Ground Support Equipment (GSE) Design

This document defines the planned electrical and mechanical ground support equipment for the 3U spacecraft. The GSE will support development, assembly, integration, test, transportation, and pre-launch operations while protecting flight hardware. Any GSE that directly interfaces with the spacecraft will be configuration-controlled and handled with the same care as flight hardware. The design is divided into Electrical Ground Support Equipment (EGSE) and Mechanical Ground Support Equipment (MGSE).

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Acronyms and Terms

Acronym or term	Definition
ADCS	Attitude Determination and Control Subsystem
APID	Application Process Identifier
BMS	Battery Management System
CCSDS	Consultative Committee for Space Data Systems

Acronym or term	Definition
CDH	Command and Data Handling
CTL	Command and Telemetry List
EM	Engineering Model
EGSE	Electrical Ground Support Equipment
EPS	Electrical Power Subsystem
F'	F Prime flight-software framework
FEC	Forward Error Correction
FM	Flight Model
GDS	Ground Data System
GNSS	Global Navigation Satellite System
GPS	Global Positioning System
GSE	Ground Support Equipment
HDLC	High-Level Data Link Control
IBF	Install Before Flight
ICD	Interface Control Document
MDB	Mission Database
MGSE	Mechanical Ground Support Equipment
MQTT	Message Queuing Telemetry Transport
OVP	Overvoltage Protection
PC/104	Embedded computer-bus form factor and interface
RF	Radio Frequency
REST	Representational State Transfer
RBF	Remove Before Flight
SDLS	Space Data Link Security
SDR	Software-Defined Radio
SOH	State of Health
SPI	Security Parameter Index
TC	Telecommand
TBC	To Be Confirmed
TBD	To Be Determined
TBR	To Be Resolved
TM	Telemetry
TTL	Transistor-Transistor Logic
UART	Universal Asynchronous Receiver-Transmitter
XTCE	XML Telemetric and Command Exchange

1. Electrical Ground Support Equipment (EGSE)

1.1 Purpose and Scope

The Engineering Model (EM) Electrical Ground Support Equipment (EGSE) provides the controlled electrical, command-and-telemetry, inhibit, thermal-monitoring, and conducted-RF interfaces needed to assemble, integrate, and test the HS-2 spacecraft. The normal non-RF command-and-telemetry path is a wired UART connection through the Battery Management System (BMS) service connector; it is deliberately independent of the spacecraft radio so that we can communicate with the satellite without radio. The UART connection is connected to the debug UART on the BeagleBone Black, which we repurpose for sending commands and receiving telemetry. A separate conducted RF path uses the EnduroSat S-band transceiver for the required RF connection, using a Software Defined Radio (SDR) and appropriate attenuation for RF signals.

The EM EGSE is the baseline for the integrated flatsat and will mature into the portable flight-support configuration. Flight-facing cables, connector savers, protection settings, instrument configurations, software versions, and procedures are configuration-controlled before use with flight hardware. The known interfaces for the EM EGSE and the spacecraft, which are elaborated in each subsection of the EGSE, are:

- An 8-pin connector with the BMS, containing: inhibit configuration/information, external power/GND, wired connection via UART, power through solar panels, and discharge to a programmable load.
- An RF connection between the EnduroSat S-Band Transceiver and a Software-Defined Radio (SDR). This requires attenuation to not supply too much power to the SDR; the math is outlined in section 1.8. This is how we interface with our communications stack.
- Other interfaces within the EGSE include: a surge protector that powers our power supplies, EGSE computer, programmable load, and the display/interface we will operate on. The EGSE computer is directly connected to the display and SDR. In addition, the BMS connector will be connected to a small PCB that serves as our single EGSE to spacecraft connection; the EGSE computer will connect with the UART pins from that PCB.
- Interfaces within the spacecraft can be found within the ICDs, but notably, all PCBs (except the PAY PCB which mounts onto the CDH PCB) use the PC104 interface, which we have a custom flatsat PCB designed for. More detail is provided in section 1.2.

1.2 System Architecture and BMS Service Interface

Figure 1 shows the current EGSE architecture. Standard 120 V AC enters one protected EGSE distribution point and supplies the host computer, regulated DC source, programmable electronic load, and bladeRF SDR. The regulated DC source is not the power source for all EGSE equipment. The BMS service cable provides the low-voltage electrical interface to the flatsat; the RF path is a separate coaxial test interface to the S-band transceiver.

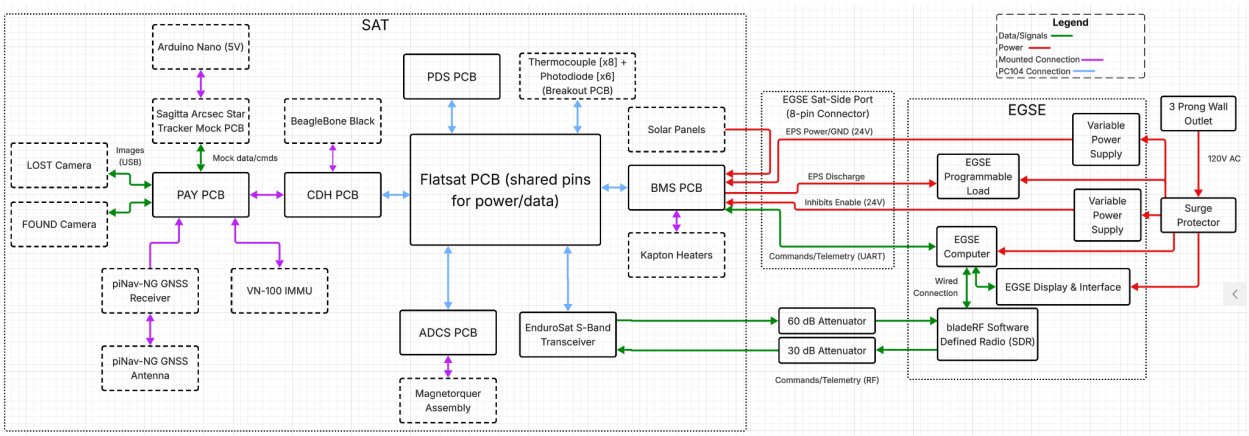


Figure 1. EM EGSE architecture and primary spacecraft interfaces.

The BMS service connector is an eight-pin spacecraft-side interface. The current design identifies the following functions. Direction is defined with respect to the spacecraft. Components are mounted as per their respective subsystem ICDs.

We have a custom Flatsat PCB that contains 6 PC104 interfaces that are all connected with each other. We use 4 of them for our PCBs (CDH, PDS, BMS, and ADCS), and one for the EnduroSat S-Band Transceiver. The PAY PCB is mounted to the CDH PCB as per the PAY ICD. The sixth PC104 interface is used for the thermocouples and photodiodes (sun sensors), since they are transmitted on the PC104 to CDH, utilizing a breakout board [TBD].

The PAY ICD plans for an Arcsec Sagitta Star Tracker to be present on the satellite; due to current funding, we are unable to purchase it for CDR/FSR. Thus, we have opted to make a mockup of the Star Tracker from their Software and Hardware ICDs, making a custom PCB with a full-duplex RS-485 transceiver, using a similar 15-pin connector as the Arcsec Sagitta StarTracker, using a 5V Arduino Nano to receive command formats and send back mock data in the same format. This allows us to test and validate that Arcsec Sagitta commands we send are still able to execute by using the same communication and wiring interfaces.

Position	Signal	EGSE function and direction
1	SYS_PIN	Controlled external system/test-power interface. EGSE supplies regulated power to the inhibited spacecraft bus for functional test, and can also be used as a discharge port by reversing current. Voltage/current limits are set according to the power budget and EPS ICD.
2	SOL_PIN	Solar-panel path test path. Not used for normal wired EGSE operation; reserved for a later solar-panel test.
3	GND	Service power and signal return.
4	GND	Service power and signal return.
5	EN	Separate, current-limited EGSE inhibit supply drives EN; high = inhibits ON and low = inhibits OFF. IBF is a removable jumper between EN and GND (EN low when installed). EGSE verifies EN state and post-inhibit SYS_PIN before power tests.

6	UART_RX	Spacecraft receive. EGSE UART TX drives this 3.3 V TTL input.
7	UART_TX	Spacecraft transmit. This 3.3 V TTL output drives the EGSE UART RX input.
8	Unassigned	No operational connection currently planned

Table 1. BMS EGSE service connector allocation.

The BMS connector will be connected to an external PCB with a molexed connector, for the purposes of strain relief, and the harness being keyed,having a positive-locking mating system, and a connector saver for repeated EM work. This external EGSE interface PCB will convert the host USB interfaces into a 3.3 V TTL UART bridge [TBR] using the 5-pin serial to UART cable that the BeagleBone Black provides, protected discrete-control circuitry for EN, voltage/current sensing, and controlled instrument interfaces. It does not expose laptop USB or raw GPIO directly to flight hardware.

Figure 1 shows the EGSE for our flatsat; the UART interfaces, inhibit interface/control, and power paths will remain the same when the PCBs are stacked and put into the structure (since they share the PC104 interface). The same BMS connector will be present and used.

1.3 Power, Charging, Discharge, and Battery Protection

The EGSE has three distinct EPS operating modes: (1) external test power at SYS_PIN supports operation while the normal flight battery path remains inhibited; (2) battery charging must reach the battery-side charge path without powering spacecraft bus loads; and (3) maintenance discharge removes energy through a controlled electronic load. Each mode is enabled only by an approved procedure with voltage, current, temperature, and time limits taken from the released EPS ICD.

Mode	**Electrical path and planned capability**	**Verification and current status**
Powering satellite while batteries are inhibited	Regulated, current-limited DC source → SYS_PIN / GND → spacecraft system bus.	Demonstrate on EM that the battery power path remains inhibited while the spacecraft operates from EGSE source power. SYS_PIN maximum voltage/current are set according to the power budget and EPS ICD.
Battery charge with satellite bus off	Dedicated battery-side charging path required; charging through SYS_PIN is not accepted as evidence because SYS_PIN is the system/test-power node.	TBD, the connector/routing on the BMS will require a v2 to support this.

Discharging batteries	BMS/EPS output → programmable electronic load. Planned capability: 0–7 W at 21 V nominal; at 7 W, constant-current setpoint is 0.333 A.	Demonstrating on EM that normal usage while the satellite bus is on, as well as configuring the programmable load, will allow for the batteries to be discharged.

Table 2. EPS operating modes and closure evidence.

The regulated source is provisioned for up to 19 W of EGSE-to-EPS service power, subject to the final EPS voltage and duration limits. The programmable load is computer-addressable for scripted tests, but its input is normally disabled; automatic shutdown occurs at the approved battery threshold or on a measured fault.

Every primary spacecraft-facing source or load line includes accessible, labeled overvoltage, reverse-polarity/reverse-current, overcurrent, and short-circuit protection. Power is off by default after connection, startup, host loss, or a detected fault. Current limits are set in hardware before a procedure begins; software may request an operating point but cannot defeat the hardware limit.

1.4 Inhibit Control, State Verification, and IBF Closeout

EN is the exposed BMS inhibit-control node. The EGSE drives EN from a separate, current-limited inhibit supply set to the EPS-qualified logic voltage (24V); EN high commands inhibits ON and EN low commands inhibits OFF. EN will be driven with a power supply (see Figure 1).

Positive inhibit verification is independent of the EN command path. Before each power test, EGSE measures EN and the post-inhibit system node (SYS_PIN) and records the BMS inhibit-state indication. The acceptance check is: EN high shall produce the inhibited/disconnected SYS_PIN condition; EN low shall produce the released/connected condition.

The Insert Before Flight (IBF) closeout is implemented as a removable jumper between EN and GND; when installed, EN is low and the inhibit state is OFF. The jumper is keyed and retained, remains externally accessible, and its presence is verified by continuity before power is applied. For bus power with a battery pack, disconnect the regulated EGSE SYS_PIN source and set EN low (inhibits OFF). For bus power without a battery pack, keep the EGSE source connected and set EN high (inhibits ON). Discharge is a normal operating mode with the bus powered as required; the programmable load draws the approved current. Charging batteries without turning on the bus is TBD, as the BMS v1 does not support this, so a BMS v2 is required. These procedures provide EGSE actuation and state evidence while preserving the spacecraft independent installation inhibit. independent installation inhibit.

1.5 Wired Command, Telemetry, and Software

F Prime Ground Data System (F' GDS) is the operator interface for the normal wired path. The host computer sends command packets through a USB-to-3.3 V TTL UART [TBR] bridge to the BeagleBone Black UART0 and receives telemetry, events, and command responses through the reverse UART path. The flight application uses the same command and telemetry definitions

for UART and RF operations; the transport below the F' framing layer changes, not the command dictionary or spacecraft deployment. The current revision of the BMS board only has the UART pins, it will [TBR] be updated to support the 5-pin serial to UART interface so we can plug in the cable to the EGSE computer.

The selected BeagleBone UART0 is dedicated to the F' transport. Linux boot-console and login/getty functions are disabled on that UART before flight-hardware use. Commands and telemetry sent/received are defined as per the CTL, CDH ICD, and COM ICD.

F' GDS displays command acceptance, BMS/EPS state, inhibit evidence, temperatures, and subsystem state-of-health telemetry. Each session records timestamped command, telemetry, instrument, and fault data. The EGSE control application separately controls the protected EN interface and programmable instruments; it does not give F' GDS direct access to power-control pins.

1.6 Thermal Monitoring

When the spacecraft is powered, the baseline thermal-monitoring path is the flight temperature system: eight thermocouple channels route through the PC104 to CDH as per the CDH ICD. They are logged through F' GDS with the normal spacecraft telemetry. This supports state-of-health monitoring and provides the powered portion of thermal sensor testing. The flatsat sensor breakout and PC/104 stack are mechanically secured by the board stack standoffs and fasteners.

Note: The below is TBD and very subject to change.

For thermal testing with the spacecraft powered off, the final EGSE/TVAC interface is a Phase B item. The TBR solution is separate EGSE thermocouples on the critical components. Separate EGSE thermocouples will have to route as dedicated pairs to an exterior-accessible spacecraft connector (potentially separate) that remains reachable in the dispenser [TBR]. This will likely involve a small breakout PCB with support for thermocouples and noise reduction, as well as having its own thermocouple amplifier that is powered by the EGSE rather than the satellite bus. This will allow us to have separate thermocouples that remain readable even when the satellite bus is turned off, while following UNP12-40 by having them terminate at this breakout PCB.

1.7 Physical Interface, Retention, and Operator Safety

The BMS service cable is the EGSE-to-spacecraft electrical connection. Its flight-facing connector shall be polarized/keyed, positively locked, strain relieved, and selected for the required voltage, current, and documented mating-cycle life, as it is connected to the EGSE connection PCB. The spacecraft-side design prevents exposed energized contacts and protects the interface against pin-to-pin and pin-to-shell short faults. Mating cycles are logged and a connector saver is used for repeated EM integration.

Other flatsat retention controls are recorded because they affect EGSE functional-test readiness: PC/104 cards are retained by the stack standoffs/fasteners; PAY-to-CDH board-to-board interfaces receive mechanical support and do not carry cable loads; USB camera plugs receive locking from FFC cable connectors on both camera and PAY PCB end; and RF coax uses threaded RF connectors. Any unlatched peripheral harness receives a dedicated strain-relief feature before integrated test [TBD].

The portable EGSE uses one standard 120 V, 60 Hz, three-prong input to a protected AC distribution point. The main switch has a visible power-on indicator. All controls, fuses, and current limits are labeled with function and rating; safety-critical controls are separated and guarded. Closing a guard for a safety-critical power control returns the associated output to off where that feature is implemented.

1.8 Conducted RF Checkout

Routine EGSE command and telemetry use the wired UART path. RF command-and-control is a separate conducted test using the bladeRF SDR and the EnduroSat S-band transceiver. This arrangement exercises the radio hardware and communication with the satellite via RF.

For the high-power direction, the initial chain is EnduroSat transmitter → high-power 30 dB attenuator → 30 dB low-power attenuator → bladeRF receiver. The EnduroSat transmitter is treated as a 12 W (approximately +40.8 dBm) maximum-power source until the radio configuration confirms otherwise [TBD]. The first attenuator is rated at least 20 W and dissipates nearly all transmitter power; the 60 dB initial total attenuation produces approximately -19 dBm at the bladeRF receiver. All components are rated for the operating S-band frequency. Calculation: $P_{RX} = 40.8\ \text{dBm} - (30\ \text{dB} + 30\ \text{dB}) = -19.2\ \text{dBm}$ (approximately −19 dBm).

For the reverse direction, bladeRF transmitter → adjustable 30–40 dB attenuation → EnduroSat receiver is the initial functional-test configuration. The bladeRF output is measured with a power meter or spectrum analyzer, and attenuation is set to the approved EnduroSat receive-input level. Reverse-path sizing is $A_{rev} = P_{TX,bladeRF} - P_{RX,max,EnduroSat}$; both limits are TBD and shall be verified before connection.

The bladeRF SDR will also need to match the RF layer frames as outlined in the EnduroSat S-Band Transceiver ICD; this includes FEC and HDLC with the same polynomial values provided. This will be done through tuning settings in the software as appropriate.

1.9 Operating Concept and UNP Verification Status

A normal integrated test begins with inspection of the closeout state, cable, connector lock, grounding, and protection settings. The operator confirms that all sources and loads are disabled, mates the service connector, establishes UART telemetry, and records the inhibit-state evidence before enabling an approved power mode. The test procedure then runs the required subsystem, full-functional, day-in-the-life, or RF communication sequence. On completion, the operator returns the spacecraft to the required safe state, disables source/load outputs, saves all logs, and de-mates the connector.

UNP requirement	**EGSE implementation**	**Current verification status**
UNP12-40, 12-76	Powered thermal telemetry is defined; powered-off TVAC sensor/connector/DAQ path is reserved.	TBD: Phase B thermal interface and procedure
UNP12-41, 12-43, 12-46, 12-48, 12-86	Keyed, locking, strain-relieved service harness; mating-cycle log; short/fault protection; controlled flight-facing cable.	Defined design; verify by drawing, inspection, and cable test

UNP12-44, 12-57, 12-58	Externally accessible IBF/RBF closeout owned by EPS; EGSE provides interface access and verification.	TBD: EPS/UNP-approved installation-inhibit design
UNP12-59, 12-72	Protected EN actuation plus independent power-path/hardware state verification.	TBR: EN interface limits and verification thresholds; demonstrate on EM
UNP12-70	Battery-side charging path with bus loads disabled.	TBD: BMS will need a v2
UNP12-71	Computer-addressable, current-limited programmable electronic load; planned 0–7 W at 21 V nominal.	TBD: discharge port, SOC threshold, duration, and procedure
UNP12-73, 12-74	Regulated current-limited SYS_PIN test power and F' GDS/UART support for subsystem, full-functional, and day-in-the-life tests.	Planned; verify in integrated EM test
UNP12-77	F' GDS over dedicated 3.3 V TTL UART service path.	Defined; verify command and telemetry transfer
UNP12-78	Conducted bladeRF ↔ attenuated EnduroSat RF test path.	Planned; verify bidirectional command and telemetry
UNP12-79, 12-80, 12-87 to 12-89	Hardware current limiting, OVP, reverse-polarity/current, overcurrent, short-circuit protection, and accessible labeled protection.	Defined; verify by schematic, inspection, and fault test
UNP12-75, 12-81, 12-82, 12-83, 12-84, 12-85	Portable EGSE with one 120 V, 60 Hz, three-prong input, protected distribution, indicator, labels, spacing, and guarded controls.	Defined; verify by inspection and functional test

Table 3. EGSE compliance approach and CDR closure status.

2. Mechanical Ground Support Equipment (MGSE)

2.1 Purpose and Scope

Because HuskySat-2 is a 3U CubeSat, the MGSE is expected to be structurally simple. It must nevertheless protect the spacecraft, provide controlled handling and access in required orientations, and support transportation, assembly, integration, and test. The planned minimum MGSE suite is a protective transport case, ESD-safe work supports or an integration stand, and any small handling or test-adapter fixtures needed to prevent loads from being applied to rails, solar panels, antennas, connectors, or other sensitive hardware.

2.2 Transport Container

We utilize a Pelican 1535 case as the baseline transport container. The final case configuration will use clean, low-shedding, ESD-compatible cushioning or a removable inner container shaped to approved structural hard points. It will restrain the spacecraft in all directions without loading deployables, exterior components, rails, or connectors. The case will include tamper indication as needed, handling orientation labels, ESD identification, and space for humidity indication or desiccant if required by the contamination-control plan.

The transport configuration will contain the spacecraft without damage when subjected to the maximum expected transportation loads. The load environment and retention method remain TBD. The design will address shock, vibration, drops or impacts associated with normal handling, and cable or accessory stowage so loose items cannot contact the spacecraft.



Figure 2: The Pelican 1535 case with a mockup model inside it

[TBD: Define maximum expected transportation load, case insert material, approved contact points, environmental controls, and acceptance criteria.]

2.3 Integration, Orientation, and Handling

MGSE will allow safe access to the spacecraft in multiple orientations and will secure it during assembly and test. Two test stands (reference Figure 3) are intended to be used: The first will allow vertical assembly and the second will allow horizontal integration as seen in Figure 4. Both are manufactured out of Aluminum T6 6061. The fixture will allow work on each required face without resting the spacecraft directly on sensitive exterior hardware.

We require the spacecraft MGSE to permit lifting in any axis. For a 3U spacecraft, a large crane fixture is not expected. Compliance will be achieved through a documented combination of approved hand-lift locations and, if required, removable handles or a small lifting cradle. The final handling analysis will determine whether dedicated handles are necessary. Personnel will not lift the spacecraft by deployables, rails, antennas, harnesses, or flight connectors.

At least two trained handlers will be used when spacecraft configuration, access, or risk makes a single-person lift unsuitable. Before each move, the destination support, orientation, path, and responsible handlers will be identified. Protective covers will be installed over exposed sensitive hardware where the design permits.

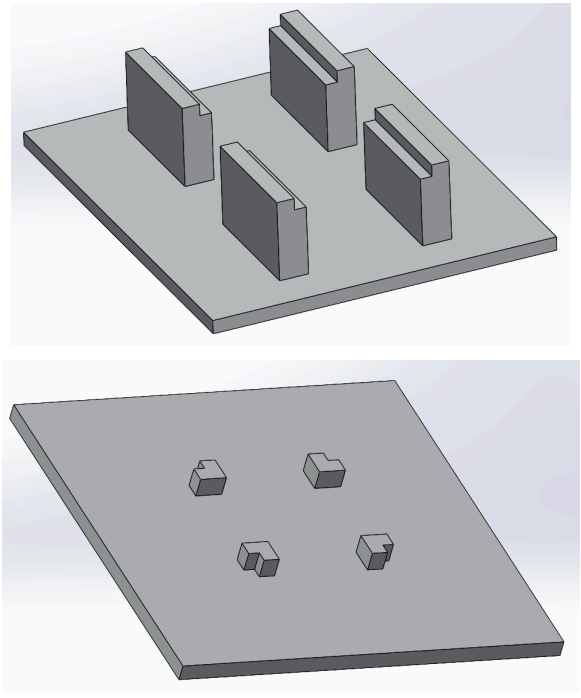


Figure 3: Test Stands

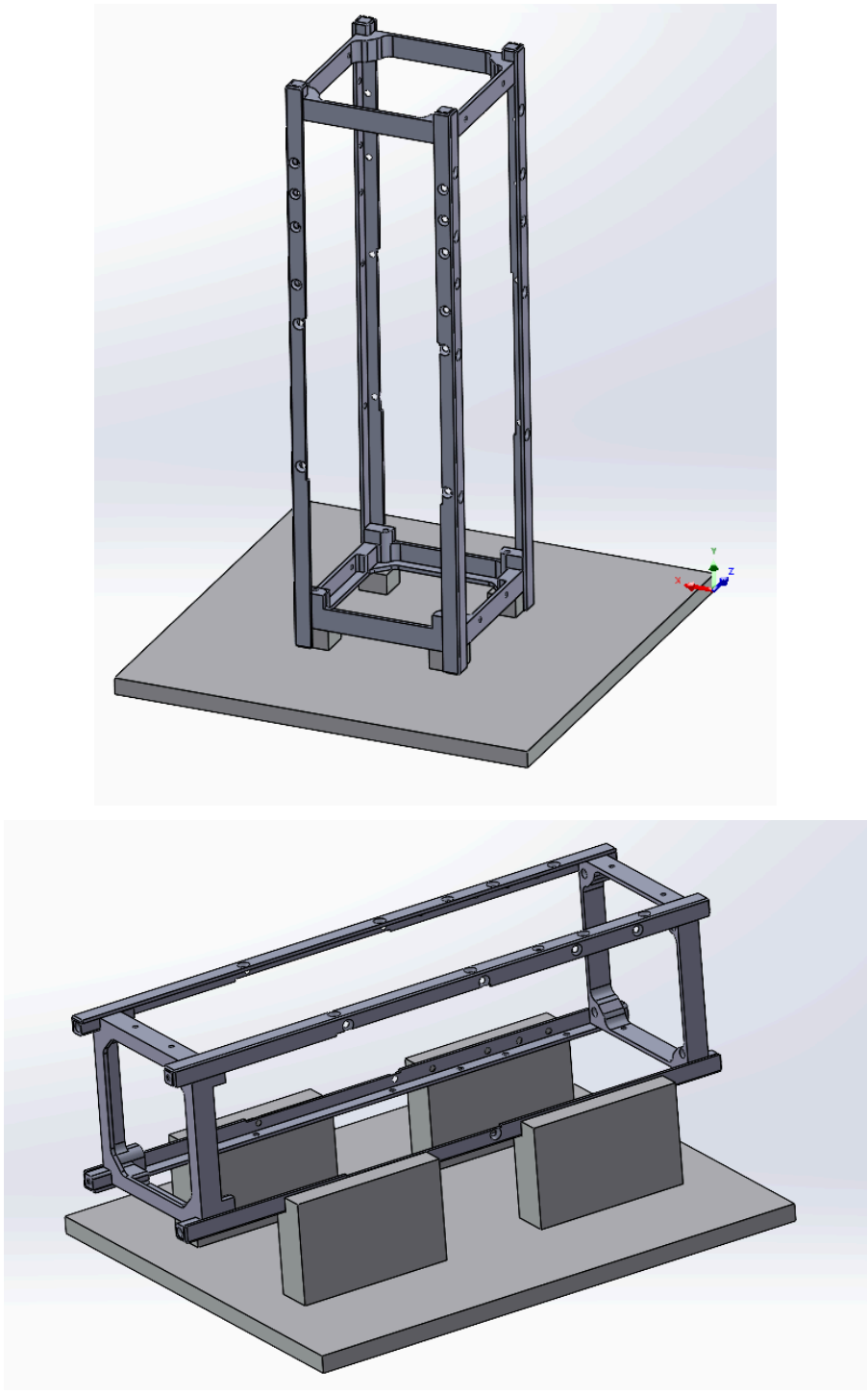


Figure 4: Stands and CubeSat

2.4 ESD, Cleanliness, and Flight-Hardware Controls

MGSE surfaces and inserts that contact or approach the spacecraft will be compatible with the project ESD-control and contamination-control plans. Conductive or dissipative features will be

bonded to the approved ESD ground where appropriate. Insulating foam or materials that generate particles or static charge will not be used near exposed flight hardware unless specifically approved. Relative humidity of the room will be measured and visible to limit possible hardware damage and increase safety.



Figure 5: An image of our current clean room

The spacecraft and MGSE used with flight hardware will be maintained in the UNP-required Class 10,000 or better facility. Our soft-walled clean room facility shall provide clean, filtered air independent of the building's HVAC system. Entry and exit procedures will be implemented to ensure limited human contaminations. This includes a specified gowning area, and published reminders to adhere to proper clean room activity practices. Activities such as drilling, soldering, cutting, etc., shall be performed outside the clean room whenever possible. If such activities must be performed inside the clean room, they shall be conducted within a laminar flow bench to maintain an appropriately controlled environment and minimize contamination. MGSE will be cleaned, inspected, bagged or covered as appropriate, and stored so that it does not recontaminate the spacecraft. Fasteners, tools, and removable MGSE parts will be controlled through shadowboxing to prevent foreign-object debris.

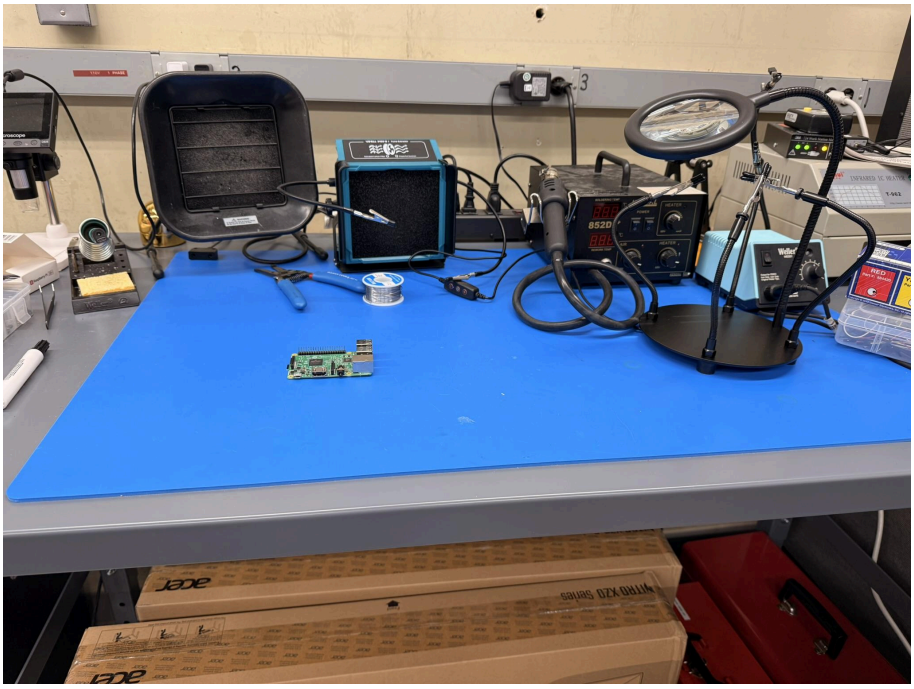


Figure 6: A showcase of our ESD-safe workstation

2.5 Structural Design and Proof Loading

Any load-bearing MGSE will be designed with an ultimate factor of safety of at least 5.0 relative to the maximum expected spacecraft load and will be proof loaded to twice that expected load. This applies to handling fixtures, stands, restraints, handles, cradles, and transport-retention elements that carry spacecraft load.

Simple work supports that only stabilize the spacecraft will still be analyzed for tip-over, slip, local contact stress, and foreseeable handling loads. Proof loading will be completed with a mass simulator rather than flight hardware. The MGSE will be inspected after proof loading for permanent deformation, loosened hardware, cracking, loss of restraint, or other damage.

2.6 Test Interfaces and Verification

Environmental-test adapters may be required for vibration, thermal-vacuum, deployment, mass-properties, or other qualification and acceptance testing. These adapters will be treated as separate MGSE items and designed when the applicable facility interfaces and test plans are known. The MGSE itself will not be described as providing vibration, thermal, center-of-gravity, or moment-of-inertia data; those data are products of the corresponding tests and analyses.

MGSE compliance will be verified through:

- Inspection of the Pelican-case configuration, support locations, ESD provisions, labels, cleanliness controls, and handling procedures.
- Demonstration that the spacecraft can be secured and safely accessed in the required orientations.
- Analysis of transport loads, restraint, contact stresses, stability, and load paths.
- Analysis showing an ultimate factor of safety of at least 5.0 for load-bearing items.
- Proof loading to twice the maximum expected spacecraft load using a representative mass simulator.
- A controlled transport or handling test, followed by inspection for movement, damage, or contamination.

The MGSE concept fulfills UNP12-28 by providing the equipment needed for assembly, integration, test, and transportation. UNP12-29 will be closed by the approved handling/lifting approach, and UNP12-31 will be closed by structural analysis and proof-load records. UNP12-94 will be addressed through facility qualification and the project contamination-control process.

Open MGSE items are the final Pelican-case insert and retention design, the maximum expected transport load, approved spacecraft contact and lift points, the need for a dedicated integration stand or handles, environmental-test adapter interfaces, and the applicable cleanroom/ESD procedures.

3. References and Closure Evidence

The following references and evidence packages control the remaining GSE design decisions. The exact revision used for acceptance shall be recorded in the final configuration baseline.

Open item	Closure evidence
Battery-side charging with the bus off	Released EPS/BMS interface and contiguous charge-cycle demonstration
EGSE source/load voltage, current, duration, and tolerance	Released EPS ICD, calibrated instruments, and acceptance-test record
UART bridge baud, framing, and throughput	Released CDH ICD and bidirectional throughput test
Powered-off thermal measurements	Thermal interface drawing, exterior connector access, and sensor/DAQ test
RF conducted-test limits and attenuation	COM/radio ICD values, part ratings, termination check, and bidirectional RF test
MGSE insert, contact/lift points, and transport loads	MGSE CAD, load analysis, proof-load record, and controlled handling test
Cleanroom, ESD, and visibly-clean controls	Facility qualification and approved handling/cleanliness procedure