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DEVELOPMENT OF A REUSABLE CUBESAT SATELLITE BUS ARCHITECTURE FOR THE KYSAT-1 SPACECRAFT

Tyler James Doering
University of Kentucky

ABSTRACT OF THESIS

DEVELOPMENT OF A REUSABLE CUBESAT SATELLITE BUS ARCHITECTURE FOR THE KYSAT-1 SPACECRAFT

This thesis describes the design, implementation and testing of a spacecraft bus implemented on KySat-1, a picosatellite scheduled to launch late 2009 to early 2010. The spacecraft bus is designed to be a robust reusable bus architecture using commercially available off the shelf components and subsystems. The bus designed and implemented for the KySat-1 spacecraft will serve as the basis for a series of future Kentucky Space Consortium missions. The spacecraft bus consists of attitude determination and control subsystem, communications subsystem, command and data handling subsystem, thermal subsystem, power subsystem, and structures and mechanisms. The spacecraft bus design is described and the implementation and testing and experimental results of the integrated spacecraft engineering model. Lessons learned with the integration, implementation, and testing using commercial off the shelf components are also included. This thesis is concluded with future spacecraft bus improvements and launch opportunity of the implemented spacecraft, KySat-1.

KEYWORDS: Embedded Systems, Fault Tolerance, CubeSat, Small Satellite, Systems Engineering

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March 31, 2009

DEVELOPMENT OF A REUSABLE CUBESAT SATELLITE BUS ARCHITECTURE FOR THE KYSAT-1 SPACECRAFT

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THESIS

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The Graduate School
University of Kentucky
2009

DEVELOPMENT OF A REUSABLE CUBESAT SATELLITE BUS ARCHITECTURE FOR THE KYSAT-1 SPACECRAFT

THESIS

A thesis submitted in partial fulfillment of the requirements for the degree of Masters of Science in Electrical Engineering in the College of Engineering at the University of Kentucky

By

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2009

Dedicated to my friends, family, and advisors for all of their help along the way of my academic career…

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1.1 Small Satellite and CubeSat Introduction

1.1.1 Small Satellites

On October 4, 1957 a new age was started. The launch of Sputnik I by the Soviet Union was the technological advancement that began what became known as the "space age." This launch marked the beginning of the utilization of space for science and commercial activity. The first artificial satellite ever launched, Sputnik I, weighed only 84 kg, and was no bigger than a regulation size basketball, being only 58 cm in diameter [1]. Shown in Figure 1, this spacecraft had a simple and short mission to study the atmosphere.

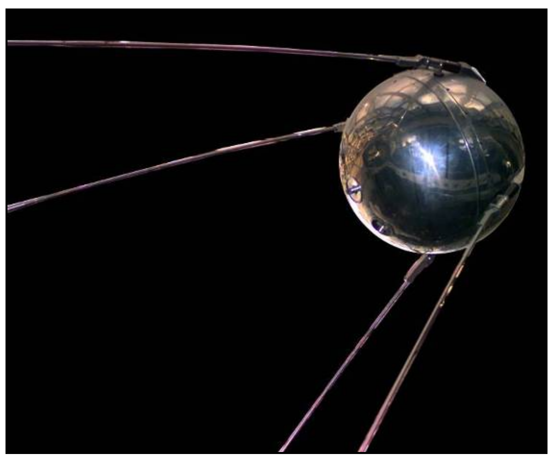

Figure 1 – Sputnik I

The launching of Sputnik I caused the United States to scramble and initiate this so called space race. On January 31, 1958 the United States responded by launching their first satellite, Explorer I. The mission of this satellite eventually discovered the Van Allen radiation belts around the Earth. Explorer I was also a relatively small spacecraft weighing 14 kg, less than Sputnik I, and was just over 80 cm long. A photograph of Explorer I is shown in Figure 2 [1]. During the Cold War, space was a prime area of competition between the Soviet Union and the United States.

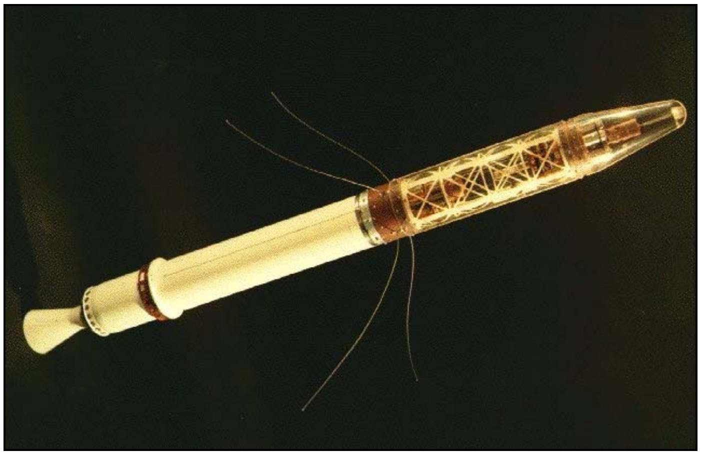

Figure 2 – Explorer I

Both of these spacecraft proved that meaningful science can be performed in a relatively small package. In 1964 the first television satellite was launched into geostationary orbit to broadcast the Olympic Games from Tokyo. Later, Russian launch activities declined while other nations ramped up their own space programs. The number of objects in Earth orbit has increased steadily by 200 per year on average [2] since these initial launches. Despite these first two small spacecraft, satellites following Sputnik and Explorer proved to be much larger, more massive, and more expensive.

Prior to about 1990 the aerospace communities focus was on much larger spacecraft with long and expensive development and mission times. It wasn't until satellites like OSCAR-10, built in 1983, and ALEXIS, built in 1989, that the realization was made that much smaller satellites can yield the same missions goals [2]. The miniaturization of technology allowed these small spacecraft to have meaningful missions with much lower development costs and development time. Because these small spacecraft have a much shorter development time and can cost orders of magnitude less, they are becoming more and more attractive to developers trying to fulfill a customers needs. The cheaper and faster a mission can be completed the more satisfied the customer will be. These small spacecraft are becoming more and more attractive as developers push the envelope to create smaller and cheaper spacecraft with extremely short development times.

There are four basic classifications of small spacecraft [3]. The largest being the "minisatellite" with a wet mass, spacecraft bus payload and fuel, between 100 and 500 kg. The "mircosatellite" has a wet mass between ten and 100 kg. Both the minisatellite and microsatellite are more generic terms and are sometimes are simply referred to "small satellites." The final two classifications, "nanosatellite", between one and ten kg, and the "picosatellite", less than one kg are sometimes grouped together as well and called "nanosatellites." The focus of this thesis will be on a particular standard of nanosatellites called CubeSats, which range between one kg and upwards of five kg.

1.1.2 California Polytechnic CubeSat Program

As a means to decrease the development time and launch availability, in 1999 Professor Bob Twiggs of (Figure 3) Stanford University and Dr. Jordi Puig-Suari of (Figure 4) California Polytechnic (Cal Poly) University collaborated to create the CubeSat Standard as a means to standardize satellite buses, structures, and subsystems. This standard is intended to provide access to space for small payloads.

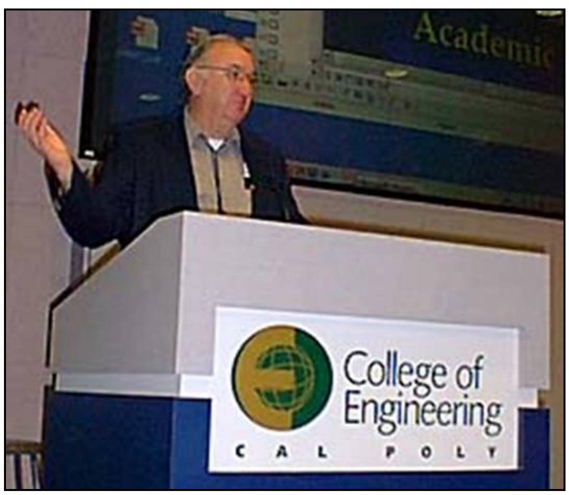

Figure 3 – Professor Bob Twiggs

Figure 4 – Dr. Jordi Puig-Suari

The CubeSat standard currently specifies three form factors available to developers. A one unit (1U) is ten by ten by ten cm and weighs less than one kg. The remaining two are two unit (2U) and three unit (3U), being ten by ten by twenty cm while weighing less than two kg and ten by ten by 30 cm while weighing less then three kg respectively. The standard is currently maintained by the CubeSat Program in the Aerospace Engineering Department at Cal Poly. As all CubeSats to-date have been launched as a secondary spacecraft, the CubeSat Design Specification's (CDS) [3] purpose is to ensure the safety of the CubeSat and protect the launch vehicle (LV), primary payload, and other CubeSats on the mission. This standard specifies dimensional and mass requirements, structural requirements, electrical requirements, operational requirements, and testing requirements to ensure the safety of the primary mission on which most CubeSats will be launched.

In addition to the other requirements, the standard also allows for a common launch vehicle interface (LVI). The most common launch vehicle interface currently used is the Cal Poly Poly-Picosatellite Orbital Deployer (P-POD), shown in Figure 5. The operation of the P-POD is conceptually similar to the action of a "jack-in-the-box." The P-POD can launch up to three units worth of a CubeSats at a time. This means there can be three different configurations of spacecraft launched at any given time, a single 3U spacecraft can be launched, a 1U and a 2U, or three 1U's can be launched. The CubeSats are designed to have separation springs at the feet or mating section of the spacecraft. The P-POD has a main pusher plate and main spring that will push the spacecraft out of the P-POD once the door is opened. The individual separation springs on the CubeSats will push the spacecraft further apart once they have deployed from the P-POD. The different coefficients of drag will separate them further and further the longer they are in orbit.

Figure 5 – Cal Poly P-POD LVI

Despite their small size and limited mass, CubeSats can perform significant science missions and carry multiple payloads. At first there was concern by the aerospace community that these small spacecraft are nothing but space debris, but many organizations are realizing the potential benefits of developing and launching a CubeSat. The National Aeronautics and Space Administration (NASA) have already joined the community with the launch three of their own CubeSats, with another waiting to be launched, all built around the same spacecraft bus. The National Science Foundation (NSF) currently has a program to launch at least one CubeSat per year to study space weather and atmospheric research [5]. Many large aerospace companies such as Boeing and The Aerospace Corporation have also joined the community with the launch of CubeSat Test Bed 1 (CSTB1) and AeroCube-2 [6] respectively.

University students developing CubeSats gain invaluable experience and are challenged with the same problems many aerospace engineers encounter in industry. In addition to the development challenges encountered in all spacecraft, their small sizes, relatively small power and mass budget of the spacecraft warrants some additional unique challenges.

1.1.3 CubeSat Launch History

The first CubeSats were launched on June 30, 2003 using a Eurockot launch vehicle. This launch included five universities and one United States Company. For this first launch two different launch vehicle interfaces were used. One deployer was the Cal Poly P-POD, and the other was an interface developed by the University of Tokyo. This mission saw the successful deployment of all spacecraft [7]. Since this first launch there have been total of eight different launches including CubeSats and there are approximately another seven more launches with CubeSats manifested in 2009.

Table 1 – Previous CubeSat Launches

Launch Vehicle	**CubeSats**	**Total**	**Date**	**Launch Success**
Rockot	CanX-1, DTUsat, AAU CubeSat, QuakeSat, CubeSat Xi-IV, TiTech - CUTE-I	6	June 30, 2003	Y
Kosmos-3M	CubeSat Xi-V, NCUBE-2, UWE-1	3	Oct. 27, 2005	Y
M-V-8	CUTE-1.7+APD, ASTRO-F	2	Feb. 21, 2006	Y
Dnepr	SACRED, ION, RINCON, ICE Cube 1, KUTESat, nCube, HAUSAT-1, SEEDS, CP2, AeroCube-1, MEROPE, Voyager, ICE Cube 2, CP1	14	July 26, 2006	N
Minotaur	GeneSat-1	1	Dec. 12, 2006	Y
Dnepr	CSTB-1, AeroCube-2, CP4, Libertad-1, CAPE1, CP3, MAST	7	April 17, 2007	Y
PSLV	CanX-2, AAUsat-2, Cute-1.7+APD II, COMPASS-1, Delfi-C3, SEEDS	6	April 28, 2008	Y
Falcon 1	PRESat, NanoSail-D	2	August 2, 2008	N

Table 1 shows all of the previous launches and the respective CubeSats that were launched. As you can see there has been two out of the eight launched that have failed to make it to orbit. Despite these set backs, all CubeSats whose respective launch vehicle has successfully made it to orbit, have been deployed successfully.

Because there has been a proven success rate of deployment and a wide adoption of the CubeSat standard, these small spacecraft are becoming increasingly more popular. Commercially available subsystems are being designed to easily and quickly implement the spacecraft bus. Government organizations are issuing requests for proposals (RFP) for these CubeSat spacecraft missions. It is apparent that a reusable bus design is necessary to implement the spacecraft quickly and reliably to fulfill a wide variety of missions both quickly and cost effectively.

1.2 The Kentucky Space Consortium

1.2.1 Kentucky Space Enterprise History and Mission

The Kentucky Space Consortium, formerly known as the KySat Consortium, was created in May of 2006. Under the lead of the Kentucky Science and Technology Corporation (KSTC), the consortium was created as an ambitious non-profit enterprise involving a consortium of universities and both public and private organizations to design and lead innovative space missions within realistic budgets and objectives in the state of Kentucky. Originally created as the KySat Consortium, the first major effort was KySat-1, a 1U pico-class spacecraft, designed around the CubeSat standard [8]. Because the launch opportunities of CubeSats were minimal, the KySat Consortium evolved into a broader mission, including near space, sub-orbital, orbital, and deep space missions. With these new missions the KySat Consortium, being only one year old, officially transitioned into the Kentucky Space Consortium in May of 2007. Figure 6 shows a detailed history of what is now the Kentucky Space Consortium.

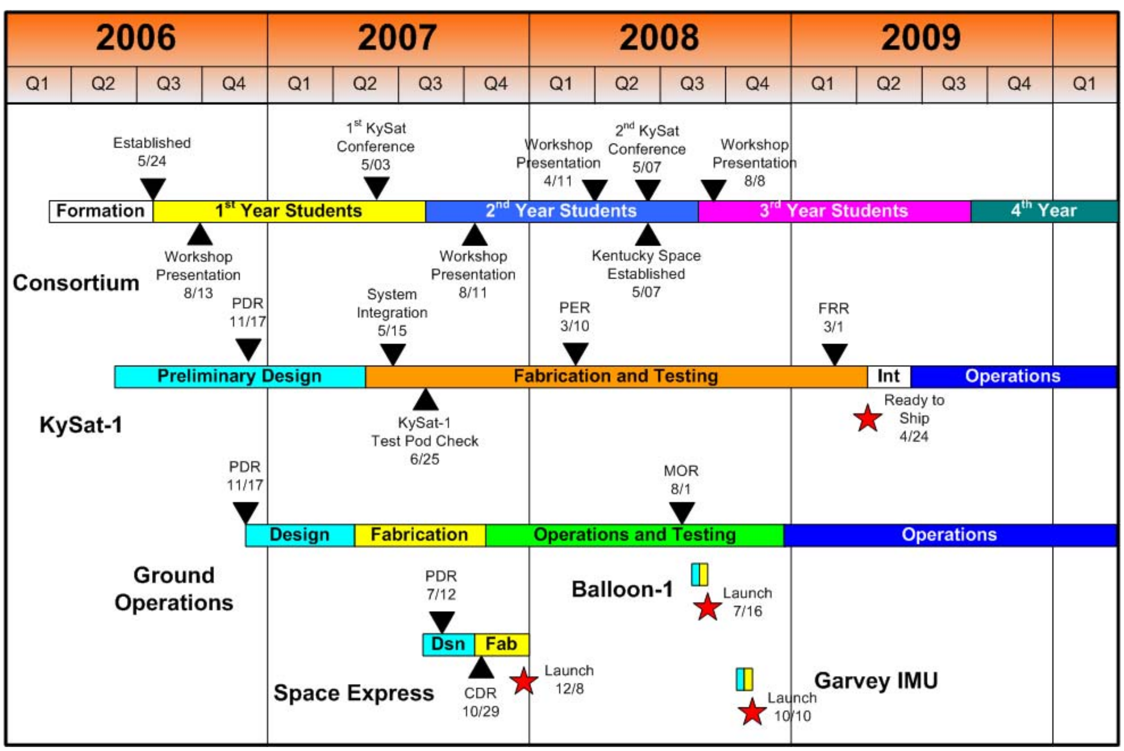

Figure 6 – Kentucky Space Consortium History

The Kentucky Space Consortium is currently made up of six public universities including: University of Kentucky (UK), Murray State University (MuSU), Morehead State University (MSU), Kentucky Community and Technical College System (KCTCS), University of Louisville (UofL), and Western Kentucky University (WKU). The non-academic members include: Kentucky Space Grant Consortium (KSGC), Kentucky Council on Postsecondary Education (CPE), Kentucky Science and Engineering Foundation (KSEF), Belcan Corporation, and managing partner Kentucky Science and Technology Corporation (KSTC). All members of the enterprise hold shares of Kentucky Space Enterprise. Students from each of the universities comprise of the Kentucky Space Design Build Team. The design build team consists of the students responsible for designing, building, and operating the different missions and spacecraft that belong to the Kentucky Space Consortium.

Kentucky Space's goal is to help realize a space exploration program that is focused on efficient costs, robust launch schedules, research and educational opportunities. From high-altitude balloon projects to orbital flights, to reaching out to artists and educators to building small satellites, Kentucky Space seeks to energize a user base far outside the boundaries of Kentucky and beyond the traditional models for space opportunities. As well as building spacecraft, a large portion of Kentucky Space's missions include a public outreach aspect. The formal vision of Kentucky Space is a collaborative, non-profit organization that is recognized for enhancing the economic vitality of Kentucky through the expansion of technology development opportunities in aeronautics and astronautics, the stimulation of business development and economic growth, and the expansion of education opportunities throughout the Commonwealth. The Kentucky Space formal mission statement is as follows:

(1) To provide the means to encourage, facilitate, and implement the development of space science and technology within Kentucky.

(2) To provide design, build, launch, and mission support services for space-related activities through a university-base, student-supported, state-wide collaborative consortium.

(3) To successfully establish a series of independent, technology-related business ventures derived from the activities of Kentucky Space to foster economic growth, provide employment opportunities, and develop the talent force.

(4) To establish an interactive program for engagement of primary and secondary education to develop interest in technology-related career.

(5) To function as an agile, innovative, collaborative, and entrepreneurial organization with the necessary process to successfully achieve the enterprise vision.

1.2.2 Kentucky Space Missions

Since the creation of the Kentucky Space Consortium, the enterprise has successfully completed three missions, with a fourth awaiting launch. Most all missions completed and future missions use the same technologies and are intended to be more advanced from the previous mission by building upon the other mission's successes and failures. To facilitate multiple missions, robust reusable hardware and software must be designed, developed and tested. The main goal from one mission to another is reuse.

The first mission completed by the Kentucky Space Consortium was named, Space Express. Space Express [9] was a rapid turn around, suborbital access to space experiment launched in December 2007. The Space Express mission was designed to help test subsystems and processes being developed for future orbital missions. The Space Express mission was launched from the White Sands Missile Range on a Lunar Rocket and Rover Shadow 1B launch system. The space express payload consisted of telemetry package that gathered temperature, pressure, and mission time on board the rocket. The telemetry gathered onboard the rocket was transmitted to three redundant ground stations using the very high frequency (VHF) amateur HAM radio band. On December 5, 2007 at 09:06:26 mountain standard time (MST), Kentucky Space Space Express was launched from the Small Missile Launch Complex at the White Sands Missile Range in New Mexico. Figure 7 shows the launch as it traveled in a suborbital trajectory to a planned altitude of approximately 115-125 km. At liftoff all systems performed nominally. Sensors in the payload detected ignition of the booster and started the mission timer, logged flight data and transmitted telemetry packets to the ground stations. At approximately one point two seconds following liftoff, the vehicle suffered a distinct roll-yaw departure from stable flight. Between one point two seconds and one point five seconds after liftoff the solid rocket booster should have expended fifty to sixty percent of its propellant and approaching mach three. The acceleration would have been over 100 Gs and temperature of the nose is estimated to have been in excess of 1000 degrees Fahrenheit. While the launch system suffered a failure during the boost phase, the payload was confirmed to function as designed by receiving telemetry approximately seven seconds after liftoff.

Figure 7 – Space Express Launch

The second mission completed by Kentucky Space was a high altitude balloon mission. This mission, launched out of Bowling Green Kentucky on July 16 in 2008, was a high altitude test of sub-systems to be used on future orbital missions. The high altitude balloon payload was launched with the help of WB8ELK Balloons. This mission included five different payloads on the flight string. Starting from the top of the string was the Kentucky Space Payload, which included a temperature, pressure, and magnetic field strength logging device, automatic packet reporting system (APRS) tracking device, continuous wave beacon, and two high resolution imagers. Figure 8, taken from one of the imagers onboard shows the curvature of the earth and some of the flight string below the Kentucky Space payload. The second payload was an amateur television (ATV) transmitter and video camera. The third payload was an experimental simplex repeater used in conjunction with the Kentucky National Guard. The fourth payload was a SPOT Satellite Personal Tracker device, used as a redundant method of recovery. The fifth and final payload was a redundant automatic packet reporting system (APRS) tracking device. As a means of public outreach, PearlSats were also flown on this mission. PearlSats are ping pong balls cut open to allow younger students to place gummy bears or other candy and items placed inside of them and then glued back together for flight. Once recovered these PearlSats are returned to the students for inspection to learn about the effects of the high altitude temperature and low pressure environment.

Figure 8 – Image taken by Kentucky Space Balloon Payload

The third mission completed by the Kentucky Space Consortium consisted of a strap down inertial measurement unit (IMU) that was launched on the Garvey Spacecraft Corporation (GSC) P-12A launch vehicle. The inertial measurement unit consisted of three axis acceleration and three axis rotation sensing. This flight was also a test of sensors planned to be used on a future orbital missions. From the data that the payload logged, the flight trajectory could be determined with sub-millisecond accuracy. This payload was launched on October 10, 2008 out of the Mojave Desert in California. Although there was an aerodynamic instability problem with the vehicle and a failure in the parachute recovery system onboard the launch vehicle, the payload was recovered, as shown in Figure 9. Data collected from the payload helped engineers at Garvey Spacecraft Corporation to determine the cause of the aerodynamic instability problem.

Figure 9 – Kentucky Space IMU Payload Recovery

The fourth mission for Kentucky Space, currently awaiting a launch, is KySat-1 as the enterprises first orbital satellite shown in Figure 10. This first orbital mission has two goals. The first is internal to Kentucky Space, developing a reusable bus to be used on multiple missions. The second goal is public outreach to interest younger students in science, technology, engineering, and mathematics (STEM). The satellite was designed with an attractive concept of operations in mind, to facilitate the spacecraft being used as a hands on learning tool for kindergarten through high school students in the areas of science, engineering, and math. The timeline to design and build KySat-1 was aggressive; therefore off-the-shelf technology was used whenever possible. As KySat-1 will be the basis for a series of CubeSats to be launched, the spacecraft bus was designed to be robust, modular and reusable. The CubeSat payload includes a low resolution imager and an experimental high bandwidth communications transceiver. The formal mission goals for KySat-1 are as follows:

(1) Create an Infrastructure to Develop CubeSats
(2) Develop a Reliable and Reusable CubeSat Bus Architecture
(3) Implement Operations that will Attract Kentucky Students to Space Technology

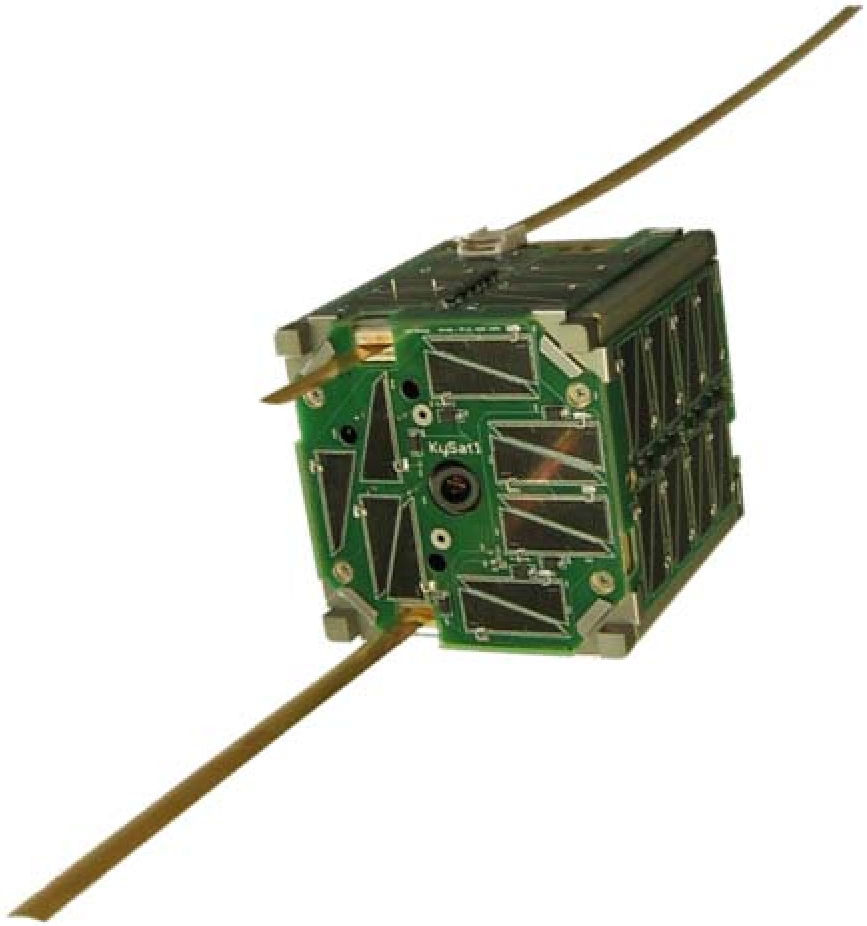

Figure 10 – KySat-1 Engineering Model

1.3 Thesis Statement and Motivation

As the Kentucky Space consortium has plans to launch a series of CubeSat missions with a wide variety of payloads it is apparent that a reusable spacecraft bus is necessary. This thesis will overview the design, development, implementation, and testing of the KySat-1 CubeSat Spacecraft bus. The main motivation of this thesis research is the implementation, systems integration, and systems engineering of the spacecraft bus. The spacecraft bus is implemented using commercial off the shelf products. This spacecraft bus design uses less than a single U of a CubeSat. Chapter two of the thesis covers the designed architecture of the spacecraft bus. It will include details of the commercially purchased subsystems along with the supplier and the custom designed subsystems used in the implementation of the KySat-1. Once the design and architecture has been established Chapter three will discuss the integration and implementation problems and solutions encountered integrating commercial off the shelf components with the custom designed hardware to implement the designed spacecraft bus, which is the main focus of the thesis research. Chapter four will then cover the experimental testing and results of the integrated engineering model spacecraft, KySat-1. Chapter five discusses the lessons learned with the integration, implementation, and testing using commercial off the shelf components. Future spacecraft bus improvements and launch opportunity of the implemented spacecraft, KySat-1 is also included in the fifth chapter. The thesis is then concluded in chapter six.

KySat-1, developed by the Kentucky Space consortium is manifested for a late 2009 to early 2010 launch. For this spacecraft there were two primary payloads. The first payload being a low resolution imager and the second being a high bandwidth communication transceiver. This mission will be to transition the transceiver from being simply a payload to part of the spacecraft bus design itself. The KySat-1 spacecraft is designed with an attractive concept of operations as an opportunity for outreach to younger students. The students will have the ability to command the spacecraft using hand held radios and antennas to take photographs, play audio files on the HAM radio, and use telemetry collected from the spacecraft to get a hands on approach to learning about the space environment.

The integration of the spacecraft was split into two large portions. The first major hurdle was to ensure the spacecraft would mechanically fit together. The second component of integration was the electrical system. For most of the commercially purchased items, this was the first time they were integrated into a spacecraft. As with any complicated design there are inevitable problems. The electrical integration of the spacecraft was one of the larger challenges encountered during the design and implementation. With integrating custom designed hardware with off the shelf hardware there were problems that had to be addressed and solved. The largest integration effort went into the integration of the electrical power system. KySat-1 was the first spacecraft bus design to use this electrical power system resulting in many integration challenges. With all off the shelf sub-systems there was some integration effort. Given the nature of satellite design, these off the shelf sub-systems are still custom hardware, just not custom hardware designed by Kentucky Space.

Thorough testing of the spacecraft is one of the most important aspects in the design life cycle to assure mission success. One of the most important aspects of this testing is to ensure you are testing like you will be flying. Testing to a space environment can be difficult due to the extreme differences in space and terrestrial environments. At the time of the writing of this thesis, testing was an on going process for the KySat-1 spacecraft engineering model. Prior to complete integration of the spacecraft the antennas were first matched and tuned. After the antennas were tuned and matched to the correct frequencies, the antenna radiation patterns were measured at an outdoor range. The first large scale test that was performed on the fully integrated engineering model was the thermal environmental testing. The engineering model of the spacecraft was placed in a thermal chamber and the temperature was cycled to mimic the temperature swings the spacecraft would see being in a ninety minute orbit, going from sun to shade and vice versa. Following the thermal environmental testing the spacecraft was taken to the outdoor range once again to measure the performance of the entire communication system. In parallel to all of this testing with the integrated engineering model a duplicate set of spacecraft bus hardware was up and running to test the spacecraft flight software.

The focus of this thesis and research performed by the author included the electrical and mechanical integration, spacecraft software design, and experimental testing done to date. This project included a wide number of students that were involved overall. The author of this thesis was the systems integrator and systems engineer for the Space Express, Garvey IMU, and KySat-1 missions. For the first balloon mission he served as an advisor to the younger students that lead them to a successful mission. His duties on the

KySat-1 spacecraft included working with the other team members and outside vendors to ensure all designed and purchased sub-systems with electrically and mechanically work in the space environment for KySat-1. He was the lead hardware designer for the custom hardware designed for the KySat-1 bus, ensuring all board schematics and layouts were correct.

2 KYSAT-1 SPACECRAFT BUS DESIGN

All spacecraft can generally be broken into two main parts, the spacecraft bus and payload. The spacecraft bus typically consists of seven different subsystems. The seven subsystems include propulsion subsystem, attitude determination and control subsystem (ADCS), communications subsystem, command and data handling (C&DH) subsystem, thermal subsystem, power subsystem, and structures and mechanisms [2]. While some spacecraft may have a more demanding mission or payload than others and include all seven subsystems, there are some missions and payloads that don't require all seven subsystems. The spacecraft bus discussed in this chapter of the thesis has six of the seven subsystems listed previously, omitting propulsion.

2.1 Design Approach and Timeline

Most previous CubeSat missions were a "one-off" design, meaning one spacecraft was built and it was very difficult to reproduce either due to component or subsystem selection. The KySat-1 bus was designed with a vision of a reusable, modular, and robust platform. The design was on a fairly short timeframe, therefore the approach taken to use existing off the shelf components and subsystems. The design approach leveraged the use of existing components that could be commercially purchased. Using commercially available components and subsystems would give the community flight heritage in these subsystems so they could be purchased and flown on many different missions in different configurations.

The design process followed a strict systems engineering approach using the NASA Systems Engineering Handbook as reference. The design of the mission started with Phase A, selecting the concept of operations. Becuase one of the main purposes of KySat-1 was the design of the spacecraft bus, there wasn't a specific payload the bus was designed around. The bus and payloads were designed independently. The selection of the payload was chosen to support the second mission goal of providing public outreach. Phase A was a short time period due to the simple yet attractive concept of operations. Phase B, the preliminary design happened primarily during a six week timeframe where most of the team members completed an internship at NASA Ames Research Center and Stanford University. Phase C, the fabrication and implementation of the spacecraft bus components, was roughly a six month period. The building and testing, Phase D, was the longest period. The systems engineering life cycle can be seen in Figure 11.

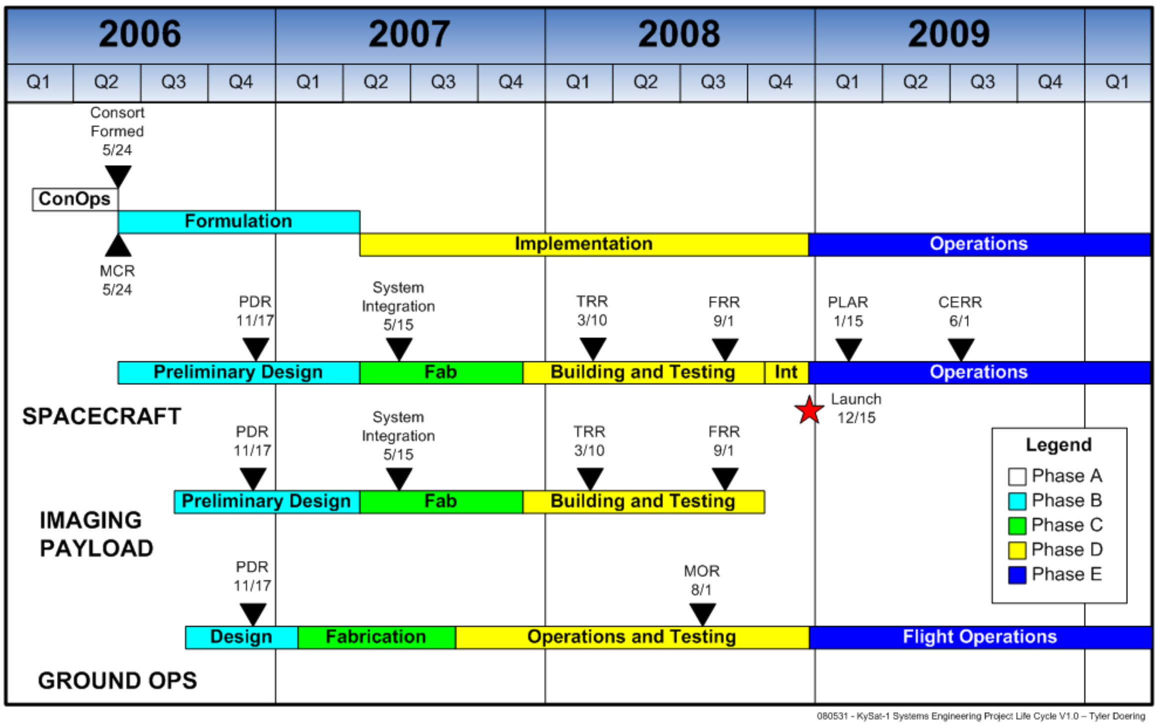

Figure 11 – Systems Engineering Life Cycle

The designed architecture was based on the CubeSat Kit (CSK) purchased from Pumpkin Inc. This kit is the only standard with respect to mechanical and electrical interfaces currently available in the community. It is designed around the PC/104 standard form factor with a custom 104-pin header to provide communication and power throughout the bus. This standard has been adopted by other companies developing subsystems to be compatible with the CubeSat Kit Standard. Figure 12 shows a detailed hardware block diagram of the spacecraft bus design implemented for the KySat-1 spacecraft. This design leverages both commercially purchased subsystems, shown in black, and custom designed subsystems, shown in blue.

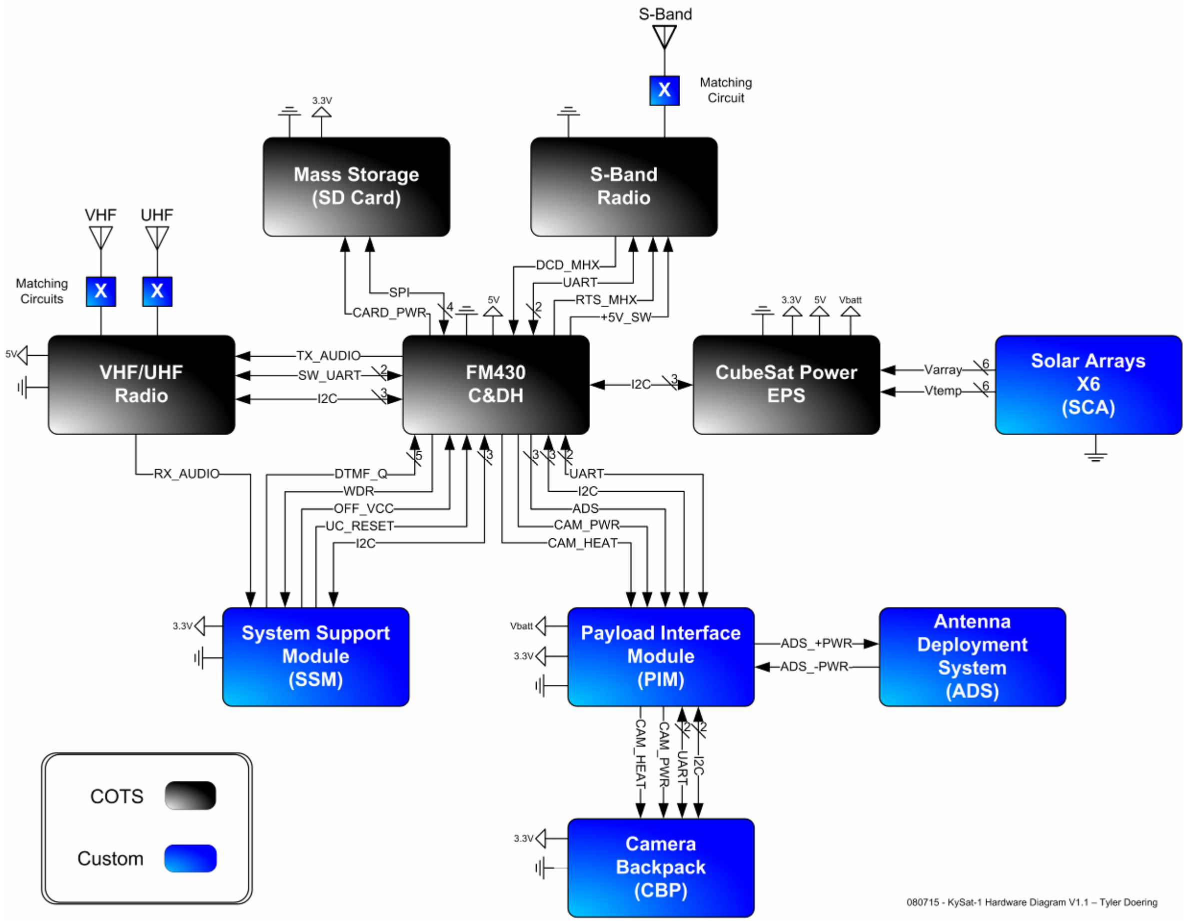

Figure 12 – Spacecraft Hardware Block Diagram

2.2 Structures and Mechanisms Design

The structures and mechanisms design of the spacecraft consists of the spacecraft frame and board stack architecture, antenna mounting and deployment, and outside solar board mounting. The whole external mechanical design is required to conform to the CubeSat Design Specification. The mechanical architecture chosen was the CubeSat Kit architecture. The CubeSat Kit is designed around the PC/104 standard form factor, with a simple pass-through connector scheme. This stacking interconnect bus header is well-suited for the tight confines of a CubeSat. The different printed circuit boards that make up the different subsystems are fastened in all four corners with standoffs in a board stacking fashion.

2.2.1 Spacecraft Frame Assembly

The spacecraft frame, shown in Figure 13, is commercially purchased from Pumpkin Inc. The structure, consisting of three main sub-assemblies, the base plate, chassis walls, and cover plate, are made from 5052-H32 sheet aluminum. All captive and loose fasteners are made from stainless steel. All surfaces that come in contact with the launch vehicle interface, CubeSat rails, are hard-anodized to prevent galling. The hard-anodizing creates a non-conductive surface, therefore to maintain the shielding of a faraday-cage; the remaining surfaces of the structure are gold alodyned to maintain electrical conductivity. The base plate, chassis walls, and cover plate are skeltonized to be as small as mass as possible, but still providing a sturdy robust structure to survive the harsh launch environment.

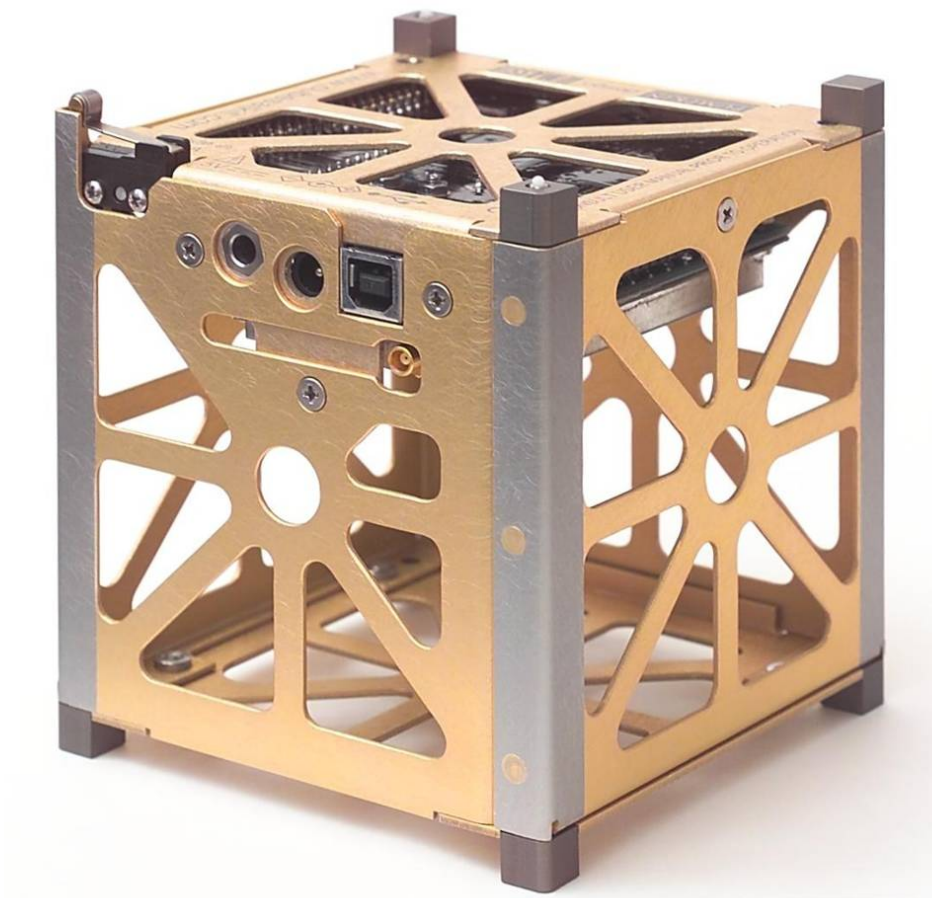

Figure 13 – Spacecraft Frame Assembly

The CubeSat Kit frame assembly also includes the CubeSat feet made of 6061-T6 aluminum. These small feet are used because the different CubeSats loaded into the launch vehicle interface are stacked on top of each other. The feet on the base plate end of the frame assembly have integrated separation springs to further push the spacecraft apart after being deployed from the launch vehicle interface. On an adjacent foot there is a separation switch. This separation switch will keep the spacecraft powered off prior to ejection from the launch vehicle interface. The separation springs and separation switch are necessary to conform to the CubeSat Design Specification.

Along with the separation switch the second power inhibit required by the CubeSat Design Specification is the remove before flight pin. This pin is used as another method for keeping the spacecraft from turning on prematurely. While the different CubeSats are being integrated into the launch vehicle interface the separation switch may become activated due to handling the CubeSat, therefore this remove before flight pin is kept in place until fully integrated into the launch vehicle interface and the separation switch can be kept deactivated.

The final part of the frame assembly are the eight solar board clips used to hold the six solar arrays captive to the frame assembly. The eight clips are simple bent sheet metal that is held captive between the cover plate/base plate and the CubeSat feet. These clips then provide a basic clamping that will hold the solar cells captive. These clips provide an extremely simple yet robust and light weight design to keep the solar arrays captive to the spacecraft frame.

Prior to being used on the KySat-1 mission, the CubeSat Kit Frame has been successfully flown on two different missions, one being the three unit frame on Delfi-C3, the first nanosatellite student project from the Delft University of Technology. The second being a single unit frame used on Libertad-1, Colombia’s first satellite. Libertad-1 was developed by eight students from Universidad Sergio Arboleda with no prior satellite experience. The CubeSat Kit frame is also being used by a number of other universities and private and public companies [10].

2.2.2 Antenna Mounting and Deployment

The spacecraft has three antennas as shown in Figure 10. These three antennas are made of spring steel. The spring steel was chosen because it can be folded or curled up then tied down. Once the tie down is released or cut the antennas will spring back to their unfolded, straight position. The spring steel is the same material found in most tape measures. The unfolding nature of the material makes it ideal for wrapping around the exterior of the cube. Due to the size needed for the antennas they need to be stowed somehow to comply with the CubeSat Design Specification.

The mounts used to hold the antenna captive to the frame are made of ULTEM-1000 material from GE Plastics. This material was chosen for its strength, small mass, long ultra-violet degradation, and low out-gassing properties. This material needs to be a non-conductive material; therefore a plastic was the best choice. Three small clamps were designed to hold the spring steel antennas. Two of the clamp designs were the same, these are both internal to the frame, and the third clamp is a different design and is on the outside of the spacecraft frame. The two clamps that hold the shorter antennas, shown in Figure 14, are attached to the chassis wall part of the frame. The stock frame from Pumpkin must be modified with two small counter sink holes to attach the antenna mounts.

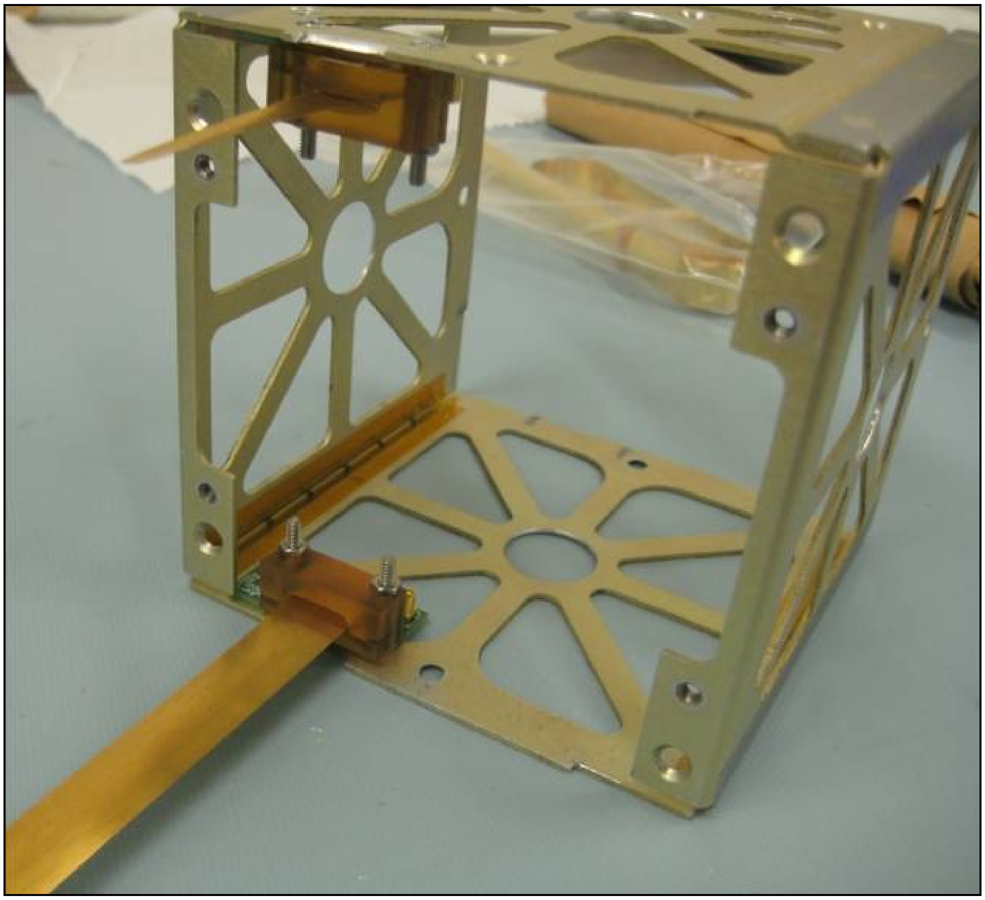

Figure 14 – Interior Spacecraft Antenna Mounts

The third antenna mount is on the exterior of the spacecraft. This mount is also used as a guide for the largest antenna as it wraps around the cube and is used as a method to stow the other two antennas. The three antennas are all held in their stowed position using a simple nylon fishing string called Sypder Wire. This deployment line is passed through a nichrome wire coil used to cut the line after current is passed through the coil. The deployment line is then tied off to an attached spring to keep ample tension on the line to keep the antennas stowed properly. The spacecraft with stowed antennas and nichrome wire cutting coil is shown in Figure 15.

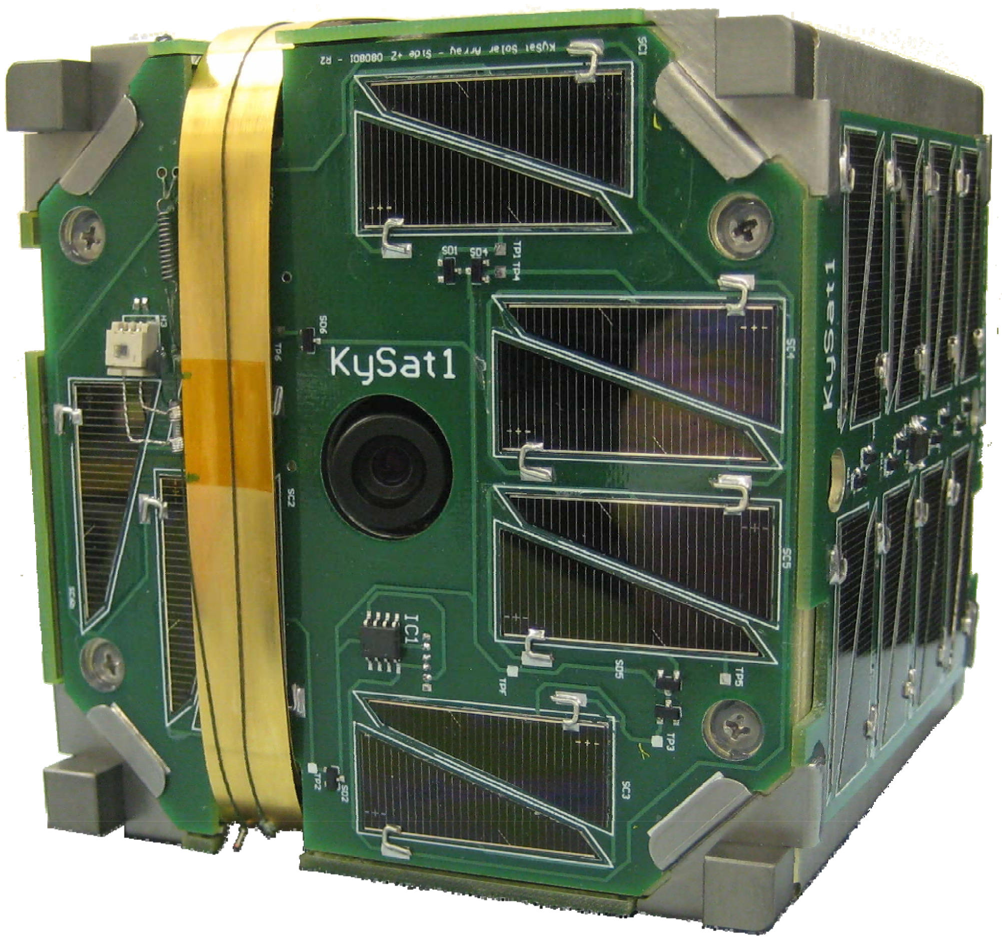

Figure 15 – Spacecraft with Antennas Stowed

2.3 Attitude Determination and Control Design

The spacecraft attitude determination and control system (ADCS) is a purely passive magnetic design. The system is designed to orient the spacecraft with the antennas tangential to the surface of the earth and perpendicular to the equator as it passes overhead. Once the spacecraft is put into a polar orbit and it passes over the North and South Pole of the Earth, the spacecraft will flip completely over. This is achieved using permanent magnets. This tumbling is damped out with the use of hystersis material. The four permanent magnets are located in the interior corners of the spacecraft chassis wall frame, as shown in Figure 14. The design is best for earth stations with lower latitudes. When the spacecraft flips over the North and South Pole the antenna nulls will be pointing directly at the earth stations located in the footprint of the spacecraft. As the spacecraft passes through the equator the antennas will be perpendicular with the equator, therefore giving grounds stations located on the equator the best view of the spacecraft in terms of antenna gain. This flipping nature of the spacecraft's passive attitude control system can be seen in Figure 16.

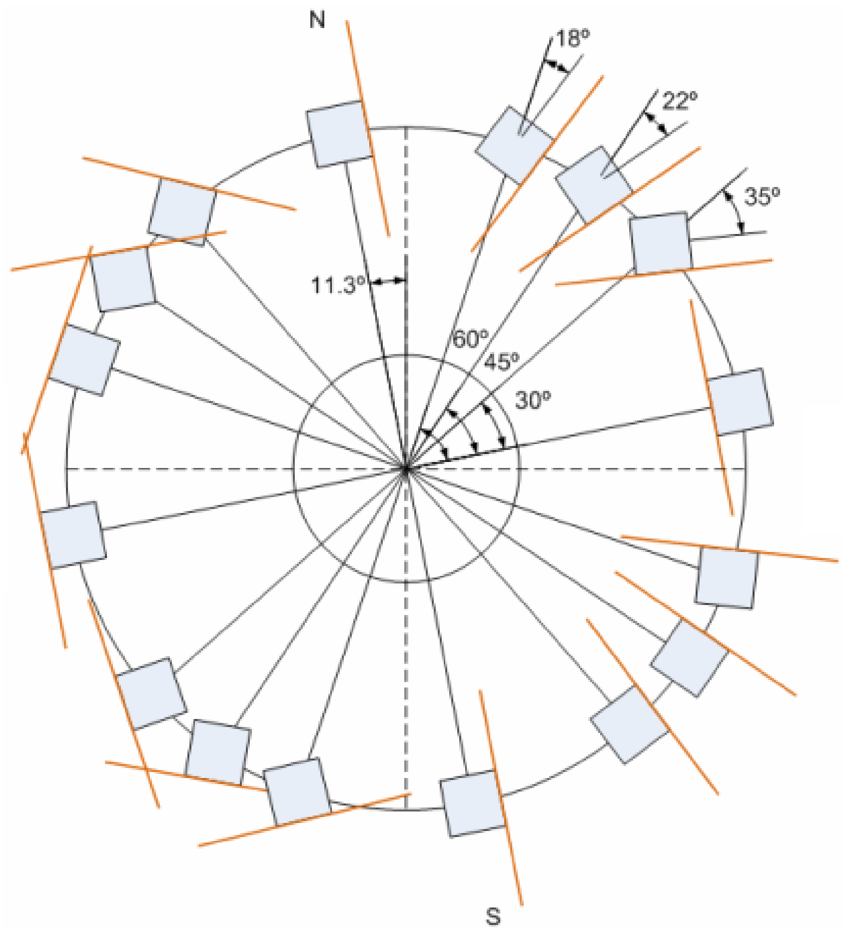

Figure 16 – Spacecraft Tumbling Nature

Because the attitude control of the spacecraft is purely passive, the spacecraft has very limited knowledge of the current attitude. The spacecraft can determine the current attitude by using the six different solar arrays as rough estimates of sun sensors. The solar panel voltage and current can be read to determine the location of the sun.

Sampling these readings can also deduce a tumble rate. This method of determination is an extremely rough estimate of current attitude.

2.4 Command and Data Handling Design

The command and data handling subsystem of the spacecraft is based on two different printed circuit boards. The main printed circuit board is the FM430 Flight Module from Pumpkin Inc. This single board computer for harsh environments is based around the MSP430 microcontroller and conforms to the PC/104 form factor. The FM430 was missing some necessary components to make the bus design complete. Instead of designing a flight computer board from the ground up it was necessary to have a supporting board. This other board that makes up the command and data handling (C&DH) subsystem is the KySat System Support Module (SSM). The system support module contains items necessary to the spacecraft bus that are not included in the FM430 Flight Module from Pumpkin Inc. The system support module is one of the five custom designed printed circuits boards that make up the spacecraft bus.

2.4.1 FM430 Flight Module

The FM430 Flight Module, as shown in Figure 17, is specifically designed to have low power consumption. The flight microcontroller, which is the basis of the FM430 Flight Module, is the Texas Instruments single-chip 16-bit MSP430 ultra-low power reduced instruction set computer (RISC) microcontroller. The specific MSP430 that was chosen as the flight computer is the MSP430F1611. This microcontroller has 50 kilobytes of flash and 10 kilobytes of random access memory (RAM). The microcontroller also includes a wide variety of on chip peripherals including forty eight input/output (I/O) pins and two universal synchronous asynchronous receiver and transmitters (USART). One includes serial peripheral interface (SPI) hardware, inter-integrated circuit (I2C) hardware, universal synchronous receiver transmitter (UART) hardware. The other includes SPI hardware and universal synchronous receiver transmitter (UART) hardware. Other on chip peripherals include a 12-bit analog to digital converter (ADC), 12-bit digital to analog converter (DAC), direct memory access (DMA) controller, watch dog timer (WDT), two 16-bit counters each with three capture/compare registers, on-board temperature sensor, and multiple clock sources.

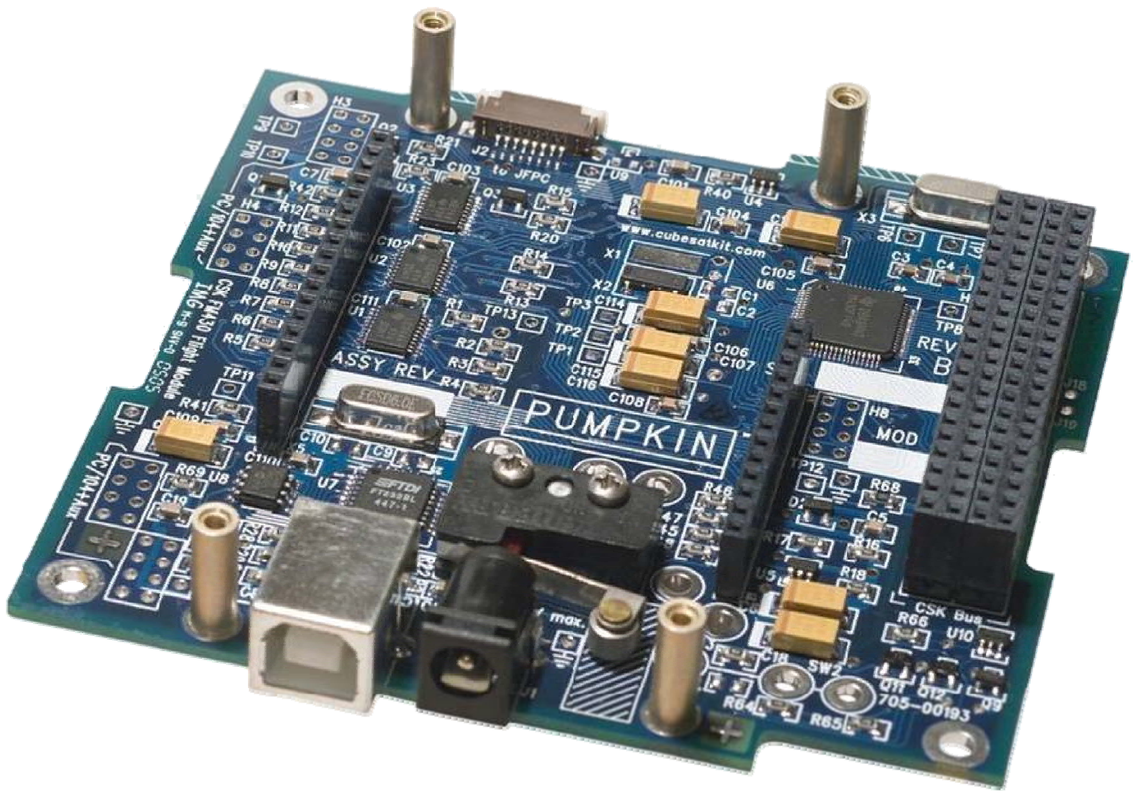

Figure 17 – FM430 Flight Module

The FM430 Flight module is powered off a single five volt supply with three point three volt input/output (I/O). It is designed with the 104-pin CubeSat Kit Bus connector that includes the MSP430's complete I/O space and user assigned signals. The board has an on-board low-dropout regulator and reset supervisor for maximum reliability. There is a secure digital (SD) card socket for mass storage devices to add an additional non-volatile storage between thirty two megabytes and 2 gigabytes. This storage is used to store large amounts of data from the spacecraft payload. There is direct wiring for a ten ampere remove before flight and separation switch. The board is fitted with universal serial bus (USB) 2.0 to universal asynchronous transmit receive bridge that can be used for pre-launch communications and configuration and battery charging while the spacecraft is being integrated and tested. For maximum fault tolerance the module incorporates comprehensive over current, over voltage and under voltage protection for reset and brown out conditions.

2.4.2 System Support Module

One of the five custom designed printed circuit boards that make up the spacecraft bus is the System Support Module (SSM). When selecting the FM430 Flight Module there were a number of items needed to have a complete modular and robust bus design. Instead of designing a flight computer from the ground up to include these missing subsystems, a separate supporting printed circuit board was designed. The system support module mainly has sub-circuits for fault tolerance. As all boards designed for the spacecraft bus, the System Support Module also has the 104-pin CubeSat Kit Bus header. This header is where all the power and communication signals are routed to the flight computer. The system support module is shown in Figure 18.

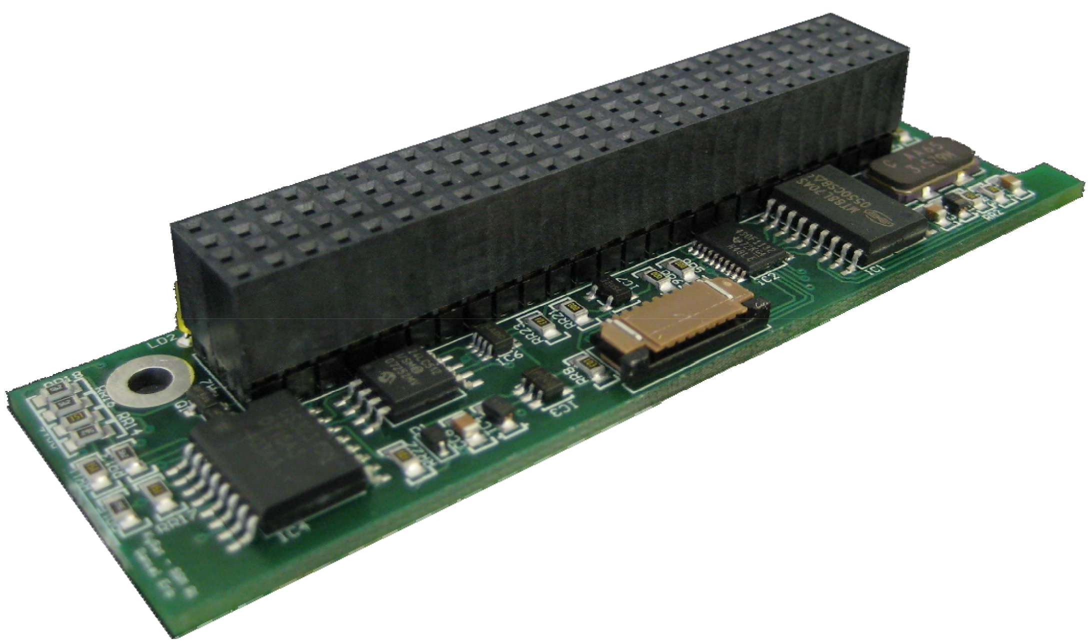

Figure 18 – KySat System Support Module

Because one the flight computer MSP430's universal synchronous asynchronous receiver and transmitters can be configured for either I^2C communication or SPI communication and this communication happens on the same physical pins of the microcontroller. This simple metal oxide semiconductor field effect transistor (MOSFET) isolation circuit is located on the System Support Module. There is also a 512 kilobit serial electrically erasable programmable read-only memory (EEPROM) chip that the flight computer communicates with I^2C communication. This EEPROM is used as extra non-volatile memory by the flight computer to store different operating parameters and spacecraft state information. The System Support Module also includes a real time clock (RTC) that communicates with the flight computer using I^2C communication. This real time clock is clocked with an integrated temperature compensated crystal oscillator (TCXO) to provide a stable clock source without drift due to temperature extremes in the space environment. The system support module has a supporting processor to the flight processor for fault tolerance purposes. If the main flight computer were to encounter a fault either to programming error or a single event upset (SEU) due to radiation, the support processor can force a hard reset to the main flight computer. This processor is also in the MSP430 family, but much smaller. Along with the support processor there is an external watch dog timer to the main flight computer located on the system support module. This external watch dog timer has a one second timeout before in forces a hard reset to the main flight computer. For another method of commanding the spacecraft, this module also has a dual tone multi-frequency (DTMF) decoder. Analog audio is sent to the system support module through the CubeSat Kit Bus header from the radio and the audio is decoded by the dual tone multi-frequency decoder and then sent the main flight computer with a 5-pin parallel bus. This bus is sent to the support processor as well to provide another level of reset of the main flight computer for increased fault tolerance and reliability.

2.5 Electrical Power System Design

The electrical power system (EPS) designed for the spacecraft is broken into three main subsystems. The first is the main electrical power system board which includes the interface to the solar arrays (to charge the spacecraft batteries), the switching regulator circuitry, and bus voltage rails with over current protection. The second subsystem is the spacecraft batteries, which are based on lithium polymer (LiPo) cell chemistry. Both of these subsystems are purchased from Clyde Space Ltd. The third subsystem that encapsulates the spacecraft power system are the solar arrays that take the suns energy and convert them to electrical power used to power the spacecraft and charge the batteries. The solar arrays are four of the five custom designed printed circuit boards that make up the spacecraft bus.

2.5.1 Main Electrical Power System Board

The main electrical power system board, as shown in Figure 19, is the bulk of the design. This board is designed around the PC/104 form factor with the 102-pin CubeSat Kit Bus header. Through this bus header the power system provides a battery voltage, 5V, and 3.3V rails to the rest of the spacecraft. The six different solar array faces are connected to the three battery charge regulators (BCR) on the perimeter of the board. Due to available board real-estate in the PC/104 form factor Clyde Space was only able to design three battery charge regulators. Because there are six faces of the spacecraft with solar arrays, each opposite side face is connected in parallel and diode protected. If the sun were to be illuminating one side of the spacecraft, it cannot be simultaneously illuminating the other side. Being diode protected the electrical power system will draw power from the solar array that provides the most power. Due to this configuration the spacecraft cannot take any advantage of the suns reflected energy off the earth, or albedo. All three battery charge regulators use an active maximum power point tracking (MPPT) design architecture with single ended primary inductor converter (SEPIC) to boost up from a minimum of three point five volts to the battery bus voltage to charge the battery. The battery charge regulators are self-sustaining and do not rely on power from the battery for their operation. This means the battery charge regulators can supply charge to the battery when the solar arrays are illuminated and regardless of the state of charge in the battery. These battery charge regulators have a flexible design and can accommodate different types of solar cells and string lengths, or solar cells in series. The battery charge regulator uses a taper charge method. The system works on the basis that when the battery voltage is below the pre-set end of charge (EoC) or float voltage, the regulator operates in maximum power point tracking mode, acting as a current source to the battery. Once the end of charge or float voltage is reached, the regulator regulates its voltage and the battery charge regulator will drift from the maximum power point and act as a voltage source. This centralized end of charge voltage controller provides the constant current, constant voltage suitable for charging lithium ion and lithium polymer batteries. One of the battery charge regulators is interfaced to the five volt universal serial bus from the FM430 Flight module to provide spacecraft battery charging and top off during integration and testing.

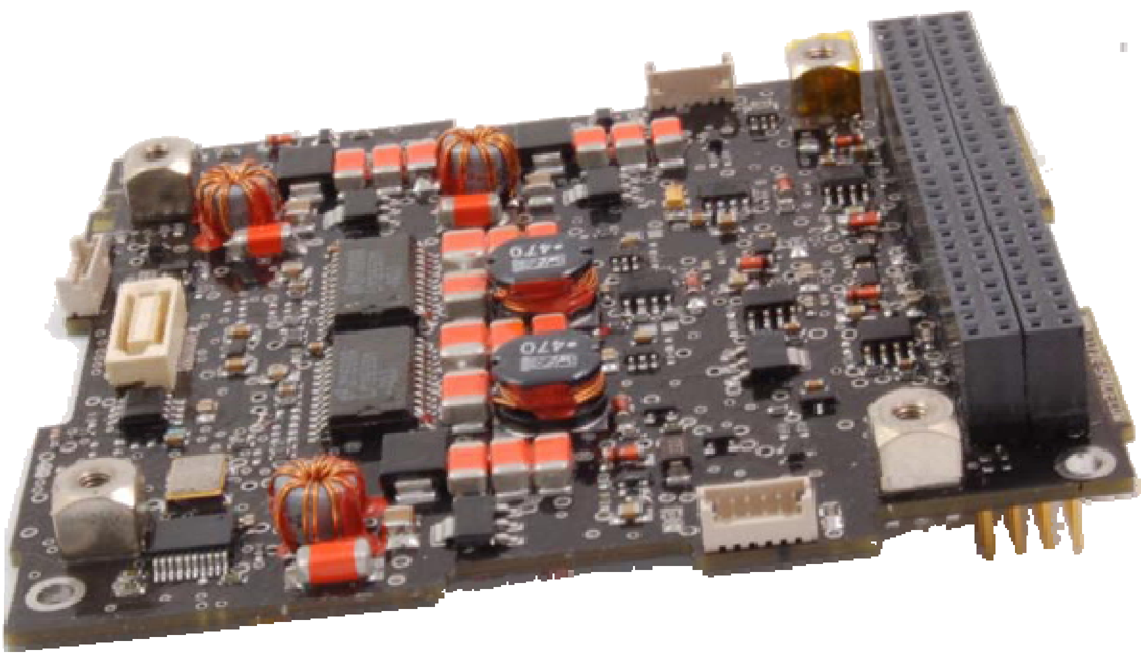

Figure 19 – Main Electrical Power System Board

To provide the spacecraft with the required voltage rails the power system has BUCK switching regulators as the main converter stage to provide high efficiency direct current to direct current (DC-DC) converter to regulate to five volt and three point three volt from the raw battery voltage. The final output stage incorporates an automatic hysteretic light mode of operation to provide seamless operation from zero load. All three bus voltage rails have an over current timed disconnection of the power bus. This timed fashion will turn off the voltage rail once an over current fault has occurred and be off for a specified time before reactivating. This on/off switching is used to correct any fault that may have happened. This timed operation also allows for ample in-rush current as different subsystems boot up without deactivating the voltage rail.

The power system also provides a wide variety of telemetry and telecommands using I^2C communication through the CubeSat Kit Bus header. The telemetries available are six solar array voltages, currents, and temperatures as well as battery voltage and temperature. The telecommands include telemetry reading, power system status, bus voltage rail pulsing, battery heater control, and soft reset of the microcontroller on board the power system.

2.5.2 Electrical Power System Batteries

The spacecraft batteries are based on the lithium polymer cell chemistry and purchased from Clyde Space along with the main power system board. A single spacecraft battery is referred to as two series cells. The two series cells are mounted flat, side-by-side on a separate printed circuit board. This daughter board architecture provides for a modular stacking approach. Two daughter battery boards can be stacked on top of each other on top of the electrical power system doubling the capacity. The complete power system shown with two stacked batteries can be seen in Figure 20.

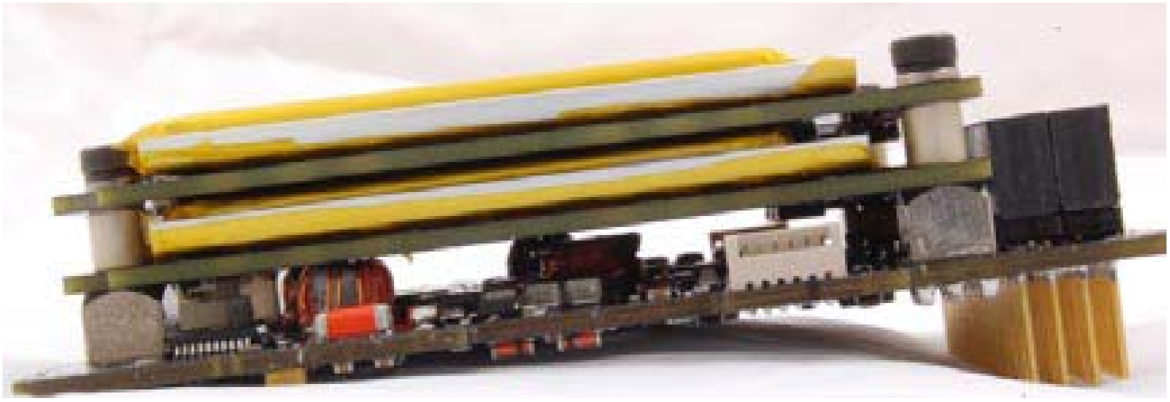

Figure 20 – Spacecraft Power System with Two Stacked Batteries

This stacking approach allows for a spacecraft bus design with two different battery capacities. A one battery board spacecraft would have a battery capacity of 1250 mili-ampere hours (mAh), and a spacecraft with two batteries is double that capacity, 2500 mili-ampere hours. This allows for a modular and redundant design. For most of these small spacecraft the limiting factor on mission duration is the batteries. With a two battery spacecraft, if one of the batteries were to fail, there would be a redundant battery to continue the mission. The battery cells are VARTA PoLiFlex lithium polymer cells. Two series cell each with a fully charged voltage of 4.1 volts provides a fully charged battery bus that rests at 8.2 volts.

The limiting factory of the spacecraft temperature extremes is the batteries. These batteries have an operating temperature of -20 °C to 60 °C while being discharged. While being charged the operating temperature range is narrower to 0 °C to 45 °C. To accommodate the extreme cold temperatures of the space environment and still be able to charge the batteries, there is an integrated resistive thermostatically controlled battery heater designed into the daughter printed circuit board which the battery cells are mounted to. This battery heater is design to keep the battery cells greater than 0 °C. The battery heater circuit has an override command from the main power system board that can turn off the heater in the event of a detected fault. Each battery board has over current, over voltage, under voltage protection. The battery board also provides battery current, battery voltage, individual cell voltage, and current direction telemetry to the main power system board.

2.5.3 Solar Array System Design

The spacecraft is fitted with solar arrays on all six faces of the CubeSat. The solar arrays make up four of five custom printed circuit boards. There are a total of six total arrays but only four unique designs. The solar arrays are based around the triangular advanced solar cells (TASC) from Spectrolab. Figure 21, Figure 22, Figure 23, and Figure 24 shows the four different designs. The solar cells are mounted to a printed circuit board substrate and treated as surface mount components (SMC) using the same reflow process as most printed circuit boards.

Figure 21 – Solar Array Design 1

Figure 22 – Solar Array Design 2

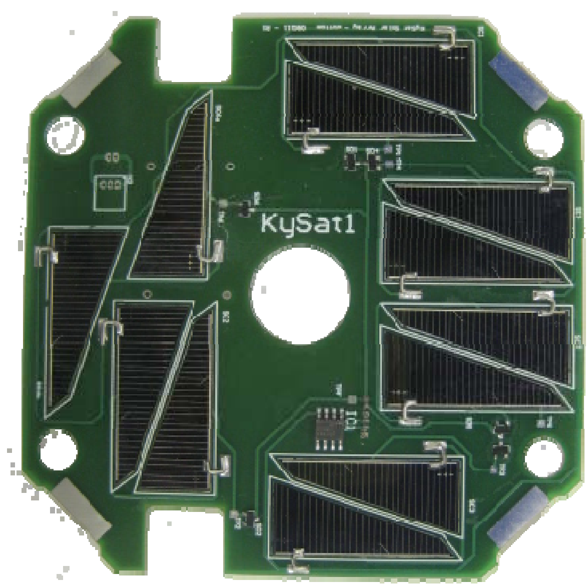

Figure 23 – Solar Array Design 3

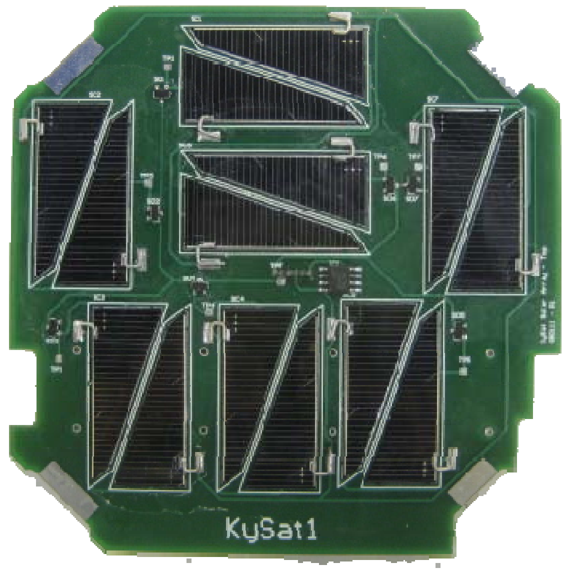

Figure 24 – Solar Array Design 4

Each of the four solar array boards is designed using improved triple-junction gallium arsenide. These cells, originally for terrestrial applications where space is at a premium are the same larger cells qualified for space applications. The larger cells are cut into squares from a circular wafer and these small triangle “scraps” are sold as terrestrial cells. These cells offer a twenty seven percent efficiency, and an open circuit voltage of 2.52 volts. The major advantage using these solar cells compared to silicon cells is that they deliver greater than four times the higher voltage. This means only one of these mulit-junction cells is required to generate the same voltage as five silicon cells connected in series. In addition when compared to a typical silicon cell, these solar cells are over twice as efficient and will delivery more than twice the power for the same area. Each of the four boards have string lengths of two cells per string and each string is diode protected in case a cell fails. Using a smaller string length will provide more redundancy giving more strings in parallel. Each board incorporates a temperature sensor to which this telemetry is provided from the main power system board. All boards conform to the CubeSat Design Specification and are designed especially for the CubeSat Kit Frame Assembly from Pumpkin Inc. The reflow processes used to bond these cells provide an extremely reliable bonding scheme for the delicate cells.

2.6 Communication System Design

The communication system for the spacecraft bus is broken into two different transceivers. One transceiver operates solely in the S-Band, and the other transceiver has a very high frequency (VHF) uplink and an ultra high frequency (UHF) downlink. The S-Band transceiver uses an industrial, scientific and medical (ISM) band and is a frequency hopping spread spectrum design, while the other transceiver uses amateur HAM bands and uses a frequency modulated (FM) transmitter and receiver. Having two different transceivers serves two main purposes. The first being a highly reliable low-bandwidth minimal earth station equipment, using a space proven design, amateur HAM radio. The other being a higher bandwidth lower technology readiness level (TRL), being a bigger risk. The second objective of two transceivers is fault tolerance and reliability. If one of the radios were to fail, there is a redundant back-up radio to replace the radio which failed. The bus can also be configured for a single radio, thus giving the bus design a more modular architecture.

2.6.1 UHF/VHF Communication Transceiver

The HAM radio transceiver, shown in Figure 25, is commercially purchased from StenSat Group LLC. This board is designed around the PC/104 form factor with the CubeSat Kit Bus header for communication and power. The transmitter is a seventy centimeter wavelength, ultra high frequency, narrow band frequency modulated transmitter. The receiver is a two meter wavelength, very high frequency, narrow band frequency modulated receiver. Both the transmitter and receiver use a frequency modulated 1200 baud audible frequency shift keying (AFSK) modulation. The data is encoded using the AX.25 protocol. The transceiver operates as a full duplex mode V/U transponder, meaning it uses a VHF uplink, and a UHF downlink, and both the transmitter and receiver can operate simultaneously. The transmitter has an output power of 30 dBm. The receiver sensitivity is between -105 dBm to -110 dBm. The receiver frequency is set by a crystal, and the transmitter frequency is set by software.

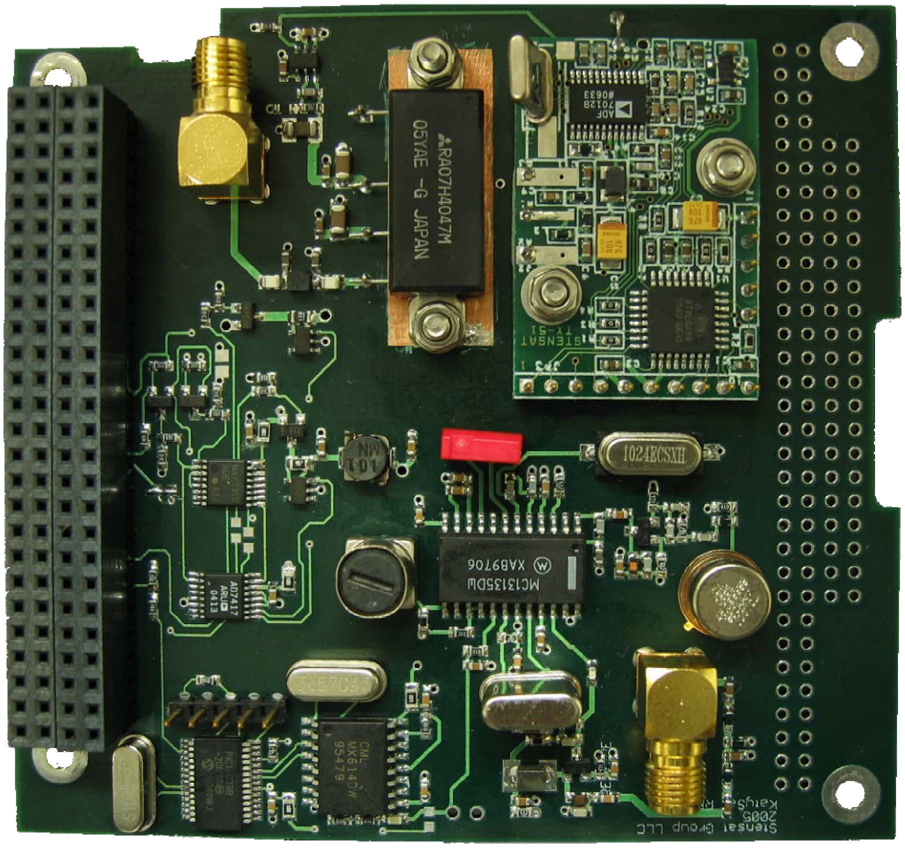

Figure 25 – UHF/VHF Transceiver

The transceiver has I^2C communication capability to provide power switching to the receiver and transmitter independently. Once the radio is commanded to power on either the transmitter or receiver over the I^2C communication there is a digital universal asynchronous receiver transmitter interface to send and receive data to and from the transceiver. The radio also has the ability to send any un-modulated audio source, and the received demodulated audio is also available on the CubeSat Bus header to the main flight computer and anything else connected to the bus. Using the AX.25 data protocol only text, punctuation, and numerical American Standard Code for Information Interchange (ASCII) characters are allowed to be sent. This is a common type of amateur HAM radio transceiver for most Orbiting Satellite Carrying Amateur Radio (OSCAR) satellites.

2.6.2 S-Band Communication Transceiver

The second transceiver, shown in Figure 26, is a much more advanced, higher bandwidth transceiver. This radio is commercially purchased from Microhard Systems Inc. Because the previous UHF/VHF transceiver operates in the amateur HAM band any data communicated must be for commanding the spacecraft and can't be used for commercial sale. If the current spacecraft bus design needed to transfer data from a proprietary payload for sale, this couldn't be done over the amateur HAM bands. Because the S-Band radio operates in the industrial scientific medical band, this proprietary data for sale can be transferred over the public license-exempt band of the radio spectrum. This communication can also happen at a much higher bandwidth than the VHF/UHF radio. This transceiver is not designed around the PC/104 form factor; however the FM430 Flight Module has a socket designed for the Mirohard MHX series radios.

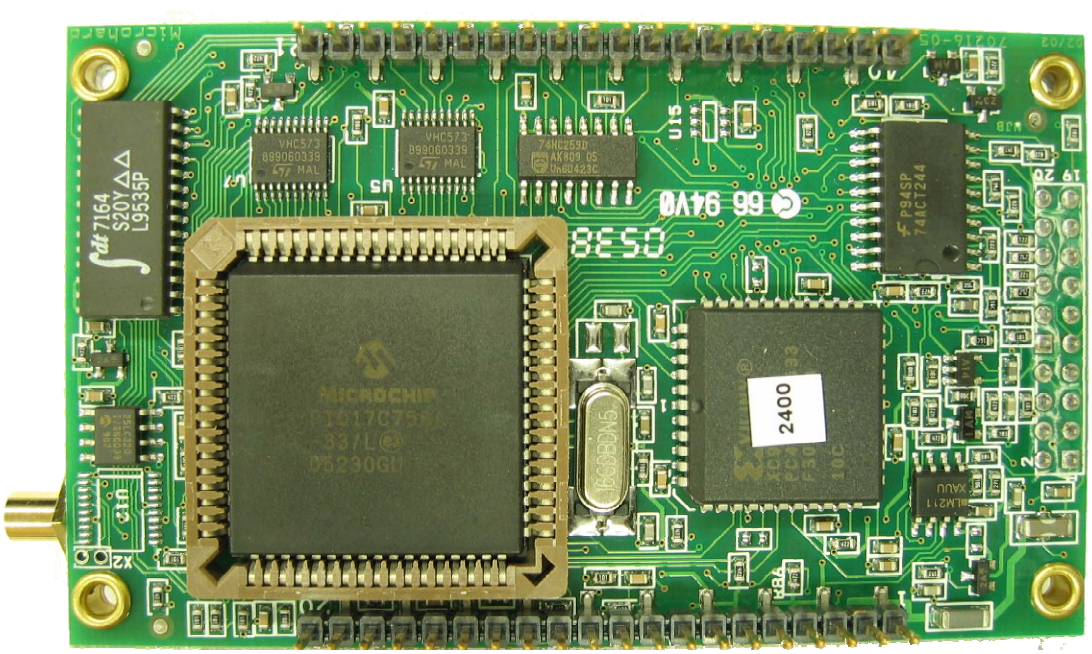

Figure 26 – S-Band Communications Transceiver

The S-Band radio is interfaced to the flight computer from the socket on the FM430 Flight Module; through this interface the radio communicates using a universal asynchronous receiver transmitter. The device also has the provisions and handshaking to implement hardware flow control, thus providing maximum throughput. This is a five volt device, but the FM430 Flight module includes the five volt to three point three volt level shifting necessary for communicating with this module. The transceiver has forty nine sets of user-selectable pseudo-random hopping patterns, intelligently designed to offer the possibility of separately operating multiple networks while providing security, reliability and high tolerance to interference. For proprietary data, the radio has an encryption key with 65,536 user-selectable values to maximize security and privacy of communications. The radio includes built in cyclic redundancy check (CRC) error detection and auto re-transmit to provide 100 percent accuracy and reliability of data. The typical mode of operation is point-to-point communications, but with multiple radios the module has the ability to be configured to create a network of various topologies including point-to-multipoint and repeater operation. Multiple independent networks can operate concurrently, so it is possible for unrelated communications to take place in the same or a nearby area without sacrificing privacy or reliability. Therefore the spacecraft bus design can act as a cluster or in constellations of many spacecraft using this radio.

2.7 Spacecraft Software Design

The spacecraft software was designed to be a modular and reusable architecture. The spacecraft bus software or satellite kernel was broken into three main sections. The software uses a commercially purchased real time operating system (RTOS) purchased from Pumpkin Inc., a file allocation table (FAT) 16 file system, purchased from HCC-Embedded, used to handle any data collected from the spacecraft payload, and the custom software written to complete the kernel [11].

2.7.1 Commercially Purchased Software

Salvo, the real time operating system purchased from Pumpkin Inc., is a fully developed co-operative priority-based multitasking real time operating system. It was originally written in assembly language and targeted to the Microchip PIC17 family of microcontrollers, since the original implementation it has been rewritten in C, providing many more configuration options and optimizations as well being completely portable.

Salvo is written to use a small flash and read only memory footprint. The operating system is about as big as a standard library `printf()` implementation. The operating system is a stack-less implementation with multitasking, priorities, events, and system timer. Salvo was written with microcontrollers with severely limited resources in mind, and typically requires as little as one fifth of the memory as other commercial real time operating systems.

The file allocation table 16 file system purchased from HCC-Embedded is a full-featured FAT file system targeted at embedded devices with limited resources available to them. The file system allows embedded systems the ability to attach personal computer (PC) compatible media to their microprocessor. The file system was highly optimized for both speed and memory footprint to allow developers to extract the most out of their system for the minimum effort. The file system can be built with many different options to easily make trade-offs between system requirements and the available resources such as flash and read only memory and overall system performance.

2.7.2 Custom Designed Software

Completing the spacecraft kernel is the custom software developed for KySat-1. Figure 27 shows a control flow diagram depicting general data flow in the system where the black rectangles represent hardware external to the microcontroller. This hardware is interfaced with the microcontroller using software drivers, represented by the green rectangles in the figure. The operating system tasks, shown in blue, pull data from the drivers, and initiates inter-task communication.

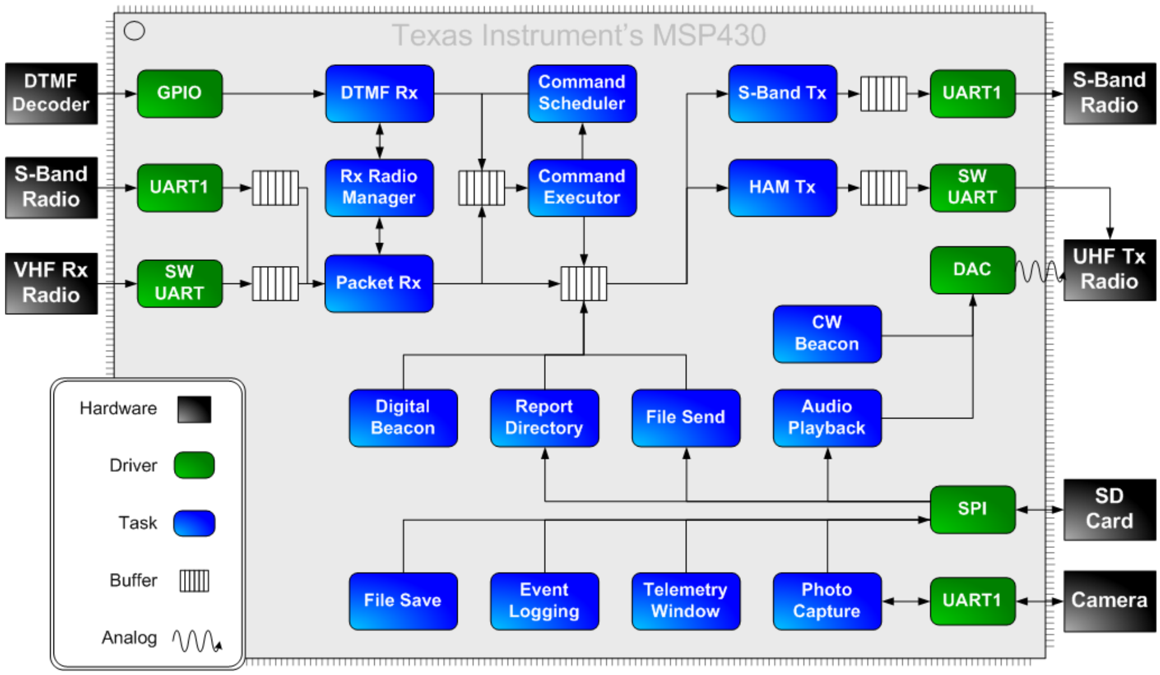

Figure 27 – Software Control Flow Diagram

Data enters the spacecraft through one of three channels, seen on the left side of the diagram, they include: the VHF receiver, S-Band receiver, or the DTMF decoder. To handle incoming digital data, a packet receiving task (Packet Rx) pulls packets from UART1 and the software UART drivers. To check for incoming DTMF tones, a DTMF receiving (DTMF Rx) task polls a driver, which reads General Purpose Input/Output (GPIO) pins. A VHF radio power manager (Rx Radio Manager) task is used to stop and start the DTMF and packet receiving tasks, because they should only run when the radio is powered. The receiving tasks alert the radio manager when a packet is received, or a DTMF tone is received, so that the receiver will remain powered during communication. After data is received either from the DTMF Rx task or the Packet Rx task, it is added to a queue where it will be parsed and executed. The Command Executor task is at the center of all software control. It is responsible for parsing data, and taking any appropriate action. This task can signal one of many other tasks, including: Audio Playback, Photograph Capture, Digital Beacon, Telemetry Window, Directory Report, File Save, File Send, Command Scheduler, and more. The executor also sends acknowledge packets back to the satellite operator. After parsing the data, if it is found to be a scheduled command, the executor passes the data onto the command scheduler, where it is held until the requested time. At the appropriate time the scheduler adds the command data back to the command executor's queue, where it be will executed immediately.

There are several tasks which are not part of the primary data path in the software. Tasks such as the Digital Beacon, Continuous Wave Beacon (CW Beacon), and Event Logging, run continuously, performing their specific task. Tasks such as Report Directory, File Send, File Save, Telemetry Window, Photograph Capture, and Audio Playback, run only when they have been commanded through the executor task. To send data out of the satellite, there are three channels: digital S-Band packets, digital UHF packets, and analog UHF audio. The S-Band Transmit (S-Band Tx), and UHF Transmit (HAM Tx) tasks are responsible for sending digital packets. These two tasks pull from one transmit queue. Depending on the current transmit radio, one of the tasks will start when a packet is added to the queue, and it will send all packets until the queue is empty, or the current transmit radio is switched. To send UHF analog audio, the CW Beacon or Audio Playback tasks place digital samples onto the Digital to Analog Converter (DAC), which is connected to the UHF radio.

3 INTEGRATION

The first spacecraft that used the previously described design was the KySat-1 spacecraft. This spacecraft, developed by the Kentucky Space consortium is manifested for a late 2009 to early 2010 launch. For this spacecraft there were two primary payloads. The first payload being a low resolution imager and the second being a high bandwidth communication transceiver, previously described as part of the spacecraft bus. This mission will be to transition the transceiver from being simply a payload to part of the spacecraft bus design itself. The KySat-1 spacecraft is designed with an attractive concept of operations as an opportunity for outreach to younger students. The students will have the ability to command the spacecraft using hand held radios and antennas to take photographs, play audio files on the HAM radio, and use telemetry collected from the spacecraft to get a hands on approach to learning about the space environment.

The integration of the spacecraft was split into two large portions. The first major hurdle was to ensure the spacecraft would mechanically fit together. All of the spacecraft bus components and payload need to fit and be securely fastened either to the inside of the spacecraft frame or on the outside of the frame assembly. The second component of integration was the electrical system. For most of the commercially purchased items, this was the first time they were integrated into spacecraft. As with any complicated design there are inevitable problems. This chapter of the thesis describes in detail the two major portions of integrating commercially off the shelf hardware with custom hardware both electrically and mechanically.

3.1 Mechanical Integration

For integrating these small spacecraft there are two different methods that can be used. One option consists of designing three dimensional models for every component of the spacecraft, putting the spacecraft together virtually to ensure everything fits and there are no clearance issues. The second approach involves physically building prototypes to ensure everything fits. Due to the lack of students with experience and knowledge of any three dimensional modeling software, most of the spacecraft was integrated using the build and fit methodology. This design and fit method caused many different prototypes and the final design took many different revisions and modifications to complete.

While integrating the spacecraft all fasteners and connectors were staked down with Scotch-Weld from 3M [11]. This material is for the assembly of sophisticated electronics where outgassing and corrosion of adhesive bonds are a concern. Scotch-Weld electronic grade (EG) epoxies are the advanced alternative to mechanical fasteners and lower-grade adhesives. This two part epoxy produces far lower contamination levels of ionic and outgassing impurities than typical epoxy adhesives.

The spacecraft solar arrays were the most mechanically complicated printed circuit board design. These printed circuit boards had complex cut outs and went through multiple iterations before they mechanically fit on the spacecraft. Because the spacecraft's antennas must wrap around the exterior of the spacecraft, the solar boards had to have the correct cut outs to allow the antennas to pass around the spacecraft frame. The solar panel on the face of the spacecraft with the access port was the most complicated design. This solar panel had to take into account both the exterior antenna mount and the spacecraft access port. These boards were designed using paper prototypes so that changes could be made based on different problems. This process was repeated until the design mechanically fit on the exterior of the spacecraft. If these boards were first designed in a three dimensional modeling software package, they could be fitted to the spacecraft frame and minimize the board re-spin process undertaken many times using the prototype approach.

Initially the system support module was a full size PC/104 form factor printed circuit board. When integrating the spacecraft subsystems it was discovered that the board stack was too tall and would not fit into the frame. There was a clearance issue between the mounting of the imager and the top board in the stack. This top board was split into two different printed circuit boards. All of the bus specific sub-circuits were redesigned onto a much smaller board that would be fitted between the CubeSat Kit Bus header and the S-

Band radio as shown in Figure 28. The remaining sub-circuits were moved to a different printed circuit that took on a U shape to account for the clearance issue. This new printed circuit board, the payload interface module (PIM) contained all the sub-circuits previously on the system support module that were mainly used to control the imaging payload.

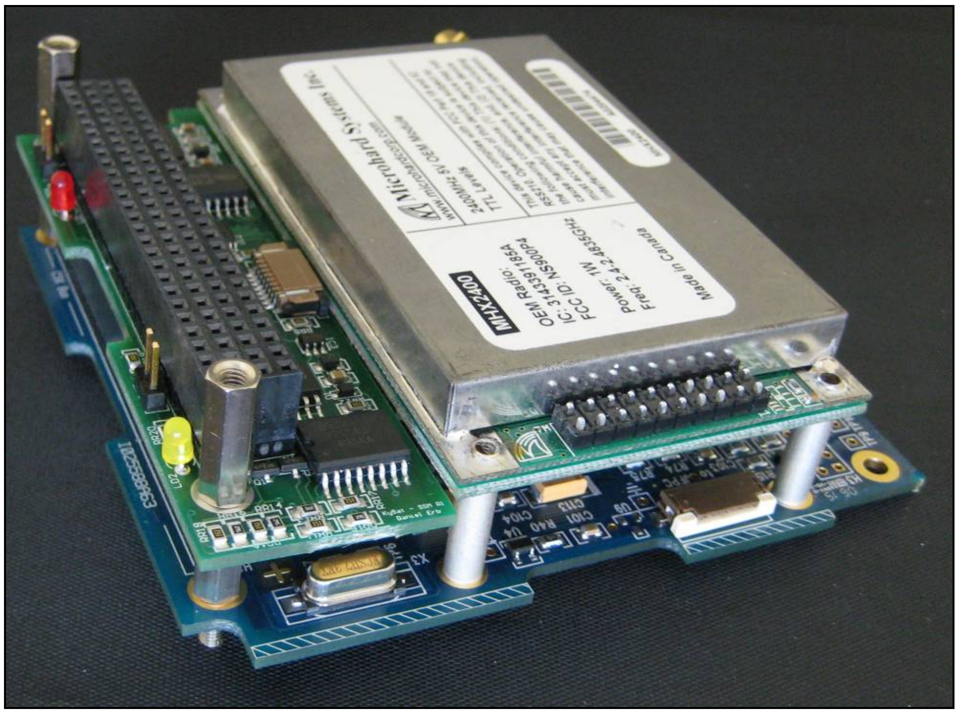

Figure 28 – FM430, System Support Module, and S-Band Radio

3.2 Electrical Integration

The electrical integration of the spacecraft was one of the larger challenges encountered during the design and implementation. With integrating custom designed hardware with off the shelf hardware there were problems that had to be addressed and solved. The largest integration effort went into the integration of the Clyde Space CubeSat electrical power system. KySat-1 was the first spacecraft bus design to use this electrical power system resulting in many integration challenges. With all off the shelf sub-systems there was some integration effort. Given the nature of satellite design, these off the shelf sub-systems are still custom hardware, just not custom hardware designed by Kentucky

Space. This off the shelf concept with respect to CubeSat sub-systems is just starting to catch on, so there are still significant necessary integration steps involved. The most mature design was the S-Band radio, which therefore took the shortest time to integrate. The rest of this chapter describes the different problem encountered electrically integration the spacecraft sub-systems. The major problems encountered consisted of the FM430 SD card interface, the I^2C communication and SPI communication isolation circuit, Clyde Space's power system design problems, and StenSat's UHF/VHF radio design problems.

3.2.1 Flight Module FM430 Integration

With the Flight Module FM430 from Pumpkin, having already flown by Libertad-1 and Delfi-C3, integration effort of this component was minimal. However some features of the FM430 were not used on the two previous missions. The two main features not used by the other developers were the SD card and I^2C communication functionalities being used simultaneously.

The FM430 has a socket to accept SD card, however on the first revision of this board the card socket was laid out with the pins reversed. This minor mistake, by the Pumpkin layout engineering, had to be corrected to be integrated into the KySat-1 spacecraft. The other issue that came up during integration and implementation was the inter-integration circuit and serial peripheral communication isolation circuit. The CubeSat Kit was originally designed for the Texas Instruments MSP430 when there were no hardware I^2C communication peripherals in the product line. As the MSP430 expanded the higher end processors in the family started to include this hardware peripheral. Because the SPI communication and the I^2C communication occurred on the same physical pins an isolation circuit must be implemented. This circuit, shown in Figure 29 is based on the Philips I^2C communication specification document. This circuit provides a means of isolating the I^2C bus of the flight computer from each CubeSat Kit module. This circuit is designed to work with master-mode and slave-mode devices on the local device side.

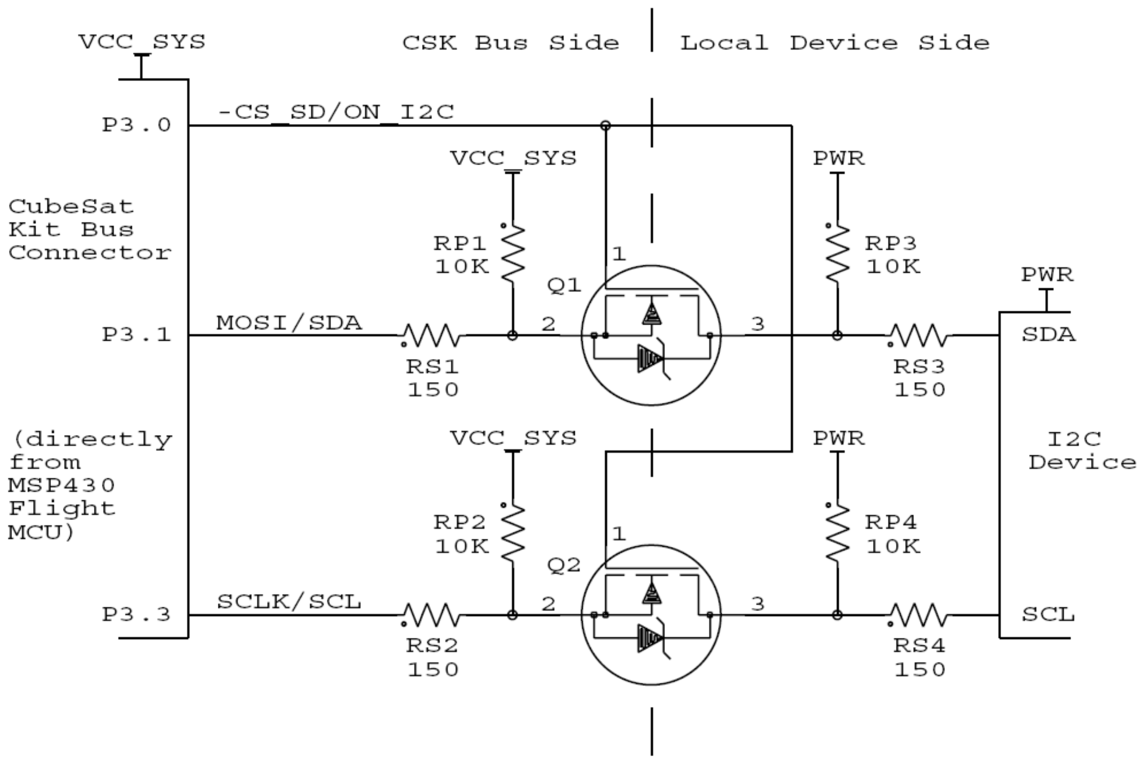

Figure 29 – I2C/SPI Isolation Circuit

The purpose of this circuit is to isolate I^2C communication from SPI communication. Devices using SPI communication have an active low chip select (CS) line which means the chip select line is driven low when the device is active. To talk to an I^2C communication device you must address the device first. Therefore any SPI devices can listen to the I^2C communication because the chip select line is set high and the device will not respond or take action to this communication. However the reverse is not true. When communicating with a serial peripheral device there could possibly be the case where the communication looks like I^2C communication and the device responds or acts upon this communication causing a fault on the bus. Every module attached to the CubeSat Kit Bus header that uses I^2C communication would require a copy of this circuit on each module. Multiple instances of these circuits in parallel on the bus were seen to cause problems with the I^2C communication, the large amount of series resistance caused very slow rise times on the serial data and serial clock lines, causing the master device to run at a much slower clock speed than normally expected on the bus.

Because this circuit didn't isolate bidirectional, there was a problem once the SD card was turned off. After turning the SD card off, the input and outputs didn't go to high impedance, as expected; rather the pins were actually pulled low. Because this isolation circuit doesn't isolate both ways, once the SD card is turned off, it will pull the serial data line low. To account for the problem the SD needs to always remain powered. This solution does increase power consumption, but while sitting idle the SD card was found to draw less than a single milliamp of current.

3.2.2 Clyde Space Electrical Power System Integration

With KySat-1 being the first spacecraft to use the Clyde Space electrical power system, there were many integration and schedule challenges along the way. The power system was the lengthiest integration process of any of the sub-systems being commercially purchased or custom designed. Because this power system was a complicated design, the schedule for delivery was perpetually falling behind. Once the power system arrived, there were both documentation and design problems that required immediate attention.

During the first mechanical integration of the spacecraft the power system design was not finished. To allow this mechanical integration to still take place a mechanical model of the power system was built by Clyde Space to ensure it would fit within the spacecraft. This mechanical fit model, shown in Figure 30, used blank printed circuit boards cut to the shape of the populated boards. Most components' masses were accounted for with bolts to ensure the mass of the model was accurate. All connectors were also populated to ensure these would be able to be integrated and not cause clearance issues. The mechanical model was fitted with right angle (R/A) connectors around the perimeter to interface with the six solar arrays. It was found that these needed to be straight connectors instead of right angle to ease in the integration.

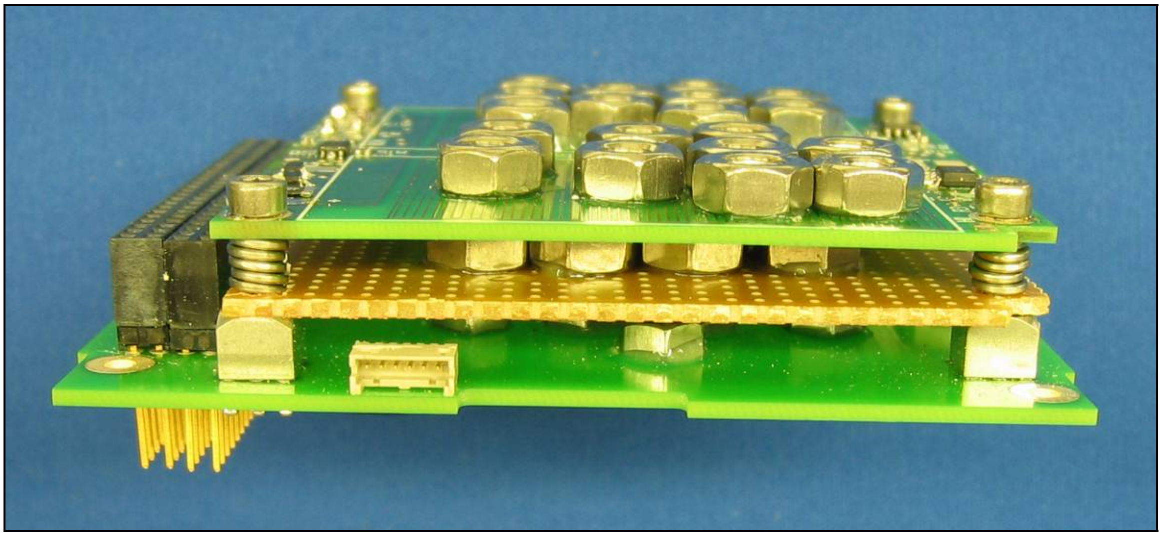

Figure 30 – Electrical Power System Mechanical Fitment Model

Once the first functional iteration of the power system was shipped to Kentucky two major design problems were encountered. The CubeSat Kit Bus is made up of two different fifty two pin connectors placed next to each other. On the first iteration of the electrical power system the printed circuit board layout had these two headers reversed, including the serial data line and serial clock line used for I^2C communication used to command and report telemetry was reversed. Both of these problems could have been avoided if proper documentation was passed along to the Kentucky Space team to review before the design continued. The power system needed to be shipped back to Clyde Space to fix the layout of the board and a new board was shipped.

Once the second functional iteration that could be plugged into the current KySat-1 CubeSat Kit Bus was received, more problems were discovered. The charging and discharging of lithium polymer batteries needs to be monitored closely and done correctly or the battery cells can become damaged and will not take on a charge. The charge limits and voltage cut off point had to be adjusted to ensure the charging cycle would not damage the lithium polymer cells. The first design of the battery board had problems with the charging method and the physical layout and connection of the batteries, when they were charged, they were damaged. Clyde Space needed to perform multiple iterations of the battery board layout to ensure the batteries would not be damaged during the charging of the cells. One of the versions of the battery board that was able to charge with out damaging cells used a cell balancing and over current protection circuit. This circuit always drew a minimal amount of current even when the battery board wasn't connected to the main power system board. This minimal current draw would over discharge the lithium polymer cells and the cells would become damaged. The final battery board design used a redesigned circuit that didn't have the constant minimal current draw.

Even after the battery daughter boards were able to be charged and discharged with the main power system board without being damaged, the Kentucky Space team continued to find problems with the main power system board that needed to be fixed. The main power system board has an active independent over current protection on the battery bus, three point three volt, and five volt rails. There was initially a design error that didn't allow ample current to be drawn from the five volt bus voltage rail. The power system needed to be sent back to Clyde Space for the circuit to be fitted with the correct components to allow the required current to be drawn. Once this circuit was fixed it was found that this circuit would react too quickly to large in rush currents. The Flight Module FM430 from Pumpkin has large tantalum capacitors used as bulk capacitors for different sub-circuits on the module. These capacitors by design have very low equivalent series resistance (ESR) causing large in rush currents. The power system board once again had to be shipped to Clyde Space for the over protection circuit to be fixed.

The final step of integrating the power system into the bus was the software used to communicate with the on board telemetry and command module of the power system. The documentation given to the Kentucky Space team for communicating with the power system was inaccurate causing for much collaboration between the Kentucky Space and Clyde Space engineering teams. There were endianess, indices, and improper command documentation problems. With proper and clear documentation this effort could have happened much faster and much more smoothly.

3.2.3 VHF/UHF Radio Integration

The StenSat Group LLC amateur band radio, having been already flown on Libertad-1, was more readily integrated by the Kentucky Space team. However because Libertad-1 had a relatively simple mission of just having a radio beacon, the major functionality of the transceiver was not fully tested. With KySat-1's concept of operations, the spacecraft and mission used the full functionality of the transceiver.

During the first mechanical integration of the spacecraft the transceivers SubMiniature version A (SMA) connectors caused interference with the spacecraft chassis wall frame. For the mechanical integration to be completed the SMA connectors needed to be rotated to fit the ends of the coaxial cable connecting to the antennas. There were also other taller components on the board that need to be replaced with some shorter versions to mitigate a clearance issue between the radio and the imaging payload.

Most electrical issues regarding the VHF/UHF radio were relatively simple to solve. To fulfill KySat-1's concept of operation, the HAM radio needed to send analog audio. The radio board had the ability to do this, but it was not implemented on Libertad-1. When implementing this on KySat-1, the team noticed the audio coming out of the receiving radio was extremely quiet. The radio was sent back to StenSat and the external modulation circuit was changed slightly in increase the modulation level from external audio being sent to the radio.

There were two hardware interfaces that were found to pose problems. Both of these problems were accounted for in the spacecraft software. The first problem was a timing and brownout issue with the transmitter processor. If the power was cycled too quickly on the transmitter when the transmitter was turned on it would not respond to any command and you were unable to send packets out of the transmitter. This problem was fixed by careful testing and integration with the spacecraft flight software. The second problem was a power leakage problem. When the transmitter was commanded to be turned off and the transmit pin of the universal asynchronous receiver transmitter interface was left high, as it should be, this pin would still power the processor. In addition when turning it back on, the processor would be left in an unknown state and not respond to any commands. This was also fixed in software by bringing this pin low whenever the transmitter is turned off.

Due to the fact some of the integration of the spacecraft occurred with fairly quick turn a rounds, StenSat put rush orders on the crystals that served as frequency basis for the receiver and transmitter. Because the factory did not properly age these crystals, they began to drift in frequency the more they were used, which caused the final transmit and receive frequency to drift as well. The transmit frequency was fixed by adjusting the clock multiplier in software on the transmitter processor, but the receiver frequency continued to drift until the crystal had fully aged.

The final integration issue that came up with the VHF/UHF transceiver from StenSat happened late in the integration and testing process. During most of the development and implementation, the same mobile HAM radio was used by all developers and this radio's terminal node controller (TNC) worked correctly with the radio board from StenSat. While doing final testing and integration the Kentucky Space team tested the spacecraft bus with the transceiver and TNC setup that would be used while in orbit. It was found that when most other TNCs were used, the spacecraft did not receive the packet being sent. The problem turned out to be that the transceiver receiver software for decoding AX.25 had a software bug. This required shipping the radio board back to the manufacturer for the receiver to be reprogrammed.

4 TESTING AND EXPERIMENTAL RESULTS

Thorough testing of the spacecraft is one of the most important aspects in the design life cycle to assure mission success. One of the most important aspects of this testing is to ensure you are testing like you will be flying. Testing to a space environment can be difficult due to the extreme differences in space and terrestrial environments. At the time of the writing of this thesis, testing was an on going process for the KySat-1 spacecraft engineering model. To ensure mission success, the Kentucky Space team decided to perform the testing described in this chapter. Prior to complete integration of the spacecraft the antennas were first matched and tuned. The spacecraft antennas were connected to a network analyzer and the antenna performance was measured. After the antennas were tuned and matched to the correct frequencies, the antenna radiation patterns were measured at an outdoor range. Both of these tests were performed without the internal electrical spacecraft bus components. The first large scale test that was performed on the fully integrated engineering model was the thermal environmental testing. The engineering model of the spacecraft was placed in a thermal chamber and the temperature was cycled to mimic the temperature swings the spacecraft would see being in a ninety minute orbit, going from sun to shade and vice versa. Following the thermal environmental testing the spacecraft was taken to the outdoor range once again to measure the performance of the entire communication system. In parallel to all of this testing with the integrated engineering model a duplicate set of spacecraft bus hardware was up and running to test the spacecraft flight software.

4.1 Spacecraft Antenna Testing

Prior to the completion of the integration and assembly of the engineering model the spacecraft was configured in a way such that the spacecraft antennas could be matched and tuned. The configuration consisted of the spacecraft frame assembly, antenna mounts, antenna, and solar arrays assembled together. This configuration used prototype assemblies of the spacecraft solar arrays. This allowed the spacecraft to be handled much more freely when tuning and matching the antennas. Stringent electrical static discharge

(ESD) and contamination procedures could be relaxed. The spacecraft was then mounted on a non-conducting fixture made out of fiberglass. The coaxial cable feeding the antennas was then brought out of the spacecraft through the same hole in which the imaging system would be mounted and was then connected to a network analyzer. This configuration and mounting is shown in Figure 31. This configuration isn't the ideal situation without the internal electronics subsystems, but it was assumed that the frame and solar arrays produced a somewhat ideal faraday cage causing the internal electronic components to be negligible. Ideally, this testing would be done in an anechoic chamber to minimize reflections off the surrounding objects, but the Kentucky Space team did not have access to a large enough anechoic chamber to perform these measurements. The purpose of the testing was to measure the overall performance of the antennas to ensure the link could be closed while in orbit.

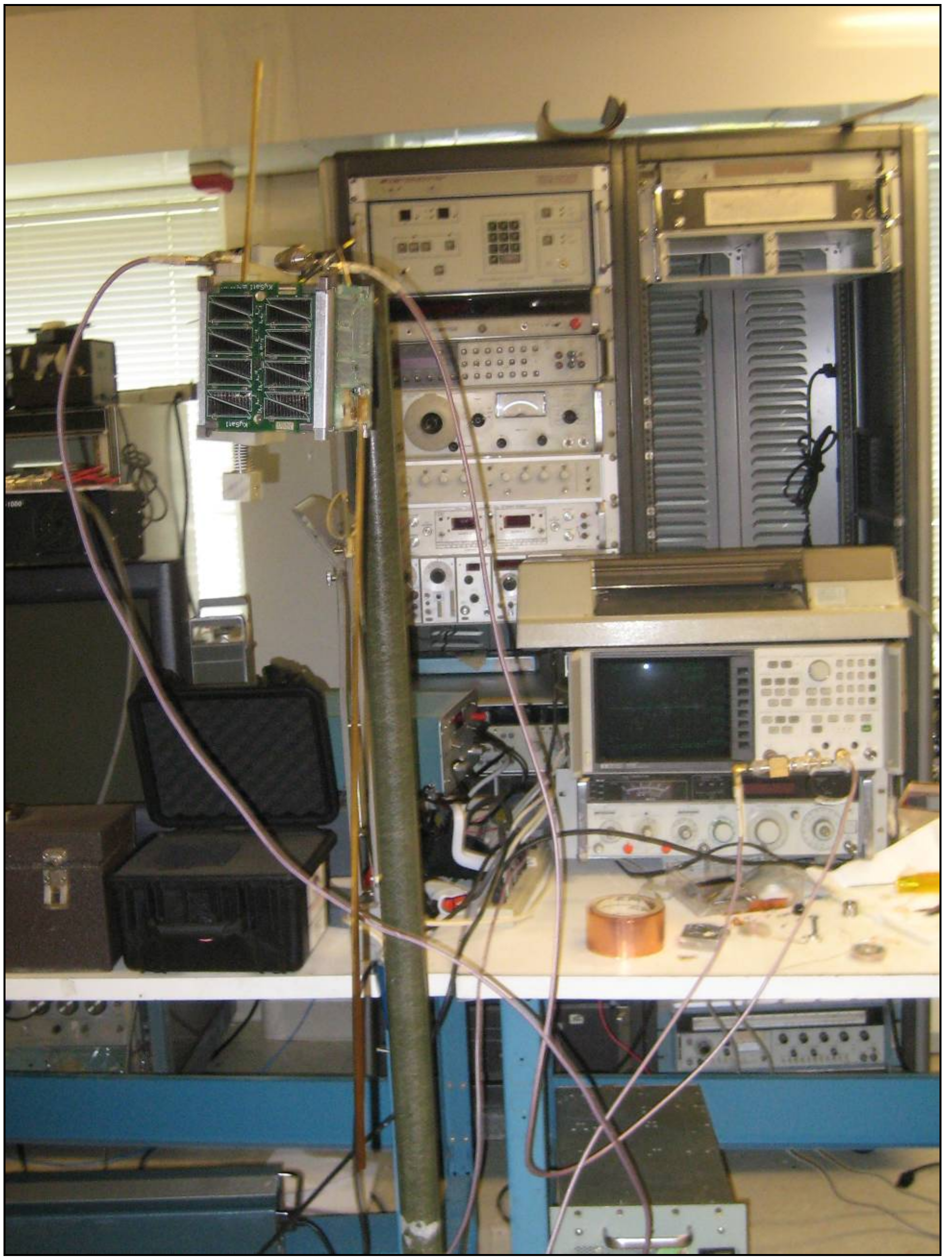

Figure 31 – KySat-1 Engineering Model Configured for Antenna Matching and Tuning

4.1.1 Antenna Tuning and Matching

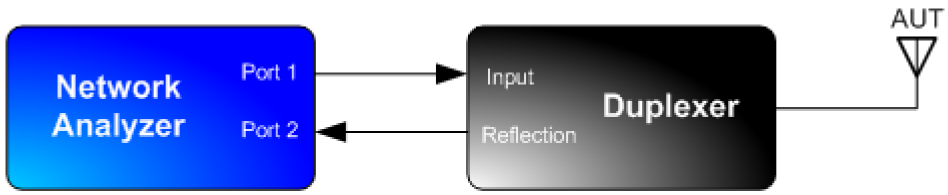

Figure 32 – Block Diagram for Antenna Matching and Tuning

Initially all three antennas were cut longer than their corresponding quarter wave length measurement. This allowed the antenna to be continually trimmed until the ideal match was made. All three antennas included a small matching circuit with series inductors and shunt capacitors. The inductor was first shorted out on all three matching circuits. The coaxial cable feed from the antenna was then fed into a directional coupler. The input and output ports of the directional coupler were then hooked up to ports one and two of a network analyzer respectively. A block diagram of this setup can be seen in Figure 32. Using the network analyzer the return loss or S21 parameter was measured. During this experimentation it was found that the UHF and S-Band antennas did not require any matching. Both antennas lengths were then trimmed to the point where the return loss was a minimum for the frequency of interest. Both the measured UHF and S-Band antenna return loss plots are shown in Figure 33 and Figure 34 respectively.

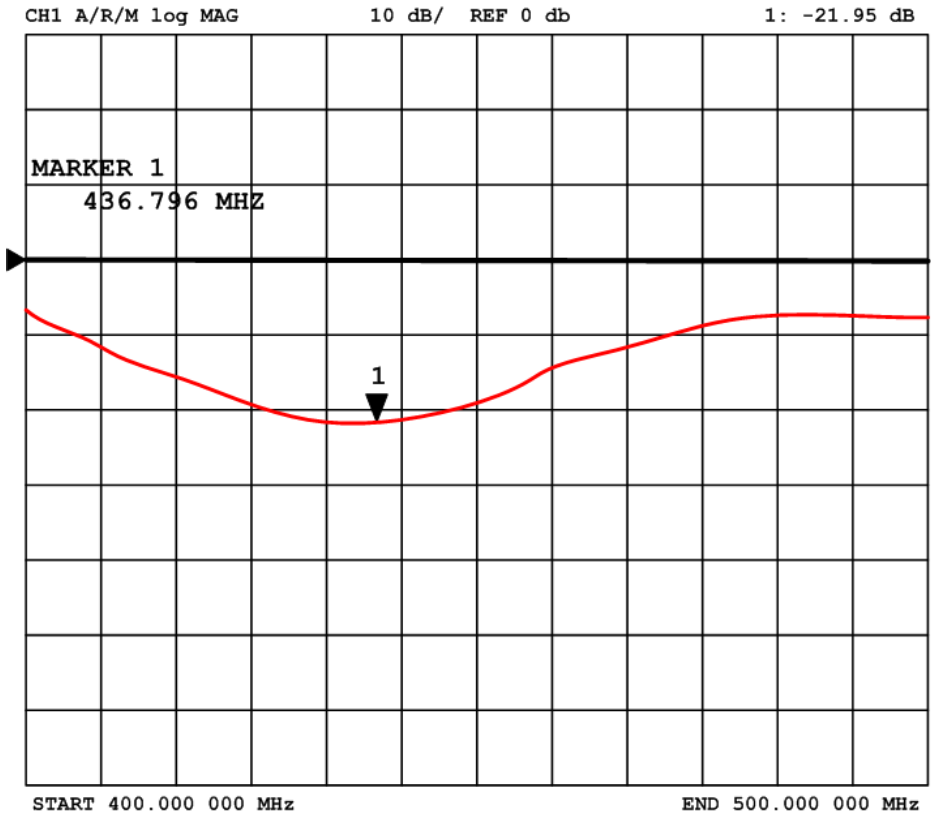

Figure 33 – Measured UHF Antenna Return Loss Plot

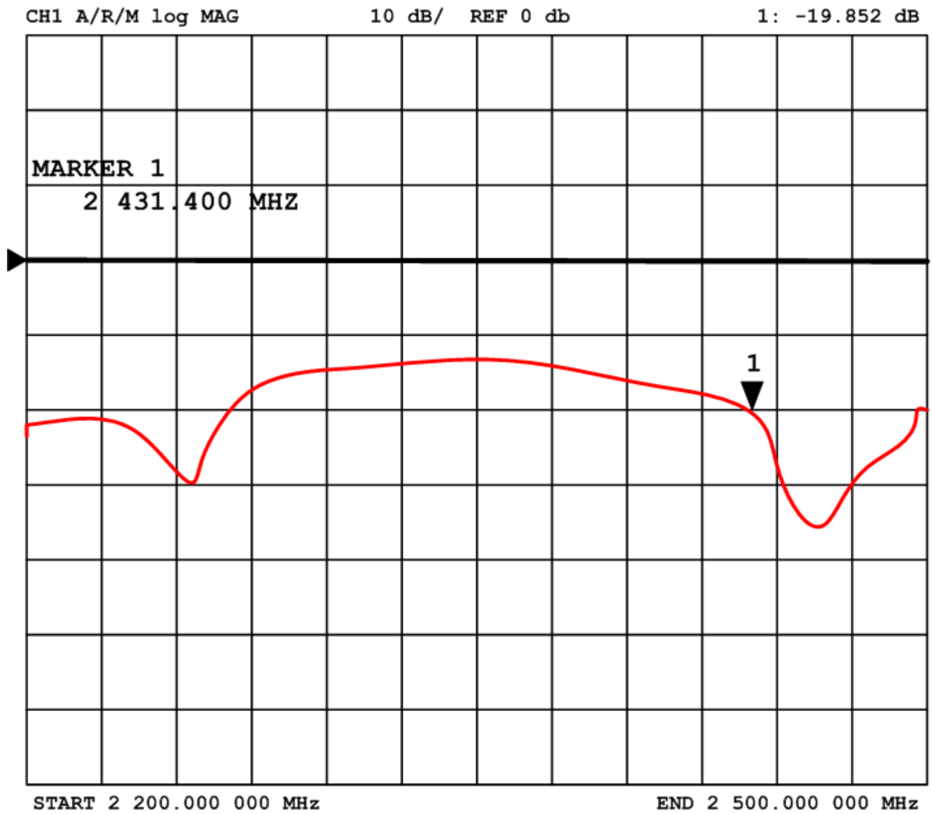

Figure 34 – Measured S-Band Antenna Return Loss Plot

During the trimming of the S-Band antenna it was found that further and further shortening of the antenna didn't shift the frequency of the minimum of the return loss plot down as expected. After further experimentation and investigation, it was decided that the length to width ratio of the S-Band antenna must be taken into account because the antenna was much shorter than the UHF or VHF antennas. To trim the antenna to the desired frequency, the antenna needed to take on a tapered shape, being wide at the base and coming to a point towards the end of the antenna. This final shape of the S-Band antenna attached to the matching circuit can be seen in Figure 35.

Figure 35 – S-Band Antenna Tapered Shape

The final antenna to be matched and tuned was the VHF antenna. After experimentation it was found that it was necessary for this antenna to have series inductance to be matched properly. Initially it was seen that with the antenna cut longer than expected to shift the minimal return loss point of the antenna down in frequency, the antenna needed to still be longer. This went against initial intuition because the antenna was already cut longer than is quarter wavelength calculation. After further experimentation it was concluded that the spacecraft body, which is used as the ground plane for the monopole antennas, is relatively small compared to the wavelength for the VHF antenna, it was necessary to make the antenna longer to account for the much smaller ground plane due to the lower frequency. This was confirmed by slightly touching the spacecraft frame, effectively adding a larger ground plane to the antenna. The final return loss plot for the VHF antenna is shown in Figure 36.

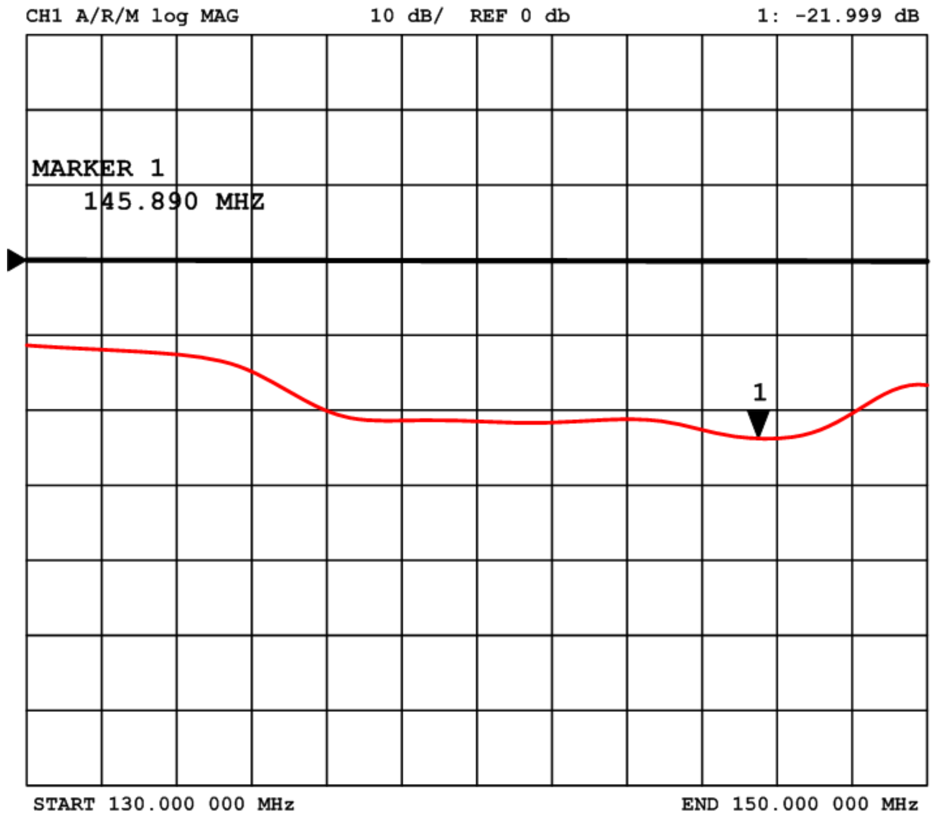

Figure 36 – Measured VHF Antenna Return Loss Plot

4.1.2 Antenna Radiation Pattern Measurement

Following the tuning and the matching of the antennas, the spacecraft in the same skeltonized configuration was taken to an outdoor range. Using the same non-conducting fiberglass mounting fixture, this assembly was then placed on a positioner. A positioner was used to rotate the spacecraft around a specified axis. The entire setup was placed on a small hill so that the receiving antenna on the ground would be looking up at the spacecraft with the sky in the background. The elevated setup helped to minimize reflections off the ground. The network analyzer and receiving antenna were then placed at the bottom of the hill, making sure it was in the far field of the antenna. The network analyzer was connected directly up to the spacecraft antenna under test and a ground receiving antenna to measure the radiation of the antenna at the selected position. The spacecraft was then rotated and another measurement was taken. The process was repeated until the spacecraft was rotated 360 degrees back to its initial starting position.

This setup with the spacecraft on the positioner on top of a small hill and a receiving UHF antenna is shown in Figure 37.

Figure 37 – Configuration used to Measure the Spacecraft Antenna Radiation Patterns

Only the vertical, or elevation, radiation pattern of the antenna was measured. The horizontal, or azimuthally, pattern of the antenna was assumed to be uniform. The spacecraft monopole antenna was placed vertical and this position is referred to ninety

degrees. Zero degrees is referred to when the antenna null, or top of the antenna, is pointing towards the receiving antenna. This all can be depicted in
Figure 38 [13].

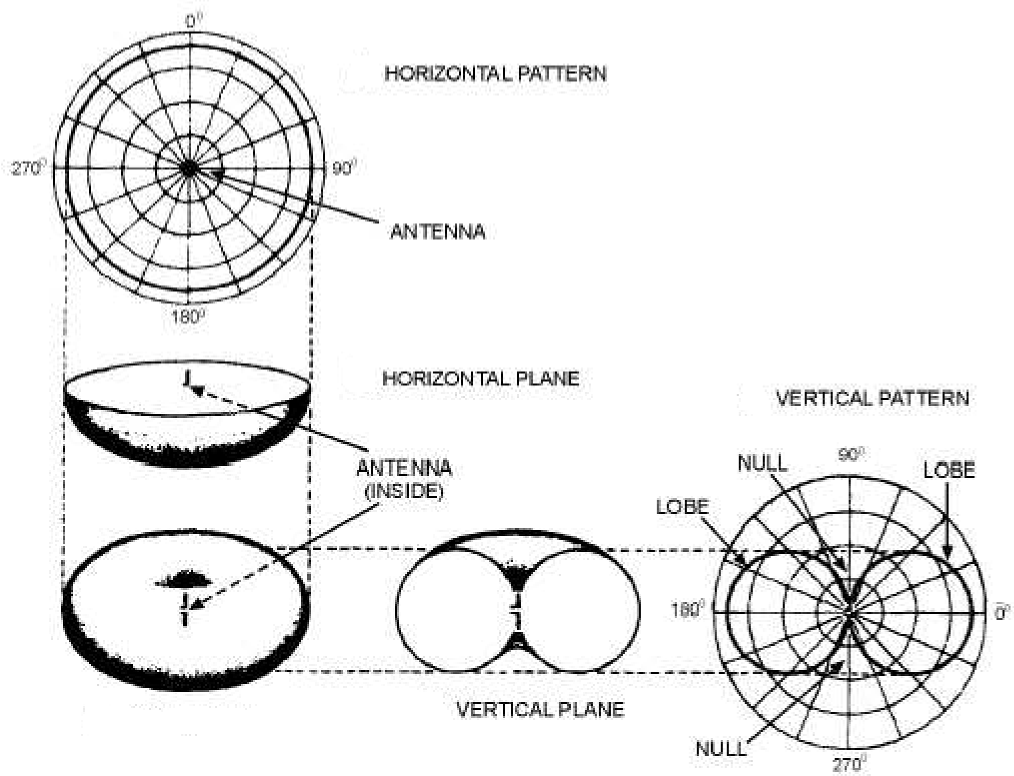

Figure 38 – Antenna Horizontal and Vertical Planes

Prior to measuring the radiation patterns of the spacecraft antennas, a reference dipole antenna pattern was measured. This reference dipole at the three different frequencies would be used as the basis for comparing the three spacecraft antennas. Using the ideal dipole antenna pattern, each measurement would take into account the receiving antenna gain and any losses due to cables and connectors used. The network analyzer was used to transmit a signal out of the spacecraft antenna. This signal was then received using the ground antenna. The received signal from the ground antenna was then fed into the network analyzer. The magnitude of these two signals was then compared to measure the radiation pattern. The following three plots shown in Figure 39, Figure 40, and Figure 41 are the measured radiation patterns of the VHF, UHF, and S-Band antenna respectively.

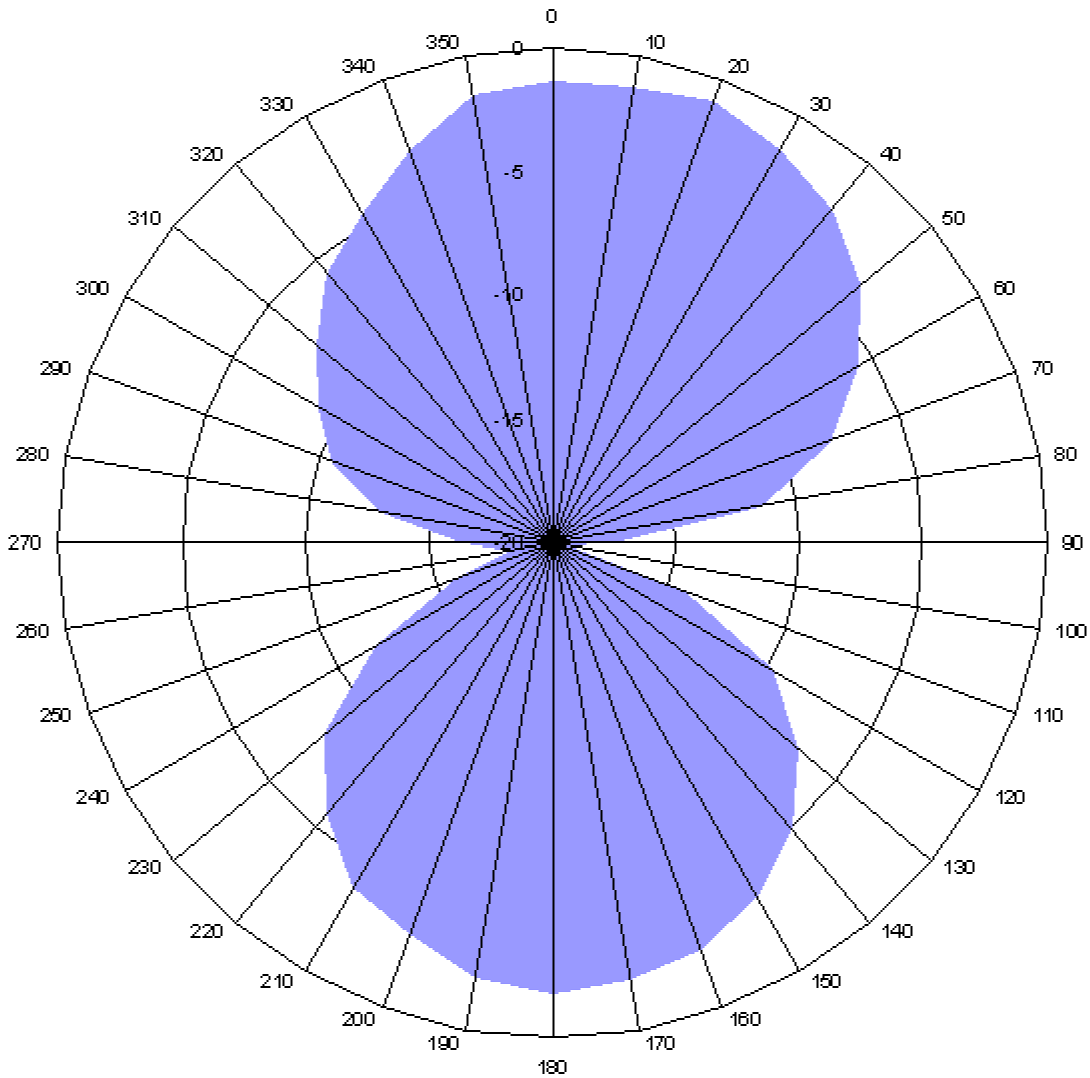

Figure 39 – VHF Antenna Normalized Radiation Pattern

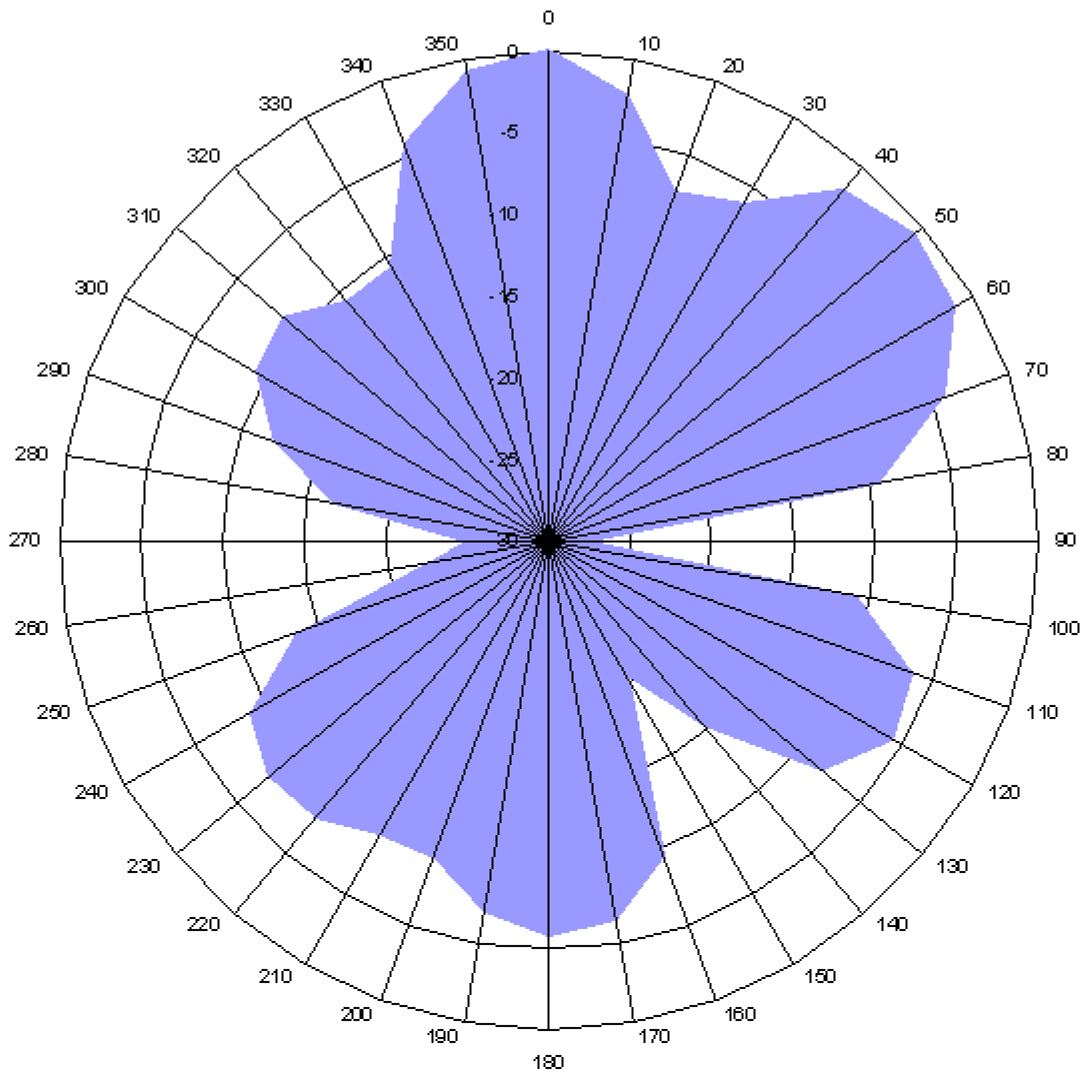

Figure 40 – UHF Antenna Normalized Radiation Pattern

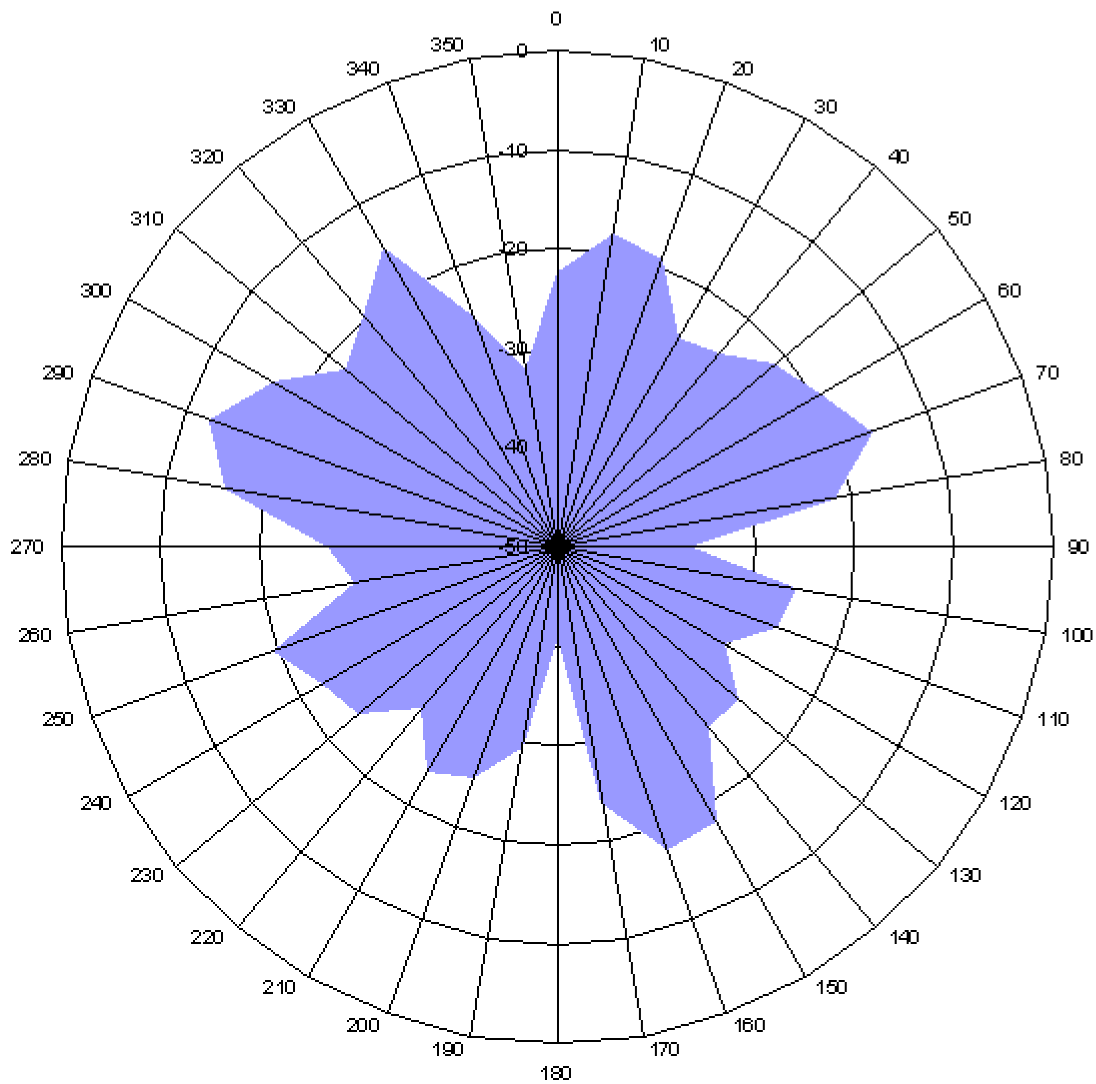

Figure 41 – S-Band Antenna Normalized Radiation Pattern

4.2 Thermal Environmental Testing

Figure 42 – KySat-1 Engineering Model inside the Thermal Chamber

After the KySat-1 engineering model was completely integrated as it would be in orbital configuration. To ensure KySat-1 will survive the thermal environment of space the engineering model was placed in a thermal chamber, as shown in Figure 42. The thermal environment of space can vary from extreme cold in eclipse to extreme hot while being illuminated by the sun. While in the thermal chamber, normal on orbit operations and command the spacecraft were performed with a ground station setup outside of the chamger. All of these operations were checked to ensure KySat-1 would perform to specification while experiencing the hot and cold of space. The most temperature sensitive device on KySat-1 is the batteries. The following list of requirements was set out prior to testing to ensure KySat-1 will have mission success.

(1) The spacecraft batteries must maintain a temperature greater than 0 °C for operation during an eclipse period.

(2) The spacecraft power system must be able to start regardless of temperature, illumination, and charge state.

(3) The spacecraft must be able to provide enough energy to the imaging payload to heat the imager above 0’C.

It would have been best to perform all these above tests under space vacuum conditions as well. At the time of this testing the thermal vacuum facilities were not capable of performing these tests. However, it can be argued that most failures will likely be due to temperature alone so testing only the thermal environment was sufficient. A ground stations with a mobile VHF/UHF transceiver, S-Band transceiver, and laptop computer shown in Figure 44, was setup outside the chamber to command and download telemetry from the spacecraft. A block diagram of this experimental setup is shown in Figure 43.

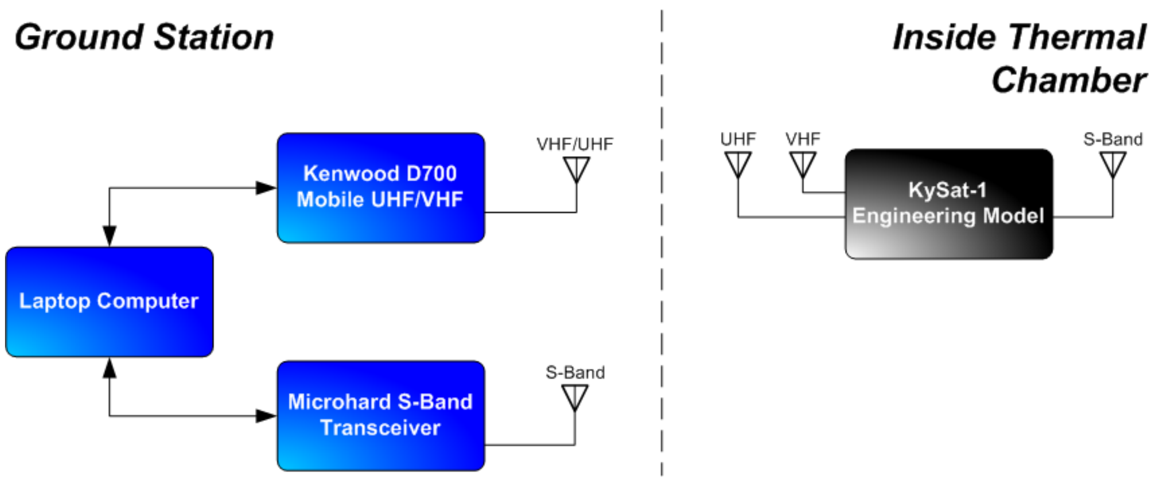

Figure 43 – Block Diagram for Thermal Environmental Testing

Figure 44 – Ground Station Setup outside the Thermal Chamber

4.2.1 Temperature Profile

The temperature profile was chosen from on orbit data provided by a current CubeSat mission, CP4 [13]. The ideal profile as shown Figure 45 is being based on a ninety minute orbit with temperature extremes being negative thirty degrees Celsius and seventy degrees Celsius. Different developer's data suggest that the temperature may not be as hot as shown in the ideal profile but in this situation it is better to over test than under test.

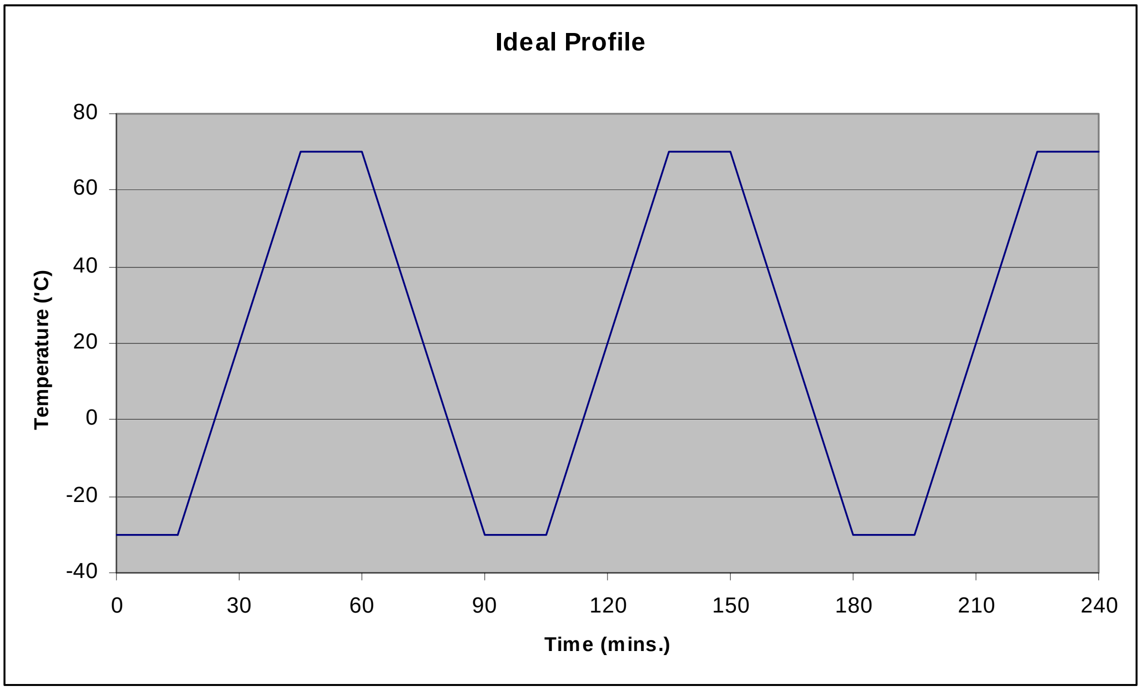

Figure 45 – Ideal Temperature Profile

However, at the time of this writing, the equipment available does not have the capability to transition the temperature of the chamber from either cold to hot or vise versa as quickly as the figure shows. To account for this, an adapted profile shown in Figure 46 was used. The motivation was to have the temperature cycling extremes but still have times where there are hot and cold soaks of adequate amount of time. From empirical testing at ramp from -20 °C to 70 °C would take approximately 45 minutes. A hot and cold soak time of 30 minutes was chosen to ensure the whole spacecraft had a chance to heat up and cool off respectively.

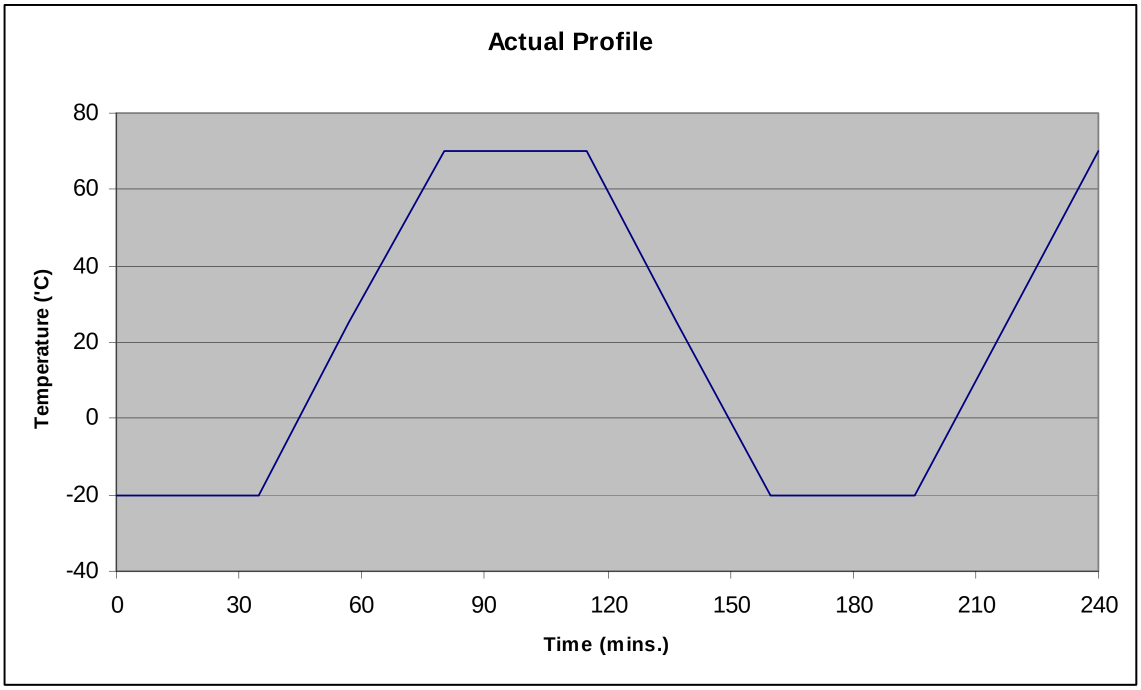

Figure 46 – Actual Profile Used

4.2.2 Battery Heater Testing

The most temperature sensitive subsystem on KySat-1 is the batteries. To charge the batteries, they must be at a temperature greater than zero degrees Celsius. If one were to charge the batteries while the temperature is below zero degrees Celsius the overall capacity would decrease. To keep the batteries warm there is an integrated thermostatically controlled resistive heater on the battery printed circuit board. This heater is designed to keep the temperature of the battery above five degrees Celsius. Shown in Figure 47 and Figure 48 is actual test data of the KySat-1 engineering model. This first thermal cycle tests were performed with USB plugged into the access port of the spacecraft to maintain and charge the batteries. Because the batteries are being charged from USB they maintain a fully charged battery voltage; however you can see small sags in the voltage when large currents are drawn due to internal resistance. The graph shows telemetry as would be taken from orbit during normal idle operation. The smaller periodic current spikes are due to the beacons being sent over the UHF radio. Once the battery temperature falls below five degrees Celsius larger more dense current spikes can be seen. These are due to the integrated heater on the battery board turn on to maintain the temperature above five degrees Celsius. There is a two degree Celsius hysteresis built into the thermostat. Therefore during the cold soak you can see the heater coming on and off to maintain the temperature of the batteries above five degrees Celsius plus or minus two degrees depending if the heater is transitioning from an on to off state or off to on state.

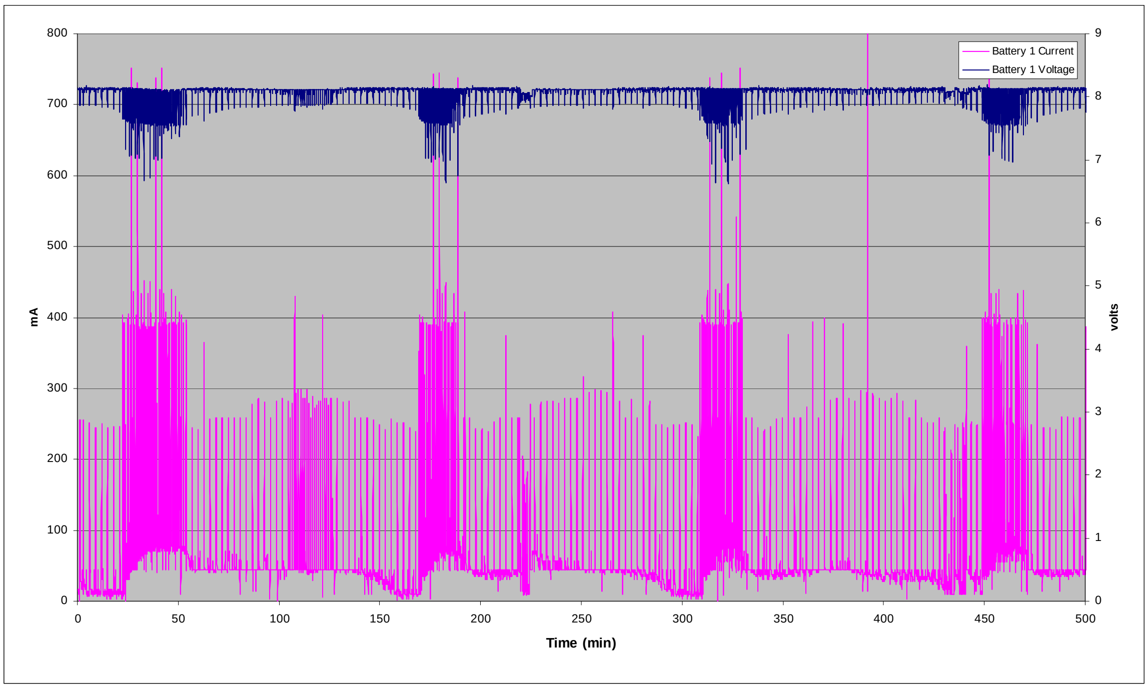

Figure 47 – Battery Voltage and Current while Charging

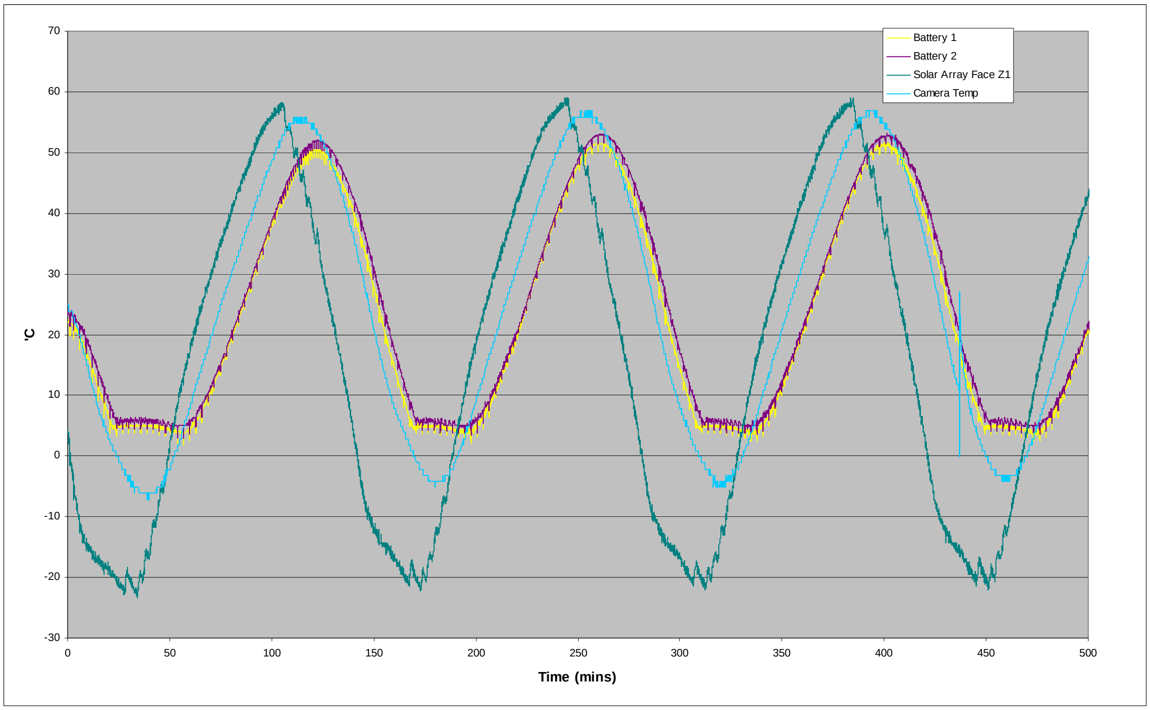

Figure 48 – Temperatures Logged while Charging

The same normal operation was also performed without external USB power being applied. This configuration would allow the team to experimentally measure the battery capacity. The discharge rate of the batteries was measured under hot and cold conditions. As you can see from Figure 49, the battery voltage starts out slightly above eight volts, a fully charged battery, and discharges to about six point five volts. The electrical power system will shut off the system around six point five volts to prevent the batteries from being over discharged. During this cycle the heater was turned on for the remaining time, so the drop off in voltage was rather steep. This is exponential decay of capacity is typical of lithium polymer chemistry batteries. One can also see a difference in total current consumption from the previous test where external USB power was applied. This increase in current is due to the current previously being drawn from the external USB power supply and batteries, and now only being drawn from the batteries.

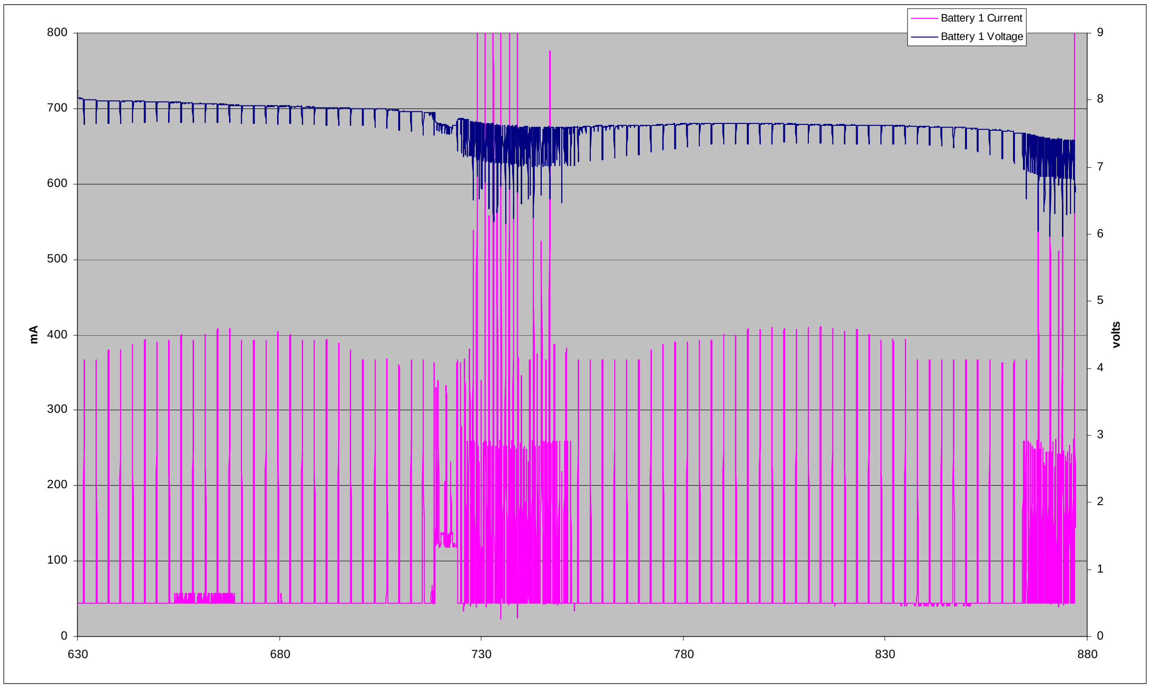

Figure 49 – Battery Voltage and Current while Discharging

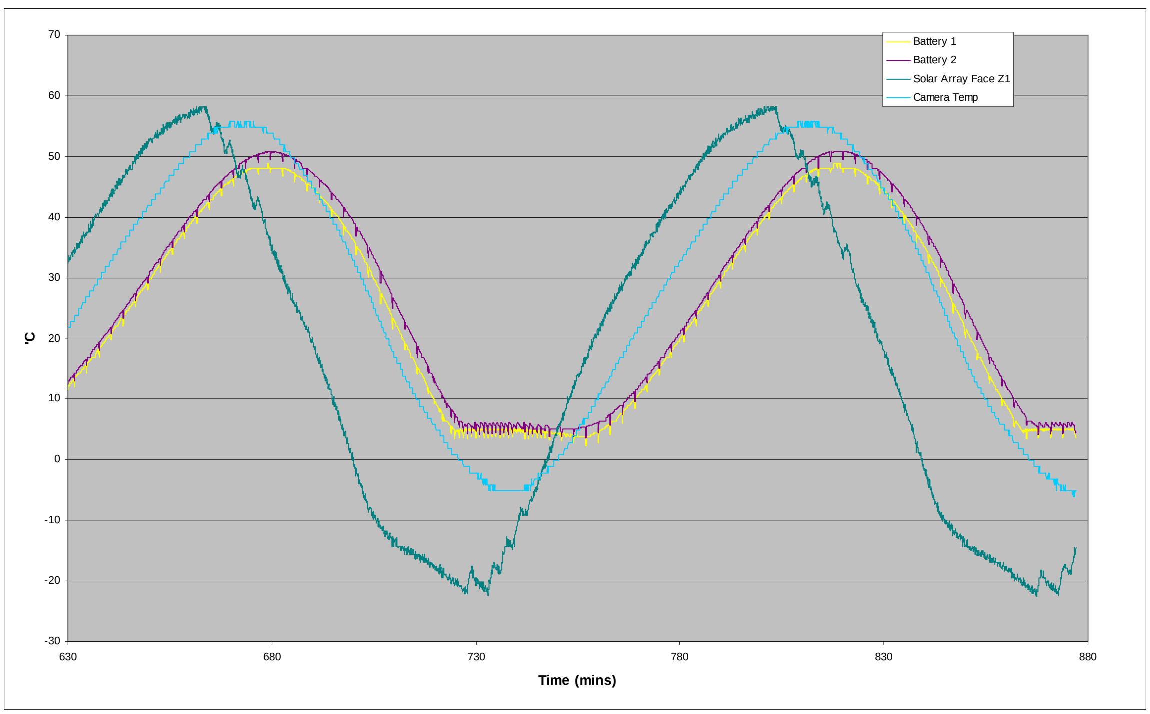

Figure 50 – Temperature Cycling while Discharging

4.2.3 Hot and Cold Power System Starts

The final electrical power system test that was performed while the engineering model was in the chamber was cold and hot starts. From previous CubeSat missions it was found that one of the most successful CubeSats, QuakeSat had problems with the power system starting while in the sun [15]. Luckily for the QuakeSat team it was deployed from the P-POD while in eclipse. For this reason KySat-1 was turned off by depressing the separation switch as it would be in the P-POD and was let to cold soak at negative twenty degrees Celsius for one hour. After an hour of cold soaking the separation switch was released allowing KySat-1 to turn on. After booting up all functionality of the power system was checked to ensure it started without any faults. This same test was done with fully charged batteries and a near turn off charge state. To ensure maximum robustness, this same matrix was also performed at the hot extreme of seventy degrees Celsius as well.

4.2.4 Imaging Payload Heater Testing

The only other temperature critical subsystem of KySat-1 is the imaging payload. According to manufactures specifications [16], the camera must be above zero degrees Celsius to take pictures. For this reason a resistive heater circuit was designed to heat the camera to a satisfactory temperature. Because the exact thermal conduction between the heater and camera is unknown the heating duty cycle and time can be reconfigured utilizing spacecraft commands. During the cold, soaks the imaging payload was turned on and commanded to take pictures to ensure this resistive heating circuit worked to specification.

4.3 Communication Systems Testing

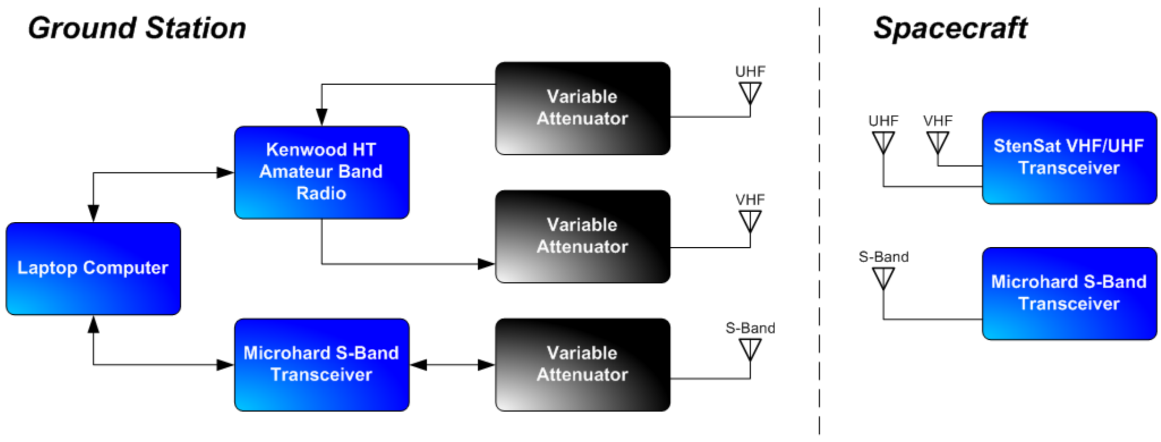

Figure 51 – Block Diagram for Communication Systems Testing

Following the environmental testing of the engineering model, the spacecraft was taken back out to the outdoor range. This time the spacecraft was configured as it was for the environmental testing, fully integrated as it would be on orbit. The spacecraft was mounted on the same non-conducting fixture as during previous testing, and then mounted on the positioner. For this test however, the fixture was then mounted on the bucket of a bucket truck. This would allow the spacecraft to be raised up to a sufficient height as to prevent reflections off the ground. Previously when measuring the antenna radiation patterns it didn't need to be this high because a much lower power signal was transmitted out of the antenna. For this experimental setup, the spacecraft radios were used to transmit. Ground station antennas were setup and hooked up to the receiving radio as would be used in orbit. Variable attenuators were placed inline with the ground antennas and the ground radios. A block diagram of this system is shown in Figure 51. The added attenuation would simulate the path loss as if the spacecraft was in orbit. A picture of this experimental setup is shown in Figure 52.

Figure 52 – KySat-1 Engineering Model Mounted to Bucket Truck

After the spacecraft was suspended forty feet in the air, the Kentucky Space team setup three receiving ground antennas 112 feet away from the bucket truck, this would give a slant range of approximately 119 feet. Using the ground station antenna and radios the spacecraft was sent ping commands. After the spacecraft is sent a ping command it will respond letting the operator know that it received the ping command. The team continued to ping the spacecraft each time increasing the attenuation between the transmitting radio and the antenna on the ground. The attenuation was increased until the spacecraft would no longer respond. Using this attenuation value, the team was able to calculate the maximum slant range before uplink to the spacecraft failed. This experimental test was run for the spacecraft when receiving a command using both an AX.25 packet and a dual tone multiple frequency tone.

The same kind of experiment was also run with the downlink. However because the spacecraft transmitter was too powerful and the slant range too close, the downlink could always be established. The antenna of the receiving ground antenna was removed completely and the spacecraft was still received using the ground station radio. This was due to the poor shielding on the ground station radio and the relative close proximity of the spacecraft and the ground station radio. Because the team was unable to move the spacecraft further away, a spectrum analyzer was hooked up to the receiving antenna with the inline attenuation. The received power could then be measured with the spectrum analyzer. From the received power the team could then calculate the equivalent isotropically radiated power (EIRP). Using the equivalent isotropically radiated power and the receiving ground station antenna gain while taking account any cable loss; the maximum slant range could then be calculated for the downlink.

The final radio on the spacecraft, the S-Band transceiver, was also tested using the same method. Because the S-Band radio has the ability to change the output power while in orbit this made this testing possible. However because the S-Band radio both on the spacecraft and the ground station used the same antenna to transmit and receive, splitting the uplink and downlink apart was not possible unless there were two receiving S-Band radios and antennas on the ground, one with inline attenuation and the other without. The spacecraft was commanded and the attenuation was increased until the link was not able to be closed. Using this number the maximum slant range was able to be calculated for the S-Band system as well. All communications systems onboard the spacecraft and the ground station performed extremely well. This gave the Kentucky Space team confidence that the link could be closed while in orbit.

4.4 Spacecraft Flight Software Testing

Software is the hardest system to verify. It is unlike any other system. The number of possible inputs can exceed the time allocated to test the software. For this reason the software testing of the flight software happened in parallel with all other spacecraft testing utilizing a number of strategies to ensure there would not be a software bug while on orbit. Once development was completed, all code was reviewed and white box testing was completed. Following white box testing, a full suite of static analysis was completed on the entire code base. Following the static analysis, a duplicate set of hardware for KySat-1 was constructed and the team undertook black box testing the spacecraft flight software prior to being launched into orbit.

Static analysis using lint was used to further verify the software by scanning for lint compliance. Every file in the code base was scanned before it was allowed to be used for flight. Static analysis revealed several coding mistakes. It also helped to improve code reliability by alerting programmers of unhandled cases. The team used LintPlus from Cleanscape to perform PC Lint compliance of the entire code base.

After KySat-1 software development was completed, and all code was reviewed and tested by the flight software team, final testing began. The task was given to new students starting the project. The goal was for them to use the satellite completely, with only the requirements and specifications documents. With any student run project, student turn-over is high, so it is very important that the documentation be complete to pass down ideas. Selecting people who are not familiar with the system for testing ensures that the documentation is complete and thorough. If any gaps or problems were found in the documentation, it was updated appropriately.

During testing, each bug that was found was documented with the date it was found, a description, and the software version number. Every bug found, was promptly corrected, and the test case which revealed the bug was rerun. By tracking the bugs, the software team was able to plot the number of bugs found versus time. In an ideal testing situation, the number of bugs found will increase exponentially in the early stages of testing, but as testing progresses, the system becomes more reliable, and fewer bugs are found. At the end of testing, the bugs found versus time curve should become flat. Figure 53 is a diagram produced from the testing of KySat-1 software, which shows that this behavior was seen during the testing process. As you can see from the plot, the curve of bugs versus time has not completely flattened out. This indicates that more testing needs to take place. The current testing is an on going process and as future environmental and vibration testing of the engineering model is scheduled to take place the software testing will continue in parallel.

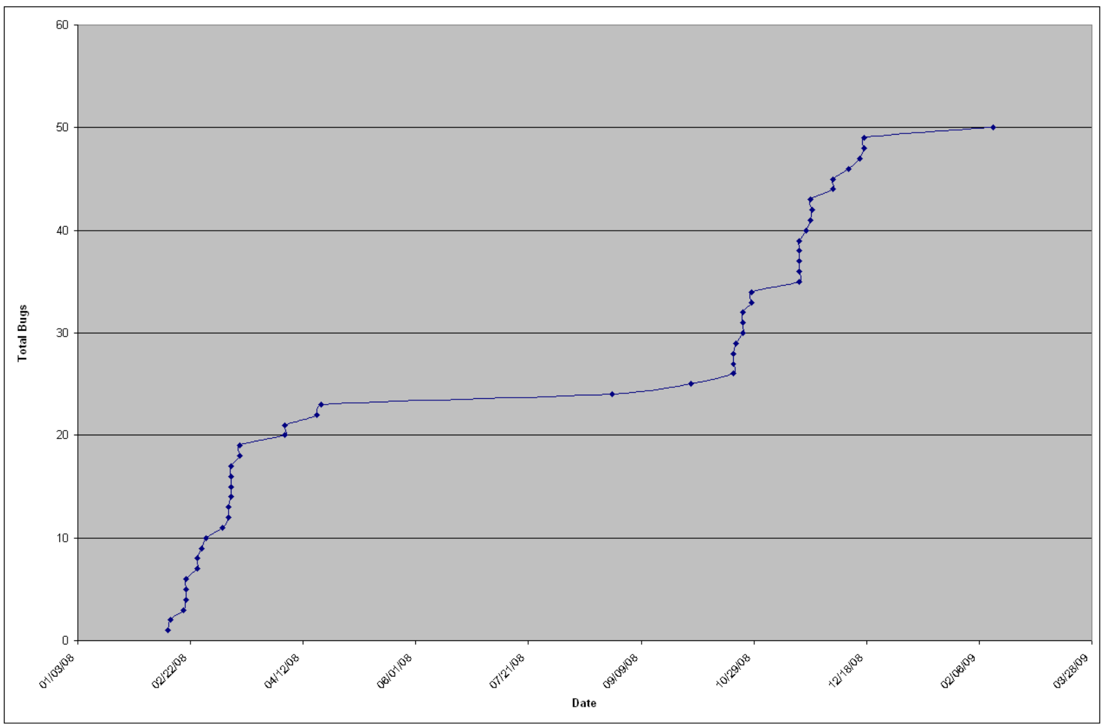

Figure 53 – Software Bugs versus Time

4.5 Future Testing

The Kentucky Space team plans to continue testing the engineering model of the KySat-1 spacecraft until delivery of the flight model to the launch integrator. The team currently plans to perform random vibration testing of the engineering model. The spacecraft will be placed in a representative fixture of the P-POD and this test pod will be bolted to a shake table. The same vibration spectrum as the spacecraft in launch is expected to be seen due to the vibrations of the launch vehicle connected to the table. This will give the team more confidence that something on the spacecraft will not fail during the launch and deployment. After the vibration testing is completed the spacecraft will then be tested in a thermal vacuum chamber. The spacecraft will then be operated as it would be in orbit similar to the thermal environmental testing performed previously, but this will be under a vacuum similar to the space environment. The final step before delivery of the spacecraft to the launch will be a hot vacuum bake out. This bake out will allow any materials or substances to outgas prior to being integrated with other spacecraft. The out gassing and vibration testing are important parts for CubeSats. Because these spacecraft are launched as secondary spacecraft on a mission, the primary spacecraft developer wants to ensure that one of these smaller spacecraft will not do anything to damage the primary spacecraft or jeopardize the primary mission.

5 DISCUSSION

Many challenges were faced and lessons learned during the design, implementation, and integration of the KySat-1 spacecraft bus. This chapter discusses the lessons learned and improvements that can be put in place to future developers of any small spacecraft. This chapter also discusses the upcoming launch opportunity for KySat-1, future spacecraft bus improvements, and future missions proposed to use this spacecraft bus. This project was also completed in a multi-university academic environment which caused problems on top of the typical design and development.

5.1 Design Methodology with Commercially Purchased Subsystems

The KySat-1 spacecraft was design around the methodology of purchasing off the shelf subsystems that were designed to be integrated electrically and mechanically together. This approach was taken to save time in the design cycle of the spacecraft bus. It was initially thought that these components, because they are commercially available, could be purchased and a new bus could be built quickly. However since the CubeSat standard is still relatively new to the aerospace community, the commercial industry is still lagging behind. These commercial suppliers are designing new revisions of their subsystems and older revisions are becoming obsolete. Overall the choice of going forward purchasing as many subsystems as possible did save significant amount of time in the design life cycle, but also added a significant amount of time to the integration and implementation of the spacecraft bus. As the completion and launch of KySat-1 comes closer and testing ends these subsystem are becoming mature and can be trusted to be “plug and play” devices. However the journey to get to this state encountered many ups and downs along the battle. KySat-1 was a spacecraft bus design that leads the way with respect to using commercially purchased subsystems. This mission and all the time put forth by the Kentucky Space design build team will lead the way for future developers to have the ability to implement a spacecraft bus in a timely fashion using commercially purchased subsystems.

5.1.1 Electrical Power System Decision

Since the initial shipment of the EPS from Clyde Space to the Kentucky Space team, it has take on many different revisions mainly due to the integration of KySat-1. The EPS that ended up being used on KySat-1 was the fifth revision, and the batteries were on their sixth revision. Between the fourth and fifth revisions the connector used mate the batteries and the main board changed making them no longer backwards or forwards compatible. The time the team spent integration the power system into KySat-1 was comparable to the time it would have taken to design one internal to Kentucky Space. However if the Kentucky Space team would have invested the time, the intellectual property would belong to Kentucky Space and future spacecraft would be cheaper because this subsystem could be reproduced much cheaper than purchasing from Clyde Space. With looking back at the complexity and final design of the EPS flown on and the schedule of development for KySat-1, going with the purchase of the Clyde Space EPS was the right decision. This EPS will only continue to improve with its increased popularity and be a solid design for years to come as the Clyde Space company grows. This EPS was the first and only product from Clyde Space at initial release, and now the company is selling solar arrays, larger batteries, and larger power systems for 3U CubeSats.

5.1.2 VHF/UHF Radio Decision

Since the initial purchase of the VHF/UHF radio from StenSat, the radio went through three revisions. The radio is now obsolete due to the end of production of the FM communications receiver integrated circuit used as the basis of the VHF receiver. The overall integration time versus time it would have taken to design the transceiver internal to Kentucky Space was not worth the time put forth by the team, especially since the radio is now obsolete. StenSat currently has a simplex UHF transceiver in the design phase but currently is not in production. The performance of the StenSat radio compared to radio designs investigated by the Kentucky Space team is inferior. StenSat is loosing popularity in the CubeSat community due to other transceiver designs by Innovative Solutions In Space (ISIS) [17] and Astronautical Development [18]. These transceiver designs are far superior to the StenSat radio.

5.1.3 S-Band Radio Doppler Shift

Due to the frequency hopping nature of the S-Band radio the link may not always be able to be closed. The S-Band radio will be able to account for a limited amount of doppler shift. The highest elevation passes of the spacecraft directly over head of the ground station will be the passes that encounter the most doppler shift during the start and end of these passes. Shown in Figure 54 is a surface plot of the elevation and azimuth angle versus doppler shift. The passes during which the S-Band radio will be most effective are passes in which the spacecraft has a maximum elevation of forty five degrees. It would be most effective to have an S-Band radio that did not frequency hop. However due to RF knowledge of the design team, designing and building KySat-1 there was not time to design and implement this radio. An S-Band radio with either quadrature phase shift keying (QPSK) or something similar would be a bus improvement for KySat-1. However due to the relative complexity of designing an S-Band radio, the choice to use the commercially available off the shelf Microhard S-Band radio was the correct decision.

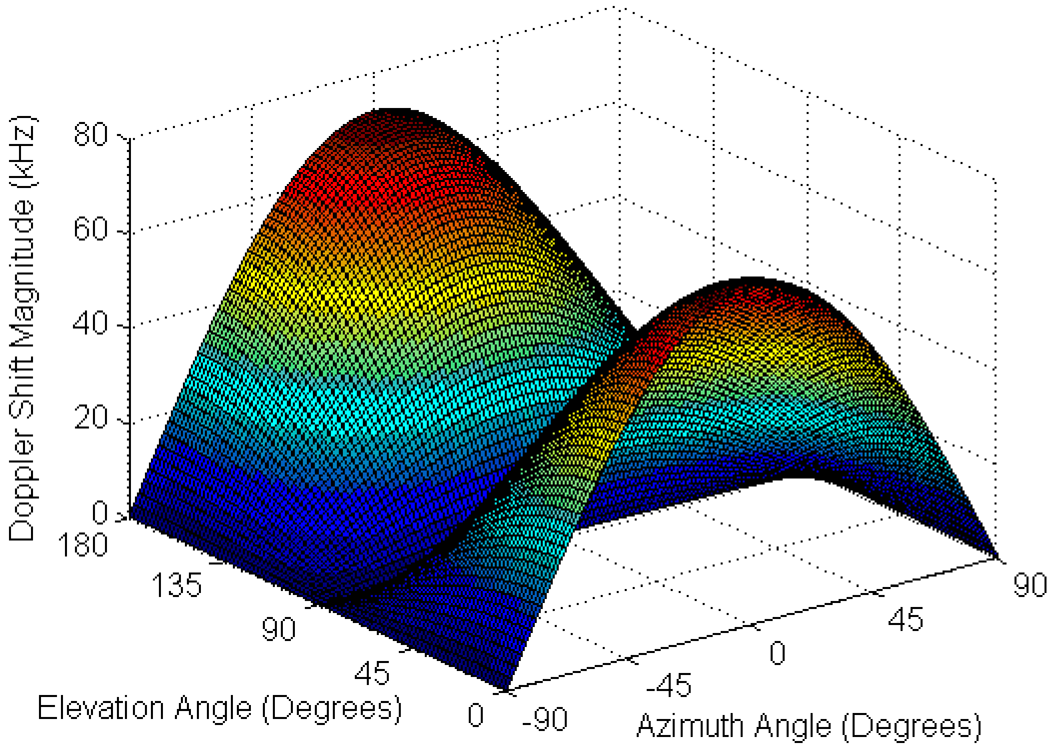

Figure 54 – S-Band Doppler Shift Plot

5.1.4 CubeSat Kit Decision

The CubeSat Kit from Pumpkin Inc., including the Flight Module FM430 and frame assembly are on their fourth revision. The KySat-1 spacecraft bus uses the third revision. The fourth revision still under design and development, the fourth revision includes multiple improvements. The Flight Module FM430 has a new architecture to have to the ability to accommodate many different user defined processor families and is shown in Figure 55. All revisions of the frame and FM430 are both backwards and forward compatible. The choice of used the CubeSat Kit architecture was an extremely beneficial decision in terms of purchase or design your own. Pumpkin is also an increasingly popular company within the CubeSat community. The company will only continue to improve with its increased popularity and be a solid design decision for years to come as they expand.

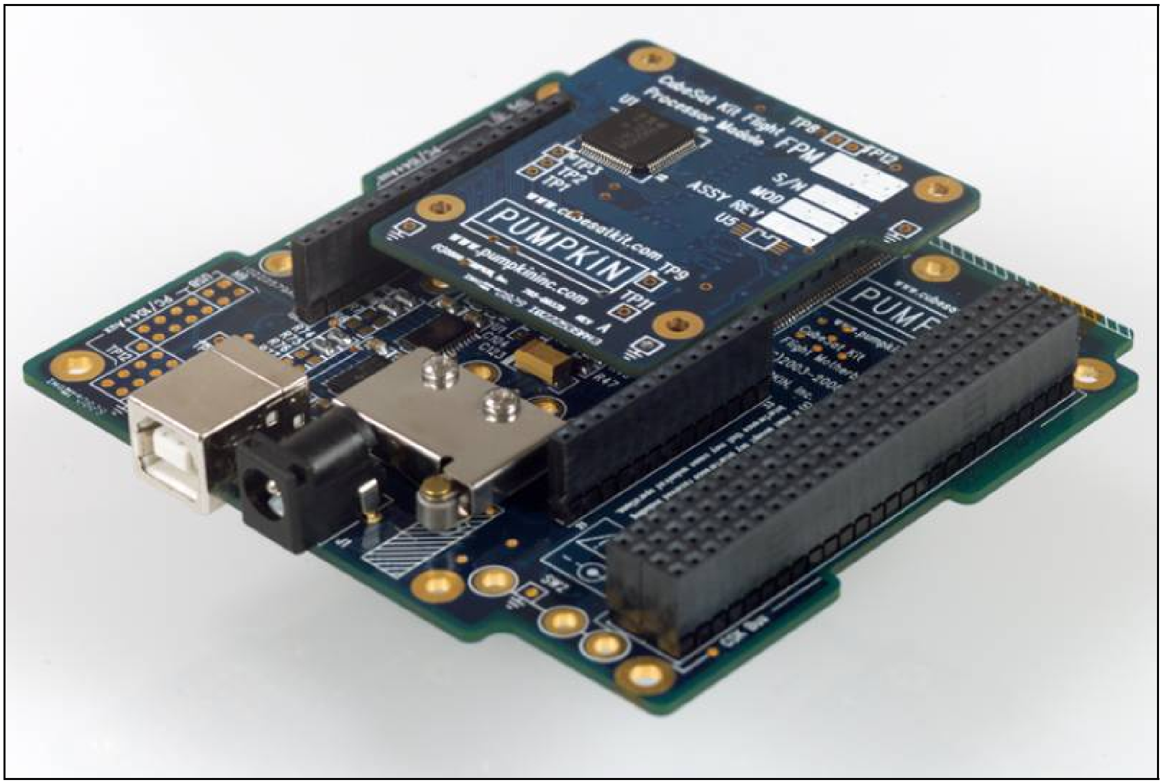

Figure 55 – 4th Generation Pumpkin FM430

5.2 KySat-1 Launch Opportunities

The KySat-1 spacecraft has been down-selected to be a primary CubeSat on the Educational Launch of Nanosatellites (ELaNa) mission. KySat-1, along with Explorer-1 [Prime] (E1P) from Montana State University, Hermes from the Colorado Space Grant Consortium (CoSGC), and SwapSat from University of Florida, Gainesville as a backup will be the first educational CubeSats launch on a NASA mission. At the time of this writing the primary mission has not been officially selected and announced, but the teams have been asked to be prepared for a fourth quarter 2009 launch. The teams are currently working with the NASA Launch Services Program (LSP) from the Flight Projects Office at Kennedy Space Center (KSC) and Cal Poly on the integration of the CubeSats onto the primary launch vehicle. The launch vehicle selected is the Taurus XL from Orbital Sciences Corporation, shown in Figure 56.

Figure 56 – Taurus XL Launch Vehicle

5.3 Spacecraft Bus Advancements

For most of the spacecraft bus design, development, and testing this was the first time most of the students worked on a spacecraft or project of this magnitude. The spacecraft bus is reusable and through testing, the best effort is being put forth to make this current mission a success, but there are improvements and lessons learned throughout the design life cycle.

5.3.1 Electrical Power System Advancements

The first improvement that needs to take place before this spacecraft bus is implemented on another spacecraft would be advancing the power generation of the solar cell arrays. The small TASC cells need to be replaced with much larger cells to have an improved packing factor. These small cells are good for first time developers because they are inexpensive, and large amounts can be purchased to experiment with, but because these cells cannot be pack tightly, the power generation is much lower. For the development of

KySat-1 many cells were damaged due to inexperienced handling of the fragile cells. The next iteration of the spacecraft bus needs to take advantage of the Improved Triple Junction (ITJ) solar cells from Spectrolab. The difference in size can be seen in Figure 57.

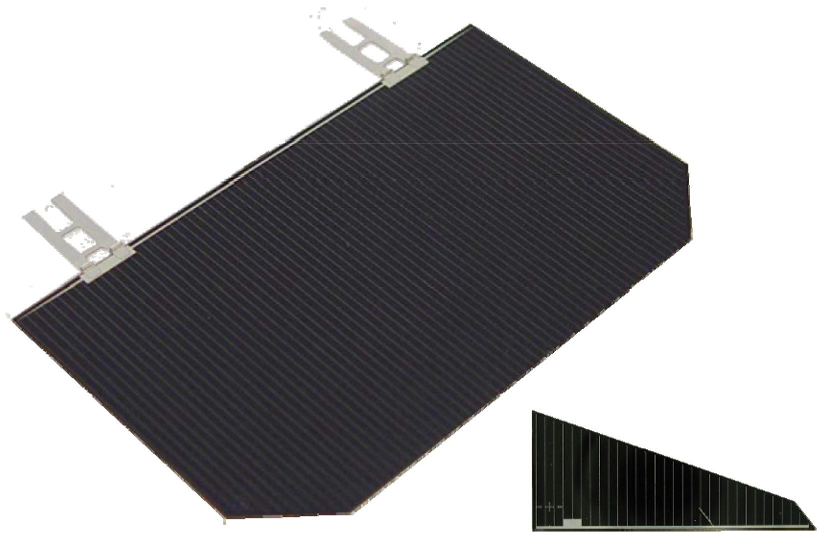

Figure 57 – Spectrolab ITJ and TASC Solar Cells

5.3.2 Communication System Advancements

The next bus advancement has to due with the VHF/UHF communication systems. This advancement will decrease the power consumption of the spacecraft bus. The UHF transmitter needs the ability to change the output power while on orbit. For ground stations with larger gain antennas the transmit power of the spacecraft can be decreased. For ground stations with smaller gain antennas the transmit power can be increased. Also the UHF transmitter needs a separate continuous wave beacon. The current continuous wave beacon is modulated on off audio tones, instead a on off carrier needs to be sent. This will about the continuous wave beacon to be received and decoded at a much lower signal to noise ratio. This continuous wave beacon can be transmitted at a much lower power. Having a separate beacon transmitter and data radio increases the modularity of the spacecraft bus.

5.3.3 Command and Data Handling Advancements

The final spacecraft bus improvements deal with the command and data handling subsystem. Once the spacecraft is completely integrated for delivery, it is not possible to reprogram the flight processor. This means all software testing needs to be completed prior to environmental testing, because the spacecraft must be taken apart to reprogram the flight processor. There are two different solutions to fix this problem. A software boot loader can be written, and the flight processor can be reprogrammed using the USB umbilical. A boot loader would also benefit the software architecture. Currently the spacecraft will be in three different modes. A current state flag and three spacecraft programs can be stored in the EEPROM. With a boot loader, once the spacecraft is turned on it can read the current state and load the correct program. The other solution would be to route the JTAG programming interface to outside of the spacecraft. This could be routed from the Flight Module FM430 to an external connector located on the solar array. Being able to reprogram the flight processor would allow spacecraft software testing to continue up until delivery to the launch integrator. This extra testing would increase the chance of mission success.

5.4 Future Missions and Bus Reuse

The spacecraft bus designed and implemented on KySat-1 was designed to be a reusable spacecraft bus architecture to serve as a basis for future Kentucky Space missions. This has already been implemented in four different proposals submitted by the Kentucky Space Consortium. The first proposal submitted using the KySat bus design was on a slightly larger form factor spacecraft. Polarization Observation Satellite (PolOSat), shown in Figure 58, is a spin stabilized forty five centimeter circular science platform that will fly in a LEO with a low inclination (nominally less than eight degrees) to study the polarization of gamma ray bursts (GRB). PolOSat used six duplicate copies of the KySat bus around the exterior of the spacecraft, with the science instrument inside a shielded cylinder in the center of the spacecraft.

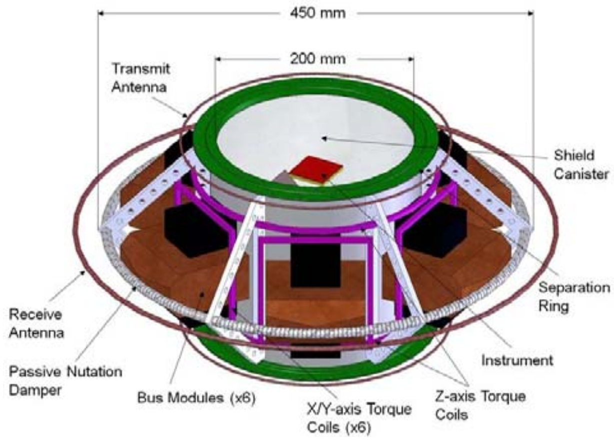

Figure 58 – PolOSat

The KySat bus has been proposed on two CubeSat missions. The first mission was a spacecraft proposed to the National Science Foundation in response to a call for CubeSats to conduct space weather research. The goal of the Danjon CubeSat, shown in Figure 59, is to measure the albedo (global reflectivity) of planet Earth. This 2U spacecraft is based around the KySat bus with four lunar telescopes and an added attitude determination system based on a sun sensor and magnetic attitude control system. The final CubeSat that uses the KySat spacecraft bus design is currently being proposed as a National Experimental Program to Stimulate Competitive Research (EPSCoR) to NASA which combines a NASA payload flown on Genesat-1 and the KySat bus.

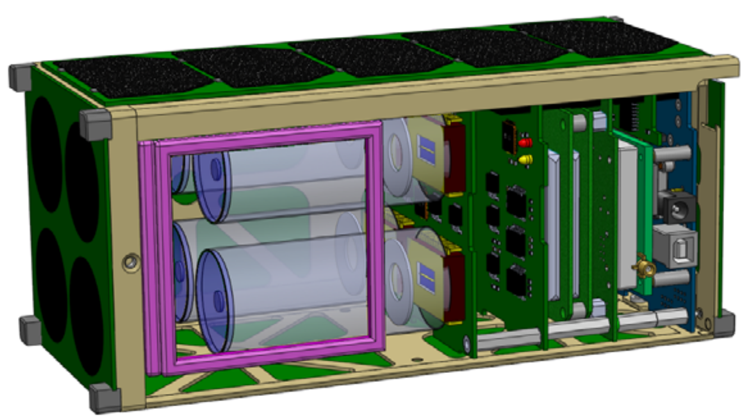

Figure 59 – Danjon 2U CubeSat

6 CONCLUSION

KySat-1 design began in the summer of 2006, the development; implementation and testing are a continuing process. The spacecraft bus was developed to be reusable and to form a basis for a bus for future missions. Portions of the KySat-1 software and hardware have already been reused for three sub-orbital missions, which took place along with the KySat-1 development. The software and hardware that was reused in these missions was done so with typical modifications, and in all three missions the software and hardware performed without problems.

This thesis has overviewed the KySat-1 spacecraft bus architecture design, development and testing. The spacecraft developers learned many lessons about meeting deadlines, interfacing with a team of developers, fault tolerance, design for reuse, using documentation to pass down ideas and much more. KySat-1 will launch in the late 2009. To follow the progress of KySat-1, and other Kentucky Space missions, visit www.kentuckyspace.com.

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VITA

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