precautions prior to applying power.

Another mishap occurred on the testbed where a C&DH board received 12 V on a 3.3 V line. The root cause of this issue was traced back to differences between the flight and testbed units and a lack of documentation about these differences. Better access to schematics as well as documentation or training about these differences could have prevented this incident from occurring.

During payload integration, a concealed wire on the APLE subassembly broke off, causing the detector current telemetry to jump to the maximum value. In ambient conditions where the payload detector was not in a low-light and properly cooled environment, the associated telemetry channel was generally saturated. As such, the detector readings were usually ignored, with no pass/fail criteria specified in the recurring system functional check-out tests. This assumption led to the damage being undiscovered until after the upper spacecraft had been completely integrated and transported to GT, at which point the repair caused a significant schedule impact. Future tests ensured this value was checked, but this scenario demonstrated the risk of failing to provide pass/fail criteria for telemetry channels.

Flight-Like Testbed

The testbed is a crucial part of any space mission, and ideally it can replicate all functionalities, modes, and operations of the spacecraft. For the Lunar Flashlight testbed, a few deviations in capabilities made it difficult to rehearse for operations or troubleshoot the spacecraft, leading in some cases to large schedule impacts.

Differences between the testbed and flight C&DH units meant that separate software and firmware releases had to be compiled for each unit. One hardware difference between the two units was the number of outputs available for driving heater enable lines, with the flight unit capable of enabling more outputs than the testbed unit. Additional logic for

these channels could not be tested on the testbed, so a bug in this firmware was not discovered until after installation on the spacecraft. Additionally, this difference made it impossible to test many command sequences on the testbed prior to installation on the spacecraft.

The spacecraft avionics stack includes an interface board that connects to many subsystems and provides heater switching and other capabilities. A similar board did not exist in the testbed during SI&T. This helped to conceal an RS-422 pinout issue on the flight interface board which was not discovered until integration of the propulsion system. It also limited the ability to test power draw and flight software functionality prior to full spacecraft integration.

Finally, the Iris Radio was represented in the testbed only through an FPGA development board interface simulator. While this allowed for testing of the command and telemetry interface, it did not recreate the actual power characteristics or behavior of the radio. For example, the Iris requires a specific power-up signal sequence while the simulator does not. As a result, flight software did not account for this sequence, which led to a power-up anomaly being identified late in the SI&T process. This necessitated a flight software update, which resulted in a non-trivial schedule impact.

These gaps in testbed fidelity meant that many development tests could not be performed on the testbed prior to running on the flight system. This placed a higher risk on the flight hardware throughout the SI&T process. While it may not be feasible to implement all functionality in a system testbed, these experiences showed that a project can avoid many issues by sharing as much functionality between the testbed and flight hardware as possible.

Data Accessibility

During subsystem and spacecraft testing, the accessibility of data played a large role in resolving issues. AMPCS,¹⁶ the JPL-developed data processing and control system in use by this project, is capable of providing telemetry and uplinked commands in a CSV format. A post-processing script was developed to compile this data into Excel spreadsheets and HTML Plotly-based¹⁷ interactive charts. These spreadsheets and interactive graphs could easily be shared without requiring any special software to use, making it easier to discuss and solve problems related to each subsystem. This post-processing software was also extended to process data from Lunar Flashlight's science instrument. This science data

consists of optical detector feedback as well as temperatures, voltages, and current telemetry from the optical system. Being able to quickly interact with this data made the setup process much easier during tests of the laser.

Cross-institution IT policies meant that many of the SI&T computers were not network-connected once delivered to Georgia Tech. This presented new data accessibility issues and hurdles to keeping configurations synchronized between computers, as information was previously funneled through a centralized GitHub instance at JPL. These issues were somewhat rectified through the use of a bare Git repository on a machine with access to the GitHub server, which could act as an intermediary between networks. However, a better private network setup would have saved time.

These experiences showed that data accessibility was key to making the SI&T process run smoothly. The more steps it took to process data and deliver it to relevant parties, the longer it took to iterate and solve problems. Development of data processing scripts helped to ease this pain and get problems solved more quickly.

CONCLUSION

Following fueling of the propulsion system at MSFC in fall 2022, Lunar Flashlight is expected to launch in late 2022 or early 2023. After launch, the Lunar Flashlight spacecraft will perform a lunar orbit insertion and enter its science phase of searching for water ice in the craters of the Moon's south pole. The Systems Integration and Test campaign for this project was an intense collaborative effort between many organizations, including the Jet Propulsion Laboratory, the Georgia Institute of Technology, and the Marshall Space Flight Center. This experience resulted in many lessons learned by all participants that will ideally inform future missions and mitigate common issues from reoccurring.

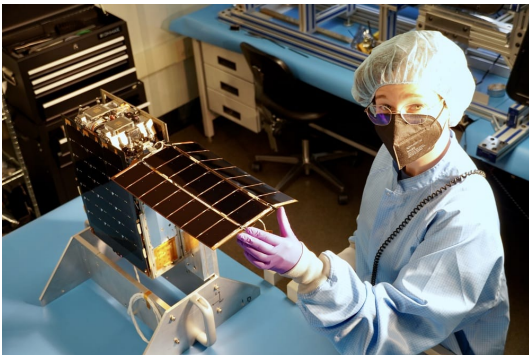


Figure 39: Spacecraft Prior to Fueling

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