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Advanced eLectrical Bus (ALBus) CubeSat: From Build to Flight

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ABSTRACT

Advanced eLectrical Bus (ALBus) CubeSat is a technology demonstration mission of a 3-U CubeSat with an advanced digitally controlled electrical power system and novel use of Shape Memory Alloy (SMA) technology for reliable deployable solar array mechanisms. The primary objective was to advance the power management and distribution (PMAD) capabilities to enable future missions requiring more flexible and reliable power systems with higher output power capabilities. Goals included demonstration of 100W distribution to a target electrical load, response to continuous and fast transient power requirements, and exhibition of reliable deployment of solar arrays and antennas utilizing re-settable SMA mechanisms. The power distribution function of the ALBus PMAD system is unique in the total power to target load capability, as power is distributed from batteries to provide 100W of power directly to a resistive load. The deployable solar arrays utilize NASA's Nickel-Titanium-Palladium-Platinum (NiTiPdPt) high-temperature SMAs for the retention and release mechanism, and a superelastic binary NiTi alloy for the hinge component. The project launched as part of the CubeSat Launch Initiative (CLI) Educational Launch of Nanosatellites (ELaNa) XIX mission on Rocket Lab's Electron in December 2018. This paper summarizes the final launched design and the lessons learned from build to flight.

PROJECT OVERVIEW

The ALBus project was an effort by early-career employees at the Glenn Research Center (GRC) to contribute to the advancement of the CubeSat platform as a vehicle for expeditious and cost-effective technology demonstration for science and exploration missions. ALBus leverages GRC core competencies in power management and distribution (PMAD) systems and shape memory alloy (SMA) materials to address the anticipated needs of the CubeSat community for advanced mission concepts while maintaining the appeal of CubeSats as inexpensive and quick development missions. The project also benefitted NASA in exposing the early career team to hands-on hardware design as well as developmental, technical, and project management practices.

MISSION OVERVIEW

The mission had two primary objectives. The first was to demonstrate the functionality of the novel SMA activated retention and release mechanism, and SMA deployable array hinges, in an on-orbit environment. The second primary objective was to assess system-level capability to charge a high capacity battery, distribute 100W of power, and thermally control the system in a low earth orbit environment. System performance was gauged by the duty cycle of the 100W power distribution capability.

The mission involved three secondary objectives. The first was to characterize on-orbit performance of a high power density 3U CubeSat in a Low Earth Orbit (LEO) environment (thermal control performance, duty cycle, etc.). The second and third included an in-house battery management system demonstration as well as a Power Point Tracking (PPT) algorithm for smart charging. However, the battery management system and PPT algorithm features were eliminated due to project schedule and constraints.

MISSION OPERATIONS

ALBus' Concept of Operations calls for a 4-6 month mission duration utilizing a Wallops Flight Facility (WFF) ground station and ground operations at NASA Glenn Research Center via an interface to WFF network as illustrated in **Figure 1**. The notional deployment concept of operations is shown in **Table 1.**

ALBus nominal orbital parameters include:

- 85 degree inclination
- Perigee: 471 km, Apogee: 501 km
- Right Ascension of the Ascending Node (RAAN): 178.9 degrees

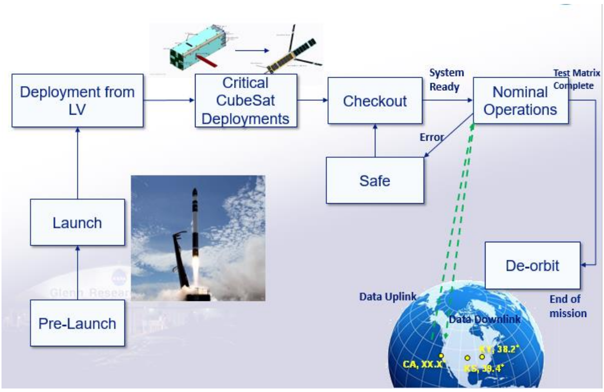

Figure 1: Launch Vehicle Concept of Operations

Table 1: Notional Deployment Concept of Operations

T = Deployment	Event
T + 0min	Deployment from launch service provider. Spacecraft power ON, deployment timer begins.
T + 1min	Spacecraft boot sequence completes and begins calculating payload data.
T + 15min	Solar array and antenna deployment sequence executes
T + 16min	Beacon begins (every 15 seconds)
T + 1 Day (estimated)	Communication link established and 1st set of data received. Beacon changes every 60 seconds. CubeSat collects and transmits data to validate thermal control and battery charging algorithm predictions.
T + 1 Week	CubeSat commanded to being nominal operations with demonstration of 100W discharge cycles
Note:	The system remains in this mode and continues to take payload data until commanded otherwise

PROJECT TIMELINE

CubeSats are habitually assembled and launched to space by several organizations including colleges/universities in timeframes ranging from six months to several years. The longer timelines are typically associated with new technologies that require extended periods of developments and testing. ALBus CubeSat is an example of such longer timeline, which consisted of developing two new technologies from basic research to flight. The project was organized into phases of reviews and technology maturation tollgates as outlined in **Table 2**.

Table 2: ALBus Milestone Schedule

Milestone	Date
Announcement of CubeSat Launch Initiative (CLI)	April 2013
Merit Review	March 2014
Technical Feasibility Review (Project Technical Review 1)	November 2014
CLI Proposal Submission	November 2014
HEOMD Acceptance Letter	February 2015
Project Technical Review 2	August 2015
Project Technical Review 3	April 2016
System Critical Design Review (CDR)	December 2016
Electrical Power System (EPS) Critical Design Review (CDR)	May 2017
Random Vibration Testing	November 2017
Fit-Check	November 2017
Thermal Bake-Out	November 2017
Mission Readiness Review (MRR)	December 2017
Pre-Ship Review (PSR)	February 2018
Delivery to Rocket Lab USA	April 2018
Launch Rocket Lab New Zealand	December 2018

MECHANISMS AND STRUCTURES

The ALBus CubeSat was based on a standardized 3U-size format, where the frame (or chassis) was obtained from a commercial off-the-shelf product. The internal and external components were modified from legacy modules or custom-built from conception, to satisfy the mission objectives. Due to the custom nature of the avionics and Printed Circuit Boards (PCBs), the assembly of the PCBs and attachment to the frame was unique. PCB assembly consisted of threaded rods with tubes acting as spacers, and various custom brackets used for frame attachment. The frame was modified slightly to accommodate several added features, mainly a radiator, antenna, and interfaces to the solar panels and mechanisms. Both body-mounted and deployable solar arrays were custom built to fit the CubeSat profile, but more importantly to facilitate the 100W electrical power system. The ALBus design was configured to use four deployable solar array panels with seven of the ultra-triple-junction type solar cells installed on a FR-4 Printed Circuit Board (PCB) substrate. These deployable solar arrays run the length of the 340 mm long CubeSat, and are to be deployed along with one of the short 100 mm sides of the CubeSat. This deployment configuration was chosen due to the absence of attitude control and determination systems in ALBus. The deployment mechanism was designed to utilize gravity gradient masses installed on the ends of the deployable solar arrays to point the CubeSat radiator down toward Earth.

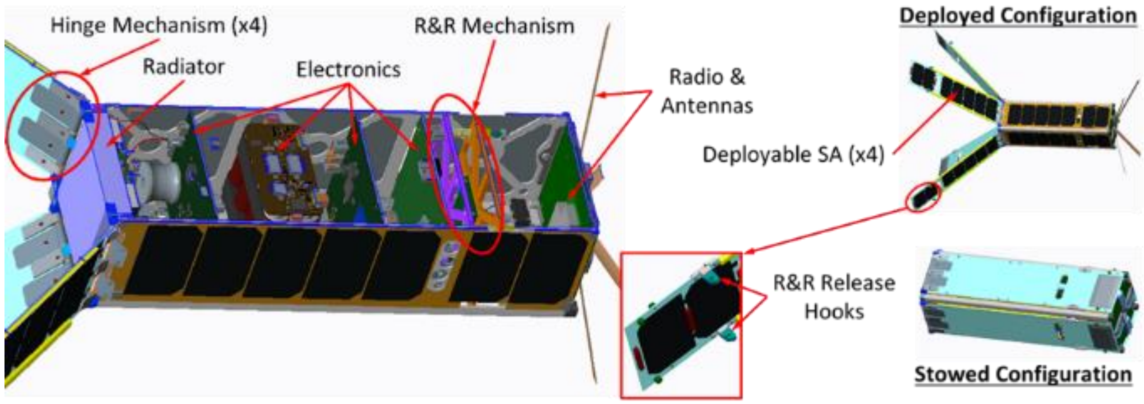

Figure 2: ALBus CubeSat Architecture – Main Internal and External Components

The final deployment angle was determined to be 135° from the stowed configuration for optimal power generation. However, a power analysis has shown that a 90° deployment angle is sufficient to recharge the batteries with acceptable power generation degradation. A summary of the ALBus configuration is shown in **Figure 2**.

The ALBus CubeSat also consisted of two unique mechanisms: the deployable solar arrays retention and release (R&R) and the hinge deployment mechanisms. The CubeSat deploys four solar arrays in addition to the body-mounted arrays on each side of the CubeSat. The SMAs were developed at NASA Glenn Research Center, and were used to deploy these solar arrays. The use of SMAs allowed the ability to test and reset the flight deployment mechanism prior to flight, which reduced the risk of in-orbit deployment failures common to CubeSats. As a result, an SMA-driven Retention and Release (R&R) mechanism and an SMA-driven hinge were designed, developed, and integrated for flight. The following gives a brief overview of the mechanisms developed for the ALBus CubeSat. More information on these mechanisms can be found in reference 1.[1]

SMAs have been used in various applications since the 1980s, including in-space hardware. CubeSats are a great way to verify and increase the capabilities of state-of-the-art SMA technology. SMAs have many advantages that can be utilized by CubeSats. In addition to being lightweight with a small footprint, SMAs are not pyrotechnics, produce low shock, do not create debris, and can be designed to be resettable. As part of the ALBus CubeSat technology development, two SMA forms were used. First, a novel thermally activated SMAs with higher transition temperatures (compared to commercially available counterparts) were used for the R&R mechanism. Second, a novel mechanically activated SMAs (superelastic alloys) were used as deployment springs to specifically engage ALBus' solar arrays and transfer the electrical power from the arrays.

The final mechanism's design converged on a two-stage SMA actively driven pin-puller type mechanism used to retain the arrays during ascent and release in orbit (R&R mechanism) as shown in **Figure 3**. The first stage is a pin-puller device driven by an SMA linear actuator (designed by Miga Motor Co using GRC's alloy). The second stage is a hook and pin design that is released by

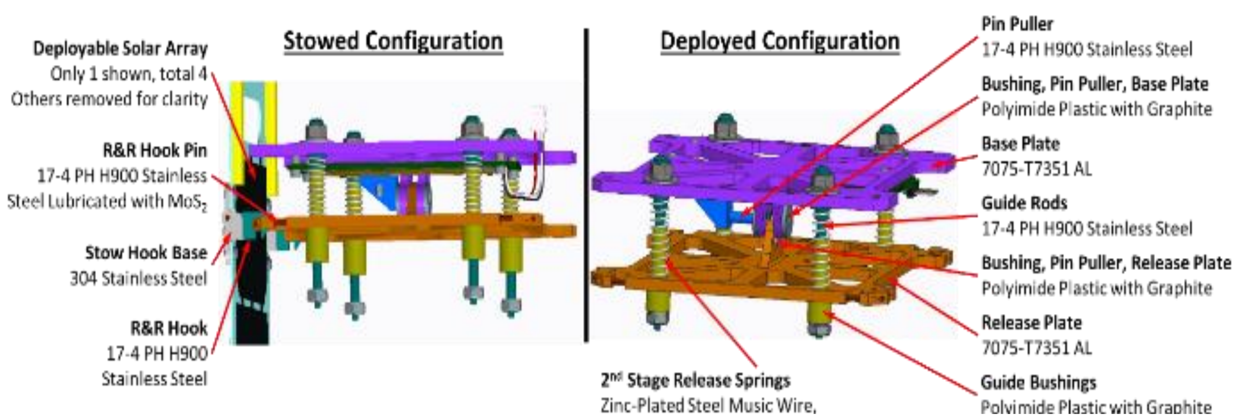

Figure 3: Retention and Release (R&R) Mechanism

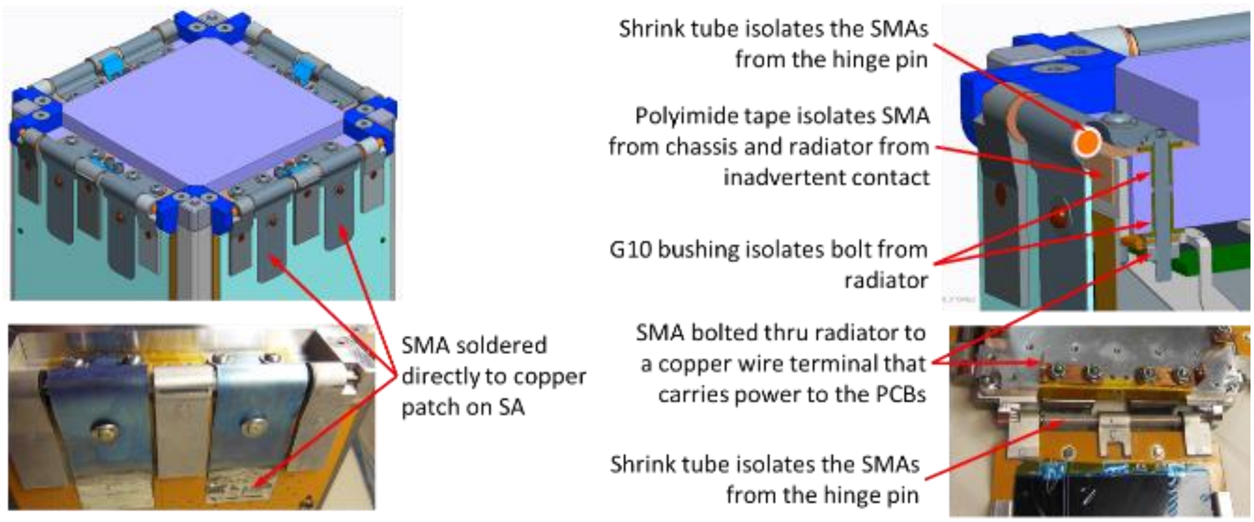

Figure 4: Hinge Mechanism and Component

a compression spring loaded plate on plain bearings. Once released, a passively driven SMA hinge mechanism, one for each of the four arrays, deploys each array to the desired deployment angle.

ALBus' initial temperature requirements for a safe solar array deployment were set to >100 °C, which exceeded any commercial SMA alloy capability. Therefore, the linear actuator consists of an SMA alloy with an atomic composition of $Ni_{19.5}Ti_{50.5}Pd_{25}Pt_5$ resulting in high transition temperatures above 100 °C, work output exceeding 15 J/cm^3, and high ductility. Thus, rods were drawn into a 0.508 mm diameter wire that was trained, cut into segments, and installed on a custom linear actuator. Five SMA wires were connected to guide rails. Once heated past the transition temperature using direct current (joule heating), each SMA wire contracts to pull its associated guide rail. The summation of the five SMA wires yields a cumulative displacement of 7.1 mm travel to pull the pin and release the second stage. Once the pin-puller releases the release plate, four compression springs move the plate, unlatching all four deployable solar arrays.

After the R&R releases, the solar arrays are free to rotate and each array is driven open by two preloaded superelastic SMAs per array (**Figure 4**). The final design of the hinge consists of two aluminum hinge knuckles that pivot over a hinge pin, two superelastic SMAs, and a latch to keep the solar array in the deployed state. In this design, a Ni-rich $Ni_{50.7}Ti_{49.3}$ (atomic %) superelastic alloy was selected to serve a dual purpose: (i) a spring load to open the arrays and (ii) a current carrying conductor to transmit power from the solar arrays. The superelastic material was rolled into a 0.2 mm thick sheet with a transition temperature (i.e., martensite start temperature) below 0°C. At room temperature, the sheets exhibited a superelastic plateau between 200 and 300 MPa, depending on the heat treatment used. This superelastic plateau denotes the effective start of the materials' stress-induced transformation from the stiffer phase known as austenite to the more compliant phase known as martensite. The superelastic sheets were machined into a flat-shape profile and then shape set to a specific U shape with a custom jig. After several iterations, shape-setting parameters were selected to be 550°C for 2 minutes followed by water quenching, which yielded the best form in terms of stiffness and reversibility after deformation. Upon deploying the arrays, a hard stop on the hinge brackets was designed to prevent the array from going beyond the required deployment angle, since the superelastic springs continue to apply a force. Once in the deployed state, a latch engages to act, as a failsafe to keep the arrays in the deployed state should an unknown or unexpected environment causing the springs to become too cold and temporarily lose their spring stiffness. The hinge design also transfers the electrical power from the solar arrays to the power management system. This is done by conducting electricity through the superelastic springs. To ensure a good electrical path and strong structural stiffness accommodations, the superelastic springs were riveted and directly soldered to the solar array panel and then attached to the radiator with screws. On the radiator end, the fasteners used to attach the superelastic springs also conduct the electricity to a copper lug. Wiring harnesses were soldered directly to the copper lug, which takes the electrical power to the power management system.

The analysis of these mechanisms was divided into three main areas: structural strength, mechanism tolerances (critical primarily to the thermal environments), and dynamic and kinematic analysis. The random vibration environment present during ascent primarily drove the structural strength of the parts. The thermal environment needed to be considered in both the R&R and hinge mechanisms. Due to the coefficient of thermal expansion mismatches between parts, the mechanism may bind at the temperature extremes if enough dimensional tolerance is not accounted for in the design. The critical analyses for these mechanisms, kinematic and dynamic analysis, were performed to ensure the mechanisms would have enough torque and force to release the arrays and deploy them at the appropriate angle. To aid in verifying that the mechanisms would deploy the solar arrays in orbit, an Automated Dynamic Analysis of Mechanical Systems (ADAMS) kinematic model was generated. The goals of the ADAMS model was to validate the design by showing all four solar arrays would deploy without adverse effects on the dynamics of the free-flying CubeSat. Moreover, the analysis was also used to evaluate some off-nominal pre-deployment rotations to see if there is a state when the arrays would not deploy or cause adverse effects on the dynamics of the free-flying CubeSat.

The frame and custom structure parts were analyzed using standard mechanics of materials methods. The components were analyzed to the random vibration environment as dictated by GSFC-STD-7000A, Table 2.4-3 to qualification level (14.1 G_{rms}). Factors of safety of 2.0 on ultimate and 1.5 on yield were used. Special attention was made to analyze the PCBs to ensure the vibration environment did not over-stress them and too many cycles to cause them to fail due to fatigue. A combination of hand calculations using plate theory and finite element analysis was used to determine the natural frequency, peak acceleration using Miles equation, the peak stress and finally estimate the time to failure.

ELECTRICAL POWER SYSTEM

Overview of System

The objectives of an advanced and flexible power management and distribution system are addressed primarily by the development of digitally controlled circuits, and associated control algorithms, for both the power management and distribution functions. The CubeSat structure was composed of a compartmented aluminum frame housing 4 subsystems, each with its own printed circuit boards (PCBs) as following: Auxiliary, Charging, Discharge, and Processor.

A PPT regulated the solar array power and charged the battery pack. The EPS provided power to a 100W load. **Figure 6**, on the next page, shows a simplified block diagram of the EPS.

Auxiliary Subsystem

Auxiliary power board created various low voltage sources for CubeSat and payload operation. The board housed one 9V DC-DC converter to provide power to the radio and one 5V DC-DC converter to provide power to the processor board. The board also provided the circuitry for remove-before-flight and footswitch logics as shown in **Table 3.** SMA R&R Mechanism is also housed on this board. It provided current heating to SMA actuator to deploy solar panels. The SMA connected to a switch as a load as shown in **Figure 5**. The last function of the Auxiliary board was the discharge enable and current sense. The processor sent an enable signal to the auxiliary board and sensed the current being drawn by the load. This circuit is shown in **Figure 7**.

Table 3: RFP and Launch Switch Logic Table

RBF Switch	Launch Switch 1	Launch Switch 2	Battery State	Mission
Pressed	Pressed	Pressed	OFF	Storage (deployer)
Released	Pressed	Pressed	OFF	Rocket
Released	Released	Released	ON	Ejected

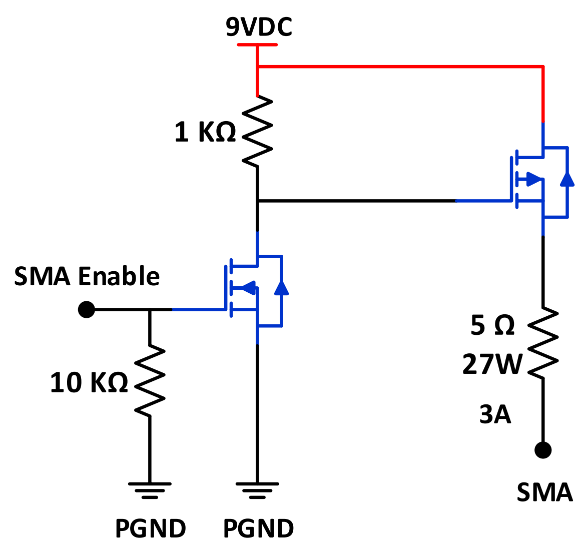

Figure 5: SMA Enable Circuit Schematic

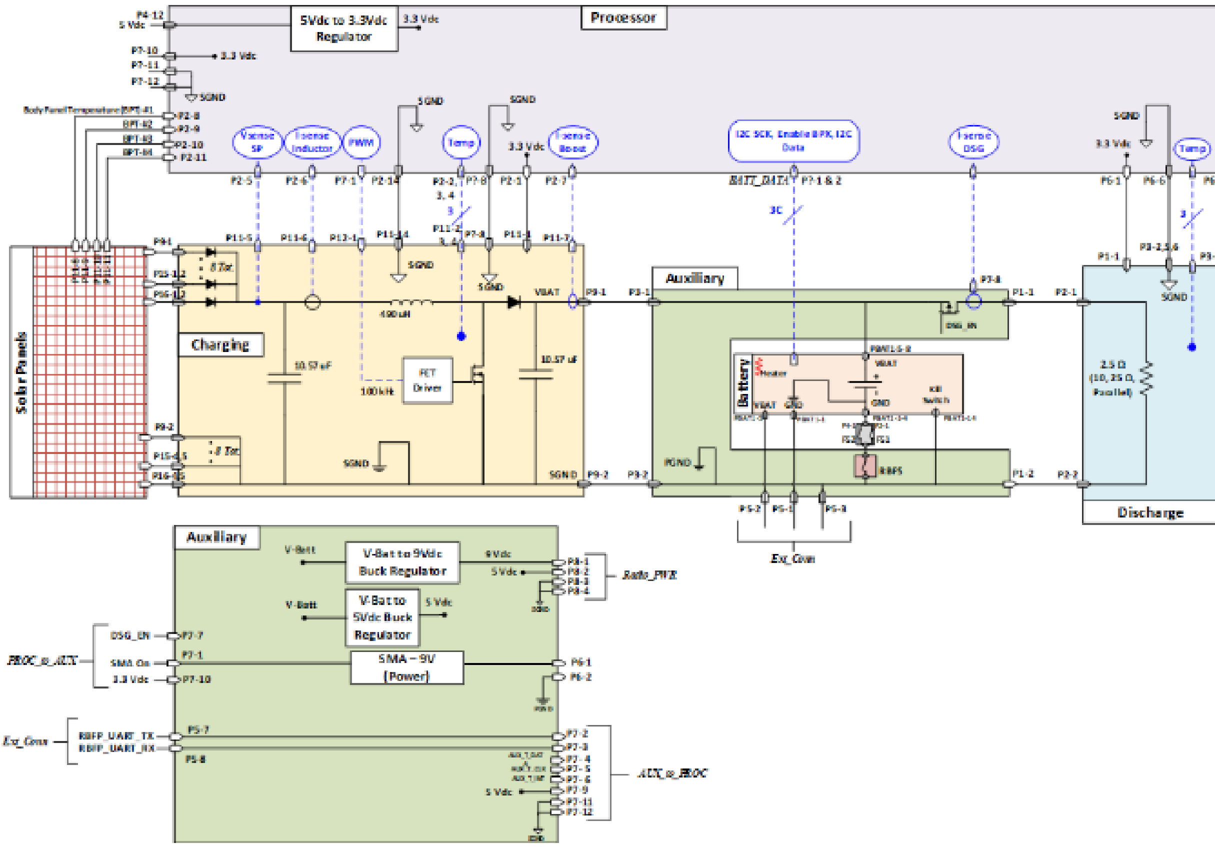

Figure 6: Block Diagram of Electrical Power System

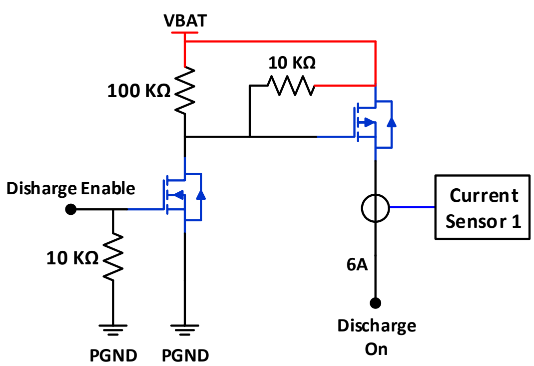

Figure 7: Discharge Enable Circuit Schematic

Solar Arrays

The ALBus solar panel configuration consisted of four body-mounted and four deployed solar panels with seven solar cells each as shown in **Figure 8**. Pumpkin designed and built the solar panels to NASA GRC's specifications (fits within 6.5 mm P-POD envelope). Additionally, ALBus utilized a Pumpkin CubeSat Kit 3U structure. While the body-mounted panels were relatively standard in their layout, the deployable panels had special design features to accommodate ALBus' SMA hinges and the SMA-based release mechanism.[2] All seven cells were ultra-triple-junction (UTJ) type solar cells with a nominal efficiency of 28.3% at 28°C. All panels were connected to a boost converter. Current sensing and temperature telemetry were reported only for the body-mounted panels while voltage sensing was reported for all panels. A protection diode was added in-line for each panel to prevent damage to the solar panels under reverse bias. Ground test of all panels were limited to I-V curves at ambient sun conductivity and illumination tests.

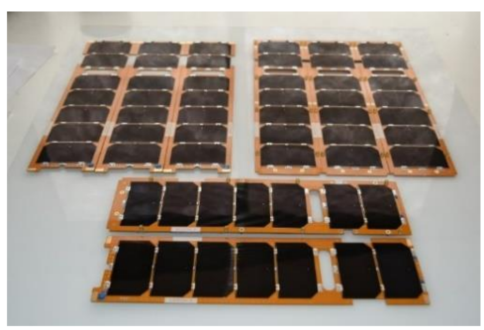

Figure 8: Custom Pumpkin PMDSAS Solar Panels P/N: 713-00825 (Deployable) and 713-00822 (Body-Mounted)

Battery Pack

The battery pack system consisted of a four series, two parallel configuration of 18650 Lithium-Ion cells. The battery pack, shown **Figure 9**, was manufactured by GOMspace, which is a manufacturer of nanosatellites for customers in the government, academic and commercial markets. The nameplate capacity of the pack was 5.2Ah over a voltage range of 12V to 16.8 V with a nominal voltage of 14.8V.

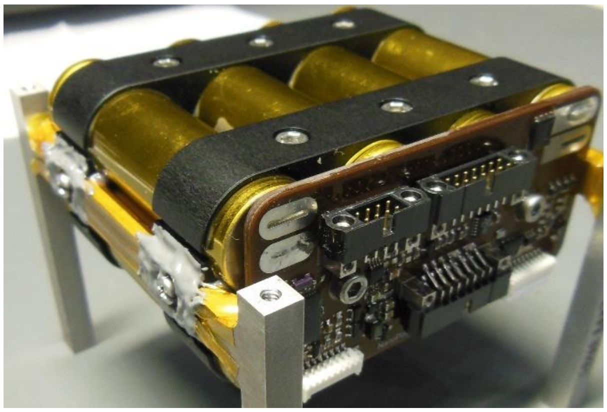

Figure 9: GOMspace Lithium-Ion Battery Pack P/N: BPX-P-2P4S-H

The battery pack's heater operation was verified in thermal chamber at GRC prior to normal ground testing. The boost converter regulated the charging and discharging of the battery pack. This protects the battery from over-current and under-voltage situations.

Charging Subsystem

The positive terminals of the solar arrays were connected through the body diode of the high side switch of a boost converter. The boost converter could be controlled for maximum power point tracking by an advanced control algorithm designed to find the optimal power point for efficient battery charging but by default, ALBus charges with constant current transitioning to constant voltage scheme. The boost converter charged the battery to a fixed voltage around 15.8V as shown in **Figure 10**. The boost converter circuitry is shown in **Figure 11**.

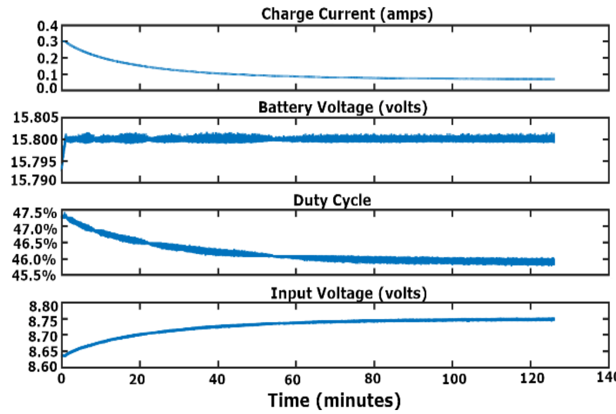

Figure 10: 1.5 Hour Battery Charge to 15.8V with Constant Current to Constant Voltage Transition

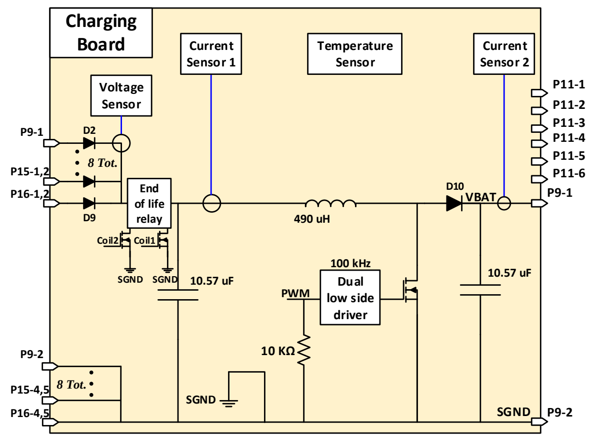

Figure 11: Charging Boost Circuit Schematic

Discharge Subsystem

Discharge subsystem takes battery voltage and produces 100W output to target load utilizing a bank of 10 high-power FET resistors in parallel. The resistors are attached to an aluminum heat sink at the end of the CubeSat to dissipate the generated heat into space.

Processor Subsystem

Processor subsystem is the main control system for ALBus. The flight computer is a Texas Instruments MSP430. Along with the processor, the board houses a 3.3V DC-DC converter to provide power to the processor and to the temperature sensors throughout the satellite.

EPS Subsystem Testing

The electrical power system went through two production iterations. **Figure 12** shows the engineering model of the EPS PCBs and its relative location within the satellite.

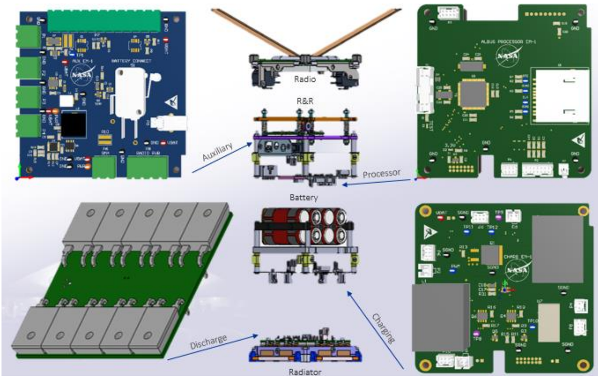

Figure 12: Engineering-Model Stack-Up

SOFTWARE

The microcontroller software was entirely written in C. The CubeSat flight computer was a Texas Instruments MSP430 running at 23.8MHz. Before flight, the microcontroller's non-volatile auxiliary flash memory would be programmed to communicate to the software that the CubeSat was in an "undeployed" state, in which the CubeSat solar panels and antennas were folded up prior to launch. After deployment from the launch vehicle, power would turn on; the CubeSat would read the flash and determine that it was in a "folded up" state. The flight computer would wait 15 minutes before opening the MOSFET to push current through the SMA to unlatch the solar panels and antenna, followed by a write to the flash memory to indicate that the CubeSat was now in a "deployed" state.

The software was responsible for charging the battery via a digital control system that runs a boost converter at a frequency of 8 kHz. Every loop (125 microseconds), two Proportional Integral (PI) loops with integrator anti-windup are executed—a constant current PI loop and a constant voltage PI loop. Each loop outputs its own new duty cycle set point. Then the minimum Pulse Width Module (PWM) duty produced from the two PI loops was used as the actual PWM duty set point, at which the PWM register would be set (PWM frequency used is ~93 kHz). The error of the constant current control system was computed by taking the maximum desired battery charging current (0.3A default) minus the actual sampled boost charge current. The error of the constant voltage control system is computed by taking a safe maximum charge voltage for a four-cell series lithium-ion battery (16.6V default) minus the actual sampled battery voltage DURING CHARGING (Note: the actual battery voltage can only be taken when charging is off). Hence, when the battery was undercharged, the constant voltage control system would saturate to max duty cycle because the constant current control system PWM duty cycle would be used. Likewise, when the battery was near full charge, the constant current control system would saturate to max duty cycle because the smaller constant voltage control system PWM duty cycle would be used. Furthermore, when input power for charging was significantly low, both the constant current and constant voltage control systems would saturate to max duty cycle, and this max duty cycle would be used as the PWM set point. A 60% max duty cycle was determined to be the most stable for this boost converter. **Figure 13** below shows a 3-minute charge snapshot with the four-cell max battery voltage arbitrarily set to 15.8V. The top graph is the boost current, which is 0.3A until after about 1 minute with the transition from constant current charging to constant voltage charging, after which the current starts decreasing. The second graph is the battery voltage bus WITH CHARGING ON, which increases to the 15.8V set point, and upon reaching this voltage, remains there with the decreasing trickle charge current. The third graph shows the duty cycle holding constant until after about 1 minute with the transition from constant current to constant voltage, after which duty cycle slowly decreases. The fourth graph shows the input voltage (measured at the output of the solar panel diode-OR circuit) holding constant until after about 1 minute with the transition from constant current to constant voltage, after which this voltage starts decreasing as the load (battery charge) current decreases.

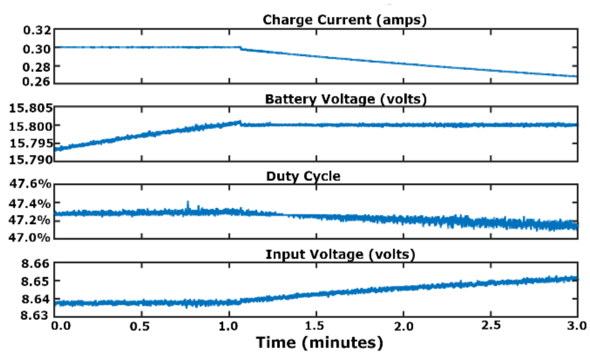

Figure 13: 3-Minute Battery Charge to 15.8V with Constant Current to Constant Voltage Transition

The charging algorithm contained a feature that could be enabled via a command for an experimental maximum power point tracking (MPPT) algorithm. If the MPPT algorithm was enabled, it would be engaged if and only if both the constant current and constant voltage P-I controllers saturated to max duty cycle (60%) due to low solar panel voltage. To save on microcontroller computation time, this MPPT algorithm tried to maximize charging current as opposed to charging power, saving the multiplication operation needed to compute the power value in watts. The MPPT algorithm was a "perturb and observe" type that would increment or decrement the duty cycle (perturb) every 50Hz. **Figure 14** shows a one-second snapshot of the MPPT algorithm in action, connected to four solar panels in parallel in the noon sun, and charging at a rate of approximately 0.33A. (Note: The max charge current was temporarily raised from 0.3A to 0.4A for the test so that the constant current algorithm would not engage (current limit) and disable the MPPT.) The top graph shows the dithering duty cycle. The bottom graph shows boost current in amps, which is being maximized. At 0.33A, the solar panel voltage dropped to about 7.1V.

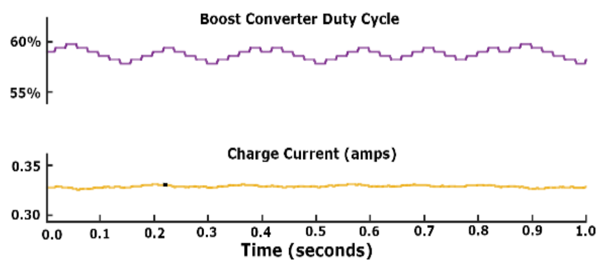

Figure 14: MPPT Charging (50Hz Perturb and Observe) Four Solar Panels in Parallel in Noon Sun

A custom MATLAB Graphical User Interface (GUI) was created to tune the charging control system and view sensor data in real-time. A custom C++ Dynamic Link Library (DLL) was written and created to serve as the bidirectional serial COM port interface between the microcontroller and MATLAB. The microcontroller was connected to the PC via a USB to serial converter. The MATLAB GUI would refresh every 1 second to update the plots. Approximately 20 different data channels from sensors (a mixture of 8, 16, and 32-bit) were received at a rate of approximately 300 samples per second using a 115,200 serial baud rate. Input data, such as proportional and integral control system constants, could be changed in real-time from GUI input fields and sent to the microcontroller. From the GUI, a step function could be set up and enabled to tune the control systems. **Figure 15** and **Figure 16** show tuning the current control system with a step function in which the reference jumps between 0.1A to 0.3A approximately every 0.2 seconds. **Figure 15** is un-tuned and overdamped. **Figure 16** is tuned and critically damped.

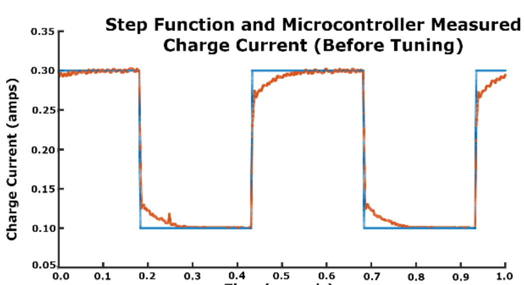

Figure 15: Tuning of the Current Control system, Un-tuned and Overdamped Tuning

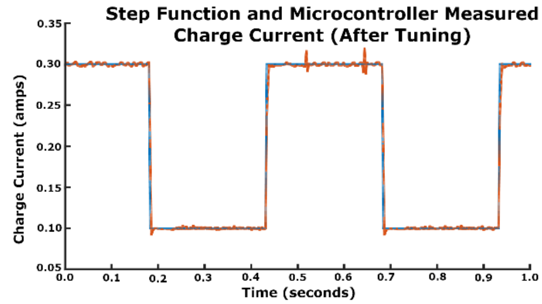

Figure 16: Tuning of the Current Control System, Tuned and Critically Damped

The CubeSat stored telemetry to a Secure Digital (SD) card over the Serial Peripheral Interface (SPI) bus. The CubeSat communicated to the ground station via an Astrodev Lithium 1 radio. The CubeSat, on boot up, would set up the radio to auto beacon every 15 seconds. The beacon would contain high priority telemetry such as battery voltage. In addition, the beacon contained a dynamic integer seed number that would be used by the ground station for generating a SHA1 hash to validate and secure the command it would generate and send to the CubeSat. Commands were issued to download telemetry stored in the SD card, as well as to enable the power discharge experiment if the CubeSat appeared in a healthy state. Discharge mode would occur if the battery was fully charged and would halt when either battery voltage or one of several thermal sensors reached a shutoff limit. Other commands include enabling an experimental maximum power point tracking (MPPT) algorithm after all primary success criteria had been met, as well as to enable a kill switch when end-of-life was reached.

The CubeSat software was architected with radio beaconing as a top priority. The CubeSat processor has access to the radio's hardware reset pin, and it will perform a hardware reset of the radio on boot up and in the event that the radio becomes unresponsive. The CubeSat processor has an internal watchdog timer, and in the event that the processor crashed, possibly due to a radiation Single-Event Upset (SEU), the CubeSat processor will fail to check in to the watchdog and will thus be rebooted in 5.6 seconds.

The CubeSat ground station GUI software consisted of a backend web server written in C++ that hosted a JavaScript webpage. This webpage listed all available command buttons and displayed telemetry received. The backend web server used a C++ software library called CivetWeb that enabled communicating with the web browser bidirectionally via WebSocket protocol. The backend web server communicated with an identical Astrodev Lithium 1 radio over a USB to serial converter using the Boost Asio library. A custom rack-mounted box containing this radio and a power supply was built and sent to the UHF team at NASA Wallops Flight Facility where ground station operations would be conducted over a Virtual Private Network (VPN). There was a requirement from the UHF facility for transmitting from the ground to the CubeSat that required toggling a relay via an applied voltage in order to disable receiving and enable transmission. To solve this issue, the radio manufacturer issued a firmware update to the ground radio that enabled the ground station software to set or clear one of the radio's General-Purpose Input/Output (GPIO) pins. This pin was connected to a MOSFET so that this relay could be toggled.

THERMAL ANALYSIS

Approach

Like most CubeSats, ALBus relies entirely on passive thermal control. The primary challenges for adequate thermal management for all SmallSats include limited external surface area for radiators to reject waste heat into space and limited thermal mass due to the small size of the spacecraft. Additionally, the high-power thermal transients (while exercising the 100W PMAD system to the internal dummy load) requires iterative analysis to predict hardware temperatures and ensure they are within component limits. Lastly, the lack of active attitude control adds additional challenges to providing adequate thermal control.

System-level thermal analysis was performed using the C&R Technologies Thermal Desktop (TD) thermal analysis software. TD is essentially a GUI pre- and post-processing package that utilizes SINDA/FLUINT, which is the NASA standard software for computation of thermal (and thermal-fluid) analysis of engineering systems. It is particularly useful in the analysis of space-based systems due to its built-in tools for calculating the thermal effects of space environments. A cutaway is shown in **Figure 17**.

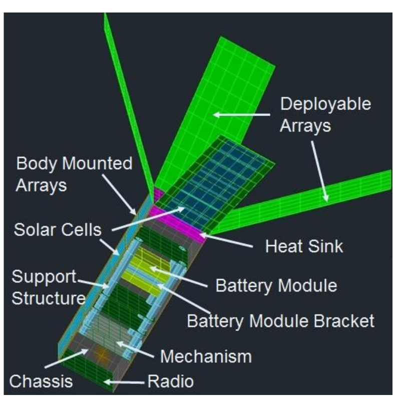

Figure 17: ALBus CubeSat Thermal Desktop Model (Cutaway)

From early in the project, simple thermal models were built (initially spreadsheet based, but later TD models) to evaluate the system level thermal impact of the conceptual design (3U CubeSat with deployable solar arrays). Basic trade studies were performed on various parameters affecting thermal control, including:

- Orbital environments – altitude, inclination, and beta angle
- Deployable Solar Array Configuration – Single/Double sided, deploy angle
- Spacecraft Attitude – +/- nadir pointing, longitudinal spin rate
- Internal configuration – board arrangement/order
- Resistive Load/Radiator – mass, placement/mounting
- Optical Properties

During most of the development, the launch vehicle and ultimate orbit of the spacecraft were unknowns. Therefore, thermal analysis was performed to ensure that the worst possible thermal environments that the ALBus might be launched into were analyzed. As with any thermal analysis, the goal is to ensure that the allowable spacecraft temperatures (survival, operational) would not be exceeded during any phase of the mission (from pre-launch to end-of-mission.)

Design

The system level thermal model contained all the major spacecraft components and subsystems.

Iterating the aforementioned design parameters, the spacecraft thermal design was defined. The operational constraints of the PMAD system placed the most limitations on the thermal-related design choices.

The PMAD subsystem waste heat (not the test load dissipation) was managed primarily by providing conductive pathways to the CubeSat frame and utilizing the solar array body panel as effective radiators. For the 100W transients, an aluminum mass was mounted at the end of the CubeSat adjacent to the deployable arrays. The exterior, or radiating surface, was covered with low solar absorptivity, high infrared emissivity silver Teflon tape. On the internal surface was mounted the 100W dummy load circuit board. As the PMAD subsystem design was modified over the development of the project, further thermal model analysis runs were required to ensure those component temperatures were maintained within the manufacturer's limits.

The isolation of the 100W load was key in managing component temperature in other parts of the spacecraft. The angle of the deployable solar arrays was optimized to maximize power and the likelihood that the spacecraft would orient itself in a +/- nadir pointing attitude (gravity gradient). This not only simplified the thermal analysis, but it ensured that the radiator surface on the 100W load bank would be pointing to earth or deep space. While both of these scenarios expose that surface to different thermal environments, the analysis confirmed that both would provide adequate thermal dissipation. The high thermal capacitance of the aluminum mass of the 100W dummy load and the bang-bang control of the load allowed adjustment of the duty cycle to ensure that electronics stayed within acceptable temperature limits for the different attitudes.

Testing

Thermal vacuum testing was performed at the NASA Glenn Vacuum Facility 10 (VF-10) as shown in **Figure 18**. The facility is equipped with a liquid nitrogen cold wall and a (clean) turbo pump system capable of pressures of below 10^{-6} torr. The cold wall is also fitted with cartridge-type heaters that allow the "cold" wall to be heated for hot case conditions (or bakeout).

Figure 18: Vacuum Facility 10 (VF-10) in Building 16 at the NASA Glenn Research Center

Development/engineering units were tested at worst-case cold and hot conditions to allow data to be collected to verify calculations and modeling used to size the 100W dummy load radiator and to provide correlation data for cold/hot temperature component temperatures.

The flight unit bakeout testing, per the project requirements, was performed in VF-10 as shown in **Figure 19**.

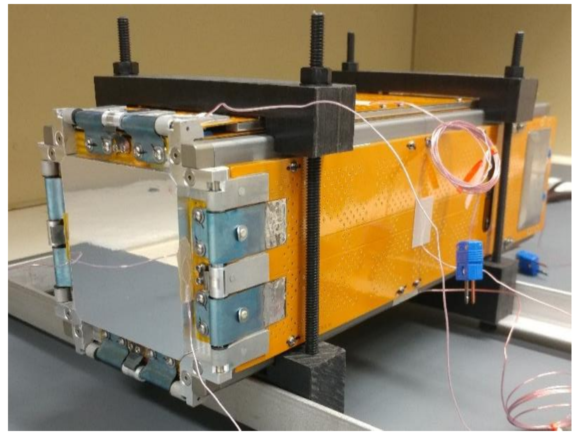

Figure 19: ALBus Flight Unit being prepared for TVAC testing VF-10

While planned, full functional and performance testing of the flight unit was not completed under thermal vacuum (TVAC). This was due to delays caused by the previously mentioned battery anomaly and other rework.

This was accepted at the pre-ship review as a large amount of TVAC testing had been performed on the engineering unit at various levels of hardware maturity. Those results, combined with the subsystem testing that was performed before-and-after the bakeout showed no anomalies and added to the confidence that the design and the hardware, as-built, would operate within specifications.

SYSTEM INTEGRATION

ALBus CubeSat's integration of its subsystems was a multiphase process. The middle stack of PCBs are assembled together as shown in **Figure 20** prior to sliding it into the chassis as shown in **Figure 21**. In parallel, the discharge board and deployable solar arrays were assembled together. They were then connected to the chassis on one end of the CubeSat. Finally, the radio was connected before the body-mounted solar arrays. The fully integrated flight system is shown in **Figure 22**Error! Reference source not found..

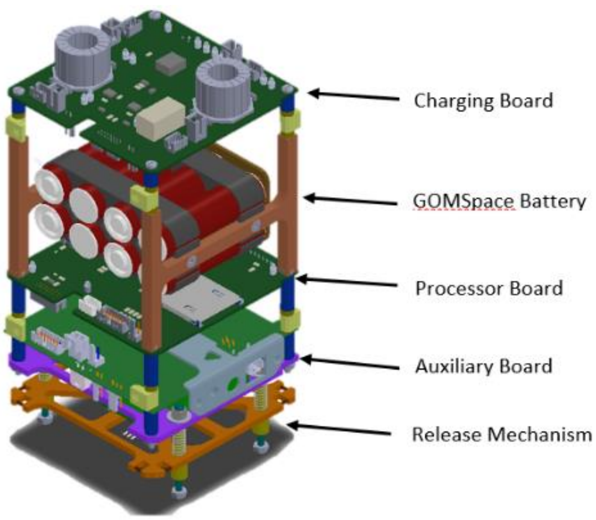

Figure 20: Middle Electrical Stack-Up

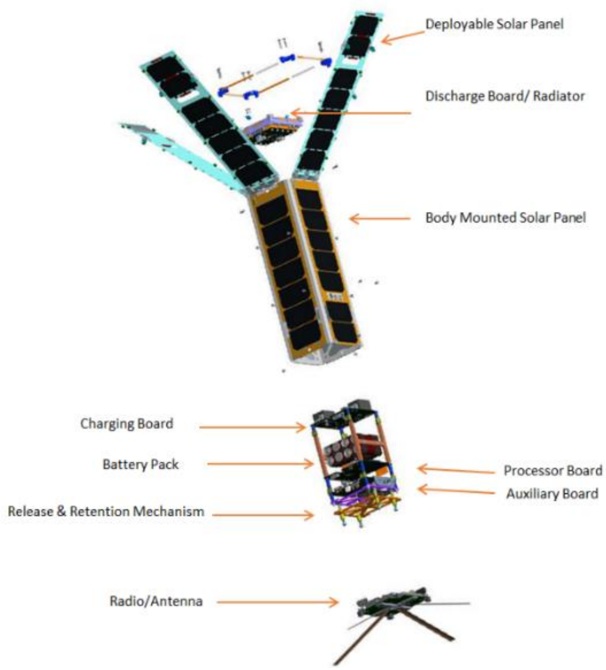

Figure 21: Full System Stack-Up

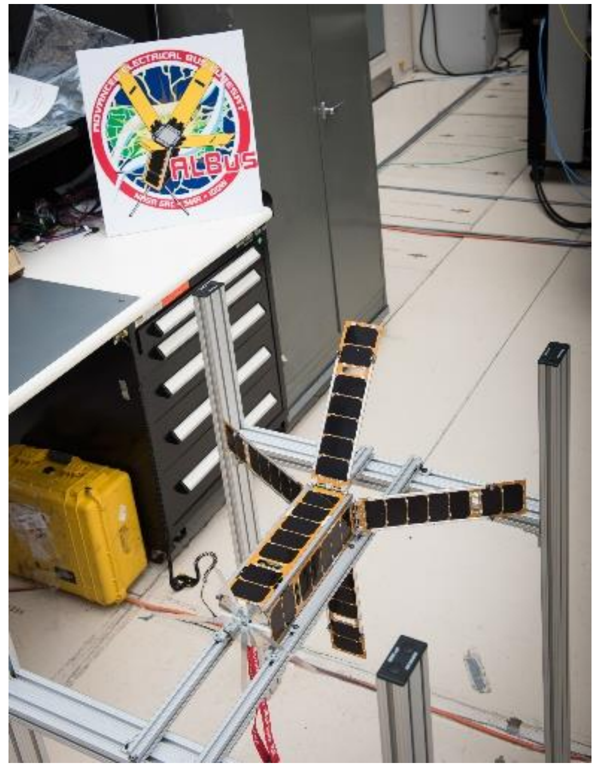

Figure 22: Fully Integrated Flight System

SYSTEM GROUND TESTING

Environmental testing and system functional testing were necessary to verify the specified requirements for the CubeSat flight unit. The CubeSat project followed guidance for Protoflight testing. The Launch Service Provider requirements document (LSP-REQ-317.01B) provided guidance and requirements on environmental testing of the Deployer and CubeSat unit defined in Table 1 - PPOD and CubeSat Test Environments Testing Table, and per Figure 1 - Dispenser and CubeSat Qualification and Acceptance Test Flow Diagram derived from MIL-STD-1540 and GSFC-STD-7000A. Additional testing requirements information is provided in Section 4: Testing Requirements of CubeSat Design Specification Document (Rev 13). The System Verification Test Flow, shown in **Figure 23** on the next page, was used for CubeSat Vibration Testing and Full Functional Test (FFT) as well as CubeSat Thermal Vacuum Testing and FFT.

The ALBus Project followed nearly all of the tests shown in **Figure 23** , except TVAC, which the Project decided not to perform. The project decided the risk in damaging

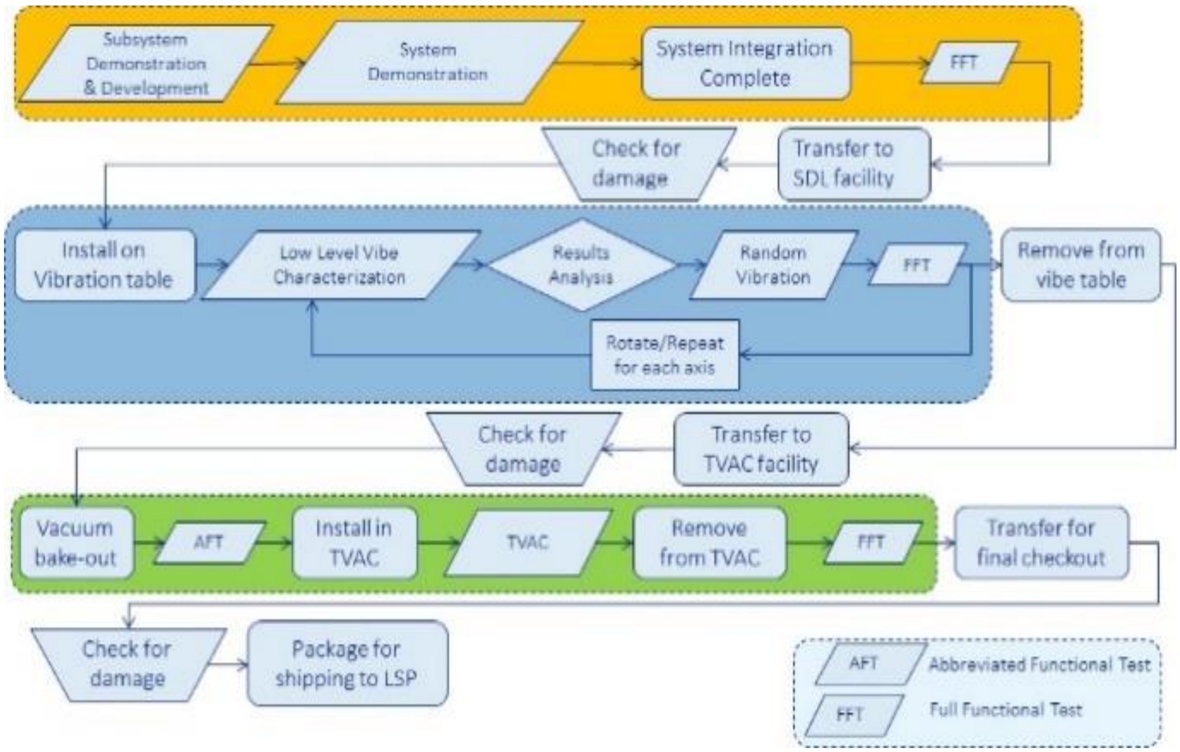

Figure 23: ALBus Verification Flow

the radio was too great. The FFT test verified the function as if the CubeSat was going through mission sequences. The FFT started with the pre-integrations tasks, such as inspections and battery charging. The deployment of the solar arrays and the antennas demonstration followed and verified proper deployment and communications. Once the communications were verified, an inspection of the software parameters were performed. Both the charge and discharge functions were tested. Lastly, the satellite software was returned to flight state and deployables were reset.

The vibration test was performed at Glenn Research Center's Structural Dynamics Laboratory (SDL) facility. The ALBus was integrated into the dispenser before installation on the vibration table. All three axis were tested at MPE + 3 dB for 2 minutes for each axes. Once the vibration test was completed, the FFT was performed.

The final major test was the thermal vacuum bake-out. The thermal vacuum bake-out test was performed at Glenn Research Center's VF 10 facility. The CubeSat was baked to 70° C at 1 x 10^{-4} Torr for a minimum of 3 hours once thermal stabilization was achieved. After the thermal vacuum test, the FFT was performed.

SYSTEM ANOMALIES

Battery Pack

During a functional test of the flight system, a component in the Commercial Off-The-Shelf (COTS) battery pack's battery management circuitry failed. The failed component was a metal–oxide–semiconductor field-effect transistor (MOSFET) that enables/disables the battery pack when commanded through a pin on the battery connector by the CubeSat footswitch or the Remove-before-Flight Pin (RBFP). The MOSFET failed during a battery charging cycle when the MOSFET was in the 'on' state. As a result, the battery pack remained enabled when the RBF pin was inserted and the footswitch engaged (non-compliance with ICD requirement). Testing confirmed that the footswitches and RBF pin switch were fully functional and sending the correct enable/disable signals through Pin 14 to the battery pack.

The damaged battery pack was safely disabled, removed from the flight system and sent back to the vendor. Vendor confirmed that only the MOSFET failed in the COTS Battery Management System (BMS) circuit. BMS circuit was repaired by the vendor by replacing the MOSFET and returned. Testing of the repaired pack confirmed that it was again fully functional. An undamaged flight spare battery pack was integrated to the flight system.

A series of tests and coordination with the battery pack vendor narrowed the most probable root cause of the failure down to missing steps in the battery charging procedure. The charging procedure used at the time of the failure did not explicitly state that the battery circuit (specifically the MOSFET) had to be enabled via an external kill switch prior to turning on the external power supply to charge the battery. This scenario was replicated with an undamaged MOSFET and repeatedly resulted in the failure experienced during the functional test. Additional precautions have been added to the battery charging procedure. Using a better power supply with better control circuitry for battery charging to avoid transient voltage spikes. Charging at a slower rate than the charge rate during the failure.

Solar Cell Damage

Cracked cover glass on body mounted solar array. Repaired solar panel by replacing damaged cell.

Flight Processor Board

A 3.3 V regulator on the processor board failed during checkout of the flight electronics boards. Failure was attributed to workmanship in the assembly process. A new processor board was assembled and checked out.

Flight Radio

A component in the COTS radio was damaged during checkout with the flight electronics. Failure attributed to incorrect orientation of connector. Connector orientation was marked for proper assembly. Radio was repaired and checked out.

INTEGRATION INTO SPACECRAFT

To prepare for integration of the ALBus CubeSat, a CubeSat Acceptance Checklist (CAC) was completed. The CAC consists of dimension and weight measurements to ensure the ALBus conforms to the Launch Service Provider's (LSP) dispenser. Meeting the CAC helps to ensure a proper jettison on orbit. Then the fully assembled ALBus CubeSat was packaged into a hard case, foam filled container and delivered by hand from NASA GRC to the LSP's integration facility. Once there, the team reviewed the CAC with the integrator and performed final integration preparations such as charging up the batteries to ensure they were full. The GRC team ended up returning two more times to the LSP's integration facility. The first time, due to launch delays, the ALBus batteries needed to be charged up. The second return was to correct a radio issue that was discovered last minute. Both returns to the LSP required removing the ALBus from the dispenser and reintegrating it. However, the radio issue was more complicated. It required shipping custom ground support electronic equipment to the LSP and deploying the CubeSat's mechanisms to communicate with ALBus' radio via its deployed antennae. This ensured the radio fix was performed correctly. The CubeSat was reintegrated successfully, shipped to the launch site for integration to the rocket, and proceeded with launch operations. **Figure 24** is a photo taken by ALBus' LSP, Rocket Lab USA, showing all CubeSats successfully completing a fit check of the CubeSat dispensers.[3]

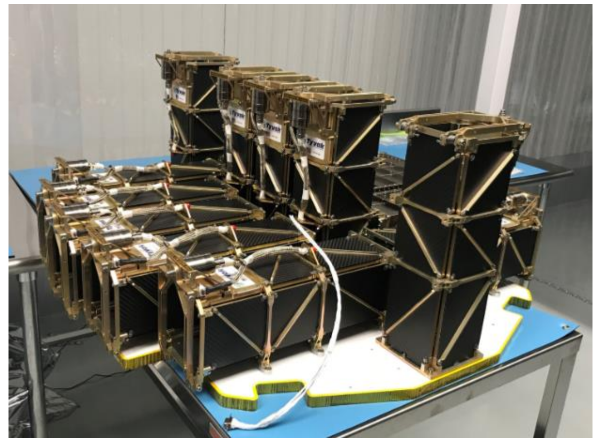

Figure 24: Rocket Lab USA's CubeSat Dispensers for ELaNA XIX

LAUNCH AND FLIGHT OPERATIONS

The ALBus CubeSat project had a successful launch Saturday, December 16, 2018 from Mahia, New Zealand via Rocket Lab on Electron. The CubeSat solar panels deployed on Sunday, December 16, 2018 at 2:42am Eastern and Wallops Flight Facility (WFF) picked up the satellite's beacon signal at 7am via their spectrum analyzer. The spectrum analyzer has a ± 7 MHz range, and the Wallops team, while searching for another satellite with a nearby frequency, saw the ALBus beacon signal coming in very strong at the exact beacon period of 15 seconds on the exact frequency of 400.4 MHz. This signal reception immediately validated the success criteria for a successful deployment of solar panels and antennas. Had the antennas failed to deploy, the radio would have most likely destroyed itself as the antennas, when stowed, are touching the CubeSat's metal chassis.

Unfortunately, that series of beacons that were received four hours after ALBus deployed was the first and last time a signal was heard from it. Due to scheduling, ALBus was not able to get a first pass scheduled to begin the search until Monday, December 17, 2018. There were seven scheduled ground station passes for the week of the 17th, and four of them were from the ELaNa XIX mission. Several attempts were made until the government shutdown on December 26, 2018, at which point the ALBus team was unable to continue searching. During that time prior to the shutdown, around ten different NORAD objects were iterated through without success, covering about three to four objects at a time per ten-minute pass. The team tried all the objects several times, looking for the beacon signal on the spectrum analyzer, but without success.

During the government shutdown, WFF ground station technicians continued to look for the ALBus 400.4 MHz beacon while they supported other missions. A few more search attempts were made after the shutdown, but eventually the search was called off early February 2019.

LESSONS LEARNED

ALBus CubeSat experienced many the common development challenges. Several lessons learned include:

- Friction forces are difficult to quantify without validation from hardware tests.[1]
- Sizing analyses such as loads, mechanisms, and kinematics should be done early on along with the design concepts even if firm inputs are not available. Do not focus only on the CAD design aspects.[1]
- Building an EDU or 3D printing hardware to test is key in any new development to quickly uncover assembly issues and evaluate actual functional performance. Do not only rely on analysis only.[1]
- Even though it can be easy to create dynamic and kinematic models for mechanisms, it may be very difficult to get meaningful correlations with the actual test data.[1]
- SMA applications should be evaluated from a system level. For example, although the hinge mechanism uses simple SMA sheets, the integration process that involved bolting, riveting and soldering proved to be very difficult.[1]
- The ALBus design used an USB-C connector as ground connect. The USB-C is compact, making this connector an excellent choice. The ALBus USB-C connector did not follow the standard pinout. Most off the shelf USB-C do not carry all the conductors. ALBus purchased female and male breakout connectors. ALBus made cables out of the breakout connector. Evaluating off the shelf cables would have been quicker and cheaper than fabricating cables. The lessons learned are to use the standard pinout and find an USB-C off-shelf cable that meets your needs.
- The ALBus design charged the battery through the protective circuit. During the first Full Functional Test (FFT), the charge was to take hours; however, the deployment happened in 15 minutes after the processor was powered. A decision was made to charge the battery while the CubeSat was deployed. During the charging, a Field-Effect Transistor (FET) in the battery protective was damaged causing the satellite to remain on with the Remove before Flight (RBF) pin. The lessons learned are to charge the battery through the unprotected circuit and include a requirement to charge the battery without powering the deployment circuit.
- The ALBus design had a resettable deployable. This feature proved invaluable during environmental testing and made an expected update to the software before launch. Having a resettable deployable is easier than replacing parts to set deployable. ALBus was integrated for launch after discovering the wrong radio frequencies were programmed. Being able to de-integrate, deploy, update the frequencies, and reset the deployable made updating the satellite much easier. The lesson learned is that resettable deployment makes troubleshooting easier.
- The ALBus design would not allow disabling the radio through the USB-C connector. ALBus decided not to perform the TVAC due to the risk of damaging the radio while in a small metal chamber. The lesson learned is include testing when designing.
- It is important to start mechanical and electrical integration early in the design to make sure harnessing and cutouts are established cutout zones.
- ALBus had two iterations of PCBs because there were various design changes needed for flight. It is important to have an engineering bench top model to test out the design to ensure functionality in flight. The more testing that can be done on the flight representative system, the better.
- Having an ability to disassemble the CubeSat in case anomalies arise is important. ALBus ran into a battery anomaly in which the entire CubeSat had to be disassembled to troubleshoot and replace the battery.
- Start thermal analysis early. Use spreadsheet modeling to establish your design envelope for parameters that have the most influence on your thermal design/performance. Use these calculations to establish what is reasonable for a certain concept or approach (e.g. if back of the envelope calculations show you need 5 kg for your proposed solution for your CubeSat, then you know you that is a dead end. Move on.)
- Do your best to define your potential (thermal) environment. You can limit your options by having to over constrain your design to fly in every possible environment. At the same time, do not go too far in the other direction.
- Work to establish good communication with the subsystems that will have the most impact on your thermal control solution. Make sure you understand what their needs are as they relate to thermal design and analysis.
- It has long been assumed that thermal-related design flaws and lack of analysis or testing is a leading cause of mission failures. Do as much analysis and testing as

your resources allow. Again, start early. Build models that sweep the design space and perform thermal development tests that inform your choices as early in the project as feasible.

- Start writing your test plan as soon as possible. It is a living document. It will change and grow as you learn. Share it with the rest of the team and get feedback. Share it with more experienced engineers and get advice.
- Once you have a reviewed test plan, do as many dry runs with the engineering unit as possible. It will pay off and give you confidence when it comes time to test the flight unit. It is also the best way to refine your test plan.
- If possible, perform a full thermal vacuum test with functional and performance testing. However, if that is not feasible for any reason, do as much as you can. Something is better than nothing and may allow you to discover a design or build flaw that would otherwise jeopardize the mission.
- Passive thermal control components may have to be removed or reworked because of rework for other subsystems. Design with ease of integration/disassembly/repair in mind.
- Thermal related or not, keep as many aspects of your design as simple as possible. No matter how simple you think it is it will balloon in complexity just with the passage of time.
- As most SmallSat projects are still educational in nature, be sure to get advice from anyone with previous experience. Ask for "sanity checks". Mine more senior engineers (or students) for as much of their experience as they are willing to give.

CONCLUSION

The ALBus CubeSat is an example of technology advancement for CubeSat applications. It attempted to demonstrate an increase of maximum power output capability to a target load as well as reduce the mechanism risk from deployments of solar arrays. While the CubeSat could not be demonstrated for the power system, the project still illustrated the potential in CubeSat applications for power where space and weight are limited.[1] It also successfully demonstrated the use of SMA as a reliable deployment mechanism.

ACKNOWLEDGEMENTS

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