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Improved CubeSat Mission Reliability Using a Rigorous Top-Down Systems-Level Approach

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Abstract

The University of Southern California's Space Engineering Research Center (SERC) developed a 3U CubeSat, designed, built, and tested by graduate and undergraduate students utilizing an engineering ”teaching hospital” environment for hands-on learning. The 3U “Dodona” mission is to send to orbit a payload for our research sponsor, providing power and data connections, as well as data forwarding to the ground using on-board UHF telemetry system. Roughly 2U of the satellite are reserved for flight processors, power systems, and attitude control system and sensors. Generally, university CubeSat efforts are challenged by lack of time, funding and personnel turnover that end up skimping completion of a full hardware checkout and in-depth failure mode analysis prior to delivery. Statistically, 50% of CubeSat missions fail, not due to a lack of knowledge but a lack of documentation and testing generally in the critical integration and test phase. To avoid this, SERC sought to develop a rigorous top-down approach from the project start, focusing on validating requirements of the sponsor and the realistic/potential failure modes from similar missions. Using a top-down approach, we were able to update the components from a legacy bus system and setup detailed test procedures to test all subsystems, both independently and integrated, in the same manner that it would operate in flight, throughout the integration and test phase of the project. As background, the Dodona CubeSat is built on the Pumpkin COLONY I bus architecture, from one of the last remaining from the initial stock manufactured in 2008. The satellite will be placed into a 650 km altitude sun-synchronous dawn-dusk orbit in order to maximize the power collected by the sun-pointed petal array, using reaction wheels to maintain a sun-pointing orientation. For the key integration and test activities for verification the student team focused on test and validation both in the lab using simulated sensor inputs for hardware-in-the-loop testing, breakout boards for new payload board designs, and software defined radios for telemetry testing. This testing campaign, combined with a series of burn-ins, allowed our team to test the system as it would be in flight with a focus to improve the reliability of the system and ensure mission success upon reaching orbit.

Keywords: Satellite, Integration, Test, University, CubeSat, Reliability

Nomenclature

DODONA .... USCs third CubeSat

PC-104 .... CubeSat Bus Interconnect Inteface

U ...... Standard CubeSat Unit (10 cm X 10 cm X 10 cm)

Acronyms/Abbreviations

ADACS ..... Attitude Determination and Control System

BCR .... Battery Charge Regulator

BEC .... Bus Extender Card

BIBO .... Bus-In Bus-Out module

CAD .... Computer Aided Design

DBB .... Dual Battery Board

DMM .... Digital Multimeter

DOD .... Depth of Discharge

DOF .... Degrees of Freedom

EPS .... Electrical Power System

GNC .... Guidance Navigation and Control

GPS .... Global Positioning System

HITL .... Hardware In-The-Loop

I&T .... Integration and Test

NASA .. National Aeronautics and Space Administration

PEC .... Payload Extender Card

PWM .... Pulse Width Modulation

RBB .... Remote Battery Board

RBF .... Remove Before Flight

SDR .... Software Defined Radio

SERC .... Space Engineering Research Center

STK .... Systems Tool Kit

TT&C .... Telemetry Tracking and Control

UHF .... Ultra High Frequency

USB .... Universal Serial Bus

USC .... University of Southern California

UTJ .... Ultra Triple Junction

1 Introduction

Dodona is the third CubeSat designed and built by students at the University of Southern California's (USC) Space Engineering Research Center (SERC). Its name is derived from the ancient Greek oracle of the same name, following tradition at USC. Dodona is the oldest of the Hellenic oracles, an ancient oak tree, and priestesses and priests would interpret the rustling of the oak leaves to provide insight and wisdom [1]. In a similar fashion, the team at USC will interpret the RF beacon packets transmitted by the *Dodona* spacecraft to determine the status of the payload and collect the science data for the mission.

As a university CubeSat mission, the SERC team was aware of the success rate of university run CubeSat missions; statistically, 50% of student CubeSat missions fail [2], not due to a lack of knowledge but a lack of documentation and testing, generally in the critical integration and test phase. To avoid this, the SERC sought to develop a rigorous top-down approach from the project's start, focusing on validating requirements of the sponsor and the realistic/potential failure modes from similar missions. Using a top-down approach, we were able to update the components from a legacy bus system [3, 4] and setup detailed test procedures to test all subsystems, both independently and integrated, in the same manner that it would operate in flight, throughout the integration and test phase of the project.

This paper will go into detail about the top-down approach used, highlighting select design choices made along the way to promote reliability, as well as the integration and test campaign used to qualify all the components individually and the system as a whole.

2 Mission Description

The 3U "Dodona" mission is to send to orbit an approximate 1U payload for our commercial research sponsor, providing power and data connections, as well as data forwarding to the ground using an on-board UHF telemetry system. The remaining 2Us are reserved for flight processors, power systems, and the attitude control system and sensors.

While the primary goal of our sponsor is to obtain validation from their payload to qualify it for space operations, the primary research goal for USC is to validate a new in-house B-dot magnetic detumble controller, and a unique rotating beacon system to maximize health and status data downlink over amateur band frequencies.

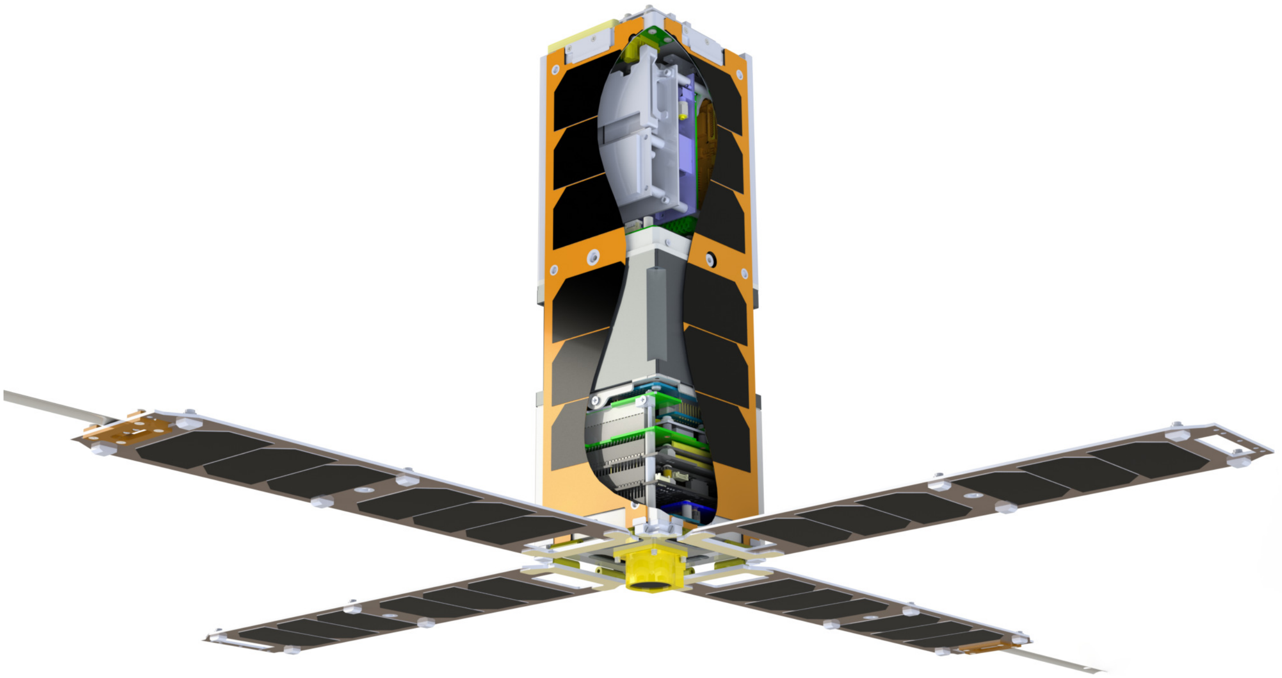

Fig. 1: DODONA Spacecraft Render (with cutout)

3 Spacecraft Overview

The design of the Dodona spacecraft is based on Pumpkin's design for their Misc II Colony I 3U CubeSat bus, utilizing the Aeneas project, USC's second CubeSat, hangar queen [4]. The modified design includes a custom ≈1.2U structure for the customer's payload modules, also allowing space for the Payload Extender Card (PEC) that provides an interface to the payloads, and a new GomSpace AX-100 UHF radio transceiver for bi-directional communication with USC's groundstation. The lower 1U is composed of the traditional Colony I subsystem, namely the Pumpkin Motherboard and PIC24 flight processor, a Clyde Space power regulation and battery system, a health and status beacon, and an interface board for the MAI-200 reaction wheel attitude control system (from Adcole Maryland Aerospace), which is mounted between the lower bus and the payload structure (the remaining 1U). Additionally, the CubeSat has fixed body-mounted solar arrays, deployable solar arrays, and a sun sensor from Sinclair Interplanetary. Fig. 1 above shows the physical appearance of the spacecraft, highlighting its deployable solar arrays, arranged in a petal configuration around the on-board sun sensor for accurate sun tracking to maximize power generation.

4 Integration and Test Plan

In order to ensure a successful mission, the Dodona student team developed a rigorous top-down approach to incorporate Integration and Testing (I&T) from the project start. The team at USC set out to emulate industry practice for integration and testing operations as much as possible in a university environment. To do this, procedures were created for all integration and de-integration operations, detailed test reports were implemented to keep records of testing, and traveler documents were used to keep track of all pieces of hardware on the satellite. The integration and test team worked with each of the satellite subsystems, as well as the systems engineer, to create a testing campaign to verify the functionality of the hardware and software, and qualify it for the mission, while minimizing the time and resources required for testing due to the accelerated schedule and limited budget of the project. During all integration and test activity, one team member was designated as the I&T lead to convey the procedure to those performing the activity and to provide independent verification of the task.

Procedures and test reports were implemented to ensure that all tests and integration operations were well defined, repeatable, and well documented for all teams working on integration. Due to the tight schedule of the project, as well as student class schedules, there were multiple teams working on integrating the hardware, so this documentation setup proved invaluable to inform teams on what tasks other team members had already completed and about any issues that may have arisen. Thorough documentation allowed the team to debug system level hardware issues by having the ability to trace all pieces of hardware used throughout the various tests, ultimately isolating the component causing the issue and correct it. In order to facilitate the use of these procedures and test reports during cleanroom operations, a set of large wall mounted monitors were installed just outside the cleanroom, visible to all and connected to the workstations in the cleanroom such that any integration diagrams or terminal window outputs could be seen by all integration team members, whether in the clean room or outside.

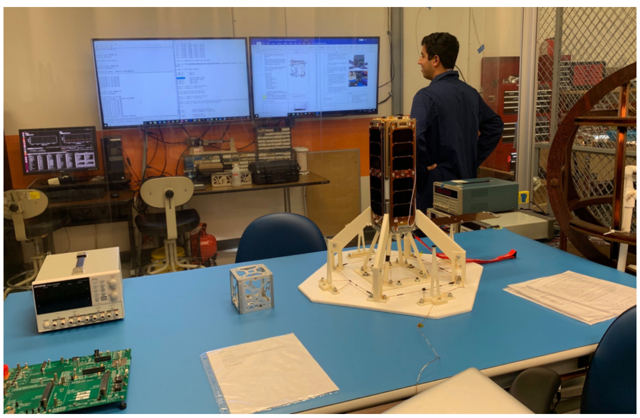

Fig. 2: Easy Display Monitors Outside Cleanroom

In addition to procedures and test reports, each piece of hardware for the satellite (including engineering models) had a document associated with it, known as a *traveler*, to document all aspects of the piece of hardware, including its serial number, related documentation or procedures, ground support equipment, any known issues or anomalies, as well as its current integration and test status. An example document can be found in Appendix A. Although traditionally a traveler is a physical piece of paper that travels with the hardware wherever it goes, the Dodona team opted to use a digital traveler system to save paper, maintain version control, and reduce clutter in the cleanroom.

The team also took advantage of collaborative platform software tools, specifically Confluence and GitLab. GitLab provided useful as multiple unit level tests could be written outside of the clean room and pushed to the Git repository. Once in the cleanroom for testing with flight hardware, the tests could very quickly be accessed, compiled, uploaded, and tested.

5 Subsystem Level Testing

Subsystem level testing was the first step in the process of hardware testing for the entire system. To eliminate unnecessary risks during hardware integration, the SERC I&T plan required that all components undergo standalone functionality testing before they could be integrated into the CubeSat stack. This enables verification of all hardware without any risk of damaging other components of the satellite during testing. As most components require data and/or power inputs from other systems on the spacecraft to perform correctly, this is handled by using simulated inputs during standalone subsystem level testing, such as power supplies and resistors to simulate battery systems, and Arduino microcontrollers to simulate data inputs.

Each subsystem team coordinated with the I&T team to build a testing roadmap, including determining what components and modules required testing, what materials were required to properly setup and execute the test, what conditions the test was meant to verify, and what data collection the test required. All this was documented in the form of testing procedures and testing results, in order to allow repeatable testing and traceability for the test results. To highlight the process the team followed in each subsystem, the following section outlines the Dodona GNC subsystem testing campaign.

5.1 Guidance, Navigation, and Control (GNC) Testing

For the Guidance, Navigation, and Control (GNC) system, a full Hardware-In-The-Loop test (HITL) is a comprehensive test that allows the testing of the GNC system as a whole replicating environmental inputs and controller outputs to ensure the hardware and software is working nominally. However, due to time and budget constraints, the Dodona GNC system was not able to reach full HITL testing and a piecewise qualification test regime was developed to qualify the components of the subsystem individually instead.

This was done by compartmentalizing the system into as many interfaces as possible (sensor to controller, controller to actuator, inter-controller) and ensuring that the correct data and operations were being passed across these interfaces and that the relevant hardware was responding correctly. If this could be proven in all modes of operation, then the GNC system could be said to be reliable.

The attitude control system for Dodona is a 3-axis stabilized control system capable of precise pointing. To ensure mission success, accurate sun pointing must be achieved so that the solar panels are fully illuminated and can continuously power the payload. As a whole, the following components function as the inputs and outputs for the GNC software:

1. 3-axis reaction wheels (contained within the MAI-200)
2. 3-axis magnetic torque rods (contained within the MAI-200)
3. a MicroMag3 3-axis magnetometer
4. 3x ADIS16260 gyroscopes
5. a Sinclair SS-411 Fine Sun Sensor

In order to use an experimental Bdot detumble controller developed at the SERC, custom control laws were developed to override the existing controller built into the MAI-200 system, allowing for different methods of detumbling. These custom routines are computed using the onboard PIC24 flight processor, and sent to the MAI-200 as override commands. All the hardware components above that interface with the MAI-200 needed testing to ensure they responded accurately to relative commands and could perform as required by the custom routines, detailed below:

1. Testing characterized the power usage of the MAI-200 at different reaction wheel speeds and the results were used in the initial tip-off detumble simulations (see Section 7.1). Full characterization was completed by running the MAI-200 reaction wheels at different steady state rotation rates, as well as running at different torque inputs, measuring the power draw of the system using a digital multimeter (DMM). In addition to the power data acquired, this test allowed the system response to custom routine reaction wheel torque and velocity commands to be observed and investigated. This data was used to refine the bit conversions inside the reaction wheel torque and velocity commands as well as put a limit on the minimum torque increment due to the limitations of the stepper motors driving the reaction wheels. It was indicated the stepper motors may exhibit 'slipping' at high torque increments, leading to a potential short circuit across the motor and a spike in current drawn by the MAI-200. This resulted in a modification of the control algorithm to avoid large increments in torque, and instead ramp up the angular rotation at a controlled rate.
2. The same power characterization testing above was performed on the magnetic torque rods, also part of the MAI-200 system. These torque rods operate using pulse width modulation (PWM) so their maximum (100% PWM) torque power draw was characterized. Thus, the power draw at torques less than the maximum were computed from these results, as they are directly proportional to the percentage PWM they operate at. This test also enabled verification that the system was responding correctly to the custom magnetic torque rod commands.
3. The MicroMag3 magnetometer was verified using a known calibrated reference magnetometer as well as a Helmholtz coil. Firstly, in order to verify the operation of the MicroMag3, two set points were measured and compared against the readings on the reference magnetometer. These were earth's magnetic field and a net zero magnetic field created inside the Helmholtz coil (see Fig. 3). These fields were measured using the reference magnetometer and output to a serial monitor before they were measured again by the MicroMag3, processed and passed out to a serial monitor. This simple test helped to ensure that the magnetometer was operating correctly and that the flight software was handling and processing the magnetometer data correctly.

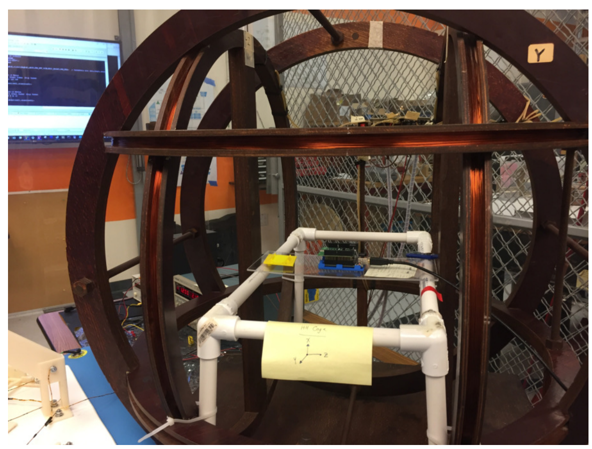

Fig. 3: Magnetometer in Helmholtz Coil

4. The Dodona ADACS uses three ADIS16260 gyroscopes to determine the spacecraft attitude, with updates from the sun sensor to account for measurement drift. These sensors were mounted physically and electrically to the flight processor module, thus functionality was verified by querying data from the Dodona microprocessor and comparing to the real world orientation of the module. The directionality and scale of the gyroscope measure-

ments as well as the processing functions could be verified by simply rotating the sensor and comparing the data readout. The accuracy of the gyroscope measurements was then verified through calibration, done by rotating the sensor through a fixed angle about a single axis while recording rotation rate and time step. This data was then integrated over time to determine the rotation angle, which was then compared to the reference angle. This calibration, done about each axis, generated offset and scale values to be used when post-processing the data to ensure a calibrated and accurate reading.

5. Individual sun sensor testing was executed first to validate the standalone component and then tested at sensor to controller level. The latter involved verifying that the flight software translated data appropriately, and testing the controller modes in which the sun sensor played a critical role, such as sun pointing and eclipse modes. Testing utilized an incandescent halogen light source with a metal cover and an aperture designed such that the light emitted was of similar angular size to that of the Sun as seen by a satellite in Low Earth Orbit [5].

   The sensor was placed on a table below the light source and placed at different locations, while the relative x,y, and z positions, with respect to the sun sensor frame, from the light to the sensor were measured to calculate a sun vector. Using manufacturer ground testing software, the queried sun sensor was compared to the calculated sun vector. Although checking out the sun sensor individually seems like a trivial task, testing separately from software that had not been validated yet was deemed essential; verifying the hardware first would help with troubleshooting during future testing involving flight software.

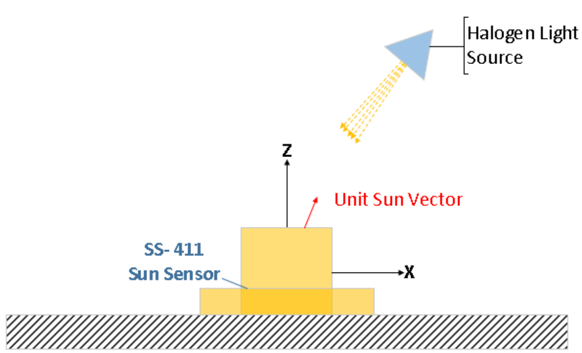

Fig. 4: Sun Sensor Test Setup

   Results from the sun sensor checkout test showed that the difference between theoretical and sensor values were within acceptable margins, so the sensor was considered qualified for operation and future testing.

   Part of the GNC control algorithm involves a transformation matrix to convert the sun sensor data, collected in the sun sensor coordinate system, into that of the operating system of the CubeSat. Testing this transformation may seem minor, but any error in this conversion process would result in an inability to sun point, and thus a catastrophic loss of input power to the system. During the process of updating the legacy software inconsistencies were found between the coordinate systems, further warranting this test. A similar procedure from the sun sensor checkout test was used, implementing the flight processors rather than ground support equipment so that the data would be processed by the transformation matrix. The sun vector (in the satellite body frame) was queried and confirmed to match the sun vector calculated manually, verifying the transformation matrix was functioning as expected.

   The final step of testing for the sun sensor, after integration into the GNC system and flight software, was to verify the pointing control functionality of the system. When in the sun-pointing mode, the satellite is expected to orient its negative-z axis towards the Sun (pointing the petal arrays at the Sun). This is done by searching for the Sun using the sun sensor, computing the angular difference between the current pointing and the desired pointing vectors, then sending torque commands to the MAI-200 attitude control system. The MAI-200 then spins up or down its three reaction wheels to reorient the satellite. Results from this test showed that, using a halogen light source to simulate a solar input, the satellite's reaction wheels turned in the correct direction, with expected wheel speeds required to rotate into the correct pointing vector. Although the satellite did not rotate during the test, due to the relatively weak torque from the reaction wheels in the presence of surface gravity, the validation of the expected wheel speeds and direction are indicative that in space the system will function as designed.

5.2 *Flight Software*

The Dodona flight software was built upon legacy code from USC's 2nd CubeSat, Aeneas. The Aeneas software drivers and communications architecture were used in Dodona since they had been flight proven on the Aeneas mission. The complexity and timeline of certifying the re-purposed software was alleviated using a series of software testing practices such as unit and systems level testing. Unit level testing consisted of individual units, or functions, usually having a one input and a single output. Unit tests are an ideal first step in verifying software changes, as bugs can be caught at an early stage of development. Unit level testing also makes the code more readable, more reusable, and is the most important level of testing because it lays the foundation for the rest of the testing levels, namely, integration and systems level testing.

The Dodona satellite has two on-board PIC-24 Pluggable Processor Modules (PPM). The lower processor (Lower PPM) was responsible for monitoring the overall health and status of the satellite. Additionally, it updated the orientation vector and sent control commands to the ADACS. The upper processor (Upper PPM) interfaced with the payload boards and the transmitter. Although using two processors allowed for more communication lines and processing power, it also made the software inherently more complex. Thorough testing of all software additions and modifications was a mission critical requirement.

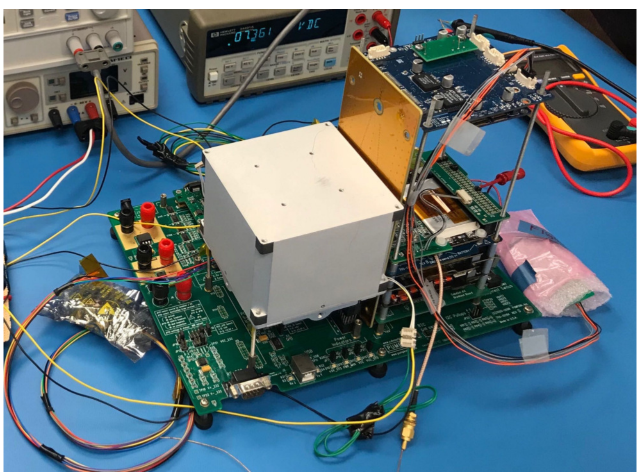

Fig. 5: Dodona Stack on Pumpkin Development Board

The goal of unit testing is to isolate each part of the program and show that it runs as intended. During unit testing, the processor, a PIC-24 micro-controller, and the electronics board of interest (EPS, Payload, batteries, etc.) were placed and run on a development board. The development board acts as a motherboard, which provides all the necessary power and communication lines between the PIC-24 and the hardware board (see Fig. 5). Additionally, the development board has a large footprint (approximately 300 mm x 200 mm) which allows for probing of voltage and communications lines if necessary for debugging. In comparison, the footprint of the boards in flight configuration is 100 mm x 100 mm with less than 20 mm in height between boards. The development board made it possible for rapid testing and troubleshooting during software testing.

Every time a software change was made the PIC-24 needed to be reprogrammed. This process was time consuming, tedious, and required handling multiple flight hardware boards. It became apparent quickly that during test campaigns it was important to allocate time for reprogramming the processors. Reprogramming was a multi-step procedure that required a development board, an MPLAB ICD 3 In-Circuit Debugger, a programming interface board, a 6 pin flat cable, a 5 volt power supply, and the MicroChip Software Development IDE. Reprogramming a PIC-24 took on average 3 minutes, in other words, over the course of one testing regime (10 unit tests) 30 minutes would be spent setting up and programming the PIC-24. One advantage to the ICD 3 debugger is having the option to set breakpoints that can be read and modified in real time using the MPLAB IDE.

Each flight board was tested individually before integration into the flight stack. This process ensured two things:

a) If there was an electrical failure or a short on one board it would be detected before integrating with other flight boards, adding a layer of protection to flight components.

b) Knowing that each board individually functioned as expected meant that it would be easier to diagnose issues during integration and systems level testing.

After all unit tests were passed, the next level of testing was at the systems level. The systems level testing can be thought of as *Black Box Testing.* Systems level tests were functional tests that were used to expose one of the following conditions:

a) incorrect or missing functions

b) interface errors

c) errors in flash chip programming

d) initialization and termination errors

Most tests conducted on the systems level were used to expose any issues with storing and transmitting health and status information.

One of the potential failure modes identified by the Aeneas mission was battery management and excessive power consumption. Post mission forensics showed the satellite entered an infinite reboot loop because the power up sequence was not regulated. Each time the processor turned on it immediately activated the magnetometer, transmitter, payload, gyroscopes, reaction wheels, sun sensor, and bus boards. Each subsystem alone did not require much power, however turning them all on at once required an amount of power greater than what could be supplied by the batteries at the time. The strain on the batteries was so great that the entire system would blackout and restart, thus entering a constant loop. The proposed solution was to program a brownout power up contingency in the flight software [6].

For Dodona, a voltmeter was used to track the power consumption during the satellite's power-up sequence. Using the voltmeter during power-up revealed a bug in the ADACS that was not previously known. It was discovered that after the processor turned on the ADACS the reaction wheels would spin up to their saturation limit (10,000 RPM), which caused a huge drain on the batteries. This anomaly prompted a deeper dive into the ADACS source code. The code was found to have a rotation matrix that was inverted, thus causing the reaction wheels to spin in the opposite direction as expected. This issue was found and fixed because of the thorough subsystem level testing using both hardware and software and by testing using flight conditions.

5.3 Avionics

Dodona's Avionics consists of multiple electronic boards that are utilized for sensing, data collection, data transmission, power distribution, control of the satellite as well as communication with and control of the payload. Most of the CubeSat's electronics were commercial off the shelf components. However, to meet the mission requirements and accommodate the unique payload, a new interface board was developed in house. This allowed for integration of the payload into the legacy system. This interface board is referred to as the Payload Extender Card (PEC)

The PEC was designed at SERC to allow for control and communication with the payload. It contains power conversion circuits for energizing the payload, data transmission circuit for switching between the multiple payloads, the burn circuit for deployment, communication channels to the transceiver and also interfaces with Pumpkin's Pluggable Processor Module which acts as the processing hub for these subsystems, the upper half of the CubeSat's brain. Thus, the PEC is responsible for functional integration of the payload as well as some auxiliary systems.

All components, both off the shelf and in-house custom components, went through rigorous standalone functional testing prior to integration in order to prevent inadvertent damage to other parts of the CubeSat. This included the test of the newly designed PEC and its associated peripherals. The PEC design went through three design iterations, each correcting the shortfalls observed in its predecessor. The following steps were undertaken to develop and test every iteration of the PEC:

1. PEC Electrical Testing

   For each newly fabricated board, connectivity tests were performed on every trace using a DMM to identify any potential manufacturing defects. Using this method, weak solder connections on the burn wire connectors and unintended shorts on the payload power supply lines were uncovered and remedied in-house, saving debugging time and preventing critical component damage. In order to verify power distribution paths, each path was tested by externally powering the lines with power supplies. Each power line was first tested individually. When no issues were observed, all power lines were energized simultaneously while monitoring power.

2. PEC Digital Input/ Output Testing

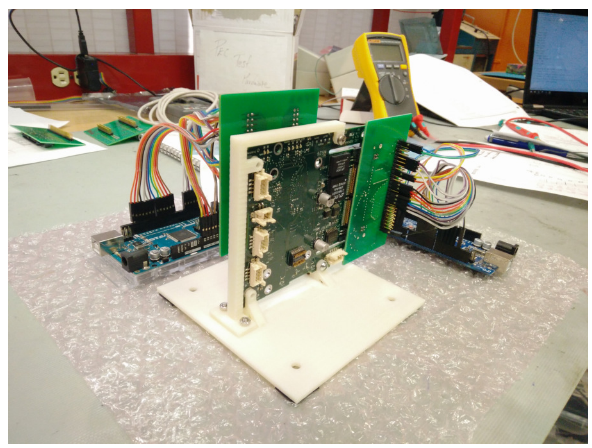

Fig. 6: PEC Breakout Test Boards

   When integrated into the CubeSat, the PEC board receives commands from the flight processors. To perform standalone testing to simulate these command inputs, breakout boards for the PEC's two interface headers were manufactured (see Fig. 6), and Arduino code was developed to allow for user control of digital inputs on individual pins on both headers of the PEC via these boards. This enabled verification of resultant digital and analog signals by simulating expected inputs from the PPMs. Analog circuits were tested by applying dummy analog signals and required control signals, observing the outputs again using the breakout boards.

In-house design and assembly of these test boards and utilization of inexpensive Arduino Megas reduced the cost of manufacturing and verification. First, every subcircuit on the PEC was tested individually by providing the associated digital signals through the breakout boards. If any unexpected behavior was observed, the associated circuit was troubleshooted. After each digital subcircuit was deemed operational, all digital circuits were tested in parallel by supplying multiple control signals through the Arduinos connected to the test boards. The Arduinos were programmed to accept commands through a serial terminal of attached computing devices for various operations like burn wire deployment or payload switching. The Arduinos would then send the digital signals based on the applied commands and the PEC would react accordingly. This validated that all digitally controlled circuits on the PEC were operational and not affected by any kind of interference.

3. PEC Payload Power Test

   The PEC was responsible for generating and distributing the power required by the payload. These include a 9 V 13 W supply and a 5 V 4 W supply for the two payloads. The payload power was generated by supplying the Battery power (6.5 V - 7.8 V) to a Boost converter (for 9 V) and a Buck converter (for 5 V). The payload power lines were switched on/ off using digital signals from the Upper PPM.

   To test this, 7 V power was supplied to the PEC using a power supply and the DC-DC converters were switched on/off by applying corresponding digital signals using an Arduino. Resistive loads were used to emulate the payload until the payload boards became available for testing. This allowed for integration and testing of the CubeSat despite the lack of actual payload hardware.

   In the first iteration of the PEC design, this test helped us realize that the generated power was not stable due to insufficient capacitance at the input and output of the DC-DC converters. This was easily fixed in later iterations by adding additional capacitors at the input and output.

   The test helped us identify the issue very early in the testing phase and allowed us to rectify the design and add in the solution in later iterations without adding additional delays to the project schedule.

4. PEC Full functional Test

   After any issues uncovered in previous steps were troubleshooted and all the subcircuits were found to be operational, a complete functional test of the PEC was conducted by running all systems in parallel. This validated the entire PEC. Finally, when the PEC was deemed to be functional and safe, it was integrated into the CubeSat and tested with flight hardware.

5.4 Power System

The power system for this mission conforms to a centralized architecture common to early CubeSat design. It is designed to support all satellite system components and any additional customer payloads during periods in and out of eclipse. The central architecture uses three major power lines to distribute power across the satellite, each subject to further regulation based on subsystem requirements. The major power subsystem components include:

- Clyde Space 3U CubeSat Electronic Power System
- Clyde Space 20 Wh Dual Battery Board
- Clyde Space 10 Wh Remote Battery Board
- Pumpkin Space Systems 3U Solar Panels (x7)

This section will discuss testing for these components, results, and any issues encountered and their solutions. Each component was tested individually to validate operation before a final systems level test. To isolate the battery system from the EPS during testing, a power supply set to a nominal battery voltage with current limited was connected to simulate the battery. Tests were performed to validate the following features of the Electronic Power System module [7]:

- Power Conditioning
- On-Board Protection Circuitry
- Battery Charge Regulator (BCR) Performance

Power conditioning tests validated the functionality of two of the three major power lines on Dodona, the 3.3 V and 5 V lines, respectively. To perform this test, the simulated battery was first connected to the EPS system, then the RBF and separation switches were closed. A digital multimeter read the regulated output from the PC-104 header on the EPS.

Protection circuitry onboard prevents under and over-voltage states. To test the under-voltage protection circuitry, the simulated battery voltage was gradually lowered below the threshold value of 6.2 V. At this point, all output power buses read zero indicating that they were shut down. Overvoltage protection starts at the end of a charging cycle and was demonstrated by gradually increasing the simulated battery's voltage above the threshold of 8.2 V and by placing an ammeter in line with the solar array input. As the voltage went above this threshold the input decreased to 0 A.

A battery charge regulator, BCR, as described in [8], is a buck DC-DC converter in two modes of operation: maximum power point tracking (MPPT) mode and end of charge mode. The system operates in the first mode during the charging phase of the battery, on complete recharge the BCR moves into its second mode where it regulates output by allowing the input voltage from the arrays to drift away from maximum power levels. BCR performance was tested to validate maximum power point tracking behavior; an oscilloscope was placed in between the solar array input and the EPS module. The output waveform observed validated MPPT and showed the panel voltage switch to open circuit values during tracking [7].

An issue uncovered during qualification was a small current draw from the battery flowing back to the MPPT on the EPS. A fix for this was provided by the manufacturer on request, a PCB which operates as a slave switch preventing backflow. The separation switch acts as the master here. This method allows system charging while on launch vehicle with the only downside that it brings down the end of charge voltage from 8.2 V to ≈7.7 V. This circuit was integrated to the motherboard per manufacturer guidelines.

To validate battery board operation, each Li-po cell was visually inspected for any damage, each board was inspected under a microscope, physical components such as fuses and resistors were tested, and, finally, all surface traces were checked. Then each board was tested as a standalone DC supply by simulating BCR input through a power supply and closing the RBF and separation switches. Connecting a digital multimeter to the positive battery bus pin on the PC-104 interface read the net voltage on the battery board.

The battery system was often the point of failure during the initial phases of ground operations testing. This was mainly because the Li-polymer cells on the boards would buckle, causing the entire system to shut down. To debug this issue, the damaged cells were inspected and tested individually for capacity retention by being put through multiple charge-discharge cycles [9], followed by a comprehensive inspection of the board. To check if the integrated cells were discharging in a uniform manner, each battery was connected to a load board with variable resistances and allowed to discharge. All cells were monitored individually using exclusive voltmeters and ammeters, as seen in Fig. 7.

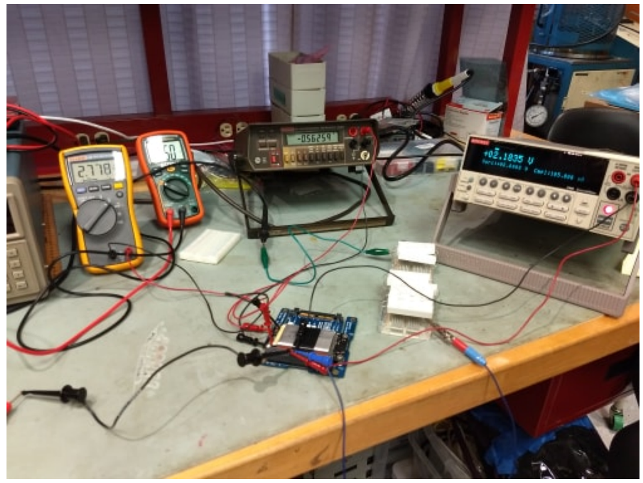

Fig. 7: Remote Battery Board (RBB) undergoing testing

During one of these tests it was identified that the battery board did not discharge as expected. It would only use certain cells on the board causing them to get overworked and eventually fail. The issue was resolved by replacing the 20 W h battery module with a flight spare. The spare was flight qualified using the same test framework.

All seven solar panels were tested individually to verify operation. Panel I-V characteristics were used to determine maximum power values [10]. A halogen light source was placed at a distance from the solar panel to recreate the solar flux environment as in space. A load board with resistances was attached to the panels to maintain them at ideal operating conditions.

5.5 *Telemetry, Tracking, and Control (TT&C)*

The Dodona Telemetry, Tracking, and Control (TT&C) system consists of two radio frequency communication units, a Stensat health and status beacon, and a GomSpace AX-100 half-duplex transceiver. Both of these modules operate in the UHF frequency spectrum using the AX.25 encoding protocol. Decoding of the data was done using a custom Software Defined Radio (SDR) setup using GNU Radio for Linux, an open source software that can be used with off-the-shelf SDRs such as the HackRF One (both GNU Radio and HackRF Ones are used in the on-campus USC ground station course).

Before integrating these components into the satellite, they were tested using simulated software commands loaded onto an Arduino microcontroller to validate the RF power being transmitted and the signal decoding using the SDR setup. This enabled easy troubleshooting of the custom decoder software without requiring complex setup of the full satellite stack to debug the TT&C system. This also allowed the TT&C team to test and verify the packetization and decoding process before receiving the customer's payload.

In-lab testing of the transceivers was executed using an HackRF One SDR with an omnidirectional collapsible antenna to receive Dodona's radio beacons, while the decoding program was developed in Python using the GNU Radio's software toolkit. The use of the open-source GNU Radio platform, along with inexpensive hardware such as the HackRF One and Arduino Mega, allowed the TT&C team to undergo extensive testing well-within the project budget.

During testing, Dodona sent preset health and status beacons according to flight code schedules from inside the cleanroom, which were captured and decoded outside the cleanroom. These tests were used to determine if errors originated sender-side or receiver-side, allowing the team to rectify issues with both components while updates were made on either side.

Fig. 8: Mobile Radio Test Unit

Flight operations are planned to use the USC Ground Station (GS) 5 m Yagi antenna setup at USC's campus in downtown Los Angeles. Following the standalone testing with the SDRs, the next step before integration into the spacecraft was to test the beacon and transceiver with the USC antenna in a far-field communications test. To do this while maintaining cleanliness standards for flight hardware, a special test rig was designed and manufactured, called the Mobile Radio Test Unit (MRTU) (or colloquially as "the LunchBox"). The MRTU consisted of a rechargeable battery, an ODROID XU4 microprocessor, a sealed housing unit for the two radios, and sensors for temperature, voltage, and current. This was all integrated along with a control panel into a pelican case for portability and durability (see Fig. 8). The sealed radio cleanbox enables integration of flight hardware in the cleanroom which, when sealed by O-rings and grommeted data and power connectors, can be integrated into the MRTU without fear of contaminating flight hardware while testing in the field. Although this form of test would normally be done with a transceiver unit designated for field testing, the budget restrictions of the Dodona mission meant that only one GomSpace AX-100 transceiver could be purchased.

While near-field testing of Dodona's TT&C capacities have been executed, far-field testing is planned to utilize the MRTU from Griffith Observatory (approx. 10 km from the USC ground station site), pending upgrades to the Yagi antenna setup to enable transmission capability for bi-directional communication testing.

5.6 *Solar Panel Deployment System*

Dodona has four deployable solar arrays, stowed during launch using a spring and burn-wire mechanism, and deployed using two burn drivers, developed in-house at SERC. The burn drivers will be used to resisitve load heat a nichrome wire and sever a nylon line that is preventing the spring-loaded deployment using tension, using two burn drivers for redundancy. For all results below, the nichrome wire is 36 BNC-A and the nylon line is 10lb monofilament, approximately 28AWG.

To implement the deployable system, the nylon wire was threaded through holes at the top end of each of the panels and each of the two burn drivers, then tied off, making one large nylon wire loop that secured all four panels in their folded stowed position. Thinner nichrome wire was then wound around the nylon line and connected to a 1.1 A , 5 V power source at two separate points, one nichrome wire per each of the burn drive boards. The burn wires' power source would be turned on and supply the burn-wires with a current at the +30 minute flight time. This was intended to sufficiently melt the binding nylon wires about which the nichrome was wrapped and thus release the four-panel array. The use of two separate nichrome wire and burn drive setups ensured an integral level of redundancy for the system and helped mitigate potential points of mission failure as only one burn wire had to operate correctly to ensure deployment of the panels.

Testing procedures were designed to match the mission requirements and multiple variations of the burn driver panel deployment system were tested in their entirety. Results from this testing not only verified which setup worked best for a consistent clean burn of the nylon wire but also gave an indication of the time required for the deployment procedure.

The variables considered included the following:

- Resistance across the burn driver
- Time to burn
- Number of loops around nylon wire

After testing with various values for the resistance, burn time, and loops, the optimal configuration to minimize the total energy required to complete the burn was found to be:

- 3.6 ohm resistor
- 8 second burn time
- 4 loops

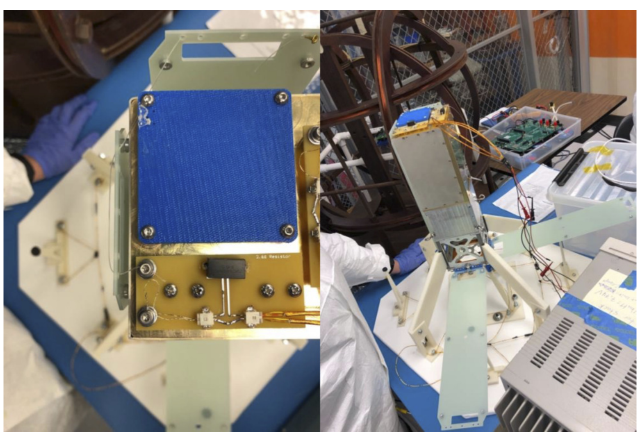

Fig. 9: Solar Panel Deployment Testing

Next, testing was conducted to investigate routing options of the nylon wire along the top panel of the payload, to select a route that will deploy all solar panels consistently. Multiple routing options were created and tested. To avoid damaging the deployable solar panels, mock-up panels were created that replicated the dimensions, spring force, and attachment points of the flight panels. The test setup comprised of integrating the full Dodona structure and attaching the mock-up deployable solar panels (see Fig. 9). Dodona was then placed in a custom made holding fixture that caught the mock-up panels at 90 degrees while they deployed. Two power supplies provided power to each one of the burn drivers.

The results from the test identified the optimal routing option and validated that the wire would not snag on any surfaces during the deployment. The test was later re-run using the actual solar panels and power and control supplied by the satellite's avionics system to verify functionality with the system as a whole.

6 System-Level Testing

Following the completion of subsystem level testing of the components, these were then integrated together in sequence to form the Dodona flight hardware stack, with verification testing executed at each step of the integration to ensure that as new hardware was added, none of the functionality of the previous hardware components was affected.

In addition to the stepwise integration verification, a few key tests were performed at the system level to verify certain satellite operations that could not be verified at the subsystem level. These are outlined as follows:

6.1 GNC Controller Verification

Given the time and budgetary constraints imposed by the project, it was not possible to do a full test of the controller with all hardware and sensor inputs integrated to qualify the Attitude Determination and Control System (ADACS). It was determined, however, that specific test cases could be developed to force the controller into its operational software modes, making use of real-world environmental effects to ensure expected functionality for mission success.

The ADACS operates by setting the Guidance, Navigation, and Control (GNC) subroutine into a pre-defined mode based on the environmental conditions of the satellite, as well as scheduled operations uploaded by the mission operations team. It was determined that the system could be verified by the testing and validation of the following five GNC modes:

1. Inertial Capture/Bdot Controller
2. Sun Search
3. Sun Pointing
4. Eclipse
5. Momentum Dumping

Each of these five operational modes were verified individually by feeding inputs to each of the active sensors in each mode, either by electronic interfaces or by external environmental inputs, and reading the outputs from the controller to verify that the correct actions were being taken by the system in each mode.

1. The detumbling controller uses a Bdot control law and was tested using a piece-wise testing philosophy. Though it is possible to build a robust test setup allowing 3 degrees of rotation with low friction, it is quite expensive to build and maintain, whereas testing specific parts of the controller using piece-wise verification is relatively low cost and simple, achieving the same outcome to qualify the system. The validation of the Bdot controller was extrapolated from the results of the momentum dumping testing due to the similarity of their control laws. They are exactly the same with the sole difference of Bdot requiring gyro inputs, and Momentum Dumping requiring Magnetometer inputs.

2. When the spacecraft is searching for the Sun, the program uses the reaction wheels to execute a 360 degree rotation about the Body X Axis followed by a 360 degree rotation about the Body Y Axis until the Sun Sensor detects the Sun. This happens whenever the spacecraft does not have sufficient data to determine the position of the Sun, so to test this the spacecraft was turned on and the resultant reaction wheel torque commands and speeds were read through the terminal to ensure this was happening and that the mode switched to Sun Pointing once the Sun was detected (supplied using an artificial light source).

3. Next in testing was the Sun Pointing mode. Sun Pointing takes in the Sun Vector returned by the Sun Sensor and the estimated inertial quaternion, and computes the error quaternion. Then it sends the error quaternion to the attitude controller to initiate a slew towards the Sun. To test sun pointing, a flat-sat setup was used with the flight software running using the Sun Sensor as an input. A lamp was used to imitate the Sun and fake Sun vector readings. This lamp was translated side to side to simulate pointing errors from the Sun, and the commanded reaction wheel torque control response from the controller was logged. These torques were rotated appropriately and fit to ensure that the rotations were in the correct direction to rotate the flat-sat, if it could rotate, towards the lamp.

4. Lastly, extensive testing was done on the Momentum Dumping mode as it is a critical function for sustained mission operations. The reaction wheels soak up any disturbance torques the spacecraft may experience, in addition to rotating the spacecraft and maintaining sun pointing. To dump any momentum gained, the torque rods generate a magnetic moment, allowing the reaction wheels to decrease in angular velocity while providing a counteracting torque to maintain the current attitude of the satellite.

   To test this, the flat-sat including the MAI-200 ADACS, the onboard magnetometer, and a reference magnetometer were used. The onboard magnetometer was placed inside a Helmholtz coil to allow the generation of specific magnetic fields. For these tests, the Earth's magnetic field was first nullified using the Helmholtz coil, and then a magnetic field was generated in a single direction. Next, a reaction wheel was spun up to above the threshold necessary to fall into the momentum dumping mode. Finally, the reference magnetometer was placed on top of the MAI-200 to measure the magnetic moments created.

   All measurements and commands were rotated into the body frame of the spacecraft and checked against the ADACS control law itself to ensure the correct magnitude and direction of the magnetic moments were being generated.

6.2 System Burn-In

After all the electronics boards passed unit level and systems level testing they were integrated for an initial boot-up test, informally called a 6 hour burn-in. The purpose of the test was to simulate the first 6 hours of the satellite's operation after being ejected from the launch vehicle, and to uncover any unexpected issues that might arise when all the hardware components are integrated and powered up. There were 5 milestones outlined in order for the test to be successful. There were a series of 5 Dominos (sequenced events) such that the n+1st Domino could only be started by the nth Domino. The Dominos were as follows:

Domino 1: Load the boot-macro, which started the initialization sequence.

Domino 2: Check the voltage on the solar panels to see if the sun was present.

Domino 3: Deploy the solar panels.

Domino 4: Turn on the transmitter.

Domino 5: Turn on the Attitude Determination and Control System and clear all Dominos, synchronize the time on the Upper and Lower PPM and save all global variables to flash memory.

The 6 hour burn-in was helpful in identifying and mitigating three main issues:

a) electrical interference between flight boards,

b) overdrawn/excessive use on power system

c) faults within the tip-off and boot-up sequence.

The 6 hour burn-in was beneficial in giving the team a good idea of how all the different subsystems integrated together and shared resources. The data gathered from the 6 hour burn-in was sufficient to show the spacecraft was configured correctly for its intended flight operations. Therefore, the most important aspect of the test was ensuring all the Dominos were scheduled (using relative and absolute time) and executed successfully, while also ensuring that the battery voltage never dropped below a medium voltage level of 6.9V. The result of the burn-in test was successful; furthermore, it helped shed more light on the operation of the system than was previously expected.

6.3 Magnetometer Calibration

The final step in ensuring proper operation of the magnetometer is calibration. Proper sensor calibration helps reduce small inaccuracies in sensor readout and is especially important with sensitive equipment such as magnetometers as they can be effected by electro-magnetic fields generated by electronics or ferrous metals. This calibration is to be done by placing the fully integrated Dodona Satellite inside a Helmholtz coil, then modulating the magnetic field inside the Helmholtz coil across a series of known set point values along all three planar axes. These known values are then compared against the magnetometer's measurements and scale and offset values are calculated that compensate for any inherent inaccuracies in the sensor as well as any inaccuracies introduced by the electro-magnetic environment created by the Dodona Satellite. Magnetometer calibration is currently awaiting final payload flight ready units before it can be completed, as this may change the electro-magnetic environment previously described.

6.4 Environmental Testing

In order to ensure mission reliability and optimal vehicle performance, a pre-flight series of tests which emulated the expected conditions on orbit were specified. More specifically, vibrational testing as well as thermal vacuum testing were decided upon and benchmarks of success researched and defined. A combination of random and sinusoidal vibrational testing was planned, whereby the fully integrated vehicle would be secured to a vibration table and exposed to random vibration levels akin to the levels expected during the launch of the vehicle. In order to maintain a conservative estimate of the vibration launch loads expected, the NASA General Environmental Verification Specification (GEVS) load qualification profile was used [11]. Although this profile far exceeded the expected launch loads presented by the launch provider, it was beneficial to qualify the vehicle to this standard in the event that the launch provider needed to be changed. Similarly, using the results from orbital simulations, a qualification profile for thermal vacuum testing was established. This thermal vacuum testing will include subjecting the vehicle to two cycles of external temperature between $-30\,^{\circ}$C and $100\,^{\circ}$C within a thermal vacuum chamber, performing a single solar panel deployment functionality test after the second cycle. Success was dependent on the proper deployment of the panel as well as full hardware checkout during thermal cycling.

The environmental testing is planned prior to final delivery.

7 Simulations and Analysis

For certain aspects of the Dodona mission, full hardware testing was not possible or practical to perform with the given budget and timeframe, including detumble operations and circuit functionality. To gather the required data validation for these cases, computer simulations were used instead. A few of these simulations are detailed below.

7.1 Initial Detumble Operations

The most critical portion of a CubeSat mission, apart from the launch itself, is the first few hours after deployment from the launch vehicle. At this point the spacecraft will be spinning at some initial random tip-off rate, and the automated ADACS will need to work to reduce this spin rate below a specified threshold, where it will be safe to activate the reaction wheel control system and reorient the spacecraft to point the deployed solar arrays at the Sun. This occurs without control from the ground operators, potentially without even the ability to signal the status of the operation to the mission controllers due to the use of a single groundstation. There is no guarantee of communication with the spacecraft, so this segment is fully autonomous and needs to be tested thoroughly with many input cases to ensure system robustness.

To perform the detumble operation, magnetic torque rods are used, which interact with the Earth's magnetic field to impart a torque on the spacecraft, transferring the angular momentum of the spacecraft to the Earth. This action requires approximately 7.6 W of power to perform, and the amount of power generated and time required depends on the orientation of the spacecraft with respect to the Earth's magnetic field at the time of the start of detumble. This cannot be precisely predicted, as it depends on rotation rates of the launch vehicle itself, its interactions with the atmosphere during ascent, and errors in the thrust control during the ascent.

In order to ensure that the autonomous system will be able to function correctly over a wide variety of initial startup conditions, computer simulations were used to test thousands of possible cases. For conservative estimates, an upper bound for the initial rotation rate was set at approximately 0.1 radians per second per axis [12]. To ensure that the combination of battery power and intermittent solar power during the spin will be sufficient to power the torque rods, simulations were run for 48 different starting orientations, spread evenly over 4π Steradians. Each orientation was computed for a Gaussian distribution of 300 initial spin rates imparted by the launch vehicle and deployment system, yielding 14,400 different simulation cases, an improvement over the initial run of 3,600 cases performed earlier in the year

The plot in Fig. 10 shows the maximum depth of discharge (DOD) of the battery for each of the attitude and spin cases considered. In order for a case to be considered a successful detumble maneuver, Dodona defined the B-Dot controller operation success as DOD not exceeding 30%.

The maximum battery depth of discharge for any of the 14,400 cases is 27.4%, below the optimal threshold goal of 30% [13]. For the most part, the runs yield DOD values of less than 10%. Aside from the few outliers, the fluctuations of DOD with respect to the Gaussian distribution of input spin rates are evenly distributed, indicating that there is a low correlation between spin rate magnitude and battery DOD during de-spin. The results of this testing helped inform the power system team what battery capacity will be needed in orbit and what testing was required to ensure full functionality of the power system in various operational scenarios.

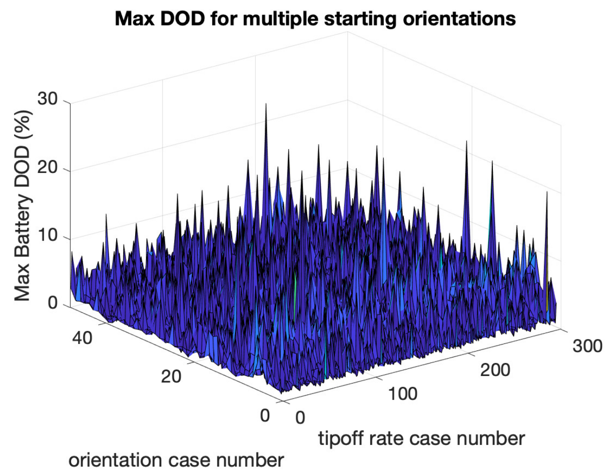

Fig. 10: Maximum Battery DOD for each case

7.2 *Electronic Circuit Simulations*

The major analog circuits on the PEC included the LTM4600 based Buck converter circuit that generated a 5V supply using the battery voltage and a LTM8054 based boost converter circuit that pumped up the battery voltage to 9V. Both converters are part of Linear Technology's µModule regulator family. These circuits used the CubeSat's raw battery power and were designed to run the payload. Failure of the analog circuits would have led to damage to the CubeSat's batteries as well as the payload resulting in failure of the mission. If the circuits were found to be inadequate, the redesign would have led to additional costs and delays, thus simulating these circuits was essential. LTSpice, a commercial circuit simulation software, was used to simulate the analog circuits and verify their operation (See Fig. 11).

The analog circuits recommended in the regulator's datasheets were recreated in LTSPice. The batteries were simulated as voltage sources and the payloads were simulated as resistive loads. The value of resistors was calculated based on the amount of power that the payload sections would require, where $Resistance = Voltage^2/Power$. The simulations allowed for verification of the circuit and helped characterise the behavior of all components involved. Thus, operation of the analog circuits was easily simulated and tested ahead of sending out the designs for manufacturing. This helped to cut down risks, development and debugging time, and redesign costs, while increasing the overall confidence in the circuits.

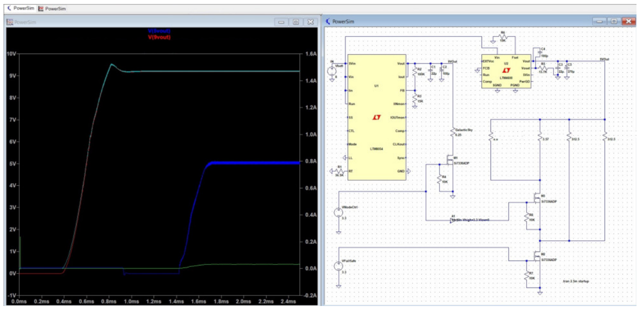

Fig. 11: Buck and Boost Parallel Operation Simulation

Performing these simulations allowed for initial design iterations before manufacturing the first hardware revision, saving considerable expense for board prototyping.

8 Conclusions

Flight hardware is very expensive, has a high manufacturing lead time, and is critical for the success of the mission. Special care needs to be taken to avoid the risk of damaging flight hardware so as to prevent unnecessary delays and expenses. Taking a step wise approach to testing each component helped reduce the risk and stress of damaging flight hardware. Testing individual functionality of every component before integration allowed us to identify, debug, and troubleshoot anomalies, manufacturing defects, and design shortfalls that had the potential of harming other flight proven critical components.

Although the Dodona CubeSat has not launched to orbit yet, the integration and test scheme used was able to verify all the critical components of the satellite and functionality with the designated ground access terminals. The integration procedures, test reports, and traveler documents all allowed quick and easy verification and documentation of all tests. This hardware traceability will be invaluable in the coming months, either to validate the effectiveness of our accelerated integration and test procedures, or to diagnose the cause of any on-orbit failure given the symptoms of the health and status beacon data.

Additionally, the wealth of documentation leftover from the project has been tremendously helpful on kick-starting USC's fourth CubeSat project, set to launch in early 2020. Given the high turnover in normal Student based projects it is hoped that the Dodona experience has set up future student teams at the SERC for good training prior to industry and success oriented missions.

Acknowledgements

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Appendix A (Example Traveler Document)

Export Control ☐

Non-Export Control ☒

Dodona Hardware Traveler

ITEM: Upper PPM

Hardware Name: PPM D1
Serial Number: 226
Date Signed in: 9/24/2018
Receiver: Sofia Bernstein
Traveler Version Number: 1

Engineering or Flight Model:	Flight
Explanation of what Upper PPM does	The Upper PPM is one of Dodona's brains. It handles communication with and control of the payload, transceiver, burn deployment boards, and the beacon. Each PPM includes a 64Mb non-volatile memory chip where global variables and flight software schedules are stored.
What Upper PPM connects to	Upper PPM connects to the PEC board.
How to identify Upper PPM	Blue half-width form factor board, snaps into underside of PEC board.

Upper PPM integrated onto Motherboard (not flight configuration)

Export Control ☐

Non-Export Control ☒

Checklist	Date	Validation Initials	Notes
Serial number/model matched expected	09/24/2018	SB	SN: 226
Package Arrival	09/24/2018	SB	Obtained from Pumpkin via FedEx
Documentation included	09/24/2018	SB	**CubeSat Kit Pluggable Processor Module (PPM) D1** DS_CSK_PPM_D1_710-00527-A.pdf
Verify export control requirement	2/14/2019	SB	Non export controlled
Condition of Packaging (damage, cleanliness etc.)	09/24/2018	SB	Brand new
Operational	2/14/2019	SB	Yes
Support Equipment			MPLab IDE Programmer, Pumpkin PIC-24 JTAG Programmer
For Programming	2/15/2019	SB	Development Board, Microchip MPLAB ICD 3 In-Circuit Debugger, 6-pin PIC24 Programming Adapter
Modifications on hardware or support equipment			None
Moved to bonded storage or cleanroom	09/24/2018	SB	Moved to cleanroom. Living in a hard case labeled "Motherboard"
Integrated into Flight Vehicle	02/20/2019	BR	Integrated in cleanroom to satellite stack

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